28 FEBRUARY 1977

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SPACE STATION SYSTEMS ANALYSIS STUDY
PART 2 FINAL REPORT
VOLUME 3
Appendixes
Book 2
Supporting Data
(7 through 18)

MCDONNELL DOUGLAS ASTRONAUTICS COMPANY

CONTRACT NO. NAS 9-14958
DPD NO. 524
DR NO. MA-04

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PREFACE

The Space Station Systems Analysis Study is a 15-month effort (April 1976 to June 1977) to identify cost-effective Space Station systems options for a manned space facility capable of orderly growth with regard to both function and orbit location. The study activity has been organized into three parts. Part 1 was a 5-month effort to review candidate objectives, define implementation requirements, and evaluate potential program options in low earth orbit and in geosynchronous orbit. It was completed on 31 August 1976 and was documented in three volumes (Report MDC G6508, dated 1 September 1976).

Part 2 has defined and evaluated specific system options within the framework of the potential program options developed in Part 1. This final report of Part 2 study activity consists of the following:

- Volume 1, Executive Summary
- Volume 2, Technical Report
- Volume 3, Appendixes
  - Book 1, Program Requirements Documentation
  - Book 2, Supporting Data
  - Book 3, Cost and Schedule Data

The third and last portion of the study will be a 5-month effort (February to June 1977) to define a series of program alternatives and refine associated system design concepts so that they satisfy the requirements of the low earth orbit program option in the most cost-effective manner.

During Parts 1 and 2 of the study subcontract support was provided to the McDonnell Douglas Astronautics Company (MDAC) by TRW Systems Group, Aeronutronic Ford Corporation, the Raytheon Company, and Hamilton Standard.

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DESIGN CONSIDERATIONS FOR A MARS SAMPLE RETURN LABORATORY MODULE FOR SPACE STATION

DESIGN CONSIDERATIONS FOR A MARS SAMPLE RETURN LABORATORY MODULE FOR SPACE STATION

INTRODUCTION

A rigorous lunar sample quarantine program was established to protect the public's health, agriculture, and other living resources from back-contamination from lunar samples and, in addition, to protect the integrity of the samples themselves and the scientific program associated with them. The program included the planning and development of special quarantine facilities, equipment, and operational procedures, with special emphasis on the design and operation of the Lunar Receiving Laboratory at JSC, where the samples were held and analyzed. These precautions were taken with regard to samples that were given little chance of containing life forms or precursors of living material because of the extreme hostility of the lunar environment. The Martian environment, however, is significantly more compatible with the requirements of life processes, and the precautions taken with regard to returned Mars samples should, therefore, be significantly greater.

The Space Station would appear to afford an almost perfect base for the initial containment and analysis of returned Mars samples, at least through the early quarantine tests and biocharacterization of the samples. The Space Station would be completely isolated from Earth. The Space Station module designed for sample holding and analysis, referred to here as the Mars Sample Return Laboratory (MSRL), would be isolated from the rest of the Space Station, and could be subjected to effective onboard quarantine procedures.

This report is an initial attempt to outline the design considerations for an MSRL, the procedures involved in the acquisition, containment, and quarantine testing of the early Mars samples, and the requirements that these operations would impose on the Space Station.

ASSUMPTIONS AND GUIDELINES

The design and procedures for the laboratory and the requirements for the Space Station are based on the following assumptions and guidelines:

A. The Mars sample(s) is contained in a sealed canister(s) onboard an Earth-orbiting capsule (EOC), which can be retrieved and brought to the space construction base by Shuttle.

B. The canisters can be removed from the EOC and introduced into an isolation chamber in the MSRL without contact with crewmen.

C. The sample canisters are sealed to prevent the loss of both Martian soil and entrapped Martian atmospheric gases.

D. Known quarantine methods and procedures shown to be effective against terrestrial microorganisms will be assumed to be equally effective against Martian life forms.

E. The MSRL module can be completely isolated from the rest of the Space Station.

F. No requirement will exist for the Mars samples or their canisters to be introduced to any part of the Space Station other than the MSRL module; and until the successful completion of quarantine testing, such introduction will be strictly forbidden.

G. No operation on the Space Station will be unduly compromised if it is necessary to isolate the crewmen assigned to the MSRL in that module for extended periods.

H. The MSRL module will contain sufficient capabilities and provisions to maintain three crewmen in isolation from the Space Station for a duration of TBD.

LABORATORY DESIGN AND OPERATIONS

The following paragraphs discuss possible laboratory operations and design characteristics of the MSRL relative to these operations. Figure 1 presents a schematic of the overall MSRL module as a guide for the subsequent discussions.

Sample Canister Acquisition

Some of the activity options available once the EOC with sample canister returns to orbit are shown in Table 1. Upon return from Mars, the EOC sample canister will be placed in earth's orbit (shuttle compatible), to be

FIGURE 1

PLANETARY SAMPLE RETURN LABORATORY SCHEMATIC

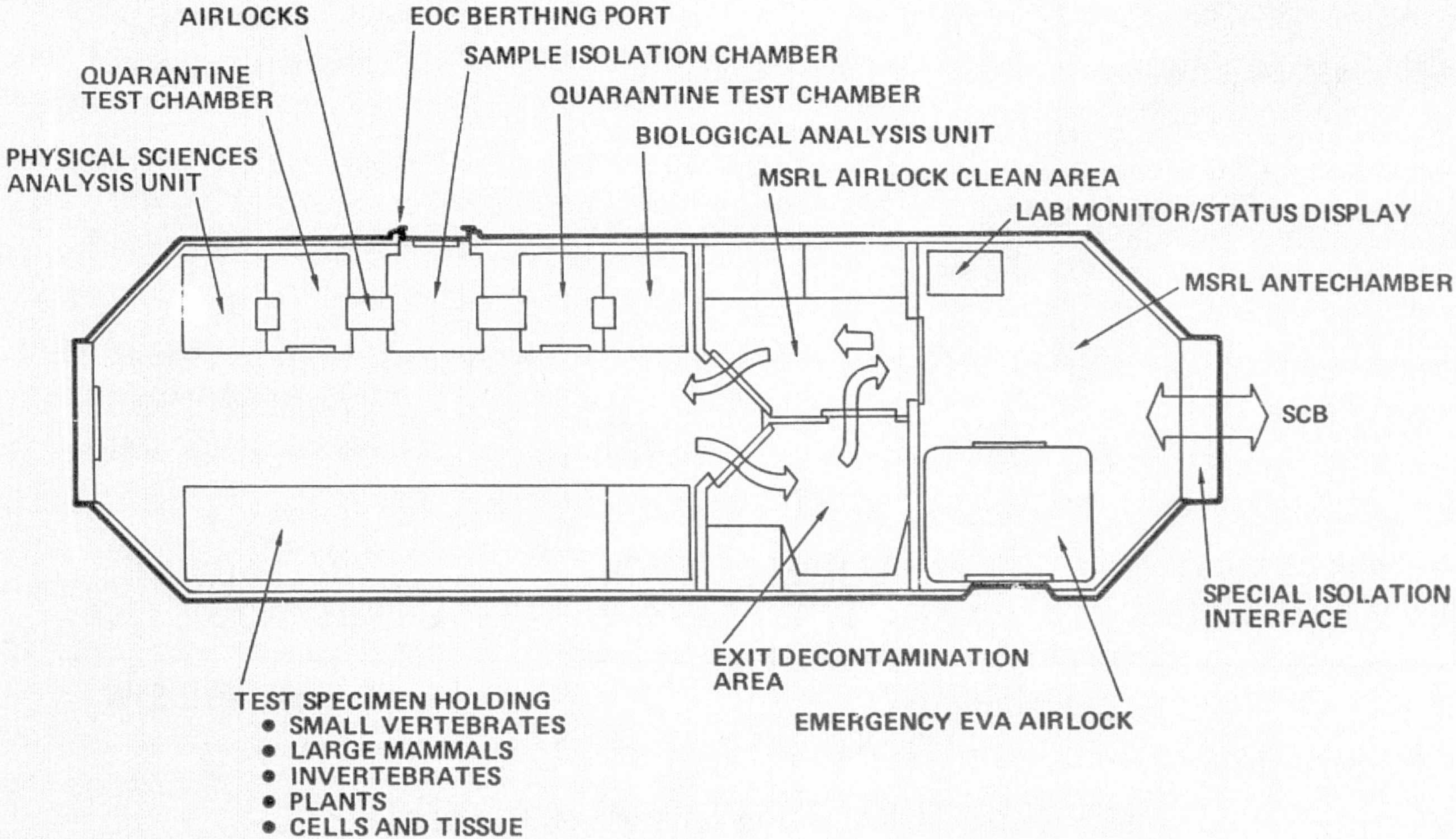

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TABLE 1

SCB RELATED ACTIVITIES IN A MARS SURFACE SAMPLE RETURN MISSION

MISSION ACTIVITY	ALTERNATIVES	
• SAMPLE CANISTER DELIVERY TO STATION	MARS RETURN VEHICLE RENDEZVOUS WITH SCB	✓ SHUTTLE RETRIEVES AND DELIVERS TO SCB
• TRANSFER OF SAMPLE CANISTER TO MARS SAMPLE RETURN LABORATORY	EVA	✓ CRANE
• TRANSFER TO ISOLATION CHAMBER	HAND CARRIED	✓ REMOTE MANIPULATOR
• INSERT INTO TEST CHAMBERS	HAND TRANSFER	✓ REMOTE MANIPULATOR
• TEST OPERATIONS	✓ MANUAL WITH STERILIZABLE AIR LOCK AND GLOVE BOX	REMOTE MANIPULATOR

retrieved by a Shuttle Orbiter and delivered to rendezvous and dock at the SCB. Clearly, this is more advantageous than to attempt a direct rendezvous. Once at the SCB, transfer and handling of the sample canister could be either by EVA or by remote mechanical means. The latter approach is preferred to minimize possible hazards to man. Thus, the sample canister(s) will be removed from the EOC and introduced into the MSRL without coming in contact with crewmen or being exposed to any area of the MSRL other than the Mars Sample Isolation Chamber. The following activities will be conducted relative to the acquisition of sample canisters. These activities are also outlined in Figure 2.

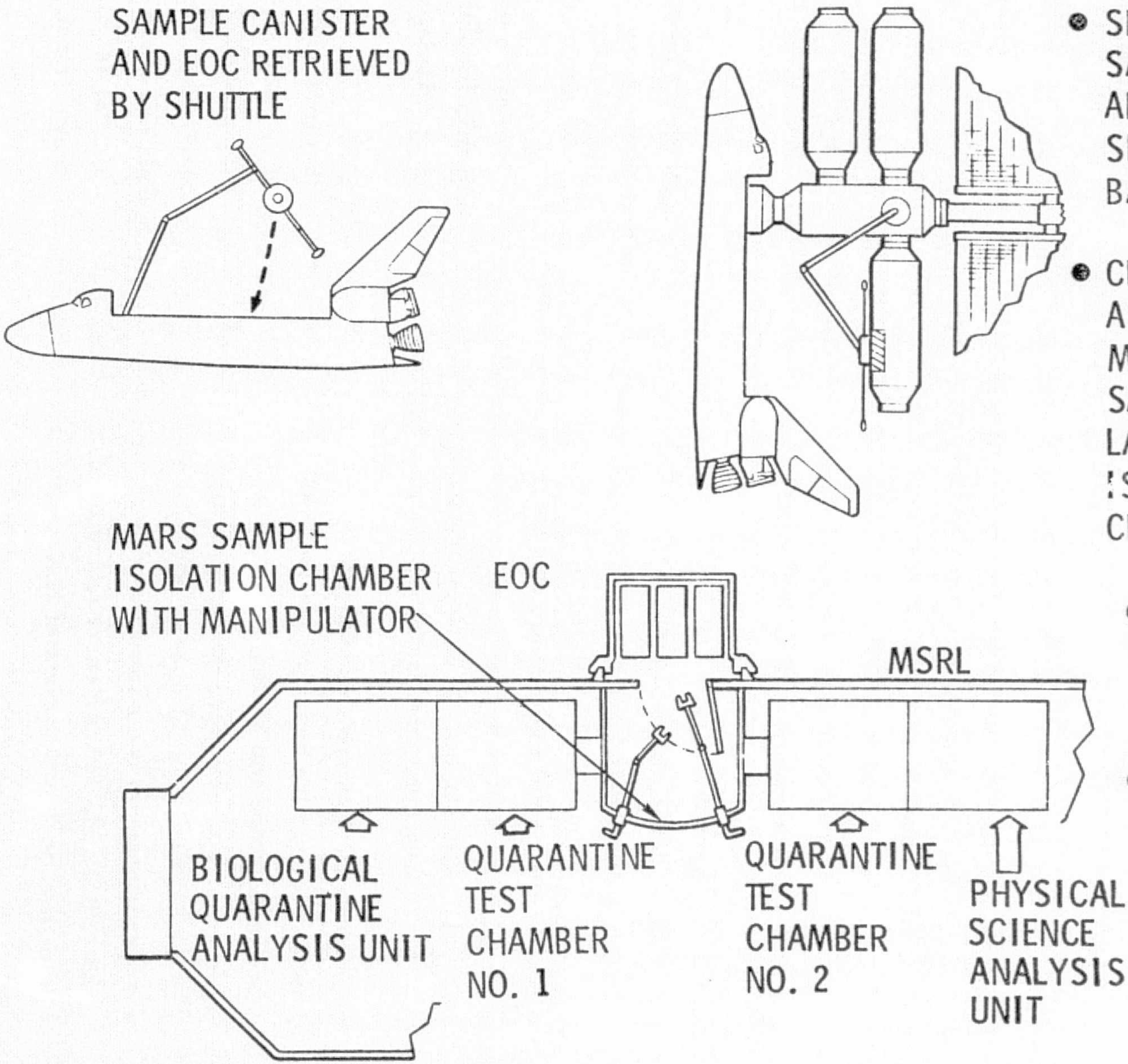
A. The EOC containing the sample canister(s) is retrieved by Shuttle and brought to the space construction base.
B. Crane moves EOC to the Mars Sample Return Laboratory and positions it outside the Mars Sample Isolation Chamber.
C. The sample canister(s) is removed from the EOC and positioned in the isolation chamber by means of remote manipulators located within the chamber. Canister manipulation is controlled by an operator within the MSRL.
D. The isolation chamber/EOC port is sealed; atmospheric gases are introduced whose composition is the same as that of the Martian atmosphere; the pressure and temperature within the chamber are reduced to simulate the Martian environment.
E. No direct access into the Mars Sample Isolation Chamber is provided from the laboratory proper. All activities within the chamber are conducted automatically or by means of remote manipulators.

Laboratory Ingress Procedures

When a crewman assigned to the MSRL proceeds to the laboratory proper, he first enters the MSRL antechamber, then the MSRL airlock, and finally, the laboratory proper, shown in Figure 3. In order to direct airflow away from uncontaminated spaces and into potentially contaminated space, the pressure within the antechamber will be 2 to 3 inches of H_2O pressure below that within the space construction base. The pressure within the MSRL airlock will be 2 to 3 inches of H_2O below that in the antechamber, and the

FIGURE 2

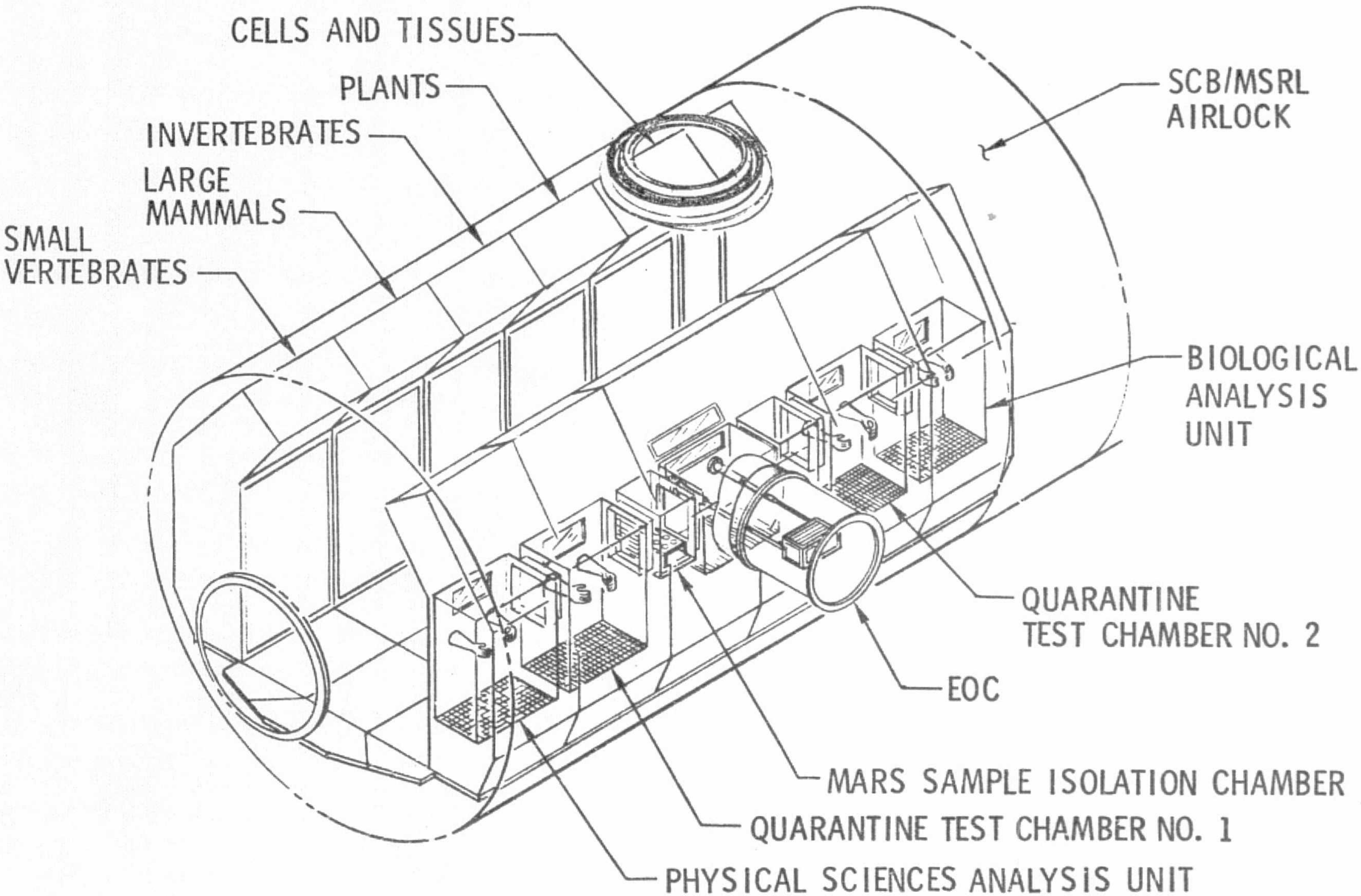
RETRIEVAL OF MARS SAMPLE CANISTER



- SHUTTLE BRINGS SAMPLE CANISTER AND EOC TO SPACE CONSTRUCTION BASE
- CRANE MOVES ASSEMBLY TO MARS SAMPLE RETURN LABORATORY (MSRL) ISOLATION CHAMBER PORT
- SAMPLE CANISTER TRANSFERRED TO ISOLATION CHAMBER
- SAMPLES REMOVED AND PLACED INTO TEST CHAMBERS

FIGURE 3

PLANETARY SAMPLE RETURN LABORATORY



pressure within the laboratory proper will be the lowest, i.e., 2 to 3 inches of H_2O below that in the MSRL airlock. The following procedures are followed for laboratory ingress:

A. Crewman in construction base opens hatch to MSRL, enters antechamber, and reseals hatch.

B. Crewman opens hatch to MSRL airlock, enters airlock, and reseals hatch. The airlock is pictured in more detail in Figure 4.

C. Crewman removes outer garments, places them in designated stowage area, removes laboratory garments from storage, and dons them.

D. Crewman opens hatch to laboratory proper, enters laboratory, and reseals hatch.

Laboratory Egress Procedures

A crewman leaving the laboratory proper to return to the space construction base is considered to be potentially contaminated and will egress through the decontamination area of the MSRL airlock. The following procedures will be followed:

A. Crewman opens laboratory exit hatch, enters decontamination area of MSRL airlock, and reseals hatch.

B. Crewman removes laboratory garments and places them in garment decontamination unit (decontamination method TBD).

C. Crewman enters personnel decontamination unit (decontamination method TBD).

D. Crewman opens hatch to MSRL airlock clean area, enters area, and reseals hatch.

E. Crewman dons outer garments, opens hatch to antechamber, enters antechamber, and reseals hatch.

F. Crewman opens hatch to construction base, enters base, and reseals hatch.

NOTE: The capability is provided for flooding both areas of the MSRL airlock, singly or together, with a disinfectant gas in case of inadvertent contamination. Choice of gas may be TBD, although ethylene oxide appears to be a good choice.

FIGURE 4

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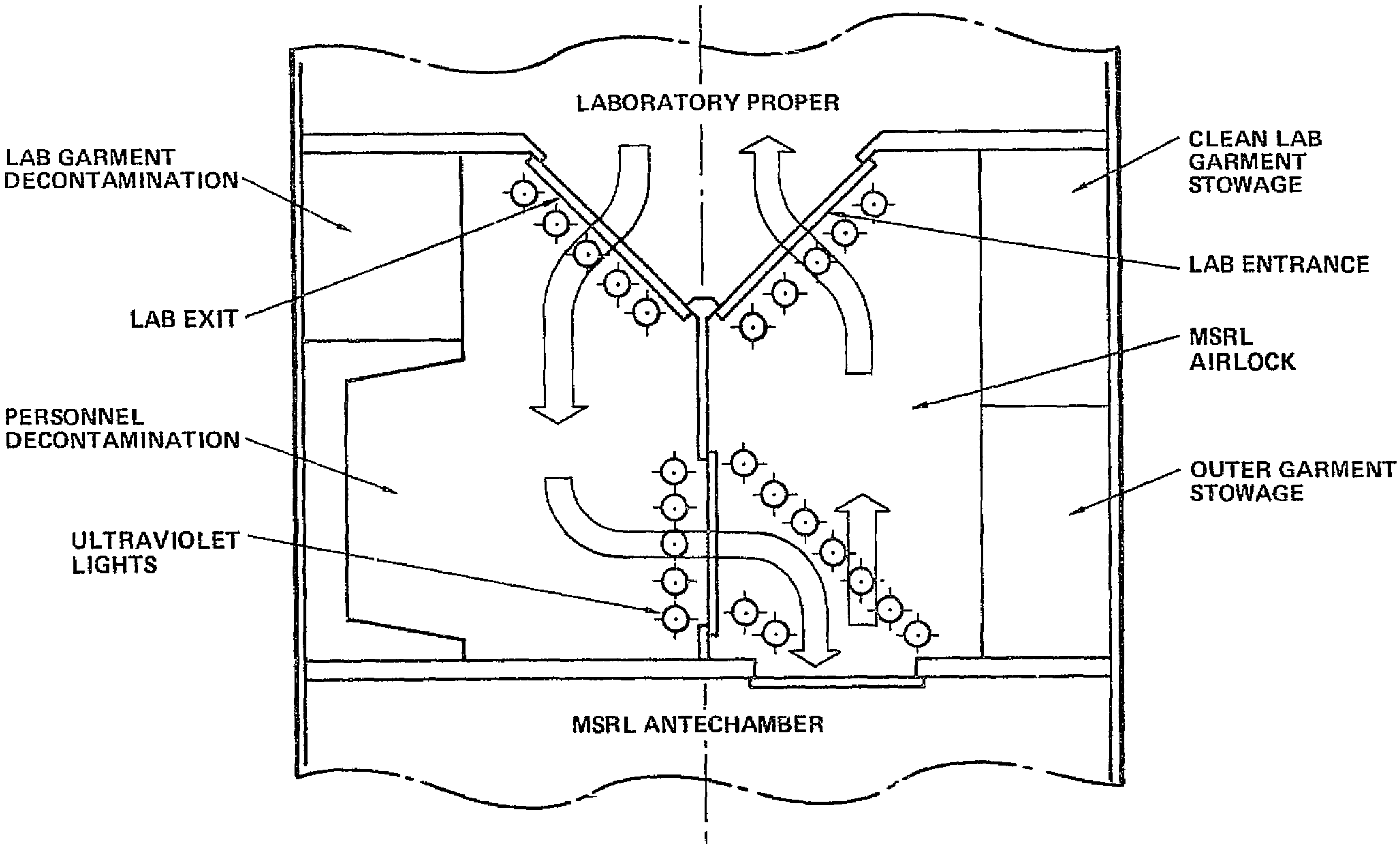
MSRL AIRLOCK CONCEPT



Mars Sample Processing and Initial Analysis

In the Mars Sample Isolation Chamber (shown in Figure 5), selected examination, measurements, and analyses will be performed on the Mars sample(s), while it is exposed to the simulated Martian environment. These activities may include the following:

A. Martian Gas Sample Analysis – Before the Mars sample canister(s) is opened, it may be desirable to draw off and analyze the gaseous contents of each container. Special provisions must be made for this, both in the container design and in the design of the Mars Sample Isolation Chamber. The gases should be drawn through highly refined microbial filters before being analyzed in the Mass Spectrometer/Gas Chromatograph analysis unit. The filters will then be cultured in various nutrient media.

B. Sample Mass Measurement – The mass of the canister and included sample will be measured on a mass measurement device. The known tare weight of the canister may then be subtracted to obtain the mass of the sample.

C. Sample Microscopic Examination – After the canister is opened, small amounts of the sample may be affixed to a microscope slide and placed on a remotely operated substage platform. The microscope may be remotely focused. The visual field may be either projected to a viewing screen or displayed on a video monitor from a video-microscope camera.

D. Sample Culturing –.Measured amounts of the Mars sample may be cultured for microorganisms in various culture media and nutrient broths while exposed to the simulated Martian environment.

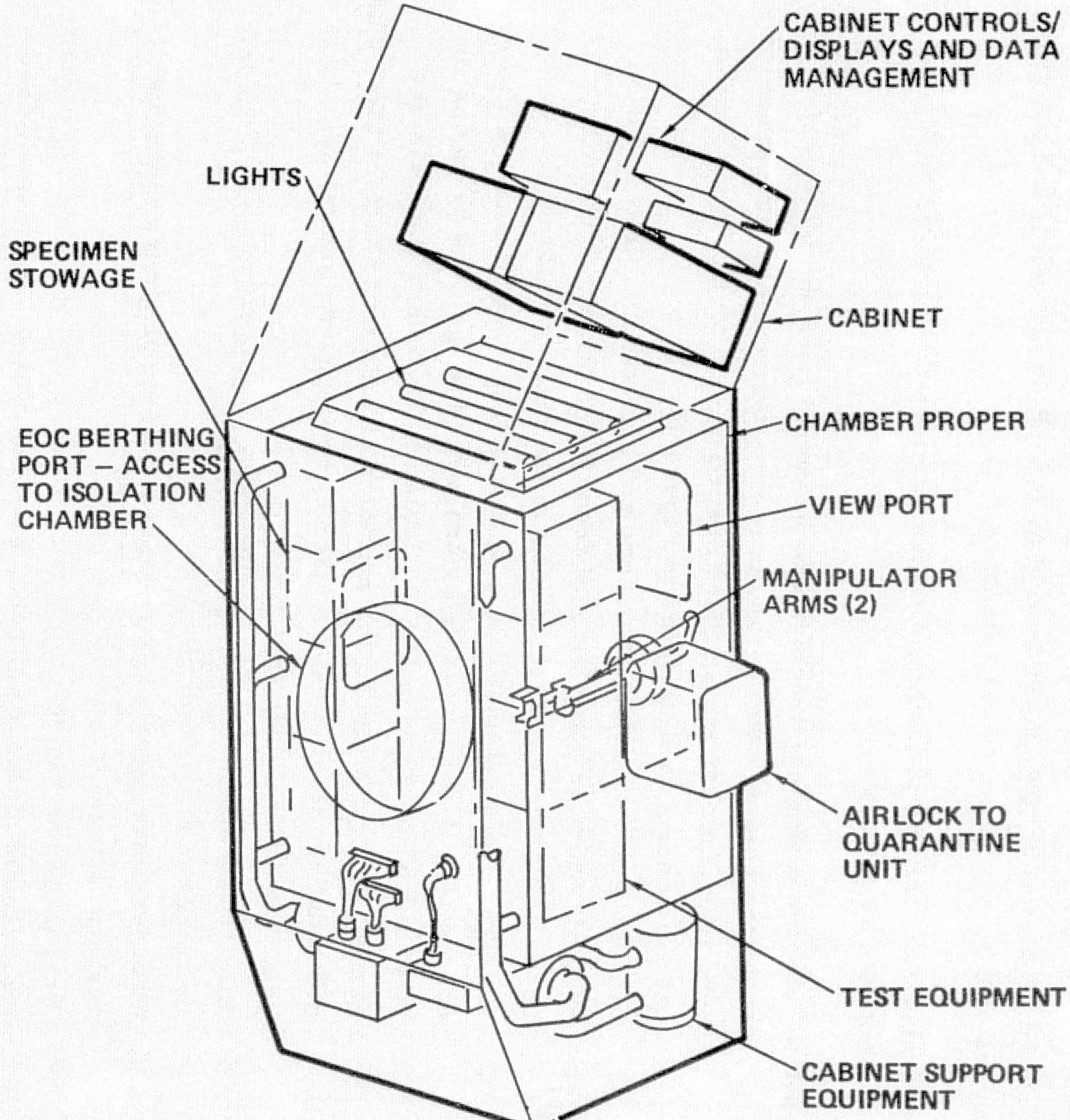
E. Other Biological and Physiochemical Analyses – Various other analyses may be performed; most extensive and complex analyses should, however, be conducted subsequent to quarantine testing and biocharacterization.

Sample Quarantine Testing and Biocharacterization

The majority of initial tests that will be performed on the Mars Sample will be those that will ensure that the samples are totally safe for terrestrial life forms. These tests will involve exposing a large number and variety of plant,

FIGURE 5

SAMPLE ISOLATION CHAMBER



animal, and protist specimens to the samples in various ways and observing the results. It is only after these tests that the major biological and physical analyses will be conducted on the samples.

Processing and handling procedures for the samples are pictured in Figure 6, and outlined as follows:

A. Transfer biological specimens to be tested from the biological specimen holding units to the quarantine test chambers together with all equipment and supplies needed for the tests. A quarantine cabinet concept is illustrated in Figure 7.

B. Seal the quarantine test chambers and activate the air circulation system, which isolates the unit from the MSRL air. The test chamber is now prepared to receive the Mars samples.

C. Transfer Mars samples into quarantine test chambers through interconnecting airlocks. (Samples are placed in airlocks by means of remote manipulators within the isolation chamber, the ports are closed, and the airlocks are pressurized with test chamber air. The port leading into the test chamber is now opened, and the sample is removed with the installed gloves.)

D. All activities within the quarantine test chamber are now conducted by means of the sealed glove ports.

E. Process Mars samples as necessary and supply to the biological test specimens.

F. Reseal the remaining samples and remove biological test specimens by means of an Isolation Transfer Unit. Place specimens in the quarantine specimen holding unit through the transfer unit.

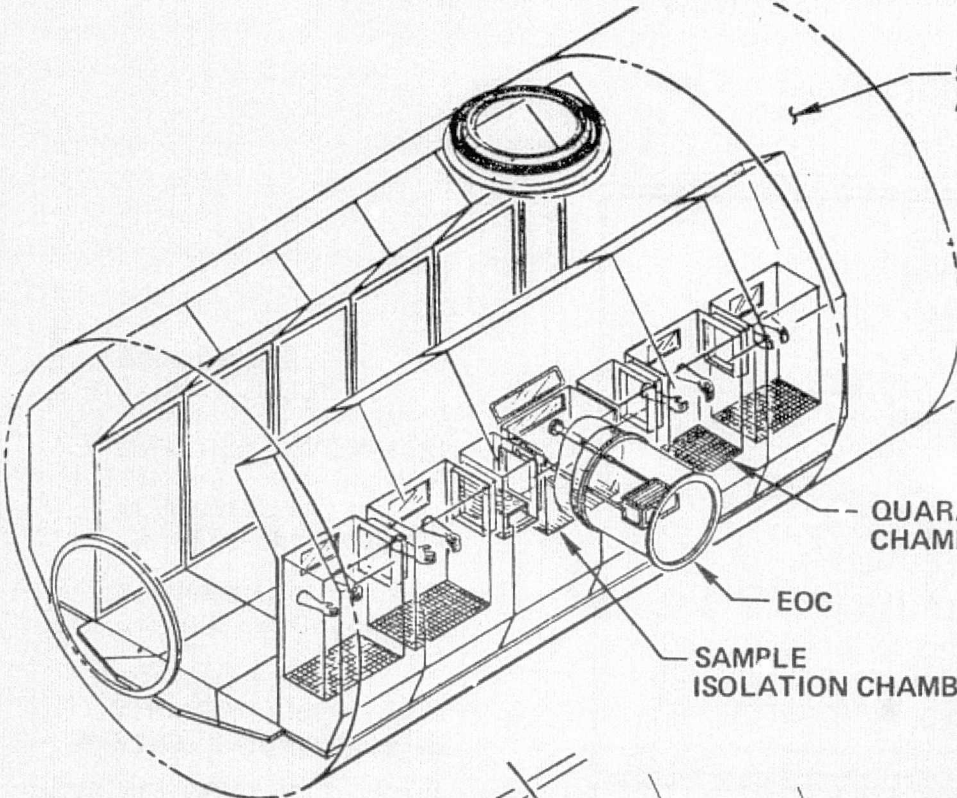
G. Decontaminate transfer unit with disinfectant gas. During all steps of the transfer procedure, specimens should remain isolated from the MSRL atmosphere.

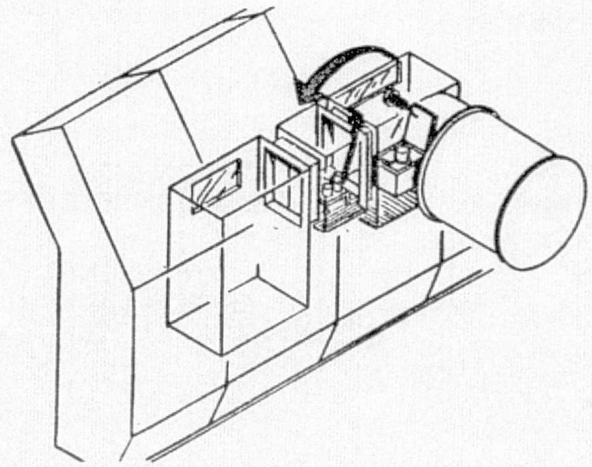
H. Observe specimens for required periods in isolation within the quarantine holding units.

I. Decontaminate the quarantine test chambers with a disinfectant gas and vent to space. Test chambers are now ready for subsequent testing.

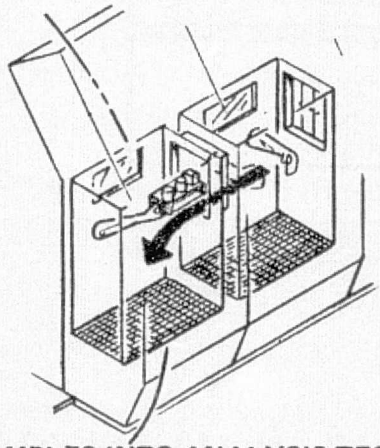
FIGURE 6

PLANETARY SAMPLE PROCESS AND HANDLING

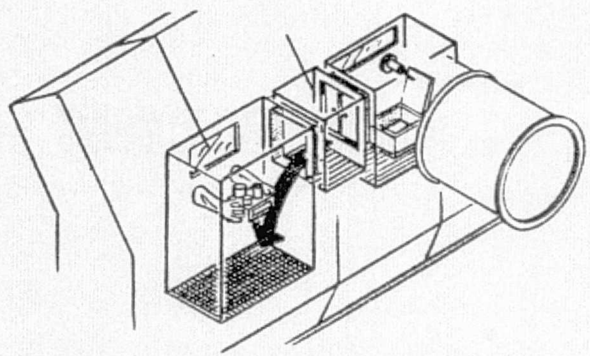




- RECEIVE SAMPLES INTO ISOLATION CHAMBER FROM EOC
- INITIAL SAMPLE TESTING
- MOVE SAMPLES THROUGH AIRLOCK TO QUARANTINE TEST UNIT



- RECEIVE SAMPLES INTO ANALYSIS TEST UNIT
- PERFORM PHYSICAL SCIENCES (AND/OR BIOLOGICAL) TESTS



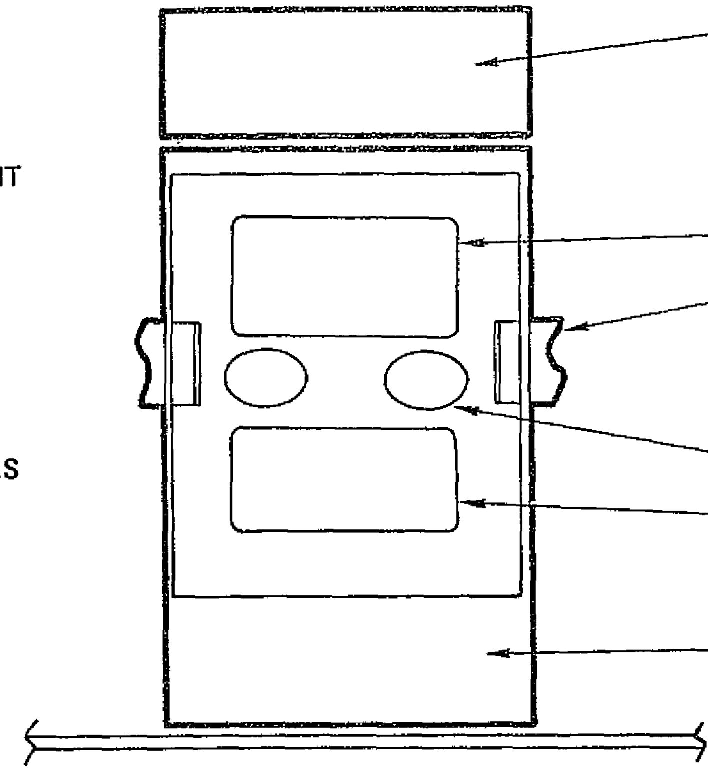
- RECEIVE SAMPLES FROM ISOLATION CHAMBER AIRLOCK
- PERFORM QUARANTINE TESTING

FIGURE 7

QUARANTINE CABINET CONCEPT

CABINET CONTENTS

- SAMPLE MEASUREMENT AND ANALYSIS EQUIPMENT
- SAMPLE PROCESSING EQUIPMENT
- SAMPLE APPLICATION EQUIPMENT
- SPECIMEN CONTAINERS
- STERILIZATION EQUIPMENT



SPACE STATION REQUIREMENTS

The design and operation of the MSRL module will impose certain requirements on the Space Station design and operations, as summarized in Figure 8. These requirements will include the following:

A. Three crewmen/scientists will be required for MSRL operations.

B. One crewman will not be allowed to work alone in the MSRL; two crewmen will be the minimum allowed.

C. MSRL operations will require approximately 16 manhours per day (24-hour period).

D. Quarantine tests and sample biocharacterization will require from 60 to 120 days per sample return. Following these tests, the sample may be exposed to the MSRL atmosphere during additional biological and physical tests.

E. Quarantine sample testing should yield negative results on samples from at least three separate returns before any relaxation of precaution should be allowed.

F. Under normal operations, all power and environmental control (both atmospheric and thermal) for the MSRL module will be supplied by the Space Station.

G. Air returning from the MSRL to the Space Station will be filtered and appropriately disinfected (e. g. , ultraviolet light) to prevent any contamination of the Space Station atmosphere.

H. The MSRL module will contain auxiliary power and environmental control units and provisions sufficient to maintain three men in isolation from the Space Station for a TBD duration during emergency decontamination procedure.

I. Evacuation of the MSRL module in case of unsuccessful emergency decontamination will be made via EVA. The MSRL EVA airlock must contain provisions for the storage of three EMU's. An airlock would not be necessary if MSRL decompression could be assured.

J. The Space Station EVA airlock, through which evacuated crewmen would reenter the Space Station, must contain decontamination capabilities similar to those of the MSRL airlock.

K. Appropriate quarantine procedures must be established in the Space Station for crewmen evacuated from a contaminated MSRL.

FIGURE 8

MARS SAMPLE RETURN LABORATORY (MSRL) REQUIREMENTS

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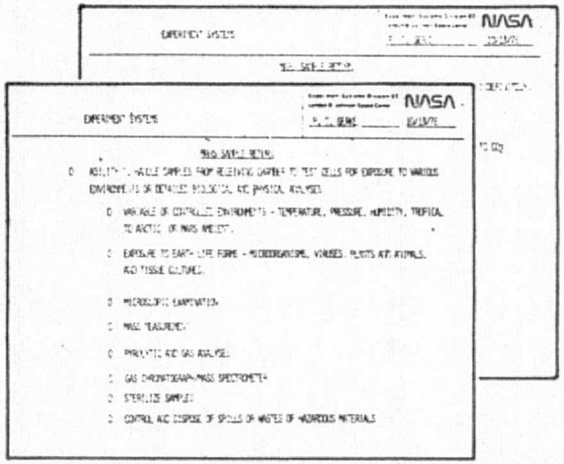
- JSC-SUPPLIED PERFORMANCE REQUIREMENTS



EXPERIMENT SYSTEMS

NASA

MARS SAMPLE RETURN

- ABILITY TO HANDLE SAMPLES FROM RECEIVING CHAMBER TO TEST CELLS FOR EXPOSURE TO VARIOUS ENVIRONMENTS OR DETAILED BIOLOGICAL AND PHYSICAL ANALYSES
  - VARIABLE OR CONTROLLED ENVIRONMENTS - TEMPERATURE, PRESSURE, HUMIDITY, TROPICAL TO ARCTIC OR MARS AMBIENT.
  - EXPOSURE TO EARTH LIFE FORMS - MICROORGANISMS, VIRUSES, PLANTS AND ANIMALS, AND TISSUE CULTURES
  - MICROSCOPIC EXAMINATION
  - MASS MEASUREMENT
  - PYROLYTIC AND GAS ANALYSES
  - GAS CHROMATOGRAPH-MASS SPECTROMETER
  - STERILIZE SAMPLES
  - CONTROL AND DISPOSE OF SPILLS OR WASTES OF HAZARDOUS MATERIALS

- ENVIRONMENT
- COMMUNICATIONS
- TESTS AND ANALYSES
- EQUIPMENT

- STUDY DERIVED FUNCTIONAL/PERFORMANCE REQUIREMENTS
  - ENTRY AIRLOCK AND MANIPULATOR FOR RETRIEVAL OF SAMPLE FROM DOCKED EARTH ORBITING CAPSULE
  - POSITIVE CONTROL OVER PERSONNEL/STATION CONTAMINATION DURING RETRIEVAL
  - SAMPLE EXTRACTION AND INSERTION INTO TEST CHAMBERS DONE WITHOUT DIRECT EXPOSURE TO MAN
  - BENIGN NATURE OF SAMPLE ESTABLISHED BEFORE DIRECT EXPOSURE TO MAN
  - ISOLATION FROM SCB MAINTAINED IN THE EVENT OF FAILURES/ACCIDENTS
  - 3-MAN CREW - 6-MONTH DURATION FOR SAMPLE EVALUATION

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L. The docking interface between the MSRL module and Space Station must be designed to permit the MSRL module to be detached from the Space Station without opening the sealed port between them.

MSRL ACTIVITY TIMELINE

Figure 9 and the following descriptions identify the major activities involved in Mars sample tests and analyses for one Mars sample return period.

Laboratory Preparation Prior to Sample Acquisition

A. Chamber sterilization.
B. Instrumentation test and calibration.
C. Establishment and adaptation of biological specimen colony.

Sample Acquisition

Activities previously described in Sample Canister Acquisition.

Sample Processing and Initial Analysis

Activities previously described in Mars Sample Processing and Initial Analysis.

Quarantine Testing and Sample Biocharacterization

A. Sample preparation for speciment exposure.
   1. Suspension for topical application and injection.
   2. Mixing with specimen food.
   3. Mixing with drinking water.
   4. Mixing in water environment of aquatic specimens.
   5. Mixing into plant nutrient mdeium (soil).
   6. Mixing into nutrient culture medium of microorganisms.

B. Specimen Exposure – Specimens exposed to Mars sample by one or more of above methods. Specimens should include various mammals and other vertebrates (rodents, carnivores, primates, fish, amphibians, and reptiles), invertebrates selected from various phyla (crustaceans, mollusks, insects, etc.), plants representative of disparate families (grasses and grains, legumes, other seed plants, seedless vascular plants, nonvascular plants), various microorganisms (bacteria, algae, molds and fungi, viruses, etc.).

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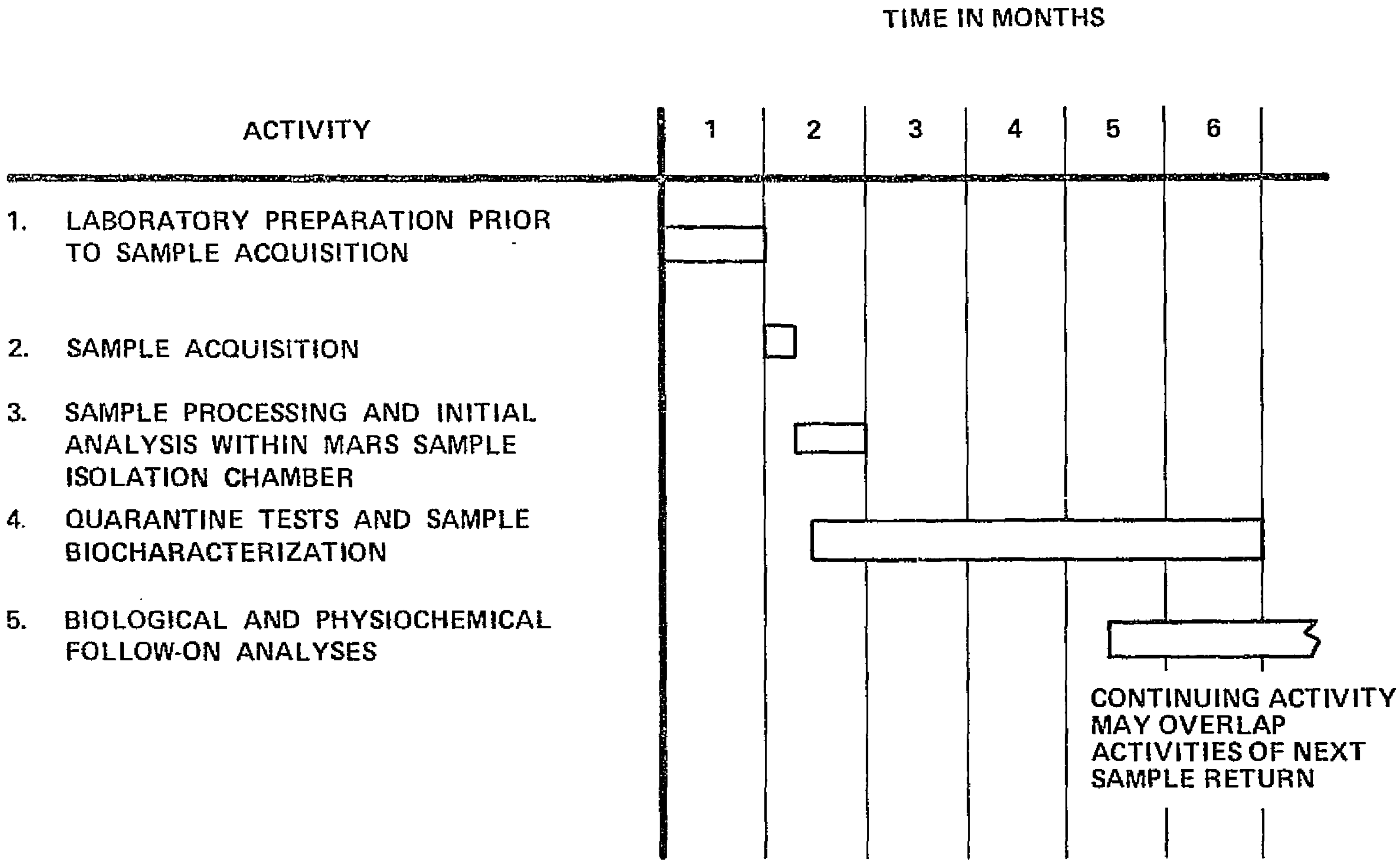
FIGURE 9
MSRL ACTIVITY TIMELINE



C. Specimen transfer to quarantine holding facilities and specimen observation and measurement.

   1. Specimen transfer.
   2. Specimen observation.
   3. Physical examination and measurement.
   4. Physiological tests and measurements.
   5. Sample acquisition and analysis.

Biological and Physiochemical Follow-On Tests

A. Tests in other controlled environments.

B. Pyrolytic and gas analyses.

C. Biochemical analyses.

D. Physical chemical analyses.

Part 8

CREW PRODUCTIVITY AS A FUNCTION OF WORK SHIFT ARRANGEMENT

CREW PRODUCITVITY AS A FUNCTION OF WORK SHIFT ARRANGEMENT

Since the Space Construction Base (SCB) is in orbit to perform specified functions (Construction, Space Processing, Experimentation), it is important that the maximum amount of productive work be done for each day the Space Station is in orbit and for each hour the crew is in orbit. The goal of any productivity effort must be maximum product output for the least cost (hours and dollars).

Station productivity may be partially defined by station use, the number of productive Space Station hours (elapsed time when construction activities are being actually performed), divided by the number of hours the station is in orbit, that is:

$$\text{SU (station use)} = \frac{\text{PH (productive hours, elapsed time)}}{\text{N (number of hours on orbit)}} \qquad (1)$$

Crew productivity may be defined as the number of productive hours the crewman puts in, divided by the number of hours he is available for productive work. For the EVA construction worker, productive work hours equals the number of hours he is actually EVA. A simple figure of merit per duty tour can be derived by the following formula:

$$CP_T = \frac{AH - OHH - LT}{AH} \qquad (2)$$

where:

CP_T = crew productivity for one tour (180 days)

AH = available hours (from groundrules = 10 hours per day)

OHH = overhead hours associated with the specific job. For EVA construction worker, OHH would include the following:

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A. Hours required for briefing to next shift and from preceding shift.

B. Hours required for pre-EVA (transfer to airlock, suit donning, suit checks, airlock depressurization).

C. Hours required for mid-shift lunch, suit doffing and donning, airlock pressurization and depressurization, personal hygiene, and rest.

D. Hours required for rest stops during EVA.

E. Hours required for post-EVA (airlock repressurization, suit doffing, initiation of suit recharging and drying, replacement of suit components such as batteries).

F. Prebreathing (if required).

LT = Lost time for illness and accidents in hours (3% of AH).

It is realized that the above formula does not account for the quality of the productive work performed. A quality factor could be incorporated into the formula, but at present no criterion data are available on which to determine the magnitude of this factor.

A complete analysis of productivity must consider all crew hours for which the Space Station Program pays, including those spent in training and in rest and recreation (R&R) between tours. The formula for career productivity might resemble the following:

$$CP_C = \frac{(AH_T - OHH_T - LT_T) \times N}{(AH_T \times N) + TH + (RRH_T \times N)} \tag{3}$$

where:

CP_C = crew productivity for 3-year career

AH_T = available hours per 6-month tour

N = number of tours during career

OHH_T = overhead hours per tour

LT_T = lost-time hours per tour

TH = training hours per career (based on 10 hours per day, 6 days per week)

RHH_T = rest and recreation hours between each tour (based on 10 hours per day, 6 days per week)

The first step in assessing construction crew productivity was to assemble a set of groundrules under which construction activities in space will be conducted. The rules are not hard and fast program decisions and are subject to change. They are, however, the basis for the results presented here, and changes in them would affect the conclusions reached.

Following are the groundrules for construction workers used in the present analysis:

- Nominal 6-day work week
- 180-day on-orbit tours
- Station crew, maximum of 12
- 60-day resupply interval
- Rotate one-third of crew each 60 days
- Nominal 10-hour work day
- Construction crew consists of two suited EVA workers plus one crane operator per shift
- Nominally 14 hours off-duty activities each day:

Eating	2. 5 hours
Sleeping	8. 0 hours
Personal hygiene	1. 0 hour
Exercise, recreation, and medical	2. 5 hours

- Control center console manned at all times except when entire crew is sleeping simultaneously
- Maximum of 6 hours actual EVA per crewman per day
- During actual EVA, a 2-hour break (lunch, rest, and personal hygiene) required after each 3 hours of EVA
- During actual EVA, a 10-minute rest period in suit is required approximately each 2 hours

- Construction workers will be recruited for 3-year careers (four 180-day tours)
- Training of construction workers will require 480 hours (3 months)
- Construction workers will be given 90 days of R&R between each 180-day tour

Using the foregoing groudrules, time lines were developed for the following three-shift arrangements:

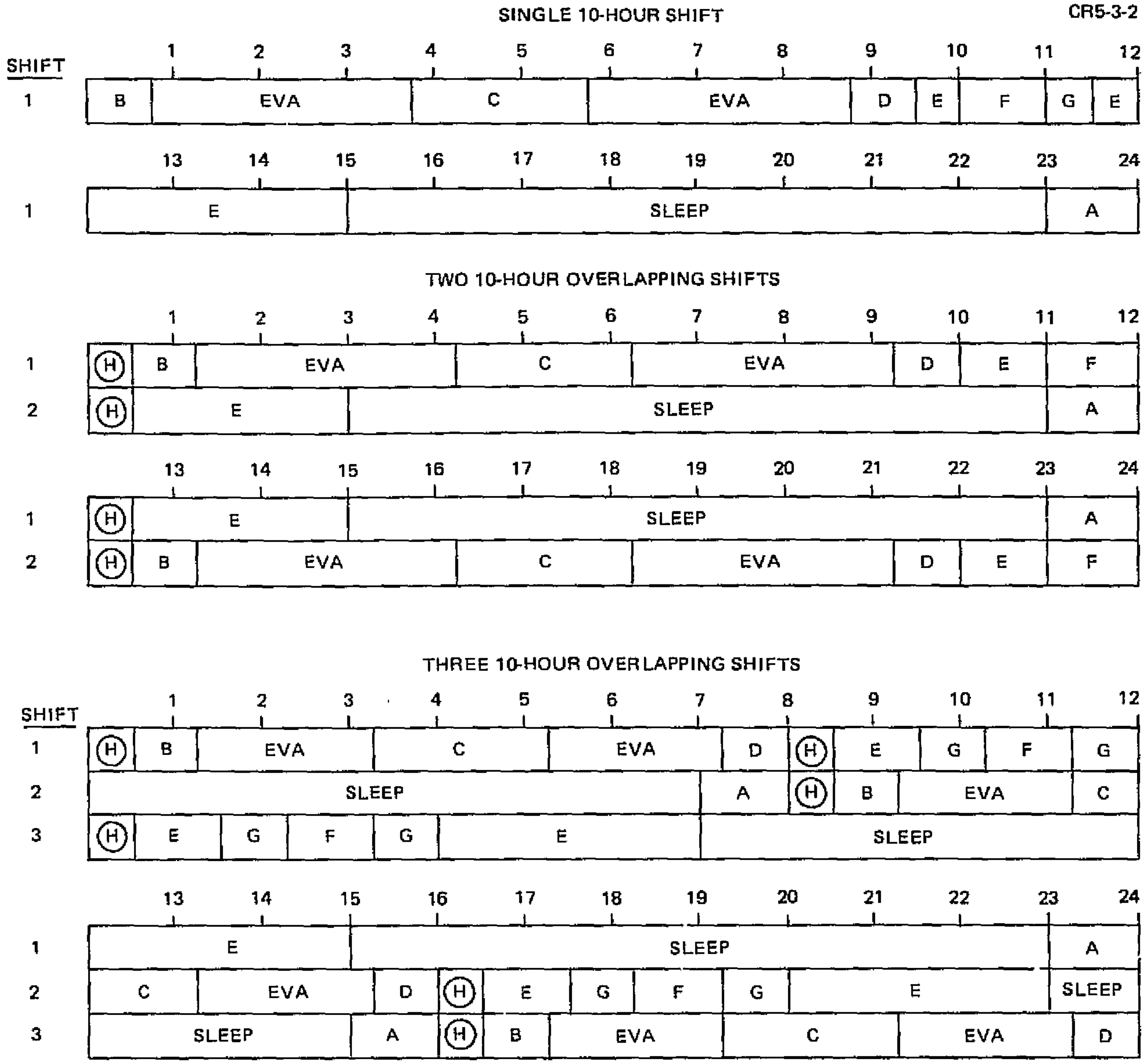
A. One 10-hour shift per 24 hours.

B. Two overlapping 10-hour shifts per 24 hours.

C. Three overlapping 10-hour shifts per 24 hours.

Twenty-four hour schedules for the three shifts are presented in Figure 1. The time blocks, though divided into 15-minute segments, represent estimates of the times required for such things as pre-EVA and post-EVA and are, in general, conservative. Actual time will in most cases probably be less, thus allowing more time for productive work. In the single-shift schedule, each crewman performs 6 hours of EVA each day, in two 3-hour periods separated by a 2-hour break for lunch, personal hygiene, and rest. The schedule is sufficiently flexible to enable extension of the EVA periods to 4 hours each if operational experience indicates that this is feasible. Using the schedule as shown, each crewman has one-half hour of his 10-hour workday available for work not related to construction.

The two-shift schedule is complicated by the necessity for an overlap period at the beginning and end of each shift for a briefing of the crew coming on duty by the crew going off. The time for briefing has been set arbitrarily at 30 minutes, which is probably conservative. In this schedule, the construction crew does not have any time remaining for nonconstruction work. In fact, they actually work 10-1/2 hours rather than 10. There is some flexibility in the schedule in that the EVA periods could be extended beyond 3 hours, but this would necessitate extending the work day to 12 hours.



LEGEND

- A — Personal Hygiene and Breakfast (1 hour)
- B — Transfer and Pre-EVA (45 minutes)
- C — Mid-shift break (lunch, personal hygiene, doff/don suit, rest) (2 hours)
- D — Post EVA (45 minutes)
- E — Exercise, Recreation, Personal Hygiene
- F — Dinner (1 hour)
- G — Nonconstruction Work (Housekeeping, food preparation, maintenance, etc.)
- H — Pre and Post-Shift Briefing (Commander and both construction crews) (30 minutes)

Figure 1. Shifts for 24-Hour Schedule

It was not possible in the three-shift schedule to schedule more than 4 hours EVA per crew and still adhere to the groundrules. Some increase in EVA time can be realized by decreasing the time required for briefings, pre-EVA, and post-EVA. A substantial increase (to 6 hours per day) can be achieved by permitting crewmen to spend 6 continuous hours EVA each day, thus eliminating the need for the 2-hour break between EVA periods. One advantage of the schedule as shown is that each crewman has 1-1/2 hours of his 10-hour work day available for nonconstruction work, thus contributing to a decrease in the number of station support personnel required.

The hours in the Figure 1 schedules were tabulated and are shown in Table 1. It can be seen that actual EVA hours per crewman is the same for one- and two-shift operations, but considerably less for three-shift operations. Total EVA hours for three-shift operations is only slightly more than for two-shift operations, at a penalty of three additional construction crew workers.

Using Formula 1 from the first page, station use can be calculated from the data in Table 1 for a 180-day period as follows:

A. One shift

$$SU = \frac{PH}{\text{hours on orbit}} = \frac{5.67 \text{ hours} \times 154 \text{ days}}{24 \text{ hours} \times 180 \text{ days}} = \frac{873.18}{4,320} \doteq 0.20$$

B. Two shifts

$$SU = \frac{5.67 \text{ hours} \times 2 \times 154 \text{ days}}{24 \text{ hours} \times 180 \text{ days}} = \frac{1,746.36}{4,320} = 0.40$$

C. Three shifts

$$SU = \frac{4 \text{ hours} \times 3 \times 154 \text{ days}}{24 \text{ hours} \times 180 \text{ days}} = \frac{1,848.0}{4,320} = 0.43$$

Station use for the three-shift arrangement would obviously be enhanced (to 0.64) if each crew worked 6 hours EVA per shift.

Table 1

DISTRIBUTION OF CONSTRUCTION WORKER* HOURS FOR VARIOUS WORK SHIFT ARRANGEMENTS (PER 24-HOUR DAY)

	One Shift		Two Shifts		Three Shifts	
	Hours Per Crewman	Total Hours	Hours Per Crewman	Total Hours	Hours Per Crewman	Total Hours
Construction Work						
Pre-EVA	0.75	2.25	0.75	4.5	0.75	6.75
Post-EVA	0.75	2.25	0.75	4.5	0.75	6.75
Actual EVA	5.67	17.0	5.67	34.0	4.0	36.0
Mid-Shift Break (Lunch/rest/personal hygiene/pre- and post-EVA)	2.0	6.0	2.0	12.0	2.0	18.0
10-Minute rest periods	0.33	1.0	0.33	2.0	0	0
Pre- and Post-Shift Briefings	0	0	1.0	6.0	1.0	9.0
Total Construction	9.5	28.5	10.5	63.0	8.5	76.5
Non-Construction Work	0.5	1.5	0	0	1.5	13.5
Total Work	10.0	30.0	10.5	63.0	10.0	90.0
Off-Duty Activities						
Breakfast and Personal Hygiene	1.0	3.0	1.0	6.0	1.0	9.0
Dinner	1.0	3.0	1.0	6.0	1.0	9.0
Exercise, Recreation, and Personal Hygiene	4.0	12.0	3.5	21.0	4.0	36.0
Sleep	8.0	24.0	8.0	48.0	8.0	72.0
Total Off-Duty	14.0	42.0	13.5	81.0	14.0	126.0
Total	24.0	72.0	24.0	144.0	24.0	216.0

(*) Three construction workers per shift: 2 suited EVA workers, 1 crane operator.

Individual crewman productivity can be calculated using the data from Table 1 and applying the groundrules listed previously. Use of Formula 2 for the three different shift arrangements provides the following figures of merit:

A. One shift

$$CP_T = \frac{AH - OHH - LT}{AH} = \frac{1,540 - 589.82 - 46.2}{1,540} = 0.59$$

B. Two shifts

$$CP_T = \frac{1,617 - 743,82 - 48.5}{1,617} = 0.51$$

NOTE: AH = 1,617 hours rather than 1,540 because the crew actually works $10\frac{1}{2}$ hours per day.

C. Three shifts

$$CP_T = \frac{1,540 - 693 - 46.2}{1,540} = 0.52$$

It is obvious from the preceding that the number of overhead hours (OHH) strongly influences crew productivity. If, for instance, the 2-hour break between EVA's (now charged to overhead) could be eliminated, the resulting productivity ratios for one-, and two-, and three-shift operations would increase to 0.79, 0.70, and 0.72, respectively.

In the three formulas above it should be noted that only the two-shift productivity figure (CP_T = 0.51) is pure construction work productivity. Both the one-shift and the three-shift numbers (CP_T = 0.59 and 0.52) include some nonconstruction work (station operations) productivity. If the nonconstruction work were subtracted from the numerator of each formula, the CP_T for one-shift and three-shift operations would become 0.54 and 0.37, respectively.

Given the groundrules listed, crew size for the SCB as a total station is a function of the number of crewmen required for production operations (construction, space processing, research), plus the number of crewmen required for station operations to support production. Table 2 summarizes the requirements, hours per day, for station operations to support the construction activities described in Figure 1 and Table 1.

Using the estimated hours in Table 2, total crew size for the SCB was developed and is shown in Table 3.

Station use and individual crew productivity were plotted and are shown in the top curves of Figure 2. Though station use for two shifts is double that for one shift, it increases only slightly for three shifts because each construction crewman performs only 4 hours EVA on three shifts as opposed to the approximately 6 hours for the other two shifts.

Table 2

HOURS PER DAY REQUIRED FOR STATION OPERATIONS TO SUPPORT CONSTRUCTION

	One Shift (Hours/Day)	Two Shifts (Hours/Day)	Three Shifts (Hours/Day)
Man Control Station**	16.0	24.0	24.0
Food Preparation	1.5*	2.5	3.5
Scheduled Maintenance	3.0*	3.0	3.0
Unscheduled Maintenance	4.0*	4.0	4.0
Housekeeping	5.0*	6.0	7.0
Trash Collection and Disposal	0.5*	1.0	1.5
Cargo Handling	0.3*	0.6	1.0
Crew Medical Care	0.3*	0.5	0.7
Berthing and Unberthing	0.2*	0.2	0.2
Space Suit Support	1.5	3.0	4.5
Totals	32.3	44.8	49.4

(*) NAA Phase B Baseline estimates

(**) Includes hours allocated for station command, manual navigation, communications, subsystem management, data management, and inventory control.

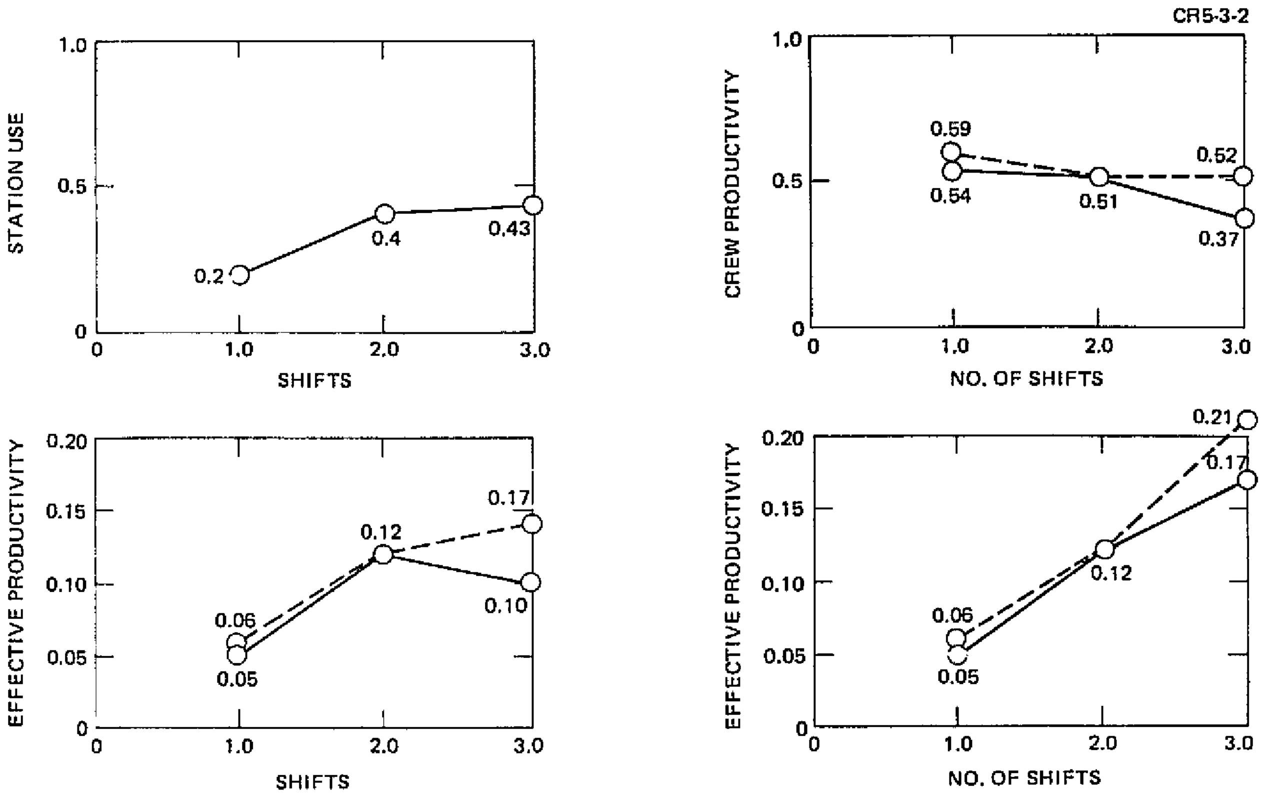


Figure 2. Relative Productivity for Single vs Multiple-Shift Construction Operations

Table 3

SCB CREW SIZE

	One Shift	Two Shifts	Three Shifts
Hours per day required for station operations (Table 2)	32.3	44.8	49.4
Less the hours per day contributed by construction crew	1.5	0	13.5
Support crew hours required	30.8	44.8	35.9
Number of support crewmen required (10-hour work day)	3.08	4.48	3.59
Actual support crewmen required	3*	4*	4*
Number of construction workers	3	6	9
Total SCB Crew	6	10	13

(*) Drop fractional crewman if equal to or less than 0.5.

NOTE: Above numbers in both tables are for one day of a 6-day work week. The seventh day will be a minimum work day with only control station manning, food preparation, and mandatory maintenance being performed.

Crew productivity is plotted for both "pure" construction crew productivity (solid line), which does not include station operations productive work, and for overall productivity (dotted line), which gives the crewman credit for all productive work performed during his 10-hour work day.

Effective productivity (EP) was calculated using the following formula:

$$EP = \frac{SU \times CP \times N_C}{N_C + N_S} \tag{4}$$

N_C = Number of construction crewmen

N_S = Number of support crewmen

The effective productivity curve on the lower left of Figure 2 is based on the time allocations given in Table 1 and shows effective productivity as the product of station use, crew productivity, and number of construction workers, divided by the total station crew size. The points on the solid line were computed using "pure" construction crew productivity, while the points on the dotted line used overall productivity. For two-shift operations, these points are the same because the crews do not perform any nonconstruction productive work.

The effective productivity curve on the lower right of Figure 2 shows the result of making what appears to be a reasonable change in the three-shift timeline. It was assumed that pre-EVA and post-EVA activities for the overlapping crews could be performed simultaneously, that pre and post-shift briefings could be done at the same time as pre-EVA and post-EVA, and that the time required for these activities at each shift change would be 1 hour. This arrangement permits each of the construction crews on the three-shift schedule to work 5 hours EVA per shift and reduces the overhead for each crew by 15 minutes per man, resulting in a dramatic increase in effective productivity. With this timeline change, the station use number for three shifts increases to 0.53, and the crew productivity numbers change to 0.47 for "pure" construction productivity and 0.57 for overall productivity.

SUMMARY AND CONCLUSIONS

- One-shift working arrangements for construction workers provide the highest individual crew productivity but the lowest station use.
- One-shift operations suffer the highest penalty in proportion of support crew required -- 100%. One support crewman is required for each construction worker.
- Three-shift arrangements suffer the least penalty in proportion of station support personnel required -- 44%. Only 4 station crewmen are required to support the 9-man construction crew, while an identical number, 4, is required to support the two-shift, 6-man construction crew.
- Two-shift working arrangements appear optimum in terms of both station use and individual crewman productivity, both being only slightly lower than for three-shift operations while requiring a smaller total crew size. The two-shift operation also has more flexibility to accommodate longer periods of EVA (if they are later found feasible) than does the three-shift arrangement.

Part 9

A PRELIMINARY ANALYSIS OF THE LOCAL LOGISTICS PROBLEM ON THE SPACE CONSTRUCTION BASE

Section 1
INTRODUCTION

This write-up documents a preliminary analysis of the local logistics problem on the Space Construction Base (SCB). The problem is basically twofold. First, how do you move modules from the Shuttle Orbiter Bay to the desired SCB ports and berth them? Second, how do you move assembly parts from the canister module to their final assembly position? An associated question is: how do you replace a module or assembly part should this become necessary? The problem is compounded by the ground rule that the Shuttle is only allowed to dock at one port on the end of the SCB. This port would be the one in the lower left hand corner (No. 12) in Figure 1-1 which shows a typical SCB configuration.

One might think of many possible ways of getting a module from this port to the opposite end of the SCB, i. e., rails, wires, etc. After qualitatively considering a number of concepts, three were accepted as worthy of a preliminary analysis. The three concepts are the mini-tug, the fixed crane, and the mobile crane. The mini-tug is a small, highly reliable, manned vehicle which is capable of safely maneuvering large objects in the close vicinity of the SCB. The fixed crane is positioned at a SCB port and is capable of extending a long mechanical arm to grasp and move objects. The mobile crane utilizes two shorter mechanical arms to move about the SCB like a two-legged spider. Each of these concepts will be considered in the sections which follow, including discussions of requirements and feasibility.

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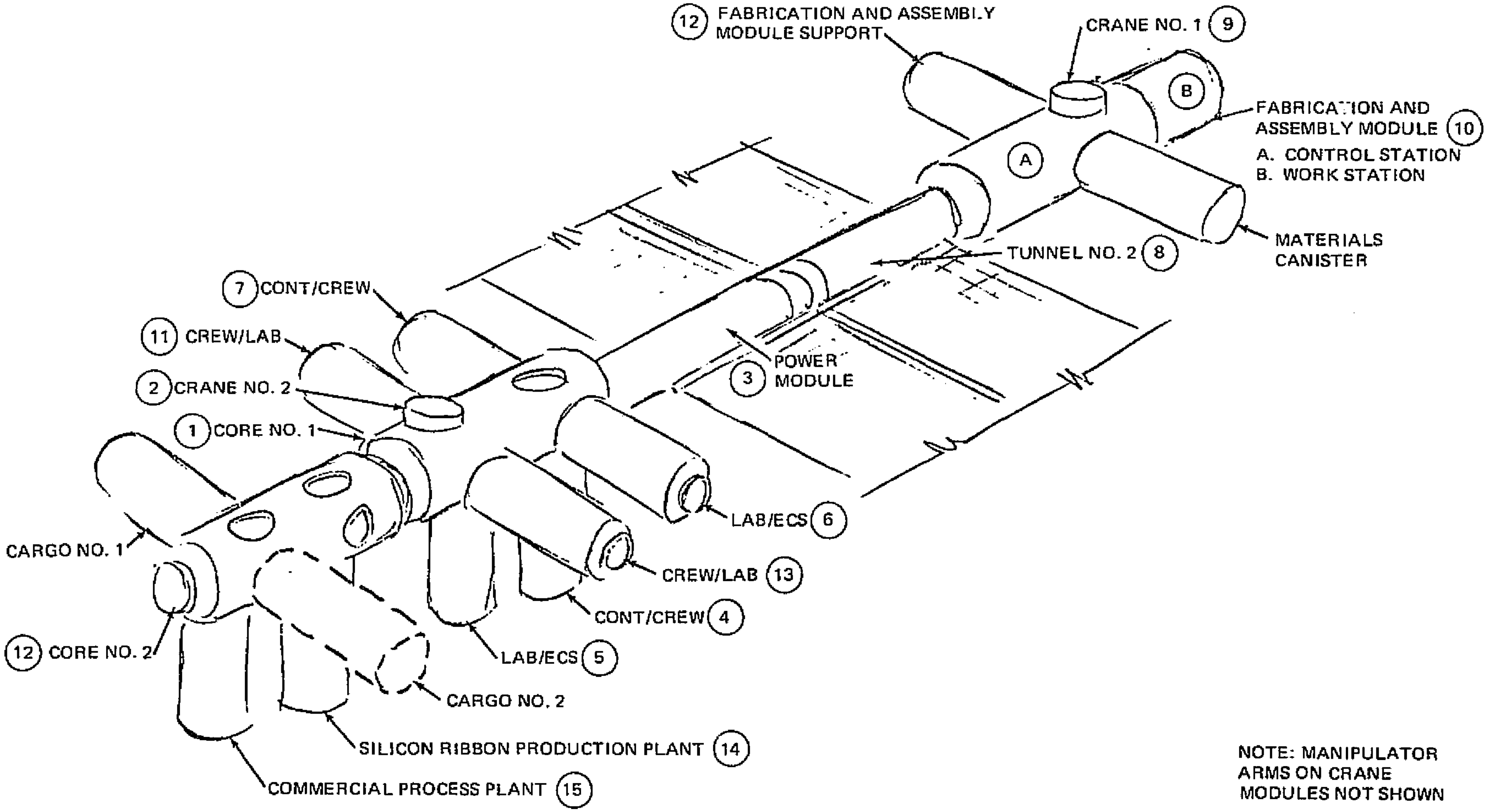

Figure1-1. Typical Large-Scale Construction Base

Section 2
MINI-TUG

The mini-tug concept considered in this analysis is a small one-man vehicle with major emphasis on maneuverability and reliability. As envisioned, the attitude control thrusters would also provide all translation. At one end, the mini-tug would have to have a docking port compatible with the SCB ports. When not in use, the vehicle could be berthed at any unoccupied port. It would probably be necessary, however, to devote one SCB port to the mini-tug with special provisions for refueling, systems checkout, etc. To avoid the problems associated with zero-g fuel transfer, it might be advisable to have replacable fuel canisters. Empties could be taken down on return Shuttle flights to be refilled. It is expected that the mini-tug would need hard-dock capability for module transport. For assembly part transport, the vehicle would need some type of remote manipulator arm. This would probably be a six degrees-of-freedom arm but could possibly be less.

2.1 MINI-TUG MODULE TRANSFER

Figure 2-1 presents a sketch of the mini-tug during a module transfer. In order to facilitate a preliminary performance analysis, a number of assumptions were made concerning the physical characteristics of the mini-tug. It was assumed to weigh 4536 kg (10,000 lbm), to be 3.05m (10 ft) in diameter, and 3.05m (10 ft) long. As illustrated in Figure 2-1, which is drawn roughly to scale, the mini-tug is quite small in comparison to a standard 14515 kg (32,000 lbm) module. This results in a combined center of gravity for the mini-tug/module configuration which is far outside the bounds of the mini-tug. Thus, when lateral translation is desired, it is not possible to simply apply a lateral thrust in the desired direction of motion. Such an action would introduce an unwanted rotation about the combined CG. In order to keep the summation of moments zero, opposing lateral forces must be applied in the front and rear of the mini-tug. The magnitude of each force must be proportional to its distance from the combined CG, with the constant

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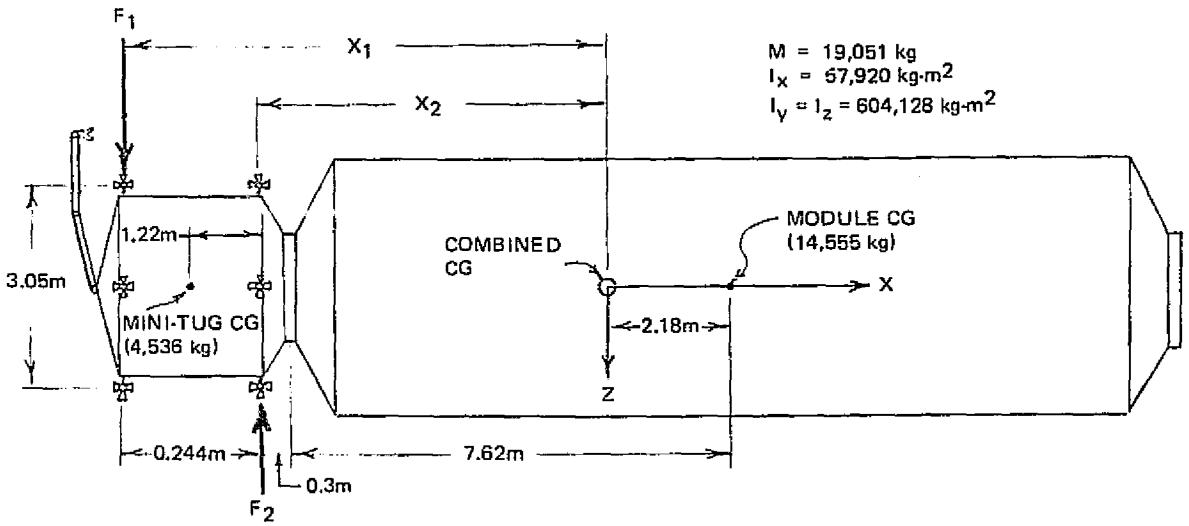

FOR NET LATERAL FORCE F_N WITH NO ROTATION

$$\left.\begin{aligned} \Sigma M = 0 \;&\Rightarrow\; F_1X_1 = F_2X_2 \\ \Sigma F = F_N \;&\Rightarrow\; F_2 - F_1 = F_N \\ &\quad\; F_{tot} = F_1 + F_2 \end{aligned}\right\} \Rightarrow F_{tot} = \left(\frac{X_1 + X_2}{X_1 - X_2}\right) F_N$$

$$\boxed{F_{tot} = 5.7143\, F_N}$$

Figure 2-1. Mini-Tug/Module Configuration

of proportionality determined by the desired net lateral force. Given the physical characteristics presented in Figure 2-1, the total of the two forces required is almost six times the net lateral force desired.

Another problem which arises in regard to moving a large module is operator visibility. A wide-angle closed-circuit TV camera (with lights) facing out of the module port opposite the mini-tug would be almost a necessity. It is very likely that the remote manipulator arm will have its own TV camera and lights. Assuming the arm is of sufficient length (approximately 3.05m [10 ft]), it should be possible to use it to look forward around the module. A full 360-deg yaw capability at the shoulder joint would allow the mini-tug pilot to check clearances around the entire circumference of the module.

The preceding paragraphs have given a brief description of the mini-tug concept. The next logical question concerns its performance. Can it transport a module in a reasonable length of time using a reasonable amount of propellant? Before this question can be answered, however, mention must be made of another factor which enters into the problem. Although the area around the SCB is often referred to as a zero-g environment, it is not in fact the same as it would be if the SCB were suspended in a void free of all forces. The zero-g results from a balance of centrifugal acceleration and the pull of gravity. This balance is exact at the center of gravity of the SCB, but not necessarily so at other locations. If an object were placed at rest with respect to the SCB coordinate system, it would in general begin to move away from that spot. This effect can be described quanitatively by the following equations

$$F_X = m\,(\ddot{X} - 2\omega\dot{Y})$$

$$F_Y = m\,(\ddot{Y} + 2\omega\dot{X} - 3\omega^2 Y)$$

$$F_Z = m\,(\ddot{Z} + \omega^2 Z)$$

The XYZ coordinate has its origin at the SCB center of gravity, with the Y-axis along the radius vector R, the Z-axis along the angular momentum vector, and the X-axis completing a right-handed system (see Figure 2-2).

The quantities F_X, F_Y, and F_Z are external forces, ω is the orbital rate of the SCB, m is the mass of the object under consideration. These are first order equations which assume the SCB is in a circular orbit and that X^2, Y^2, $Z^2 << R^2$. Since these conditions are well satisfied for our problem, the equations should be more than adequate. Using these equations, one could apply an external force history and then integrate to obtain a state history for the problem at hand, however, it is much simpler to assume a state history and solve directly for the external force history required. This can then be easily converted to a propellant requirement.

In order to evaluate the magnitude of these orbital effects on the mini-tug, a hypothetical state history for a module transfer was assumed. The assumption is that the mini-tug would dock with a module located 30.5m (100 ft) aft of the SCB CG. The mini-tug/module would back out from its initial position 10.7m (35 ft) from the centerline to a 29m (95-ft) distance to provide transfer clearance. During the first half of this maneuver (segment A, Figure 2-2), the vehicle would be under a constant acceleration outward, and during the second half (segment B), the acceleration would be reversed. The mini-tug/

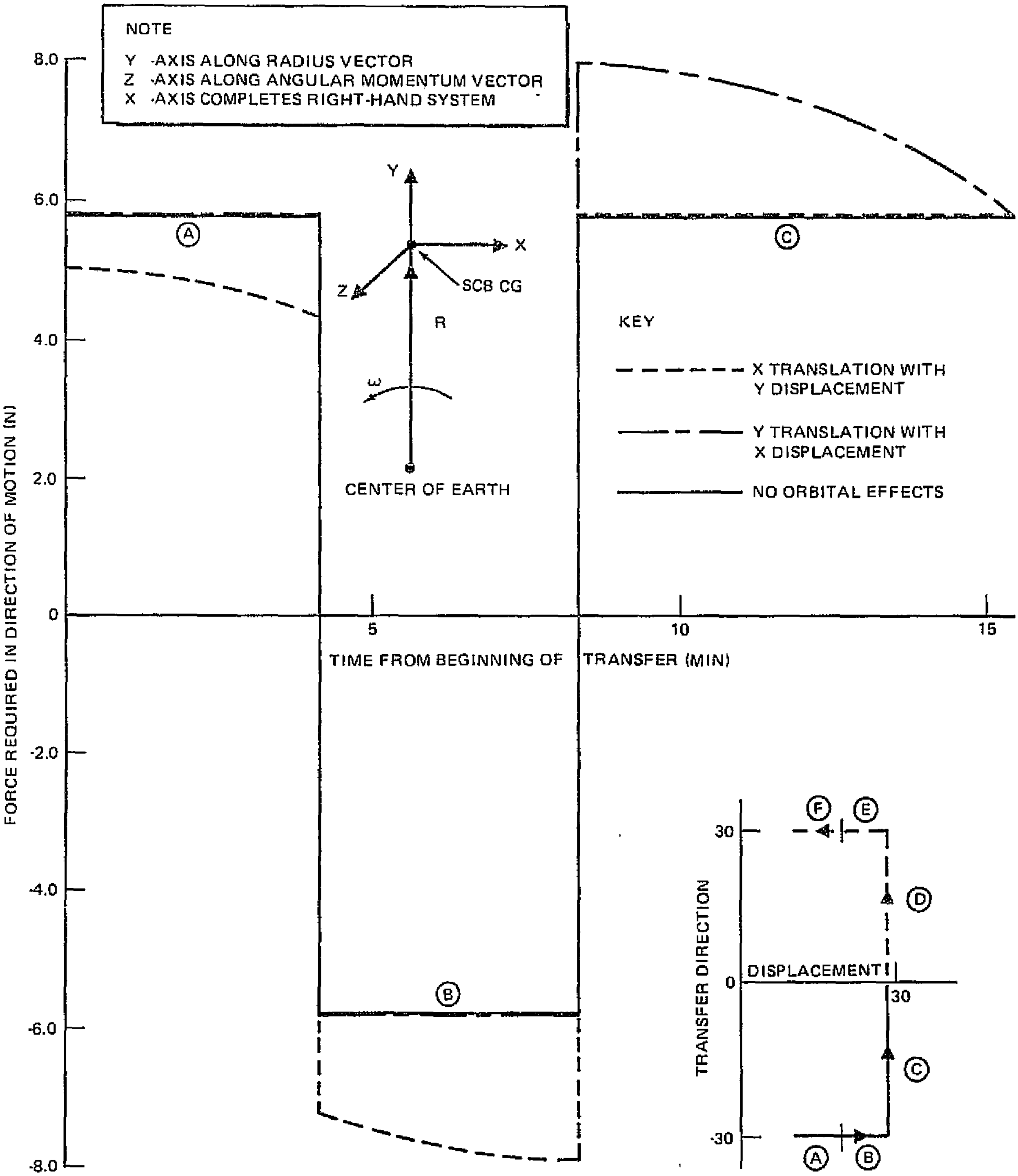

Figure 2-2. Module Transfer with Mini-Tug Orbital Effects in Direction of Motion

module would then turn and begin accelerating forward at the same acceleration along a path parallel to the SCB centerline (segment Ⓒ). Segments Ⓓ, Ⓔ, and Ⓕ would reverse the process in order to berth the module 30.5m (100 ft) forward of the SCB CG. An acceleration magnitude of 0.0305 cm/sec^2 (0.001 ft/sec^2) was assumed throughout. It was expected that this acceleration could be provided by 22.25N (5 lbf) hydrazine thrusters with the effect of variable thrust achieved by controlling pulse timing.

Figure 2-2 shows the axial force histories required for the first half of the module transfer previously described. The solid line is the axial force history assuming the SCB were in a void. It requires a constant force of 5.8N (1.3 lbf) with only the direction changing with the segments. The dashed line represents a case where the SCB centerline is aligned with the X-axis of the orbit system and the module centerline is initially aligned with Y-axis. Axial force required for movement along Y for clearance displacement is effected to some extent, but for the translation along X (segment Ⓒ), it is not. The line made up of long and short dashes represents a case where the SCB centerline is aligned with the Y-axis of the orbit system and the module centerline is initially aligned with the X-axis. Here the displacement along X is uneffected, whereas the translation along Y is effected to some extent. Notice that in both cases the differences due to orbital effects are relatively small and can either increase or decrease axial force requirements.

The situation is quite different when lateral forces are considered. If the SCB were in a void, there would be no lateral forces at all. Orbital effects, however, introduce lateral forces which are then magnified by the induced rotation problem discussed earlier. This effect is illustrated in Figure 2-3. The solid line represents the required lateral force magnitude history needed to counteract orbital forces for translation along the X-axis with Y displacement for clearance. Induced rotation problems dictate that the net lateral force be vector sum of two opposing forces. The magnitude sum of these forces is represented by the dashed line and is almost six times greater than the net force required. Note that it is also several times as great as the axial force requirement. Figure 2-4 presents the same data for Y translation with X displacement. The increase in lateral force was such that it was necessary to change the scale by a factor of two. In conclusion, it can be said that orbital effects combined with induced rotation problems

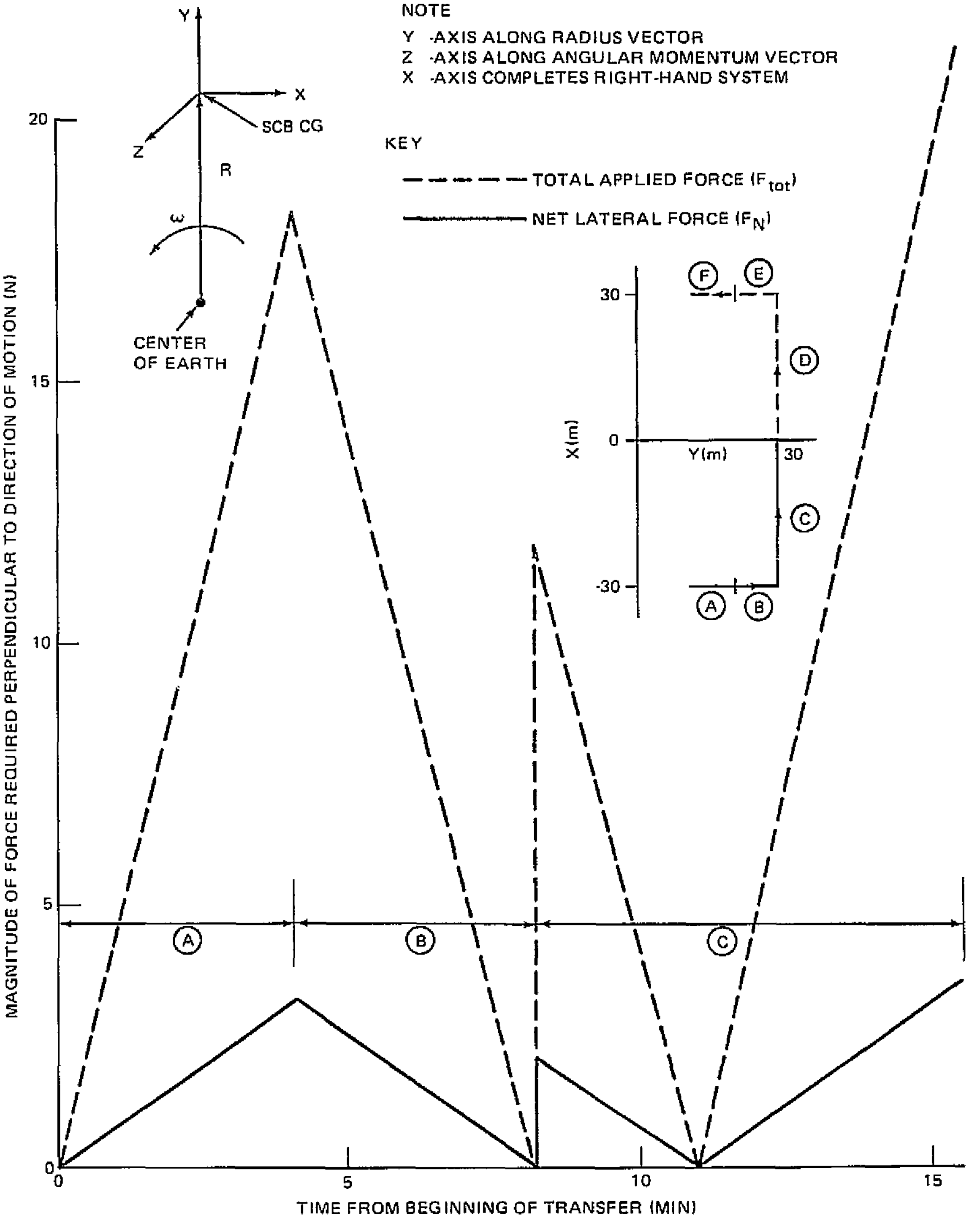

Figure 2-3. Module Transfer with Mini-Tug Orbital Effects Perpendicular to Direction of Motion - - X Translation with Y Displacement

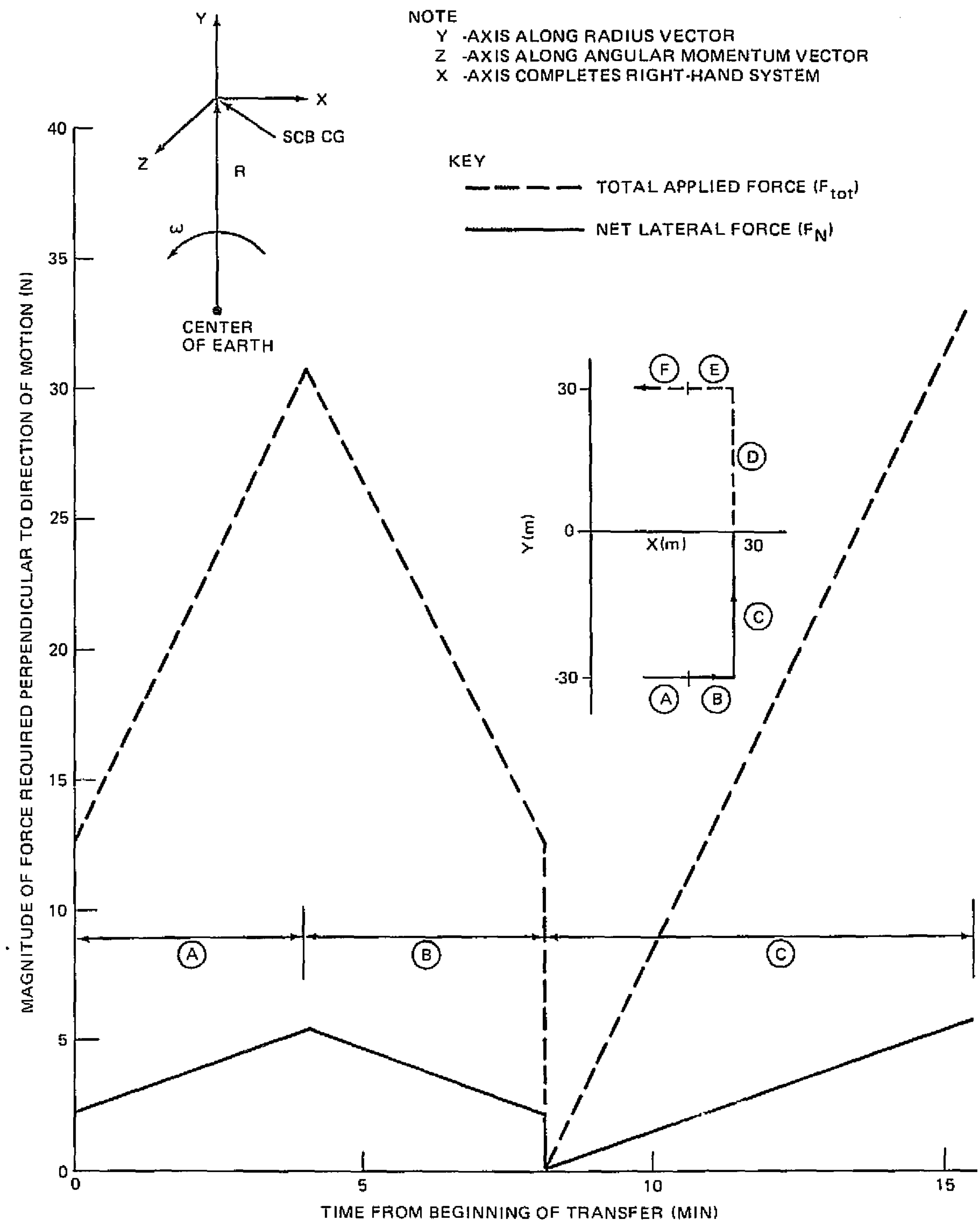

Figure 2-4. Module Transfer with Mini-Tug Orbital Effects Perpendicular to Direction of Motion - - Y Translation with X Displacement

significantly effect propellant requirements for translating and that these effects are highly dependent on transfer path.

In an actual module transfer, there will also be periods when the mini-tug is simply maintaining a fixed position with respect to the SCB. For this case, the equations reduce to the following form

$$F_X = 0$$

$$F_Y = -3m\omega^2 Y$$

$$F_Z = m\omega^2 Z$$

Note from these equations that maintaining a position above or below the SCB CG requires three times as much force as maintaining a position the same distance out of plane. Maintaining a position in front or behind the CG requires no force. The force required is proportional to both the displacement from the SCB CG and the mini-tug/module mass. This is plotted up in Figure 2-5 in the form of propellant flow rates ($\dot{w}_p$) assuming an exit velocity of 1956 m/s. The in-plane and out-of-plane flow rates are additive. No account is made here for extra propellant required due to induced rotation problems. When this is taken into account, propellant requirements could become almost six times greater for a worst-case orientation. Thus, hovering with a module at a safe clearance distance above or below the SCB CG could require propellant expenditures of almost 1/2 kg per minute.

During the course of a module transfer, it will be necessary for the mini-tug to rotate the module to obtain the proper orientation. Figure 2-6 presents propellant requirements as a function of rotation time and angle. This assumes 22.25N (5 lbf) thrusters, with a separation distance of 2.4m (8 ft) and an exist velocity of 1956 m/s. Note that as rotation time increases, propellant requirements not only become less but become more linear with respect to rotation angle.

The final drain on propellants considered was attitude control. This analysis assumes a two-sided deadband with a minimum pulse time of 20 ms on the 22.25N (5 lbf) thruster. Results are presented in Figure 2-7 for the mini-tug alone, as well as for the mini-tug/module configuration. It requires

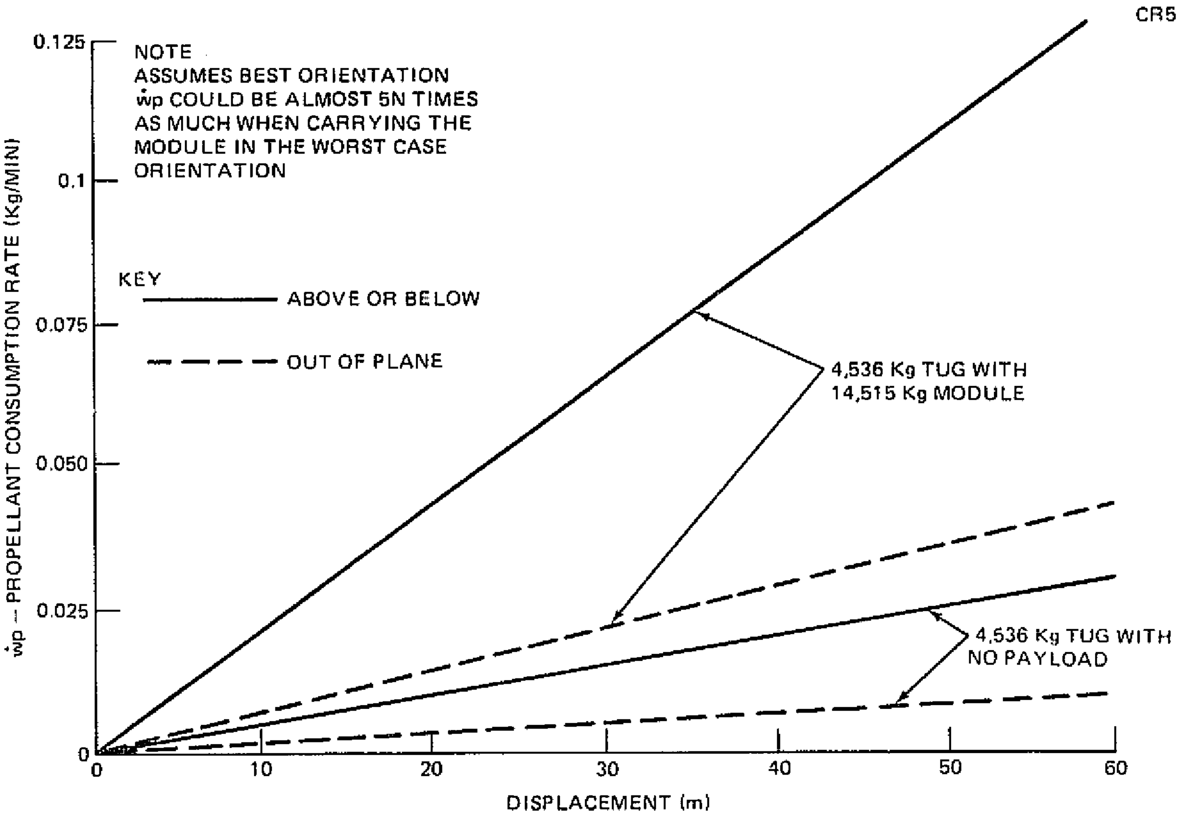

Figure 2-5. Mini-Tug Station-Keeping Propellants

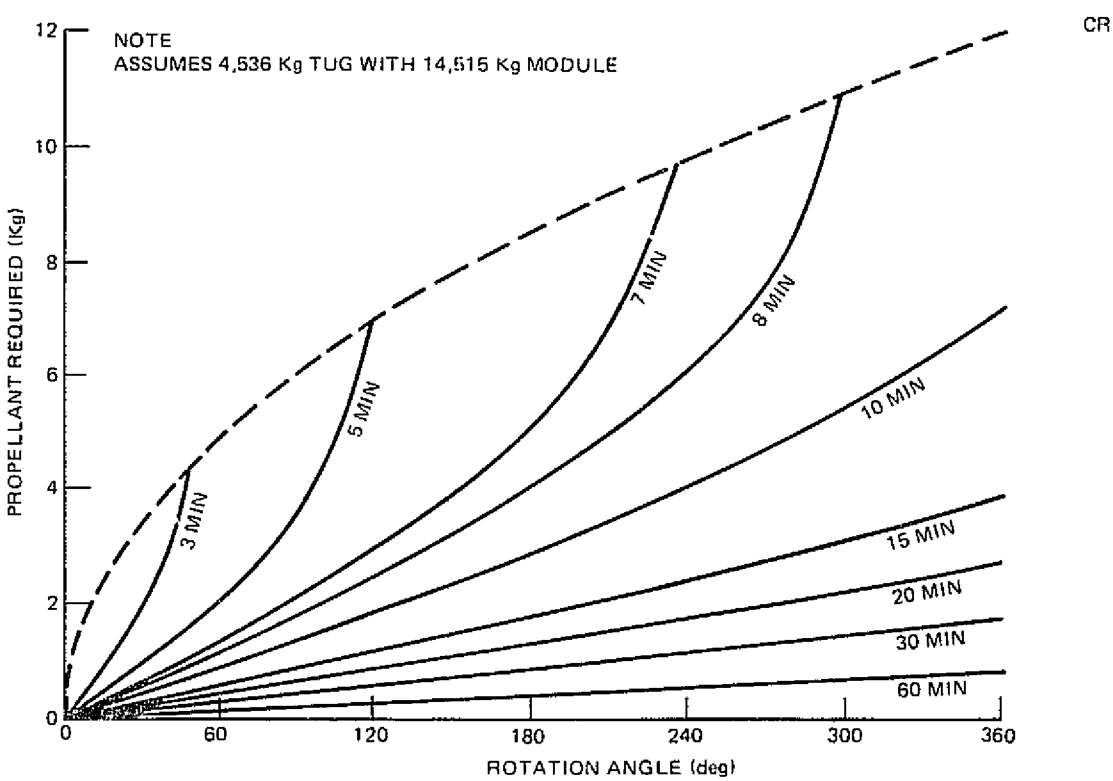

Figure 2-6. Mini-Tug Rotation Propellant Requirements

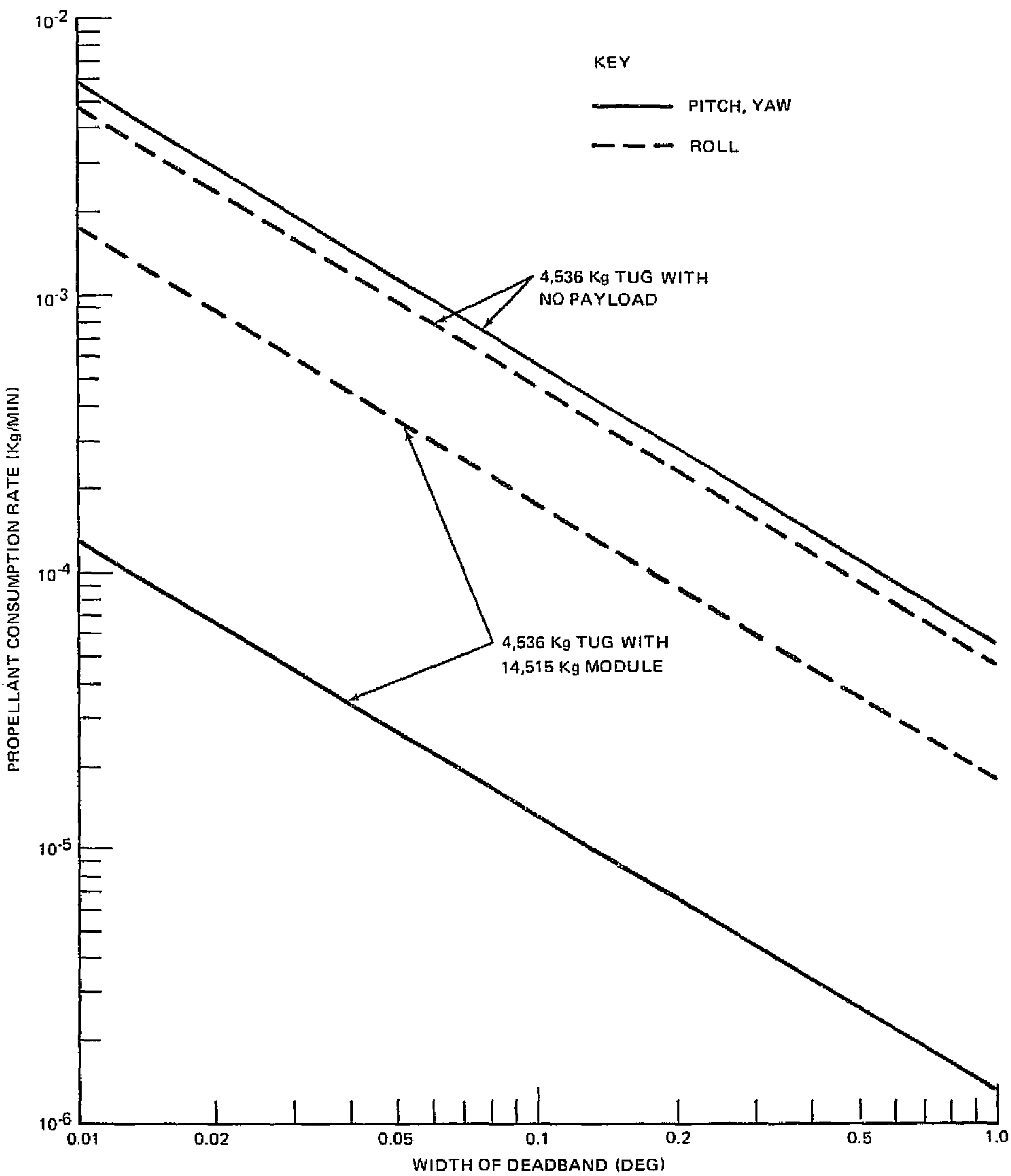

Figure 2-7. Mini-Tug Attitude Control Propellants

less propellant to control the mini-tug with a module than without, and less propellant to control roll than pitch and yaw. The higher the moment of inertia, the slower the angular velocity resulting from the pulse, the longer the time between pulses, and consequently the lower the propellant requirement. Even though propellant requirements vary two orders of magnitude in going from a very loose to a very tight deadband, attitude control propellants are not significant compared to other propellant drains.

Table 2-1 tabulates total propellant requirements for the hypothetical module transfer under consideration. Propellant drains are broken down by type and listed down the left-hand column in the chronological order in which they occur, with the exception of attitude control which is a continuous drain. Data is tabulated for each of six possible transfer paths which differ only in orientation with respect to the orbit system previously defined. The tabulated propellants include effects due to induced rotation. Even though a tight bandwidth of 0.01 deg was assumed, the attitude control propellant is insignificant. Translation times reflect an acceleration of 0.0305 cm/sec^2 (0.001 ft/sec^2), and other time were selected according to what seemed reasonable. Note that the total propellant requirement varies by a little more than a factor of two, depending on orientation.

2.2 MINI-TUG ASSEMBLY PART TRANSFER

A second function of the mini-tug is to transfer and position assembly parts. The assembly part selected for analysis, a multiple beam lens antenna element, is one of the largest and most massive that is presently under consideration. This assembly part along with a hypothetical transfer path is illustrated in Figure 2-8. The remote manipulator arm is used to grasp the assembly part, position it for transfer, and hold it fixed with respect to the mini-tug during transfer. The mini-tug/assembly part configuration in Figure 2-8 shows the assembly part in its fixed position during transfer. The physical characteristics given reflect same. Note that the pitch moment of inertia (I_y) is now less than the yaw moment of inertia (I_z) due to the asymmetry of the assembly part. Since the combined center of gravity now lies between thrusters, it is no longer necessary to waste energy by thrusting in opposite directions in order to avoid rotation. This can be accomplished now by simply balancing the thrust from the forward and aft thrusters for zero moment.

Table 2-1

TYPICAL MINI-TUG PROPELLANT REQUIREMENTS FOR MODULE TRANSFER

	Time (min)	X Transfer with Y Displacement	X Transfer with Z Displacement	Y Transfer with X Displacement	Y Transfer with Z Displacement	Z Transfer with X Displacement	Z Transfer with X Displacement
Displacement Translation for Clearance	8.2	3.83 kg	1.41 kg	6.84 kg	4.53 kg	4.04 kg	4.15 kg
90-deg Pitch Up	10	1.39 kg	1.39 kg	1.39 kg	1.39 kg	1.39 kg	1.39 kg
Station Keeping		2.60 kg	0.88 kg	2.73 kg	3.61 kg	0.93 kg	3.53 kg
Translation Along SCB	14.9	6.75 kg	10.58 kg	10.89 kg	11.25 kg	2.43 kg	7.84 kg
90-deg Pitch Down	10	1.39 kg	1.39 kg	1.39 kg	1.39 kg	1.39 kg	1.39 kg
Station Keeping		2.60 kg	0.88 kg	2.73 kg	3.61 kg	0.93 kg	3.53 kg
Translation to Remove Clearance Displacement	8.2	3.83 kg	1.41 kg	6.84 kg	4.53 kg	4.04 kg	4.15 kg
Station Keeping	20	1.92 kg	0.64 kg	5.46 kg	6.14 kg	1.86 kg	3.76 kg
Total Time	71.3 min						
Attitude Control BW = 0.01 Deg	Pitch	0.009 kg	0.009 kg	0.009 kg	0.009 kg	0.009 kg	0.009 kg
	Yaw	0.009 kg	0.009 kg	0.009 kg	0.009 kg	0.009 kg	0.009 kg
	Roll	0.13 kg	0.13 kg	0.13 kg	0.13 kg	0.13 kg	0.13 kg
Total Propellant Requirement		24.46 kg	18.73 kg	38.42 kg	36.6 kg	17.16 kg	29.89 kg

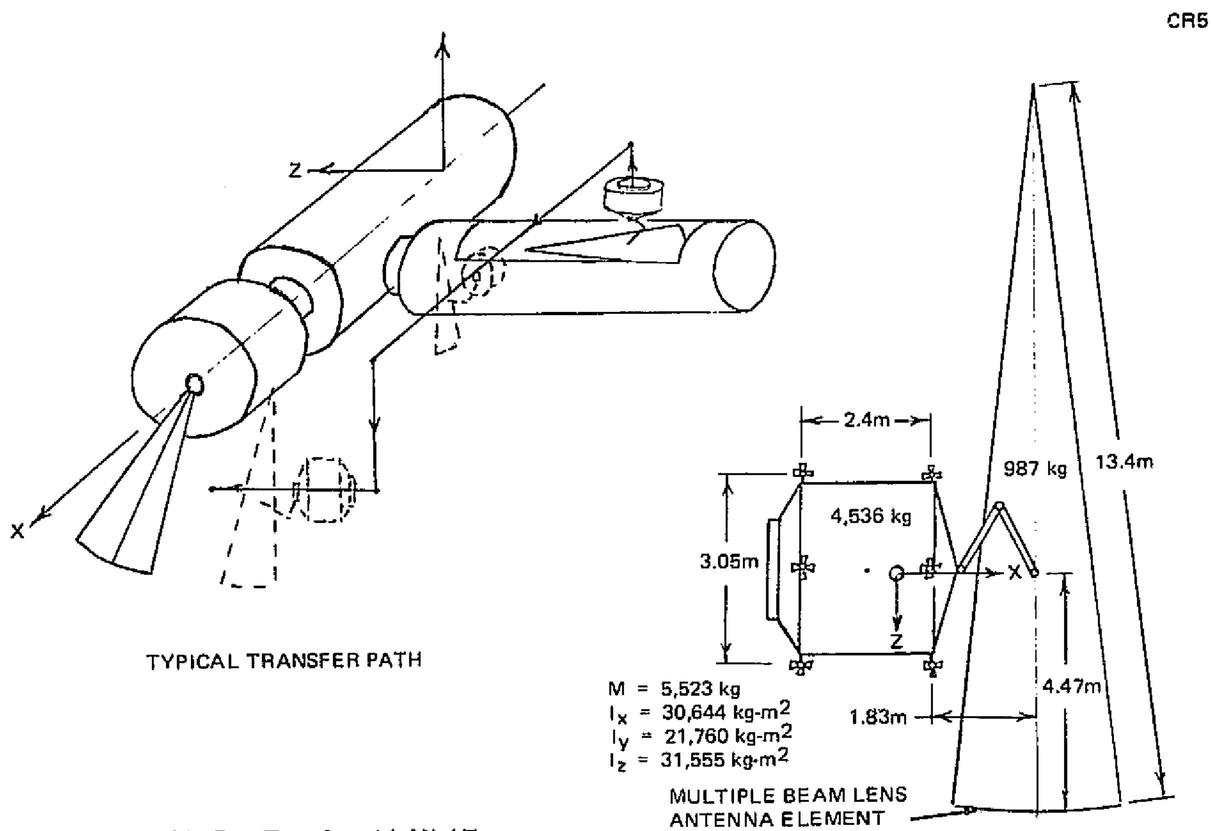

Figure 2-8. Assembly Part Transfer with Mini-Tug

The hypothetical transfer path includes backing the antenna element out of its canister and up along the Y-axis to provide clearance. This is followed by a roll and pitch which orients the vehicle to move forward along the X-axis. After the forward translation, another pitch and roll orients the vehicle to move back down below the centerline. After the downward translation, the vehicle pitches up and moves in to position the assembly part. The exact sequence of operations, including position and attitude histories, is given in Table 2-2. Times and propellant requirements are tabulated for a slow (a = 0.0305 cm/sec^2) and a fast (a = 0.305 cm/sec^2) transfer. For the fast transfer, other times were speeded up in proportion to the translation times, and 67N (15-lbf) thrusters were used in the rotation and attitude control calculations. Note that attitude control propellants have more than doubled even though the time during which control is required has

Table 2-2

TYPICAL MINI-TUG PROPELLANT REQUIREMENTS FOR ASSEMBLY PART TRANSFER AND POSITIONING

	Slow Transfer ($a = 0.0305\ cm/sec^2$)		Fast Transfer ($a = 0.305\ cm/sec^2$)	
	Time (min)	Propellant (kg)	Time (min)	Propellant (kg)
+Y Translation (Y: +3 → +15; X = +24; Z = -8)	6.67	0.39	2.11	1.12
90-Deg Roll (Counterclockwise)	(3.00)	0.18	(1.00)	0.58
90-Deg Pitch Up	(3.00)	0.24	(1.00)	0.76
Station Keeping (X = +24; Y = +15; Z = -8)	6.00	0.07	2.00	0.02
+X Translation (X: +24 → +30; Y = +15; Z = -8)	4.71	0.25	1.49	0.78
90-Deg Pitch Down	(3.00)	0.24	(1.00)	0.76
90-Deg Roll (Clockwise)	(3.00)	0.18	(1.00)	0.58
Station Keeping (X = +30; Y = +15; Z = -8)	6.00	0.07	2.00	0.02
-Y Translation (Y: +15 → -9; X = +30; Z = -8)	9.43	0.60	2.98	1.62
90-Deg Pitch Up	(3.00)	0.24	(1.00)	0.76
Station Keeping (X = +30; Y = -9; Z = -8)	3.00	0.02	1.00	0.01
+Z Translation (Z: -8 → 0; X = +30; Y = -9)	5.27	0.28	1.67	0.86
Station Keeping (X = +30; Y = -9; Z = 0)	20.00	0.11	5.00	0.03
Total Time	61.08 min		18.25 min	
Attitude Control BW = 0.01 deg	Pitch	0.16		0.44
	Yaw	0.24		0.63
	Roll	0.21		0.57
Total Propellant Requirement		3.48 kg		9.54 kg

been reduced to less than a third. This is because of the larger thrusters. Attitude control propellant, however, still makes up a very small fraction of the total propellant requirement and could be further reduced by relaxing the deadband. The total propellant required appears to be roughly inversely proportional to the transfer time.

2.3 MINI-TUG REQUIREMENTS

The requirements listed in Table 2-3 are for the most part preliminary in nature and are probably not exhaustive. They should, however, provide a good starting place for future study. A number of the requirements are obvious and others have already been discussed. This section contains some explanatory comments on the remaining requirements.

Requirements 4 and 5 which relate to determination and control of position and orientation with respect to the SCB are of particular significance. The relative state vector is what is needed and on-board accelerometers give the inertial state. Taking the difference of the inertial states of the SCB and the mini-tug would give the relative state but would introduce accuracy problems. An alternate or supplemental approach might be to navigate with respect to beacons fixed on the SCB. Requirements 6, 8, 9, 12, 13, 14, 15, 17, and 18 are directly concerned with safety. This is particularly important since the mini-tug would be working in very close proximity to thin-skinned manned modules with shirt-sleeve environments. The attitude control system would have to be designed to prevent a stuck thruster from throwing the vehicle into a spin. Manipulator arm joints should be designed to lock should the joint motor fail. Operator visibility is very important. Collision avoidance software would be highly advantageous in case of operator error. A constraint on maximum distance from the SCB should be incorporated along with an emergency radio beacon and reserve life support capability.

Table 2-3

MINI-TUG REQUIREMENTS

Requirement No.	Description
1	Able to dock or berth at standard module port.
2	Able to rotate, translate, and control modules up to 15,422 kg (34,000 lbm) and 15.24m (50 ft) long.
3	Able to manipulate and position assembly parts up to 1,134 kg (2,500 lbm) and 15.24m (50 ft) long.
4	Able to know its own position within TBD ft and orientation within TBD deg with respect to SCB.
5	Able to control attitude (± TBD deg) and relative position (± TBD m) while carrying maximum module.
6	Multiple failure capability on attitude control thrusters.
7	Mechanical arm (~3m) to grasp and position parts.
8	Unobstructed view for arm operator.
9	Collision avoidance and maximum distance software.
10	Software to control maneuvers accounting for rotating central force field and translation-rotation coupling.
11	Thruster exhaust should not interfere with experiments.
12	On-board radio and emergency homing beacon.
13	TV and lights at end of manipulation arm.
14	Portable TV and lights which can be placed at opposite end of module being transported.
15	Fuel and life support to operate TBD hrs., plus TBD hrs. life support reserve.
16	Able to refuel or easily exchange fuel tanks while berthed.
17	Arm able to pivot 360 deg around mini-tug axis to allow a look around the module carried.
18	Automatic joint lock on arm in case of motor failure.

Section 3
FIXED CRANE

The fixed crane constitutes an approach to the local logistics problem which is drastically different from the mini-tug approach. Rather than moving about the SCB with modules and assembly parts, it stays fixes at one port and uses one or more long mechanical arms to grasp and move objects.

3.1 FIXED CRANE CONCEPTS

Figure 3-1 is a sketch of the fixed crane moving a module. This is intended as a functional representation only, and it is by no means to scale. The fixed crane is envisioned with an operator control station permanently mounted at a selected module port. The control station would have two very long (35m), six degree-of-freedom, remote manipulator arms attached on opposite sides. Each arm will be functionally similar to a human arm with pitch and yaw freedom at the shoulder; pitch at the elbow; pitch, yaw, and roll at the wrist; and open-close capability on the grasper.

In addition to being able to control the remote manipulator arms the operator will be able to rotate the entire crane control platform about an axis perpendicular to the module centerline and passing through the center of the port. This should have definite advantages in terms of operator visibility and coordinated arm use. Operator visibility will be further enhanced by placing a closed-circuit TV camera with lights on the end of each manipulator arm. Camera views are displayed to the operator on CRT inside the crane control station.

Control of the remote manipulator arms is not a simple task, although many possibilities are open. One possibility might be to have a small lever associated with each degree of freedom. Lever displacement could be used to command the joint angle, angular rate, or angular acceleration. The number of degrees of freedom involved, however, would make this an impossible task for the operator.

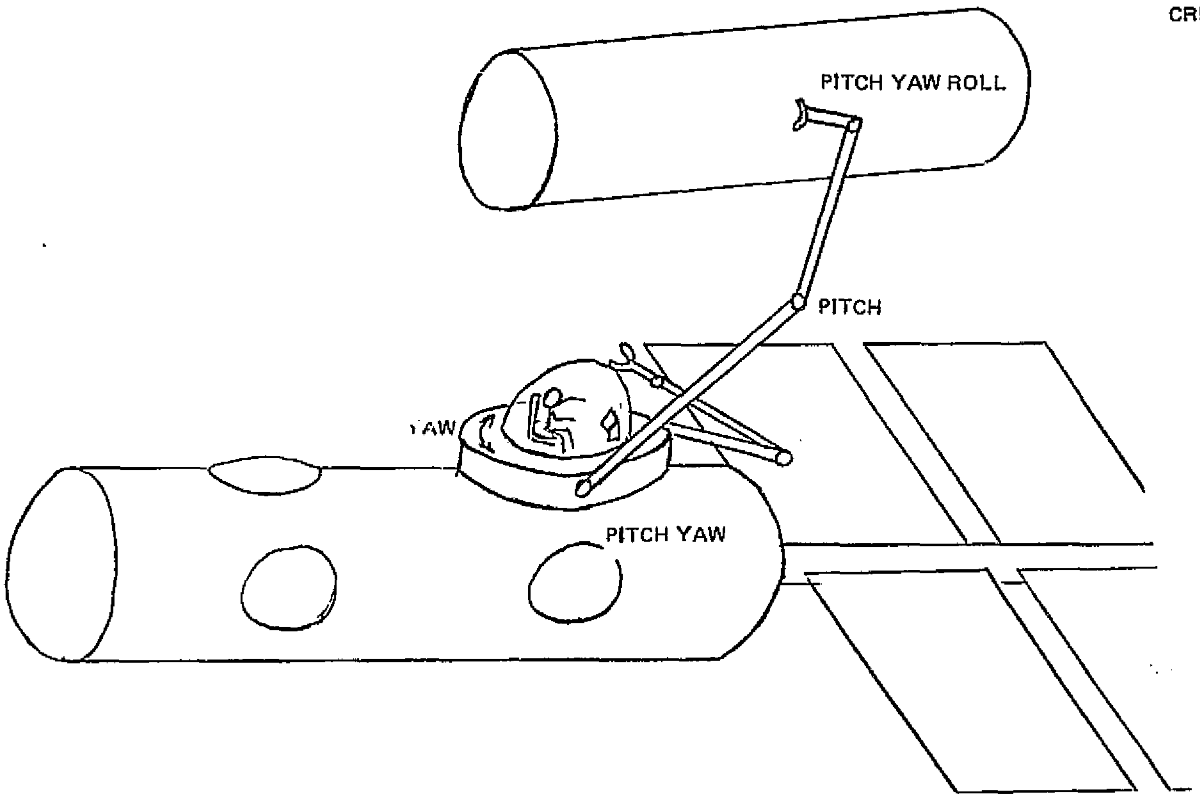

Figure 3-1. Fixed-Crane Concept

One interesting approach to simplifying the operators task is a Waldo (sometimes called exoskeleton) control. This concept (see Figure 3-2) takes advantage of the functional similarity between the crane arm and the operator's arm. A device is placed around the operator's arm which can sense joint angles. These are then converted to commands for the remote manipulator arm. The operator would have both direct visual feedback and closed-circuit TV. It could also be set up so the arm constraints would provide a force feedback when the crane arms made contact with an object. The Waldo concept would allow one man to operate both crane arms simultaneously. This is perhaps the only concept which makes this mode of operation feasible. It is doubtful, however, that simultaneous operation of the two arms will be employed since (1) sequential operation can perform the same tasks with less complexity and (2) operations analysis to date has revealed no requirement for such an operational mode. Waldo control has the disadvantages of requiring a good deal of operator space and limiting angular excursions of the crane arms to those available with the human arm.

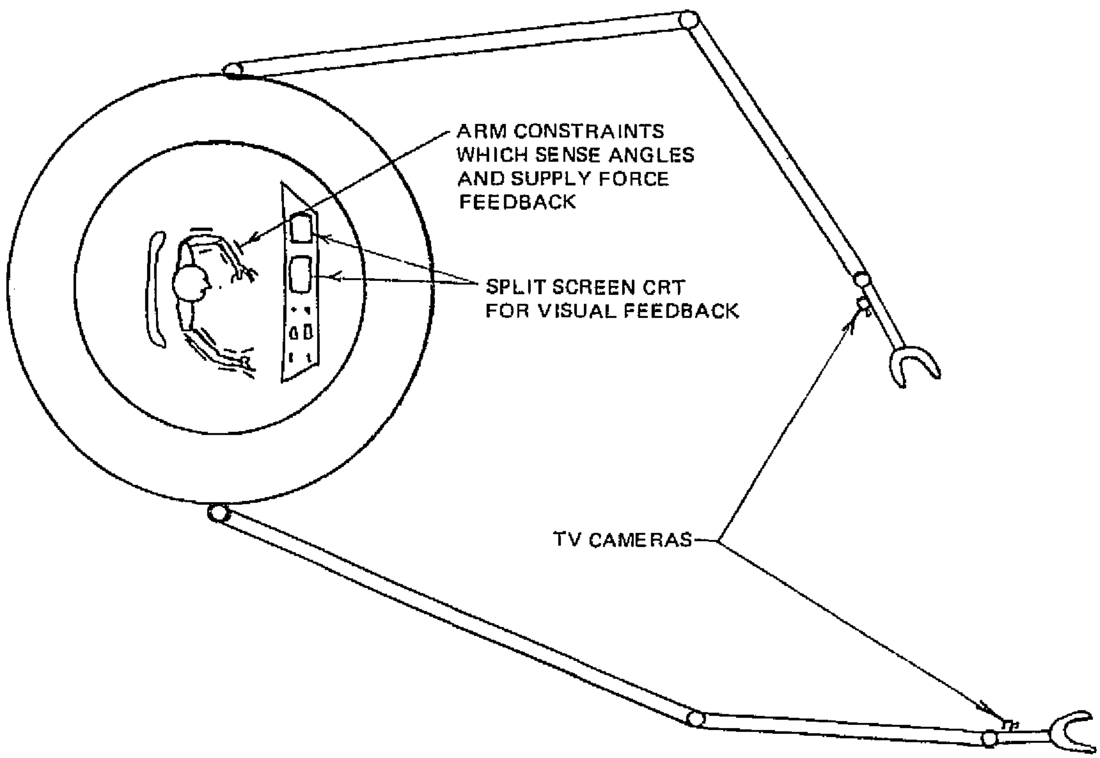

Figure 3-2. Waldo Control

Illustrated in Figures 3-3[1] and 3-4[1] are two other types of controllers which might be employed. The first of these is the replica controller (see Figure 3-3). Here the operator grabs a handle attached just beyond the wrist on a scaled replica of the crane arm. As the operator translates and rotates the handle the replica arm follows. The replica arm angles are then used as commands for the crane arm. This concept is very appealing since the remote manipulator arm can be operated with one hand. It also has the disadvantage, however, of requiring a great deal of travel space for the replica arm. The concept illustrated in Figure 3-4 is the one being used for the Remote Manipulator System (RMS) on the Shuttle. It consists of two hand controllers. One commands rotation rates and the other commands translation rates. Since these are rate commands, very little space is needed for operator hand movements, which is the principle reason this controller was choosen for the Shuttle. There are, of course, many other possible controllers but those mentioned appear to be the most promising.

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1. Shuttle Remote Manned Systems Requirements, Martin Marietta Corp., MCR-73-337; NAS 8-29904, Final Report, Vol. II, February 1974.

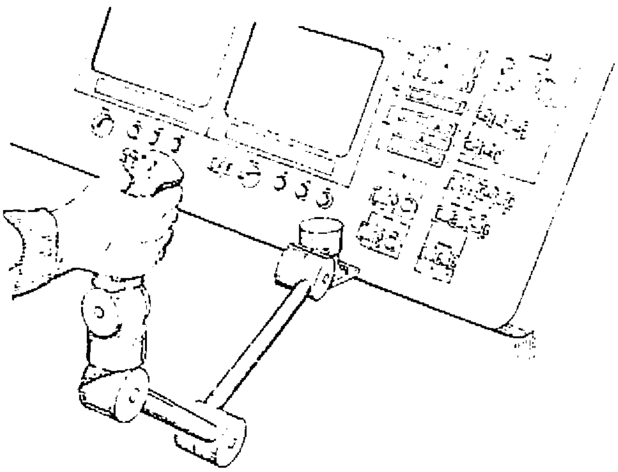

Figure 3-3. Control Console with Replica Controller

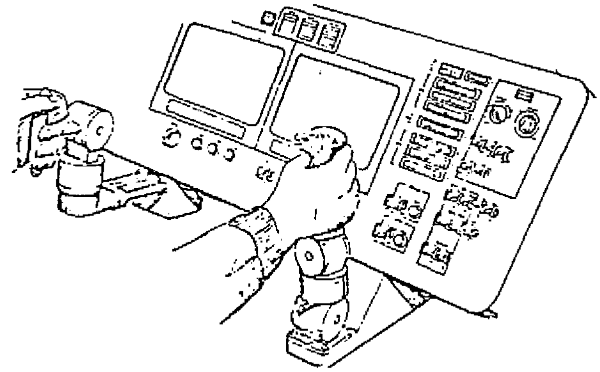

Figure 3-4. Control Console with Translational and Rotational Hand Controllers — RMS Control

Figure 3-5 illustrates a concept which could be used to great advantage with the fixed crane. For lack of a better name, this will be referred to as a cherry-picker module. Its purpose is to allow the crane arm to be used to position a worker in a remote spot for delicate adjustments, maximum visibility, etc. Before entering the cherry-picker module the worker would use the main crane controls to lock the grasper of one crane arm onto a special fitting on the cherry picker. This would complete an umbilical connection to hook up auxiliary crane controls inside the cherry-picker module. The worker would then enter the module and begin to move himself about by commanding the attached crane arm. Once in position he could flip a switch on his auxiliary control panel which would freeze the arm to which he was attached and allow him to control the other arm. There would be no necessity for an operator at the main crane control station. This concept is further illustrated in Figure 3-6, with some differences. One arm holds a cherry-picker cage in which the worker is EVA. The second arm is in a gooseneck mode which will passively hold the assembly part in any position in which the worker places it.

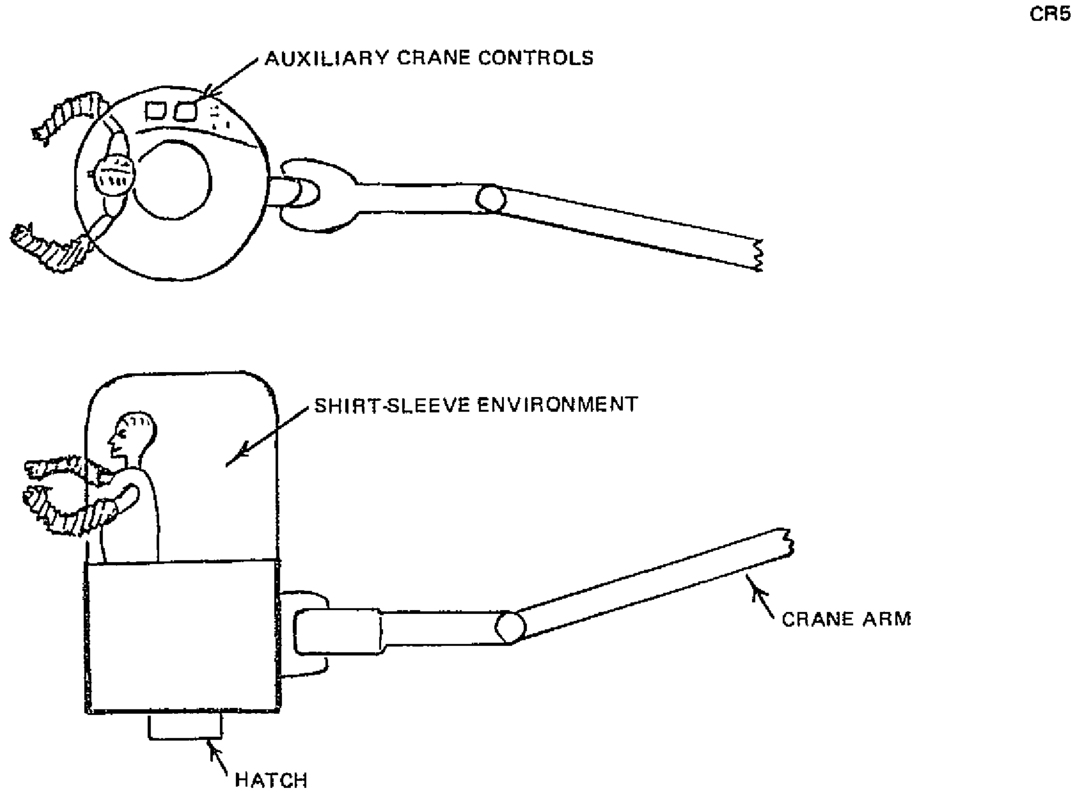

Figure 3-5. Cherry-Picker Module

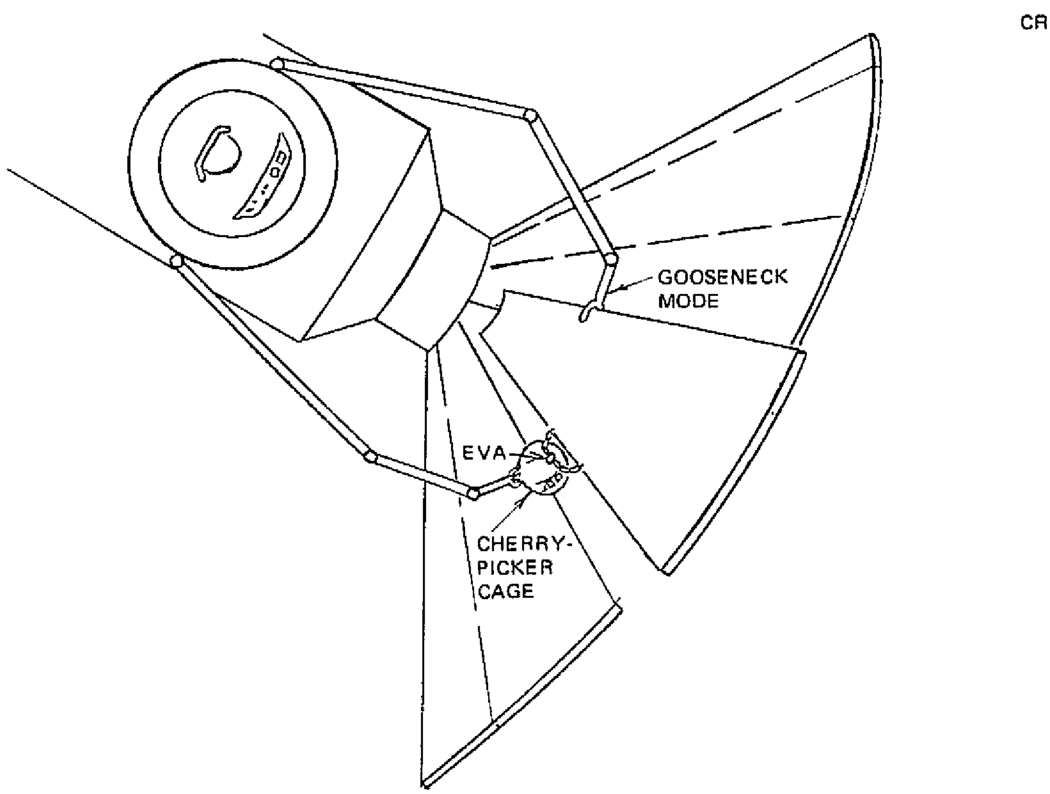

Figure 3-6. Fine Positioning

3.2 FIXED CRANE REQUIREMENTS

Up to this point the discussion of the fixed crane has been quantitative in nature. This section takes an order-of-magnitude look at the quantitative side. The expression order-of-magnitude is used primarily because a very simplified arm motion is assumed. It is assumed that a 14,515 kg (32,000 lbm) mass is swung through 180 deg with a rigid, fully extended 35m crane arm. It is highly unlikely that a transfer would be made in exactly this manner. It should, however, provide a conservative estimate of torque, power, and energy requirements as well as a basis for parameterization of transfer time and stopping distance. The effects of flexibility in the arms will be considered during parts of the study.

Figures 3-7, 3-8, and 3-9 present torque, power, and energy requirements in that order. These figures do not, however, include orbital effects. Time required for the hypothetical transfer is parameterized from 5 to 90 min. Continuous torque as well as torque applied only at the beginning of the transfer are considered. In the latter case the distance through which the mass travels while the arm is under torque is also the distance which would be required to stop the motion. Safety considerations would favor a relatively short stopping distance. Distances of 0.61, 1.52, and 3.05 m were considered. The continuous torque case corresponds to roughly a 55m stopping distance. An examination of Figure 3-7 shows that shoulder torque, and its associated normal tip force vary over three orders-of-magnitude for the range of transfer times considered. Note that for a given transfer time the effect of stopping distance on torque requirement is highly nonlinear. Torque and tip force for a constant stopping distance vary in a manner inversely proportional to transfer time squared. As expected, the shorter the stopping distance the higher the torque requirement. The power requirements in Figure 3-8 are more drastically effected by transfer time. They are inversely proportional to the transfer time cubed. As a result, the variations in Figure 3-8 cover almost six orders-of-magnitude. The relative effect of stopping distance is approximately the same. When energy requirements are considered in Figure 3-9, everything is reversed. The shorter the stopping distance, the less the total energy required for the transfer. For a given transfer time, the effect of stopping distance is not as nonlinear as it is with torque and power requirements. For a constant stopping distance, energy requirement varies in a manner inversely proportional to transfer time squared.

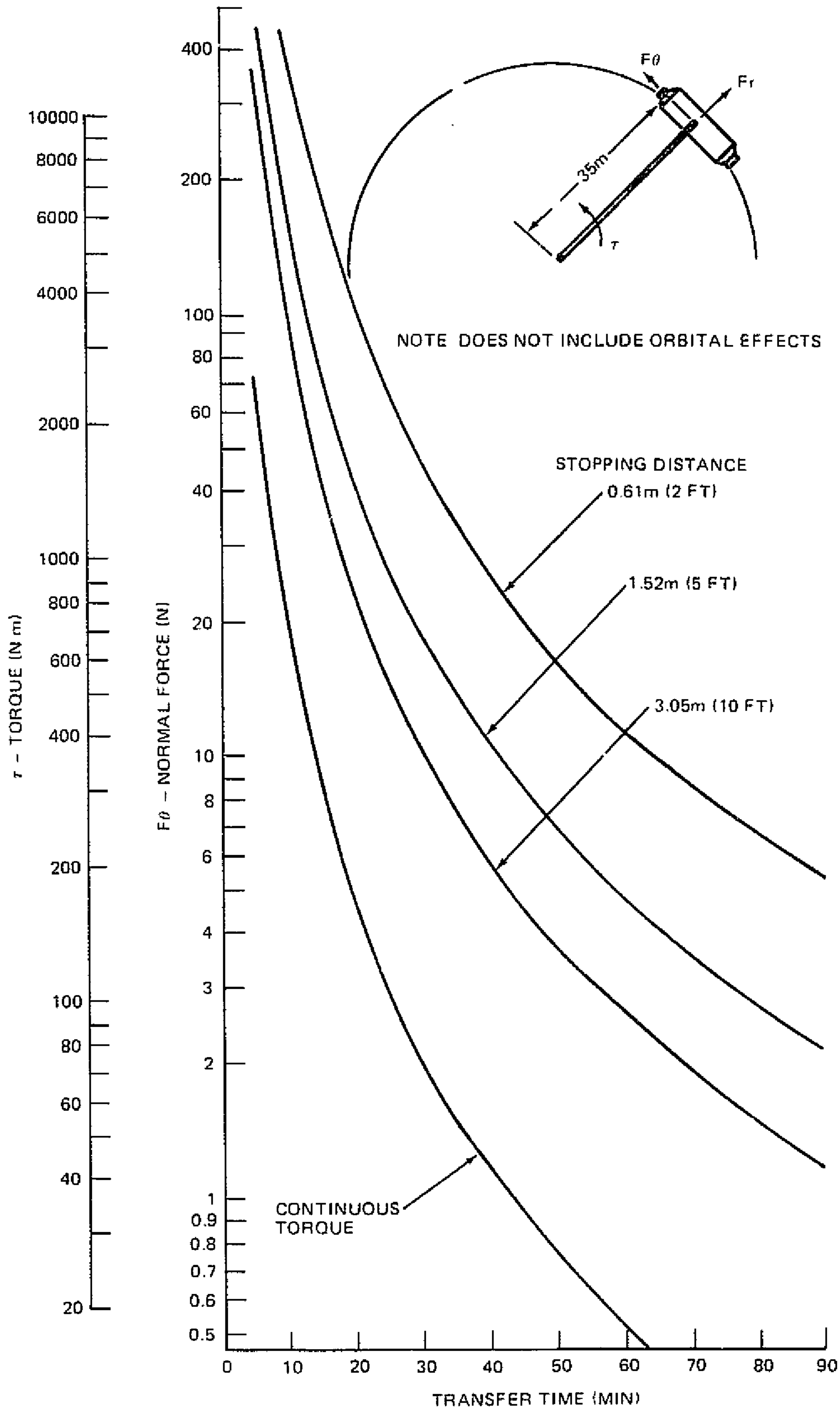

Figure 3.7. Zero-G Crane Order-of-Magnitude Torque and Tip Force Requirements

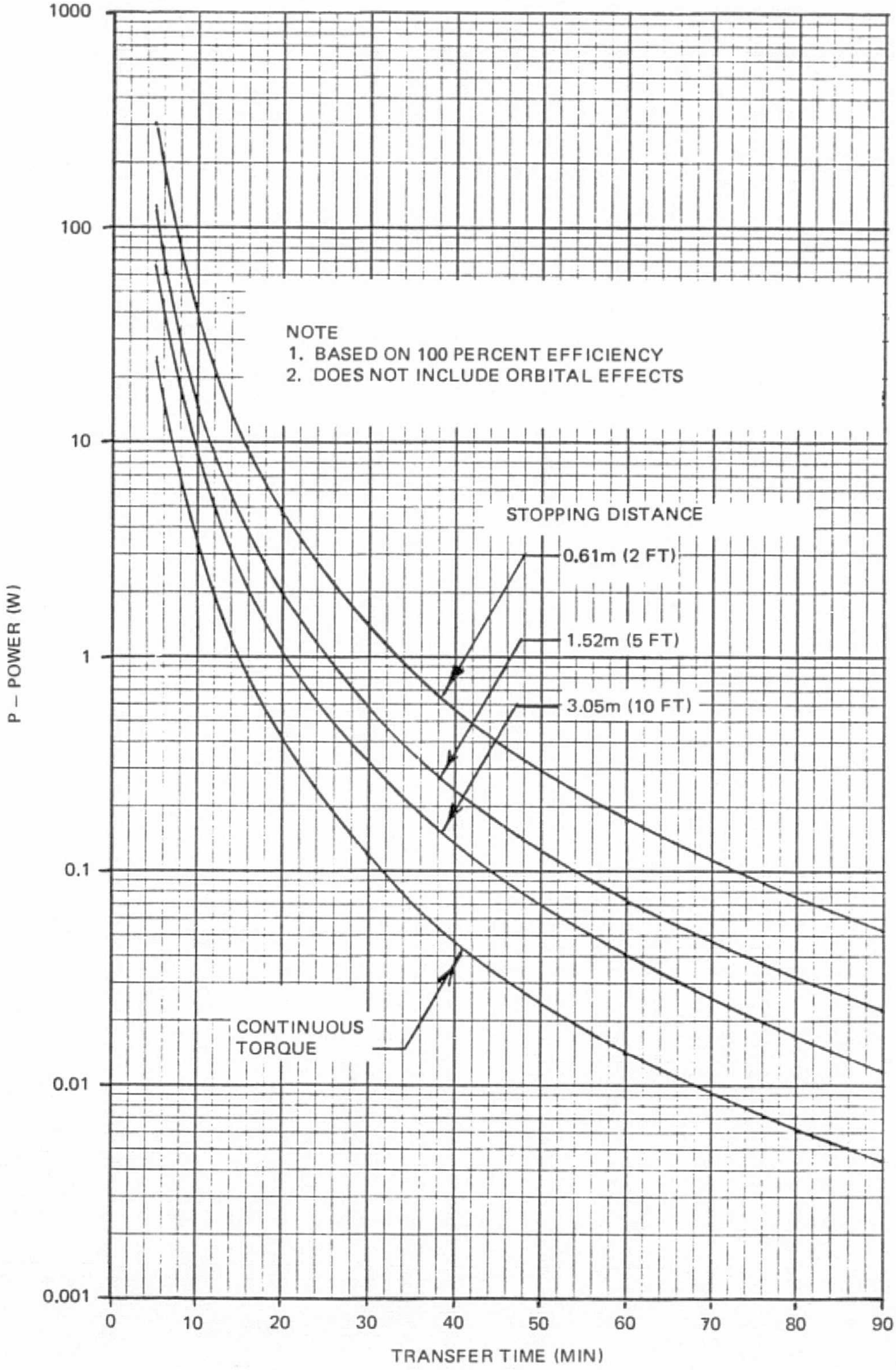

Figure 3-8. Zero-G Crane Order-of-Magnitude Power Requirement

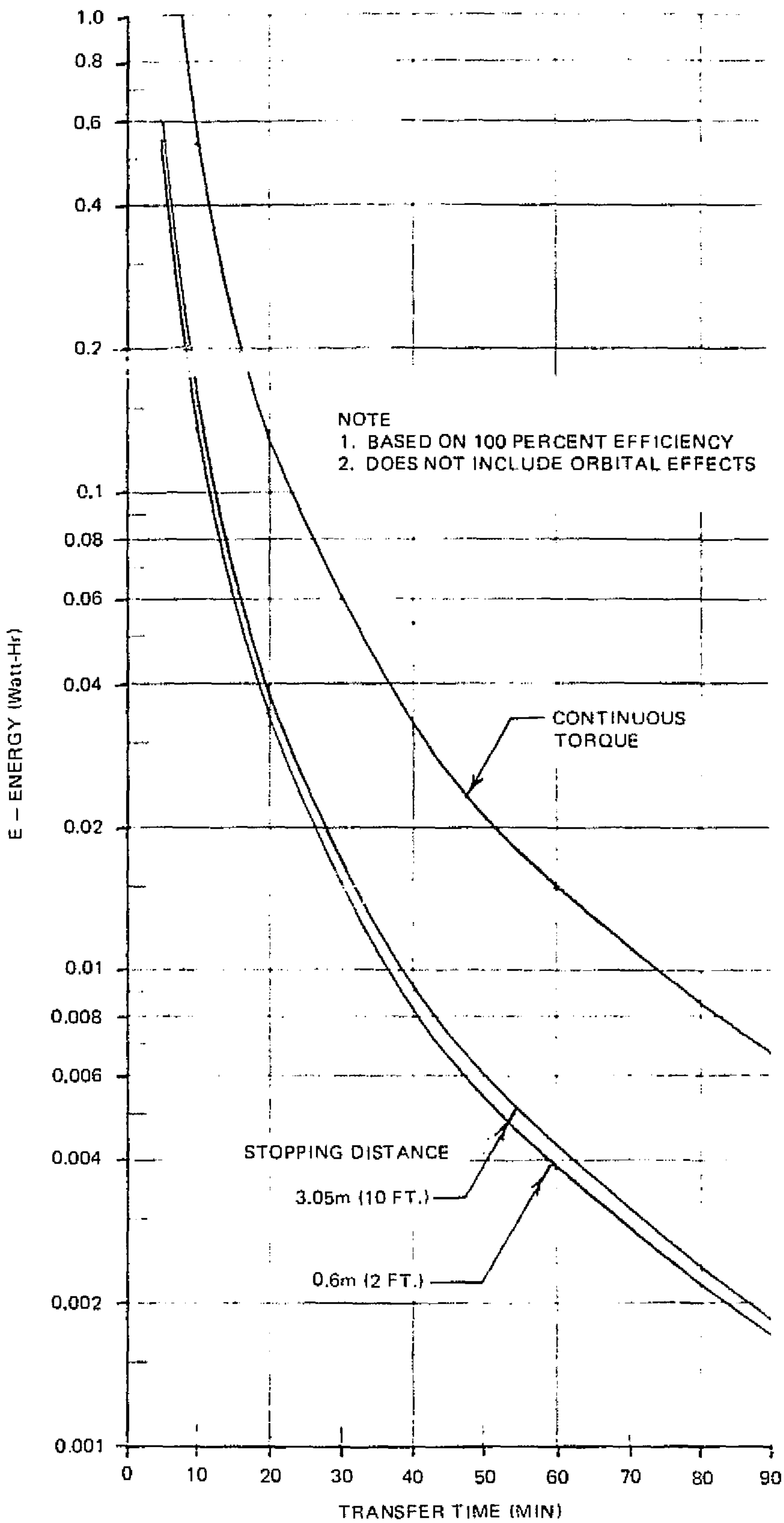

Figure 3-9. Zero-G Crane Order-of-Magnitude Transfer Energy Requirements

Up to this point no orbital effects have been considered in our hypothetical fixed crane transfer. Figure 3-10 represents an effort to evaluate how significant these effects might be. The figure presents tip force perpendicular to the crane arm (F_θ) as a function of time for a 30 min transfer with a 0.61m (2 ft) stopping distance. Due to symmetry only the first half of the transfer is considered. Segment (A) is under a constant angular acceleration and segment (B) is at a constant angular rate. If the SCB were in a void, F_θ would be a constant during segment (A) and zero during segment (B) as is shown by the solid line in the figure. In order to evaluate orbital effects, use was made of the same orbit coordinate system and linearized equations discussed in Subsection 2.1. The dashed line corresponds to F_θ for a translation along the Y-axis of the orbit system with a displacement in the X direction. During segment (A), the force required is almost identical to the force required in a void. During segment (B), the force required due to orbital effects rises to about 0.9N and then goes back down. Although this is very small compared to the segment (A) torque it is continued for a much longer time. The total area under the force/time curve is increased by almost 50 percent. For a Y translation with Z displacement (long and short dashes) the effect is more pronounced with 150 percent increase in area. In conclusion, the orbital effects appear insignificant in terms of maximum torque and power requirements but quite significant in terms of energy requirements.

Table 3-I is a list of requirements for the fixed crane. As was stated with regard to the mini-tug requirements, these are for the most part preliminary in nature and probably not exhaustive. They should, however, provide a good starting place for future study. Some of the requirements are obvious and others have already been addressed. The discussion which follows relates to some of the more significant requirements. The requirement for a 35m reach is a direct result of the fact that the crane is fixed. Present ground rules required that it be able to reach across the solar arrays and position a second crane on the opposite side while grasping the crane body. It must also be able to reach back to the port next to where the Shuttle docks and to reach around to the belly side of the SCB. The 89N (20-lbf) tip force requirement is really a fairly soft number based on the data in Figure 3-7. The requirement that the SCB corridor on which the fixed crane is positioned be kept open for crane operations is of particular significance since it cuts down the number of modules which can be berthed on the SCB by 25 percent.

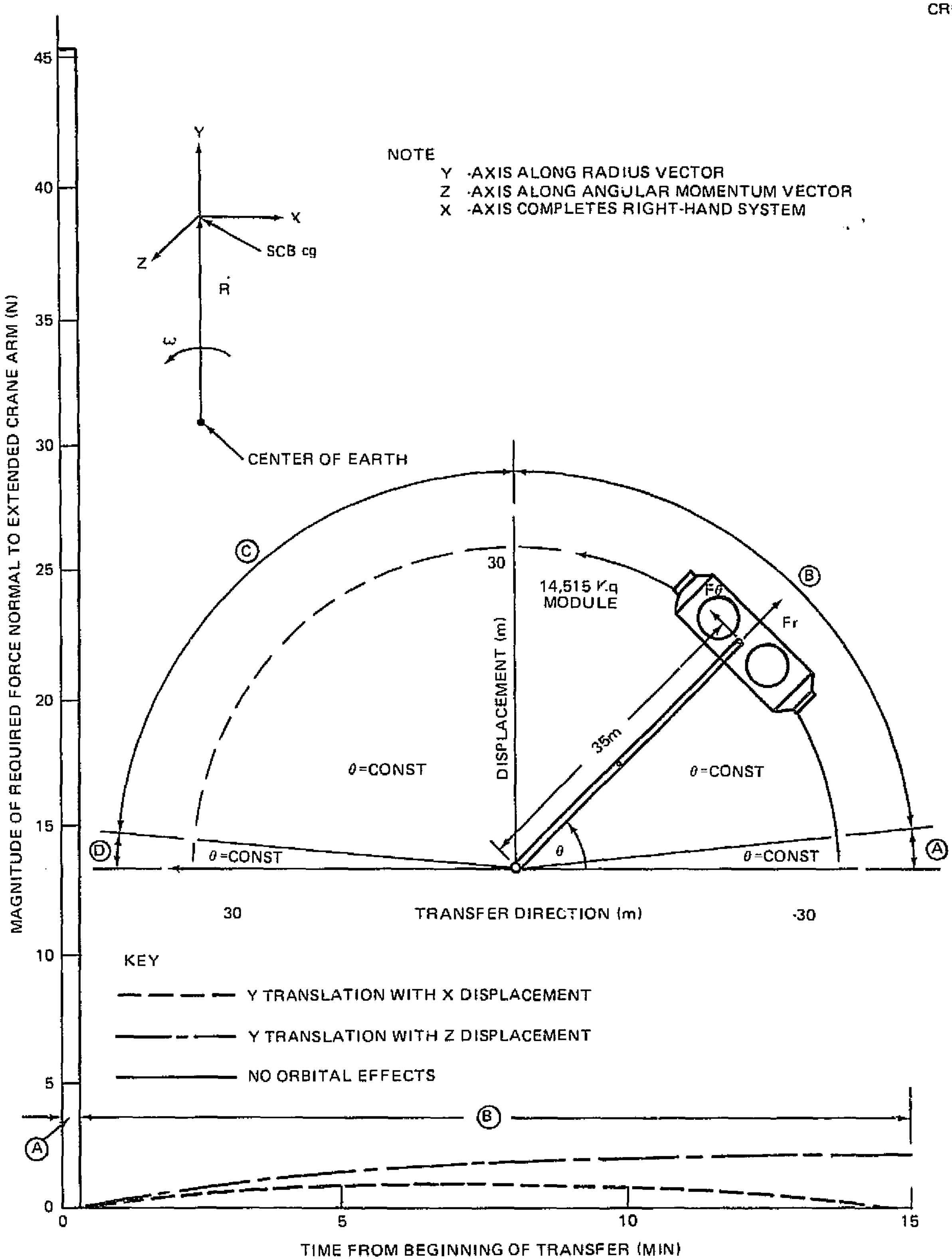

Figure 3-10. Module Transfer With Crane Orbital Effects Perpendicular to Crane Arm

Table 3-1

FIXED CRANE REQUIREMENTS

- Able to manipulate and berth modules up to 15,422 kg (34,000 lbm) and 15.24m (50-ft) long with one arm.
- Able to manipulate and position assembly parts up to 1,136 kg (2,500 lbs) and 15.24m (50-ft) long with one arm.
- 35-m reach and general grasping capability.
- Degrees of freedom:
  - Crane body (yaw)
  - Shoulder joint (pitch and yaw)
  - Elbow joint (pitch)
  - Wrist joint (pitch, yaw, and roll)
- Arm tip force capability of 89N (20 lbf).
- Arms operated sequentially, but not simultaneously.
- Manual and automatic six DOF control of each arm.
- Gooseneck or vernier control for fine positioning.
- Auxiliary control from cherry-picker cage.
- TV camera and lights on each crane arm as well as remote.
- Unobstructed view for crane operator.
- Two or more handholds per module.
- Open corridor in -Z direction of XZ plane.
- Solar arrays locked or angle limited during transfer across.
- Collision avoidance software and/or max torque override.
- Automatic joint lock in case of joint motor failure.

Section 4
MOBILE CRANE

Perhaps the most severe requirements associated with the fixed crane are (1) the extreme length of the crane arms and (2) the open corridor on the SCB. The first can be relaxed considerably and the second eliminated entirely by introducing the concept of the mobile crane. As was noted back in Figures 3-8 and 3-9, power and energy requirements for a module transfer with a crane are very low. With a reasonable efficiency factor and orbital effects thrown in, they still remain low. In fact, the power required for the CRT is considerably higher than the power required for module transfer. When all these factors are taken into account, it appears that the crane could be operated off internal battery power. Thus, it should be possible to made the crane autonomous from the SCB. It could then not only move freely about the SCB, but also out onto large space structures under construction.

The mobile crane would be very similar to the fixed crane except that it would be smaller and the control station where the operator sits would not have to be fixed to a port. In addition, a third grasper would be added to the back of the control station as shown in Figure 4-1. As it is shown here, this grasper is being used to anchor the mobile crane to the SCB, leaving both arms free. In this mode, the mobile crane could operate in a manner analogous to the fixed crane with the added advantage that it could be positioned anywhere it was needed. Because of this capability, the arm length could be reduced to about 15.24m (50 ft). This would involve much less development risk since remote manipulator arms of this length are presently being developed for the Shuttle. Since sequential use of the arms would probably be the most likely mode of operation for the mobile crane, it might be possible to use the Shuttle RMS controller as well. The important thing to note is that no single operation with the mobile crane would be any more complex than those with the fixed crane. The sequence of operations possible, however, could result in much greater capability.

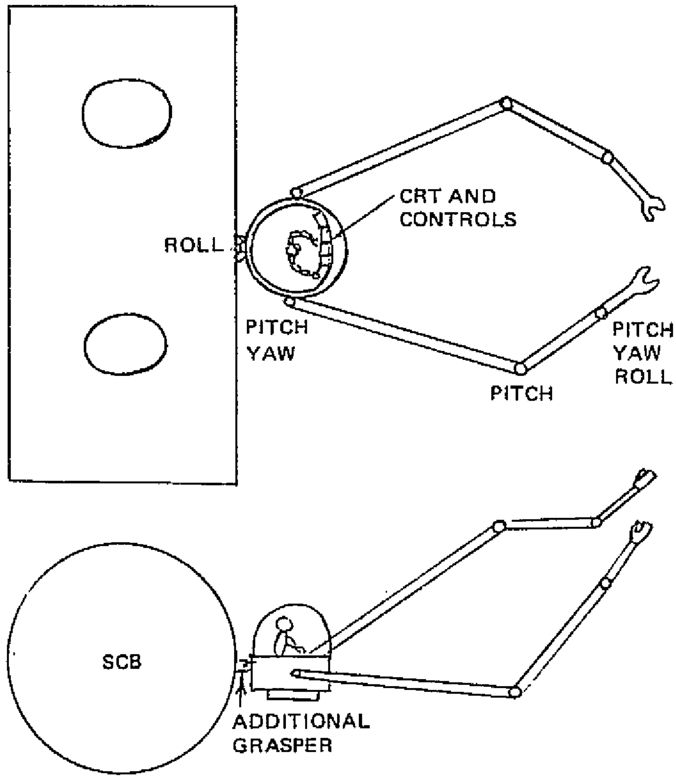

Figure 4-1. Mobile Crane – Two-Arm-Free Mode

Some of this additional capability is illustrated in Figure 4-2, where the mobile crane is shown performing an operation which would be impossible with the fixed crane. With one crane arm firmly attached to a handhold on the SCB, the other arm is used to move a module around an obstacle to its new berthing port. During the entire sequence, the operator is in an excellent position for visibility. The capability illustrated in Figure 4-3 is even more impressive. While anchored to the SCB with one arm, the mobile crane uses the other arm to position the module being transferred so that it can be grasped by the grasper behind the operator. This grasper then rotates the module as desired for clearance and the mobile crane carries it along the length of a fully stacked SCB. The two free arms move along from handhold to handhold with very low torque requirements, as a swimmer would move along a horizonal bar under water. The arm movements are sequential (not simultaneous) and at any given time one arm is always firmly attached to the SCB. It is very likely that control of the repetitive motion of moving between handholds could be automated.

CR5

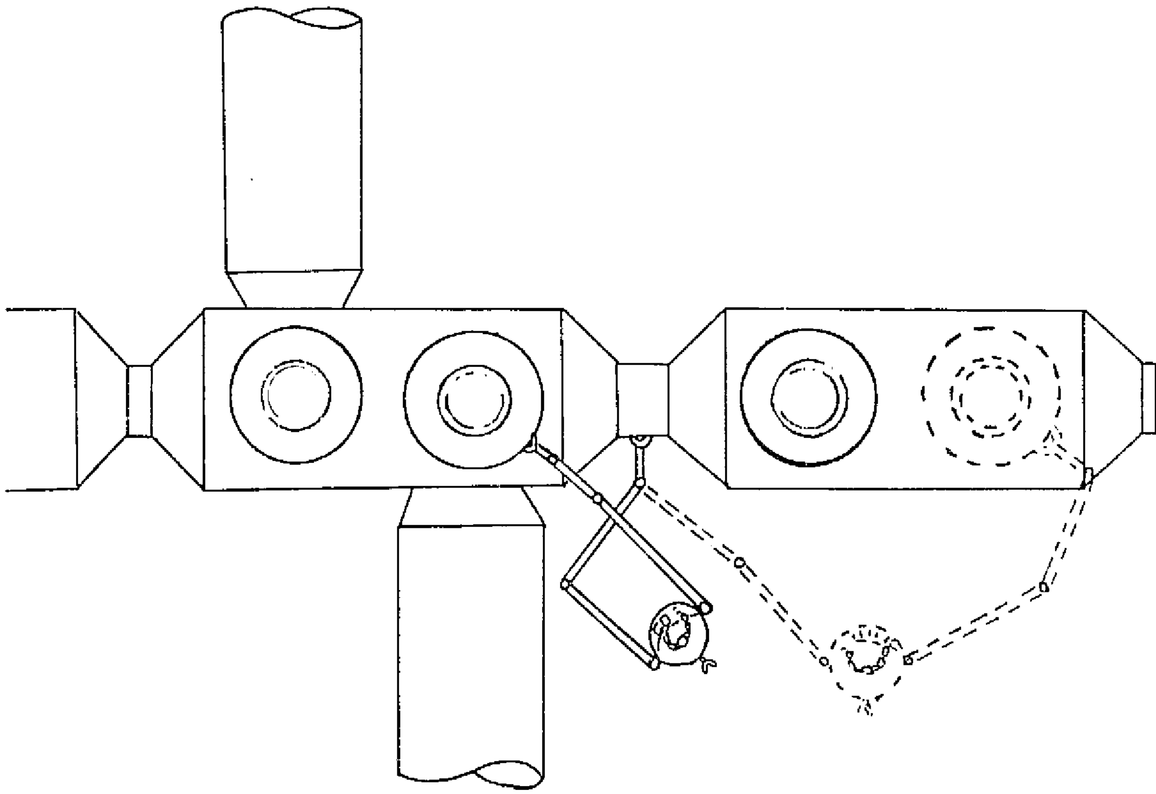

Figure 4-2. Mobile Crane – Without Stowing

CR5

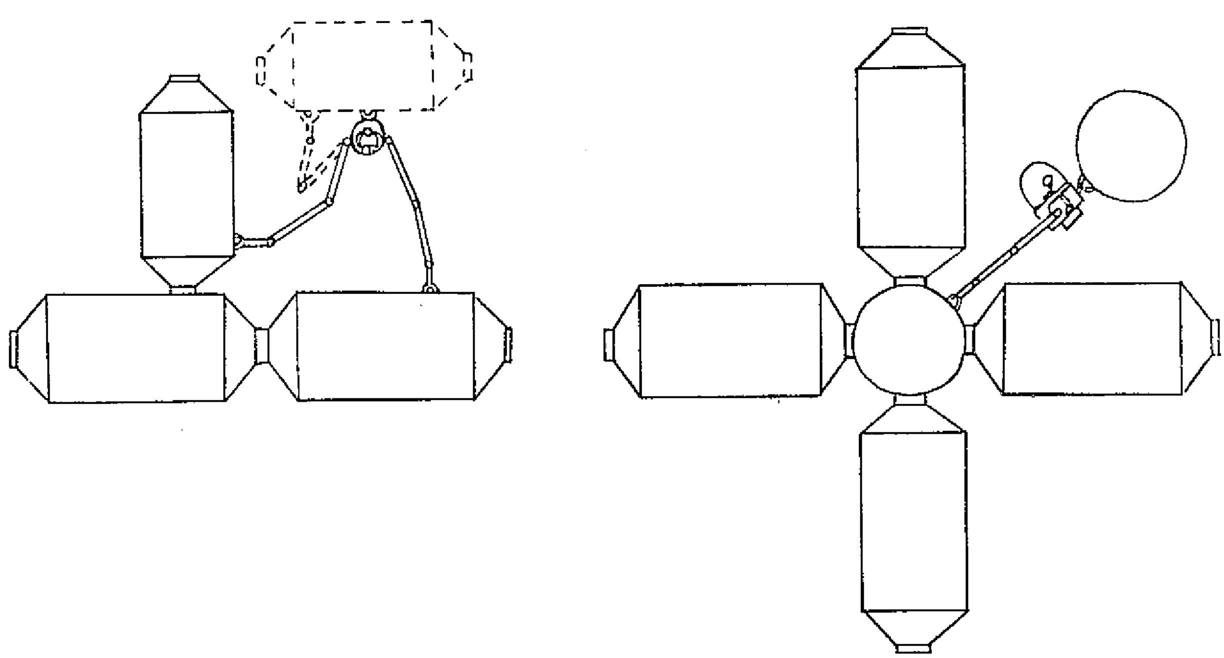

Figure 4-3. Mobile Crane – Stow and Walk

Table 4-1 is a summary of requirement differences associated with going from a fixed to a mobile crane. The mobile crane would need rechargeable batteries, on-board radio, and life support to give it autonomy. It would also be necessary to provide additional handholds so that the mobile crane could move about freely. There would be increased safety requirements, including circuitry to prevent accidental release of the last grasper, life support reserve, and an emergency homing beacon. On the other hand, arm length could be reduced to 15.24m (50 ft) and the requirement to keep an open corridor on the SCB could be dropped.

Table 4-1

MOBILE CRANE - REQUIREMENT DIFFERENCES

Increased or Added Requirements
• Twelve handholds per module.
• Rechargeable batteries (1,500 W-hr).
• Circuit to prevent accidental release of last grasper on base.
• On-board radio and emergency homing beacon.
• Life support to last TBD hr, plus TBD hr reserve
Reduced or Eliminated Requirements
• Arm reach reduced to 15.24m (50 ft).
• Open corridor not required.

Part 10

MISSION HARDWARE CONSTRUCTION OPERATIONAL FLOWS AND TIMELINES

MISSION HARDWARE CONSTRUCTION OPERATIONAL FLOWS AND TIMELINES

The construction-related objective elements which received emphasis during Part 2 were:

A. SPS Test Article-1 (TA-1)

B. SPS Test Article-2 (TA-2)

C. 30m Radiometer

D. 27m Multibeam Lens Antenna (MBL)

The requirements for both TA-1 and TA-2 dictated that they demonstrate on-orbit fabrication techniques and that TA-2 be prototypical of a model SPS being considered by JSC. TA-1 and TA-2 were designed accordingly, and the on-orbit construction sequence developed as illustrated in this appendix (Figures 1 and 2). Timelines (Figures 3 and 4) for these processes were developed and are also included. Assembly of a ground-fabricated TA-1 configuration was also analyzed. Though the two different approaches to construction of TA-1 (Figures 5 and 6) do not compare directly, (each assumed a different set of construction support tooling) some interesting observations can be made (see Figure 7): (1) either fabrication or a fabrication and assembly approach to TA-1 requires considerable EVA (in the assembly case it was for assembly; in the fabrication and assembly case it was for setting up the tooling) and (2) delivery of preassembled sections, because of their low density, require significantly more Shuttle logistics flights.

The configurations for the 30m radiometer and MBL were analyzed with respect to whether or not they could be fabricated on orbit. Deployment was ruled out as a result of study Part 1. The result was that the tight tolerances require ground fabrication with final assembly done on-orbit due to the size of the antennas. Some support structure could be fabricated on orbit however. The approach considered for the 30m radiometer and illustrated by the flow (Figure 8) and timeline (Figure 9) included in this appendix involve assembly

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on a turntable located at the end of a strongback. An EVA scaffold is erected and the work rotated by the turntable to the EVA astronauts on the scaffold. Subsequent analyses revealed that some savings in time could be realized if the scaffold was replaced by a cherry picker on the end of a crane arm. This would allow the EVA astronauts to maneuver themselves the work— the turntable would still be used however.

Assembly of the MBL was analyzed assuming that the work was located on a turntable, but not out on a strongback. EVA access was considered to be provided by a cherry picker arrangement. The resultant flow (Figure 10) revealed some awkward operations, and the conclusion is drawn that like the 30m radiometer, the MBL should be assembled on a turntable at the end of a strongback.

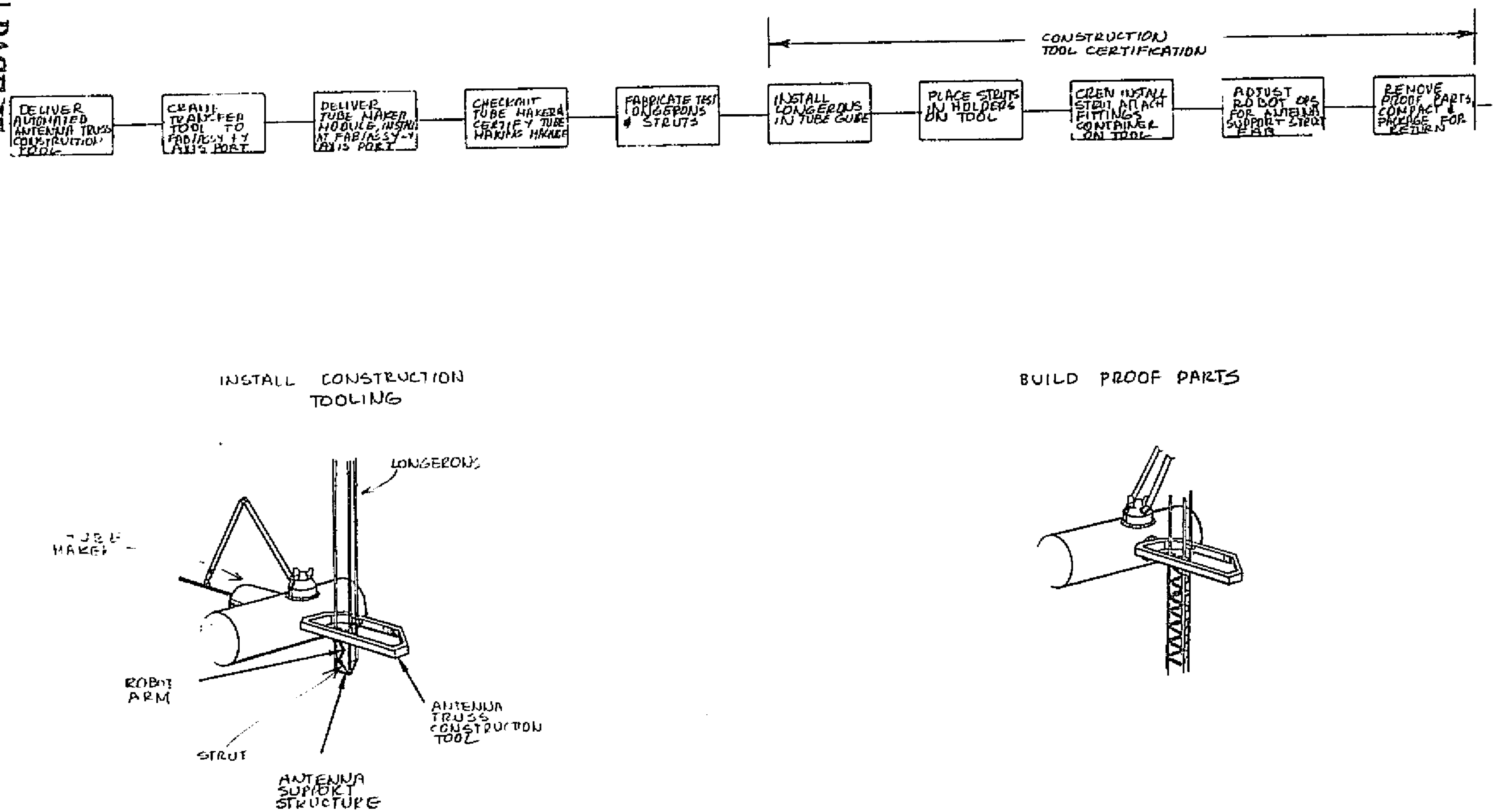
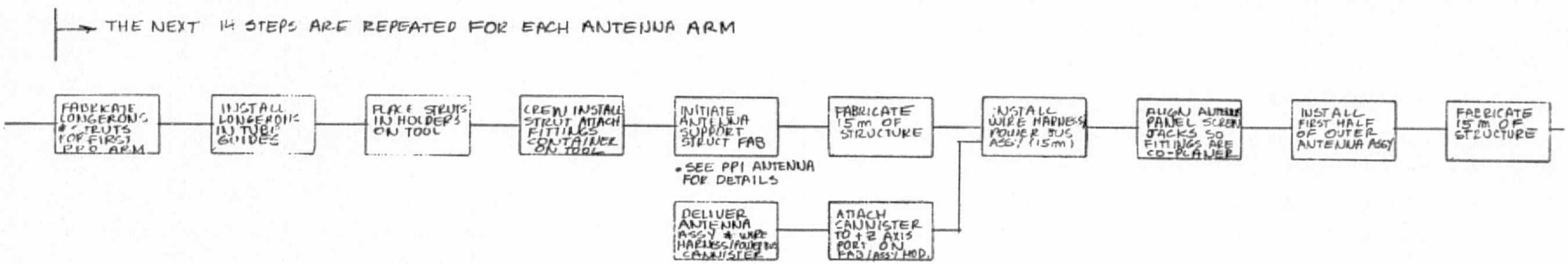


Figure 1. SPS Test Article 1 — Construction (1 of 4)



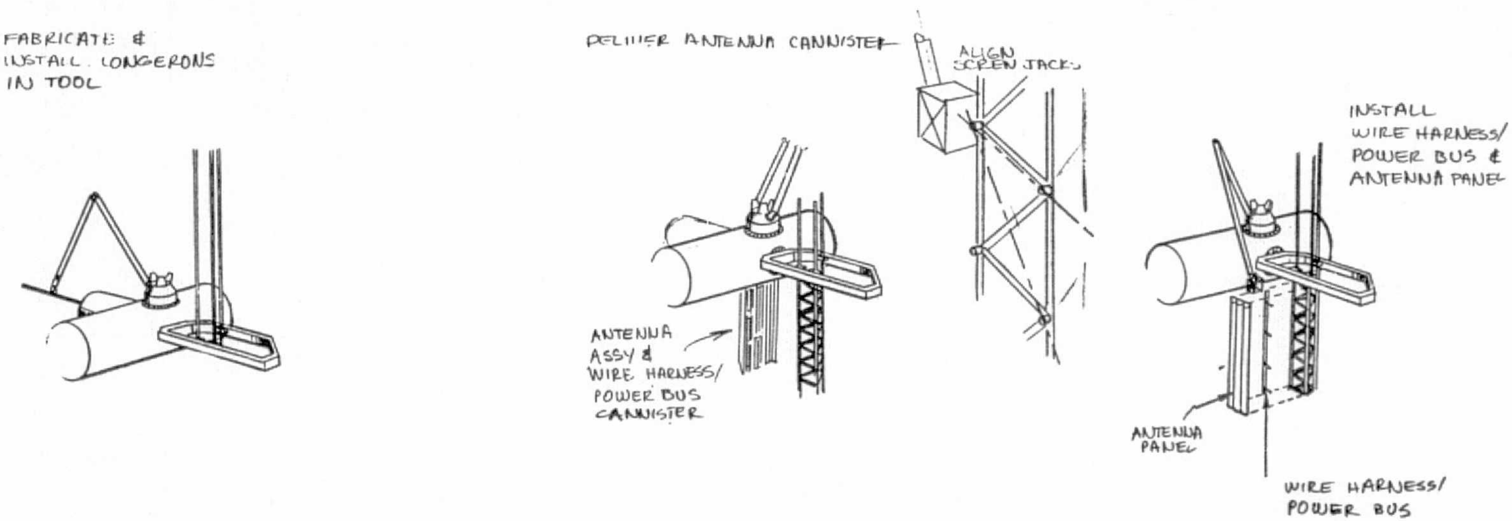


Figure 1. SPS Test Article 1 – Construction (2 of 4)

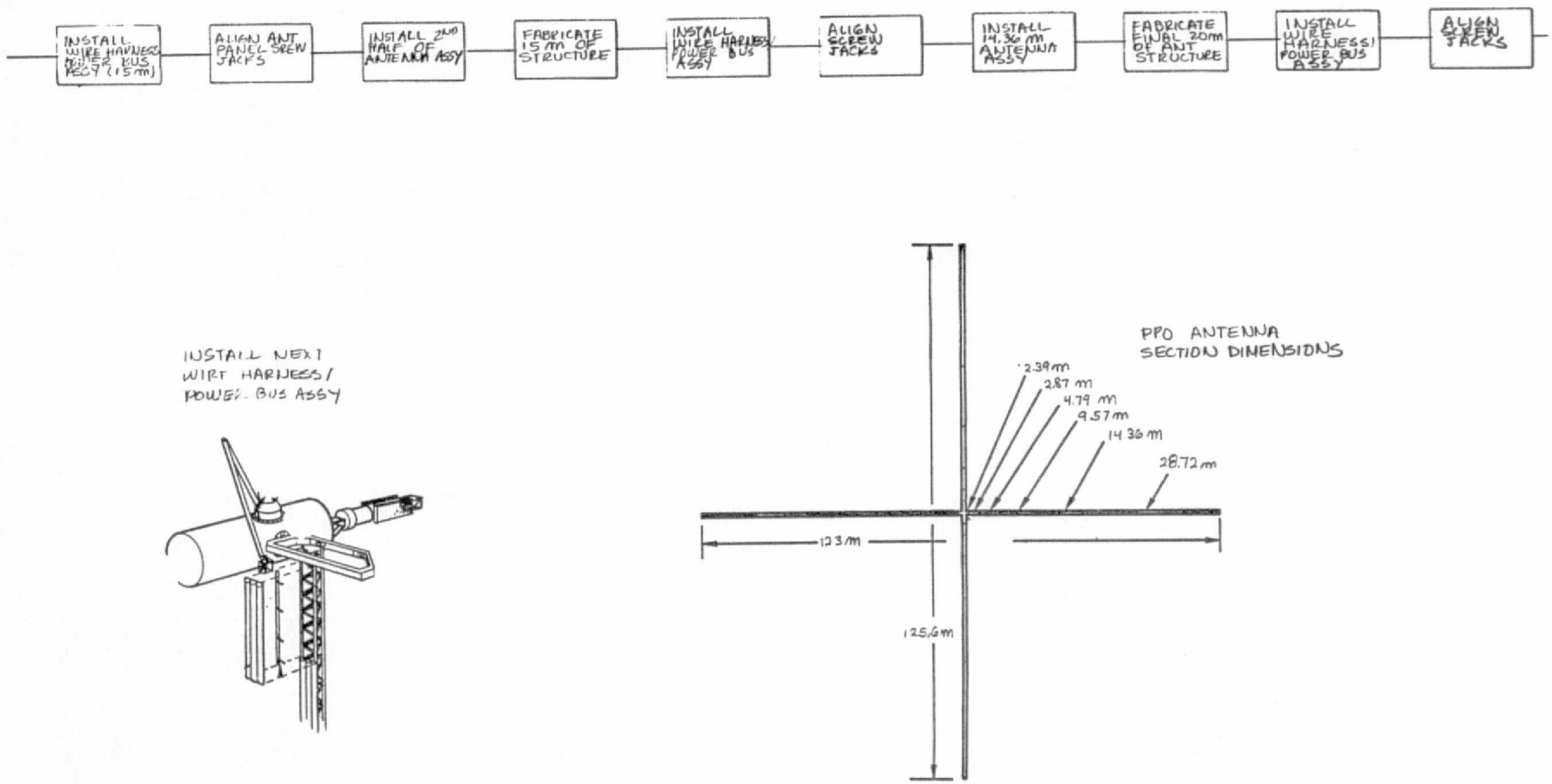
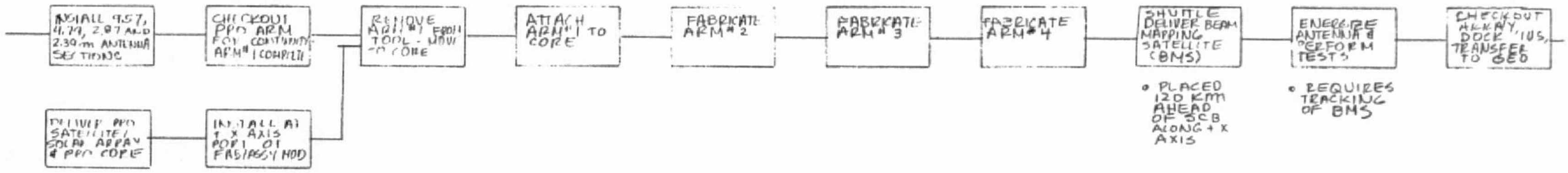
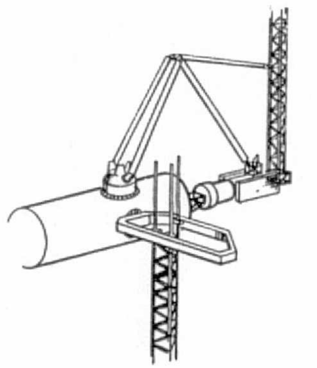


Figure 1. SPS Test Article 1 – Construction (3 of 4)



INSTALL ARM #1 IN CORE



FABRICATE & INSTALL ARMS # 3, 4 & 5

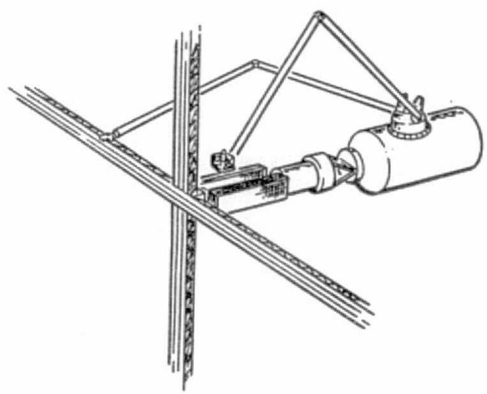


Figure 1. SPS Test Article 1 – Construction (4 of 4)

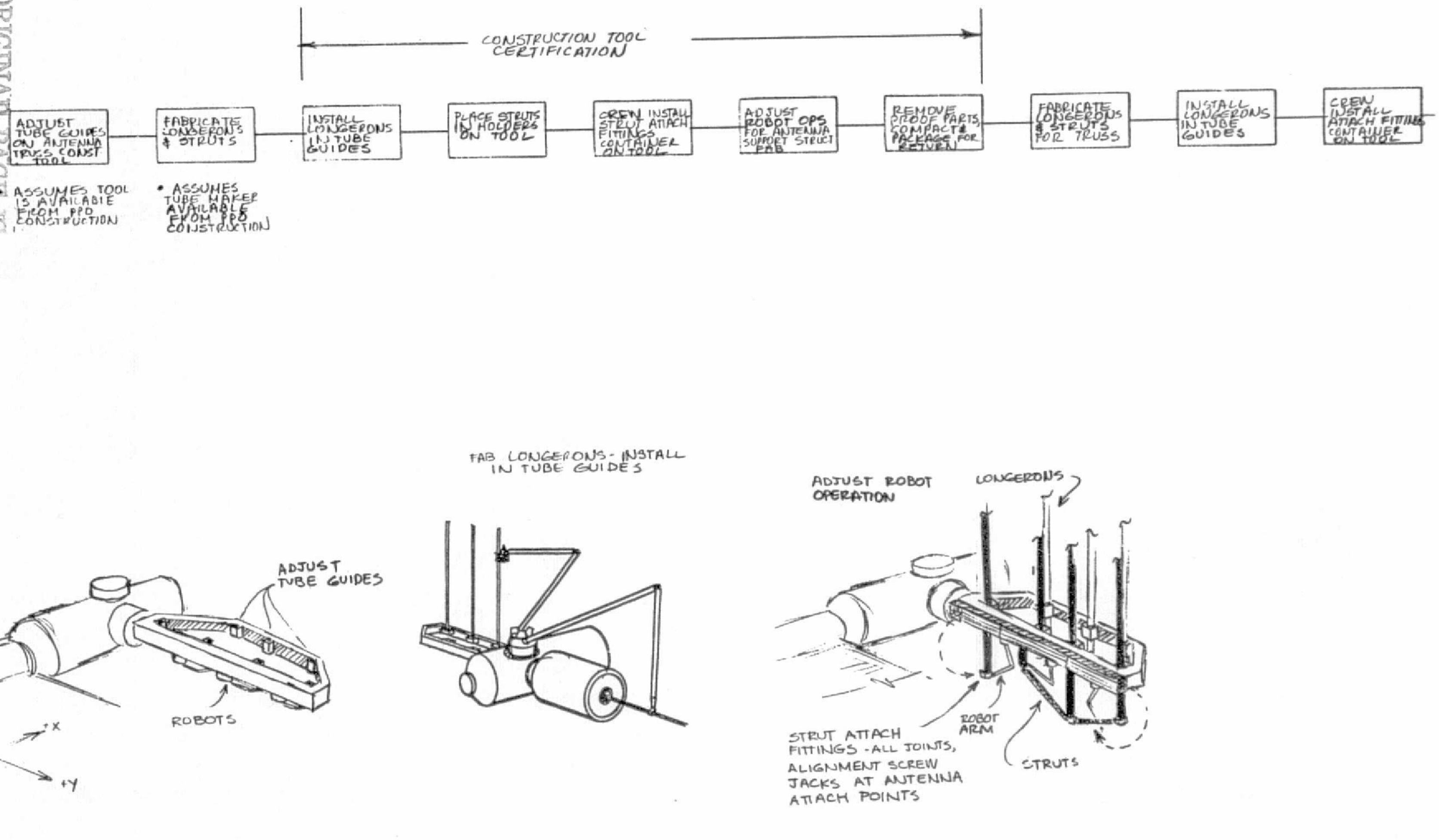


Figure 2. SPS Test Article - 2 Construction (1 of 7)

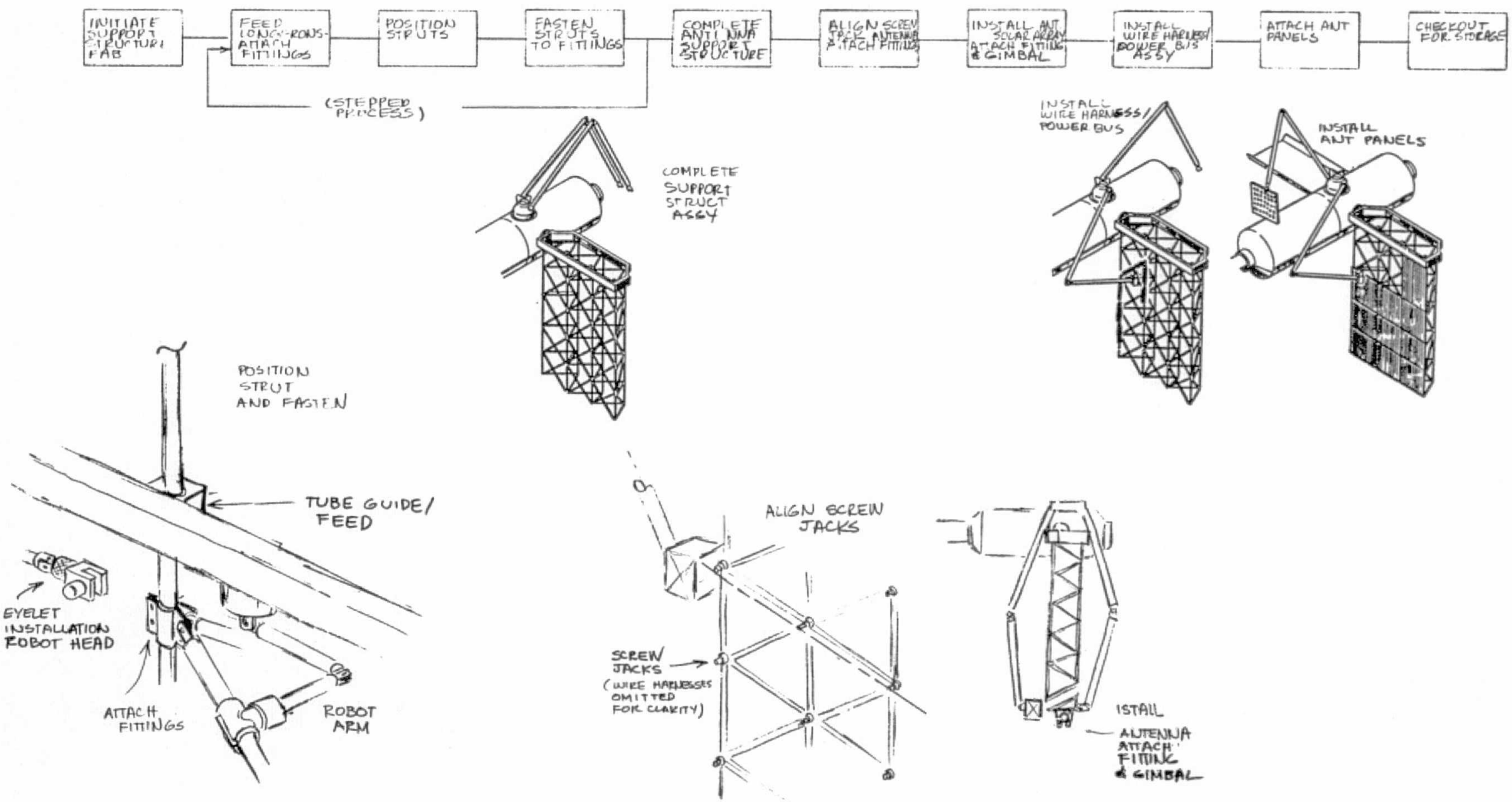


Figure 2. SPS Test Article - 2 Construction (2 of 7)

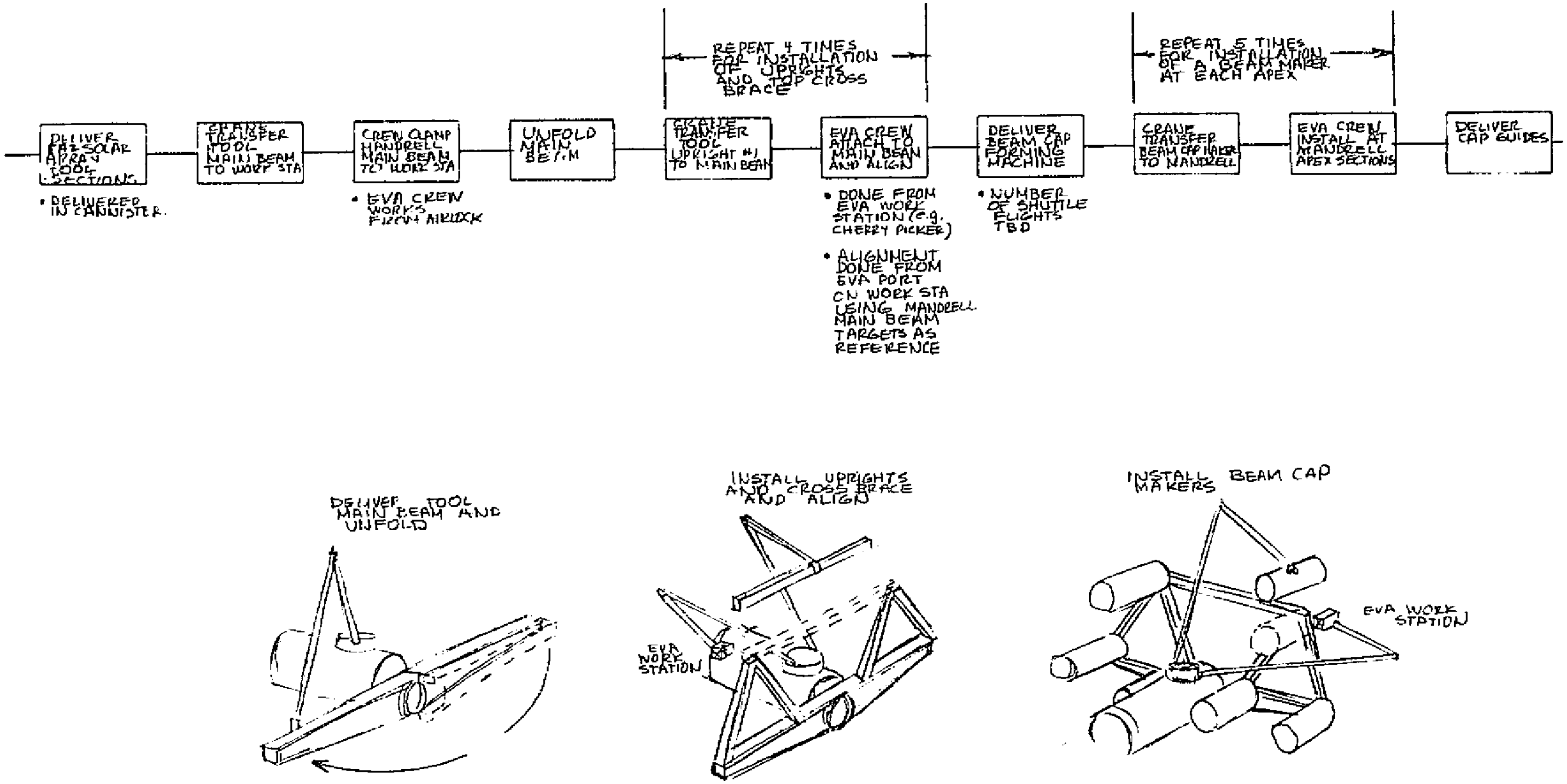


Figure 2. SPS Test Article - 2 Construction (3 of 7)

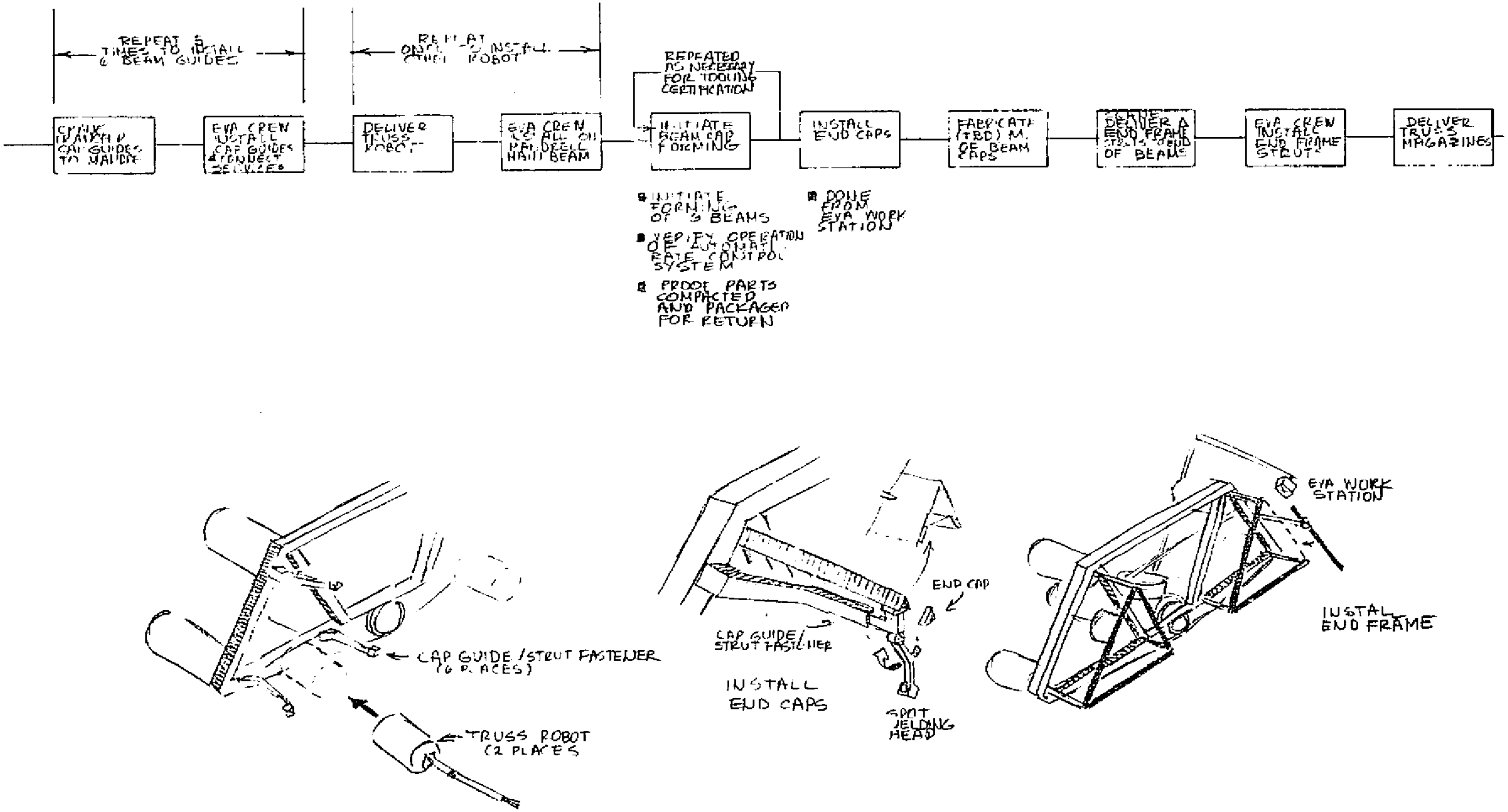


Figure 2. SPS Test Article - 2 Construction (4 of 7)

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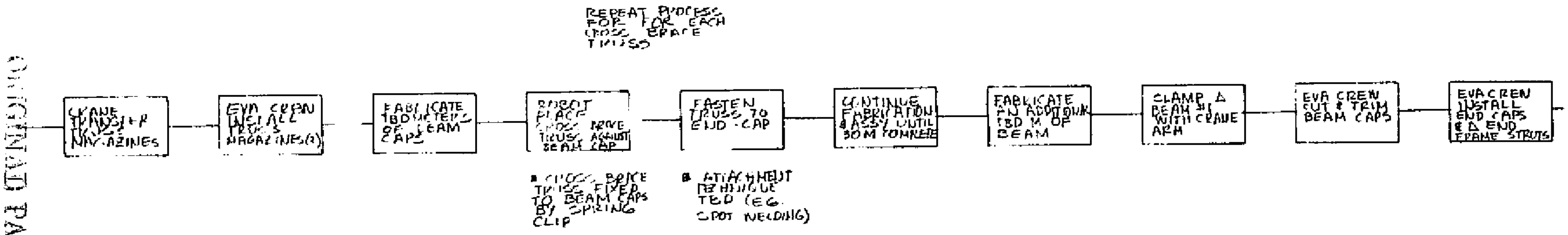

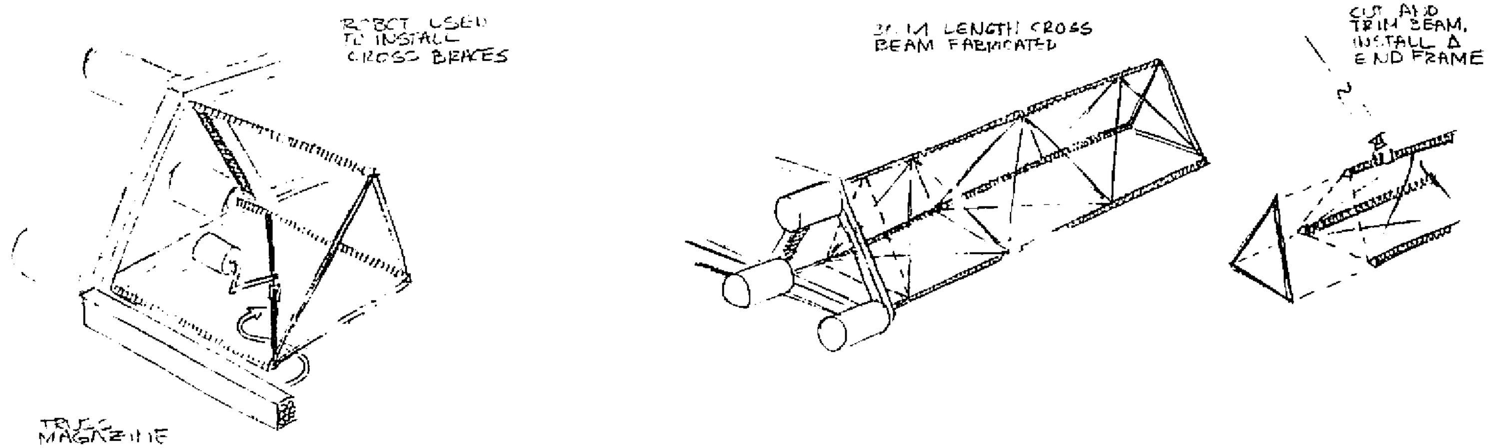

Figure 2. SPS Test Article - 2 Construction (5 of 7)

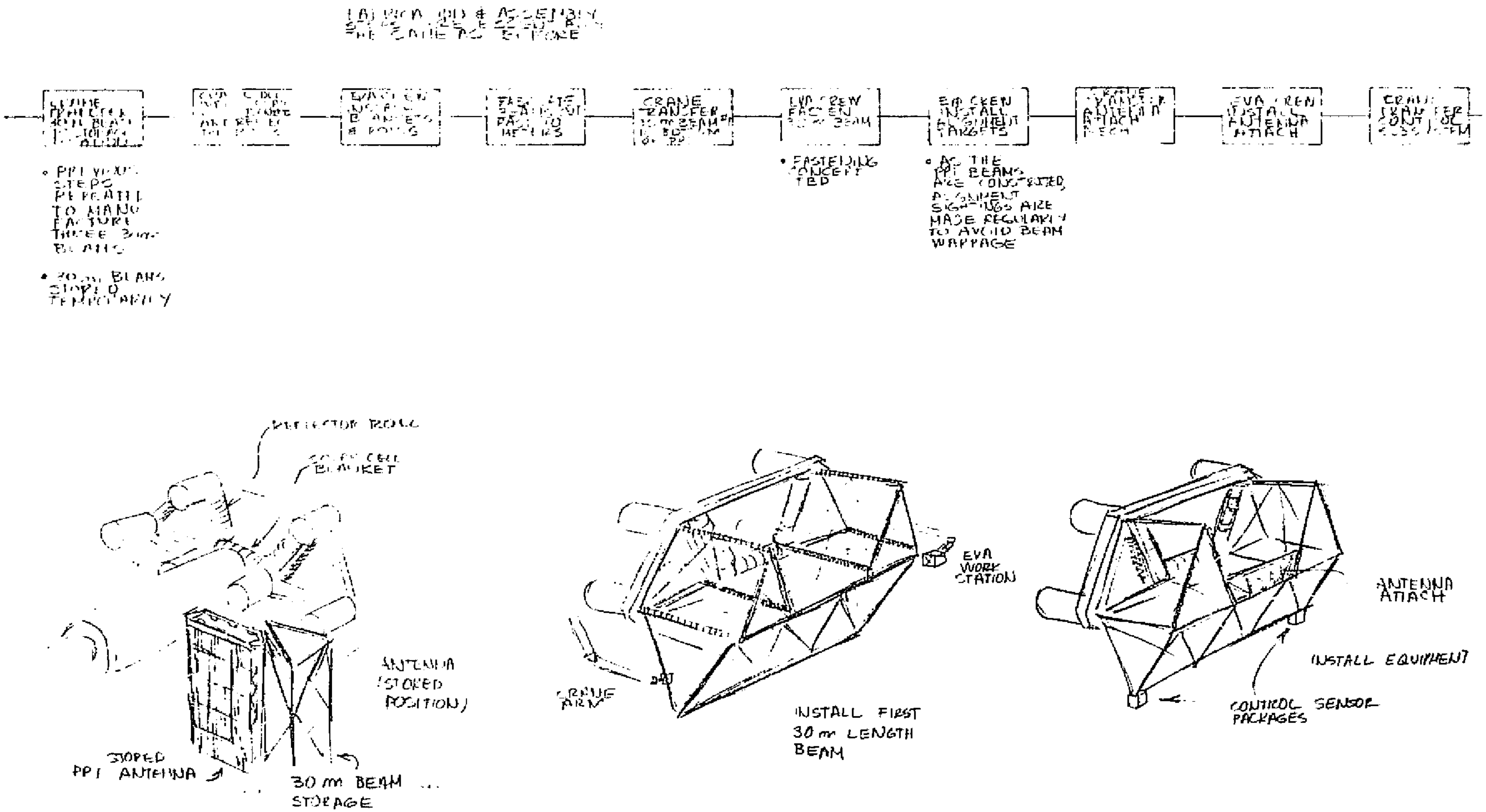

Figure 2. SPS Test Article - 2 Construction (6 of 7)

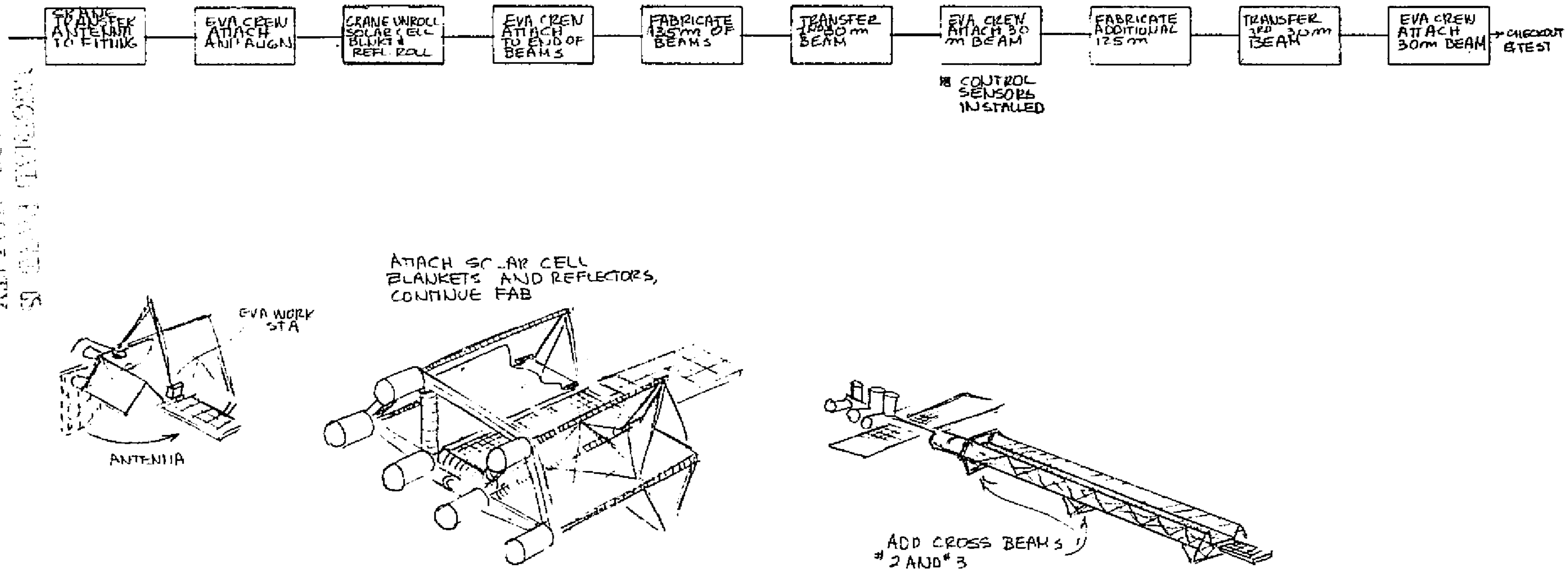

Figure 2. SPS Test Article - 2 Construction (7 of 7)

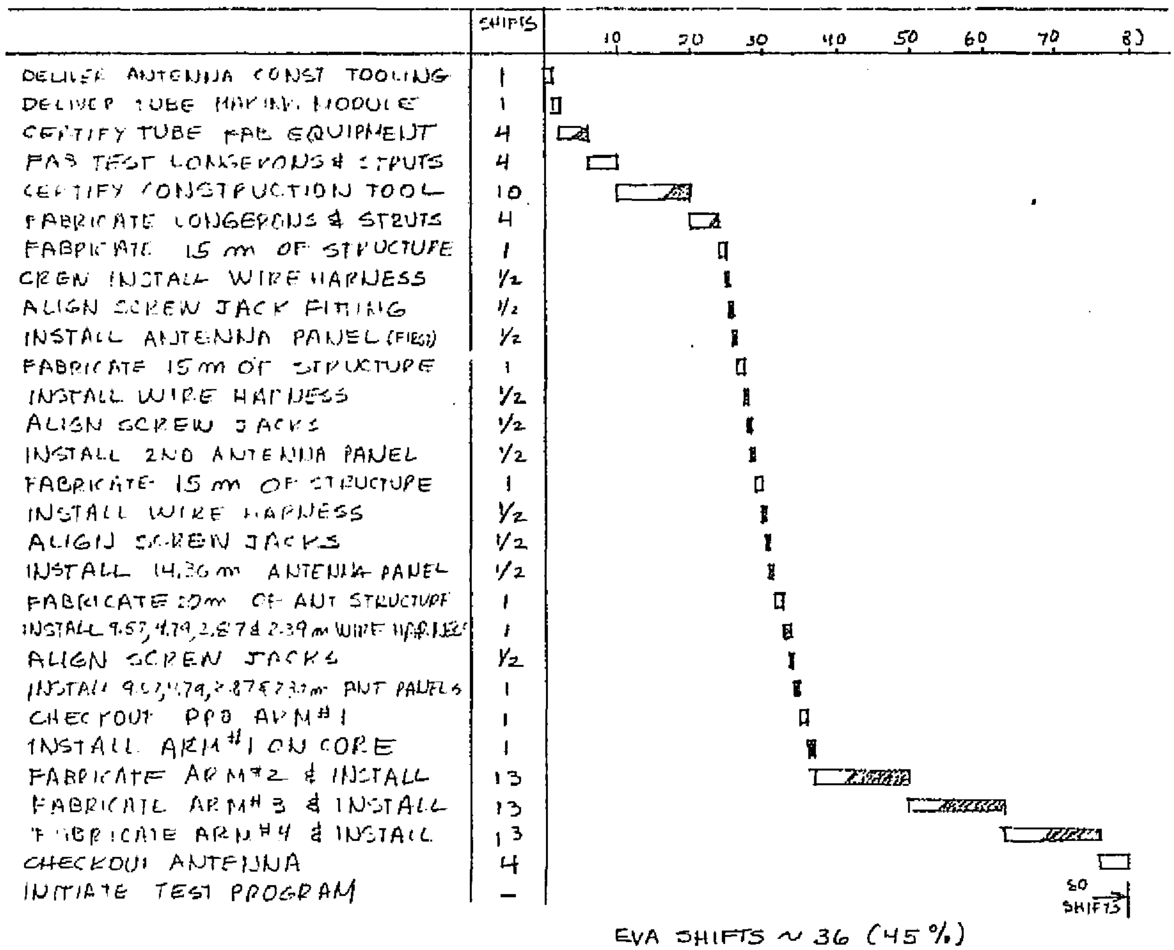

Figure 3. TA-1 Construction (FAB)

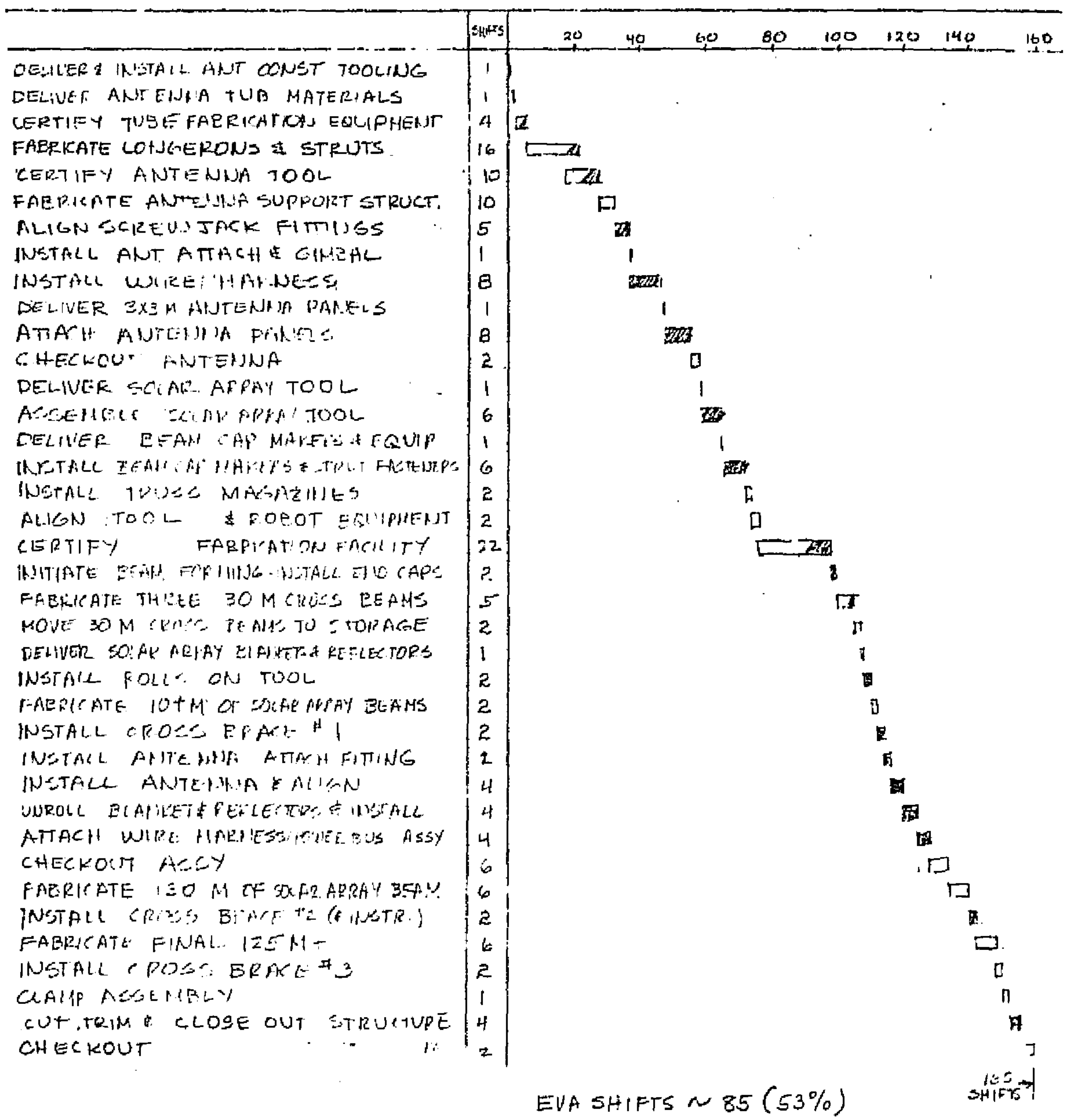

Figure 4. TA-2 Construction (FAB)

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	SHIFTS
DELIVER ARM SECTIONS 1-1, 1-2, 1-3, 1-4	1.0
INSTALL SECTION (1-1) ON STRONGBACK	1.0
INSTALL MOBILE SCAFFOLD	1.5
DEPLOY SECTION 1 & LOCK	.5
ALIGN SCREW JACK FITTINGS	1.5
INSTALL WIRE HARNESS/ELECTRONICS ASSY	1.5
INSTALL ANTENNA PANELS	1.5
CHECKOUT SECTION (1-1)	1.0
INSTALL ARM STORAGE ATTACH STRUT (ASAS)	1.0
TRANSFER (1-1) TO ASAS & CLAMP	.5
INSTALL SECTION (1-2) ON STRONGBACK	1.0
DEPLOY SECTION (1-2) & LOCK	.5
ALIGN SCREW JACK FITTINGS	1.5
INSTALL WIRE HARNESS/ELECTRONICS ASSY	1.5
INSTALL ANTENNA PANELS	1.5
CHECKOUT SECTION (1-2)	1.0
INSTALL SECTION JOINING SCAFFOLD	1.5
TRANSFER (1-2) AND JOIN TO (1-1)	.5
TRANSLATE (1-1, 1-2) COMBO AFT	.5
INSTALL SECTION (1-3) ON STRONGBACK	1.0
DEPLOY SECTION (1-3) & LOCK	.5
ALIGN SCREW JACK FITTINGS	1.5
INSTALL WIRE HARNESS/ELECTRONICS ASSY	1.5
INSTALL ANTENNA PANELS	1.5
CHECKOUT SECTION (1-3)	1.0
TRANSFER (1-3) AND JOIN TO (1-2)	.5
TRANSLATE (1-2, 1-3) COMBO AFT	.5
INSTALL SECTION (1-4) ON STRONGBACK	1.0
DEPLOY SECTION (1-4) & LOCK	.5
ALIGN SCREW JACK FITTINGS	1.5
INSTALL WIRE HARNESS/ELECTRONICS ASSY	1.5
INSTALL ANTENNA PANELS	1.5
CHECKOUT SECTION (1-4)	1.0
TRANSFER SECTION (1-4) AND JOIN TO (1-3)	.5
TRANSLATE (1-3, 1-4) COMBO AFT	.5
DELIVER TA-1 SATELLITE/SOLAR ARRAY	1.0
ERECT ARM TO SATELLITE SCAFFOLD	1.0
TRANSFER ARM SECTION TO SATELLITE	1.0
ROTATE +90° ABOUT Z AXIS	1.0
ASSEMBLE & ATTACH ARM #2	36.5
ASSEMBLE & ATTACH ARM #3	36.5
ASSEMBLE & ATTACH ARM #4	36.5
CHECKOUT TA-1	4.0
ROTATE STRONG BACK HUB -90° FACILITY	1.0
INITIATE TEST PROGRAM	—

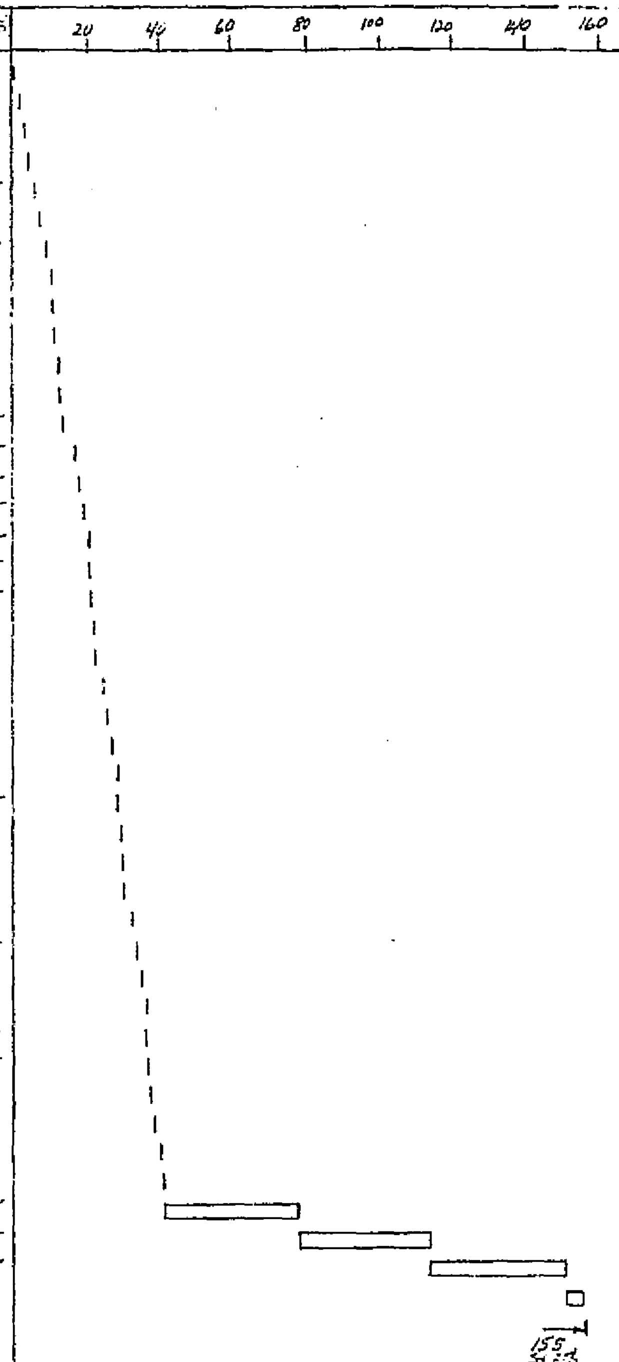

Figure 5. TA-1 Assembly — Shuttle-Tended Strongback Concept (Arm Sections Deploy on Orbit)

	SHIFTS
DELIVER ARM #1 Section #1 (1-1)	1.0
INSTALL Section (1-1) ON STRONG BACK	1.0
INSTALL MOVABLE SCAFFOLD	1.5
ALIGN SCREW JACK FITTINGS	1.5
INSTALL WIRE HARNESS/Electronics ASSY	1.5
INSTALL ANTENNA PANELS	1.5
CHECKOUT SECTION (1-1)	1.0
INSTALL ARM STORAGE ATTACH STRUT (ASAS)	1.0
TRANSFER (1-1) TO ASAS & CLAMP	.5
DELIVER ARM #1 Section #2 (1-2)	1.0
INSTALL Section (1-2) ON STRONG BACK	1.0
ALIGN SCREW JACK FITTINGS	1.5
INSTALL WIRE HARNESS/Electronics ASSY	1.5
INSTALL ANTENNA PANELS	1.5
CHECKOUT SECTION (1-2)	1.0
INSTALL SECTION JOINING SCAFFOLD	1.5
TRANSFER (1-2) AND JOIN TO (1-1)	.5
TRANSLATE (1-1, 1-2) COMBO AFT	.5
DELIVER ARM #1 SECTION #3 (1-3)	1.0
INSTALL (1-3) ON STRONG BACK	1.0
ALIGN SCREW JACK FITTINGS	1.5
INSTALL WIRE HARNESS/ELECTRONICS ASSY	1.5
INSTALL ANTENNA PANELS	1.5
CHECKOUT SECTION (1-3)	1.0
TRANSFER (1-3) & JOIN TO (1-2)	.5
TRANSLATE (1-3 & 1-4) COMBO AFT	.5
Deliver ARM #1 Section #4 (1-4)	1.0
INSTALL (1-4) ON STRONG BACK	1.0
ALIGN SCREW JACK FITTINGS	1.5
INSTALL WIRE HARNESS/Electronics ASSY	1.5
INSTALL ANTENNA PANELS	1.5
CHECKOUT SECTION (1-4)	1.0
TRANSFER (1-4) & JOIN TO (1-5)	.5
TRANSLATE (1-3 & 1-4) COMBO AFT	.5
DELIVER TA-1 SATELLITE/SOLAR ARRAY	1.0
ERECT ARM TO SATELLITE SCAFFOLD	1.0
TRANSFER ARMS & JOIN TO SATELLITE	1.0
ROTATE +90° ABOUT Z AXIS	1.0
ASSEMBLE & ATTACH ARM #2	37.5
ASSEMBLE & ATTACH ARM #3	37.5
ASSEMBLE & ATTACH ARM #4	37.5
- CHECKOUT TA-1	4.0
- ROTATE STRONG BACK HUB -90° ABOUT Y	1.0
INITIATE TEST PROGRAM	—

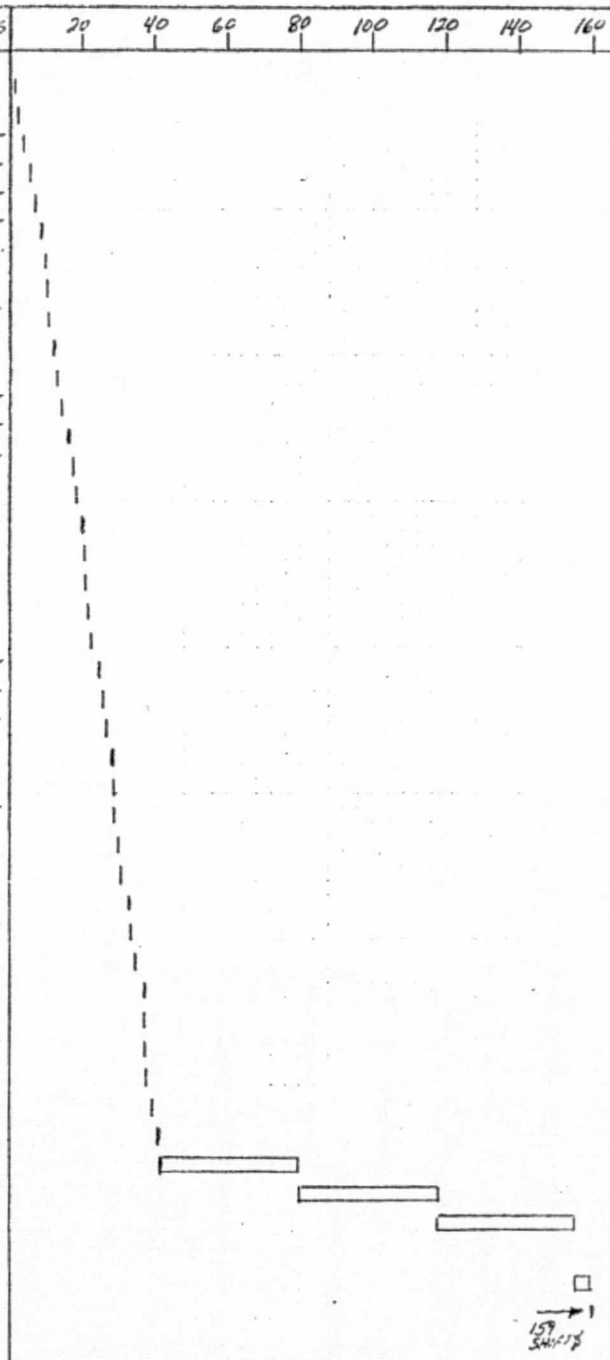

Figure 6. TA-1 Assembly — Shuttle-Tended Strongback Concept (Arm Sections Pre-fabbed on Ground)

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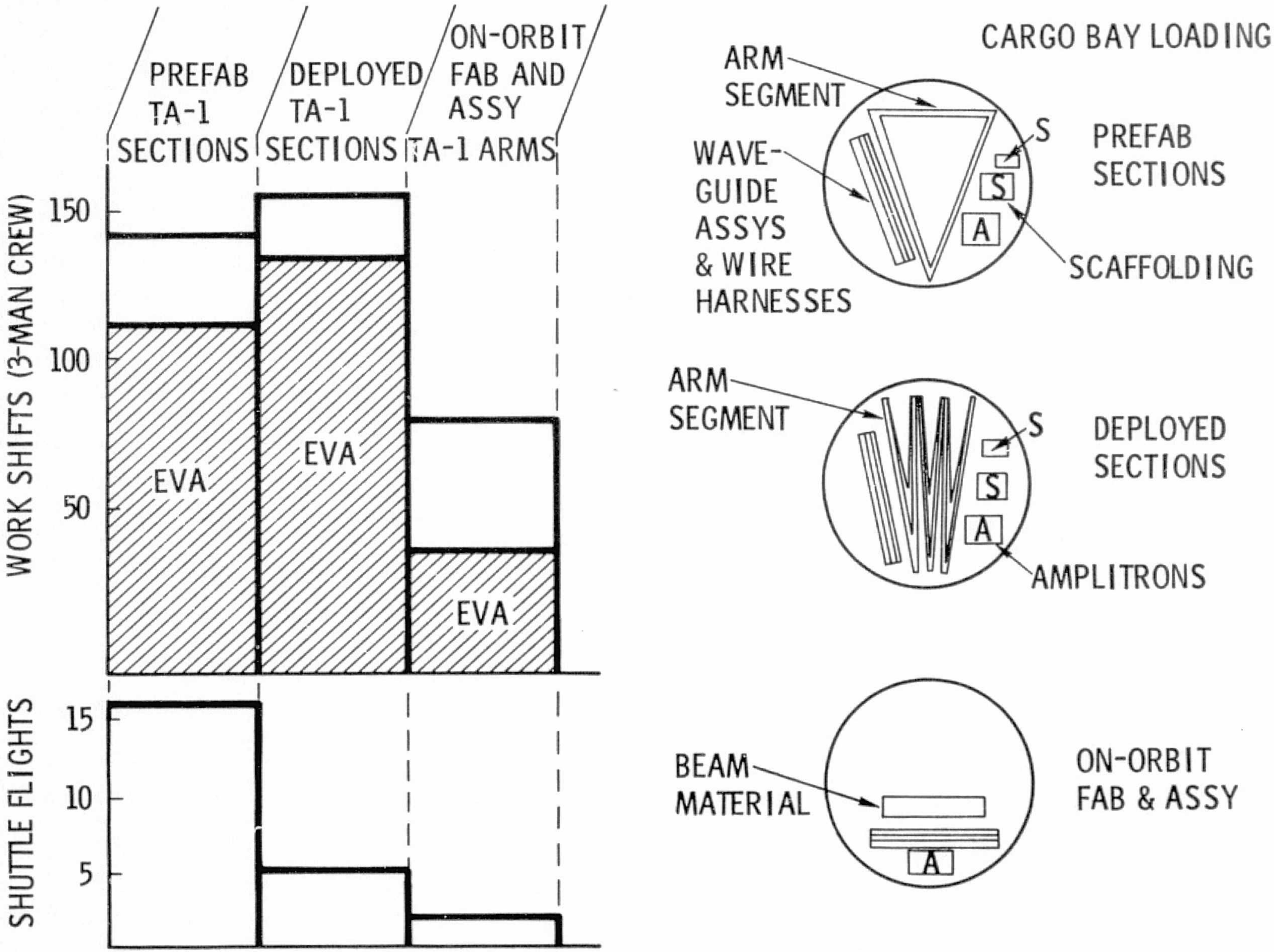


Figure 7. Requirements for TA-1 Construction Concepts

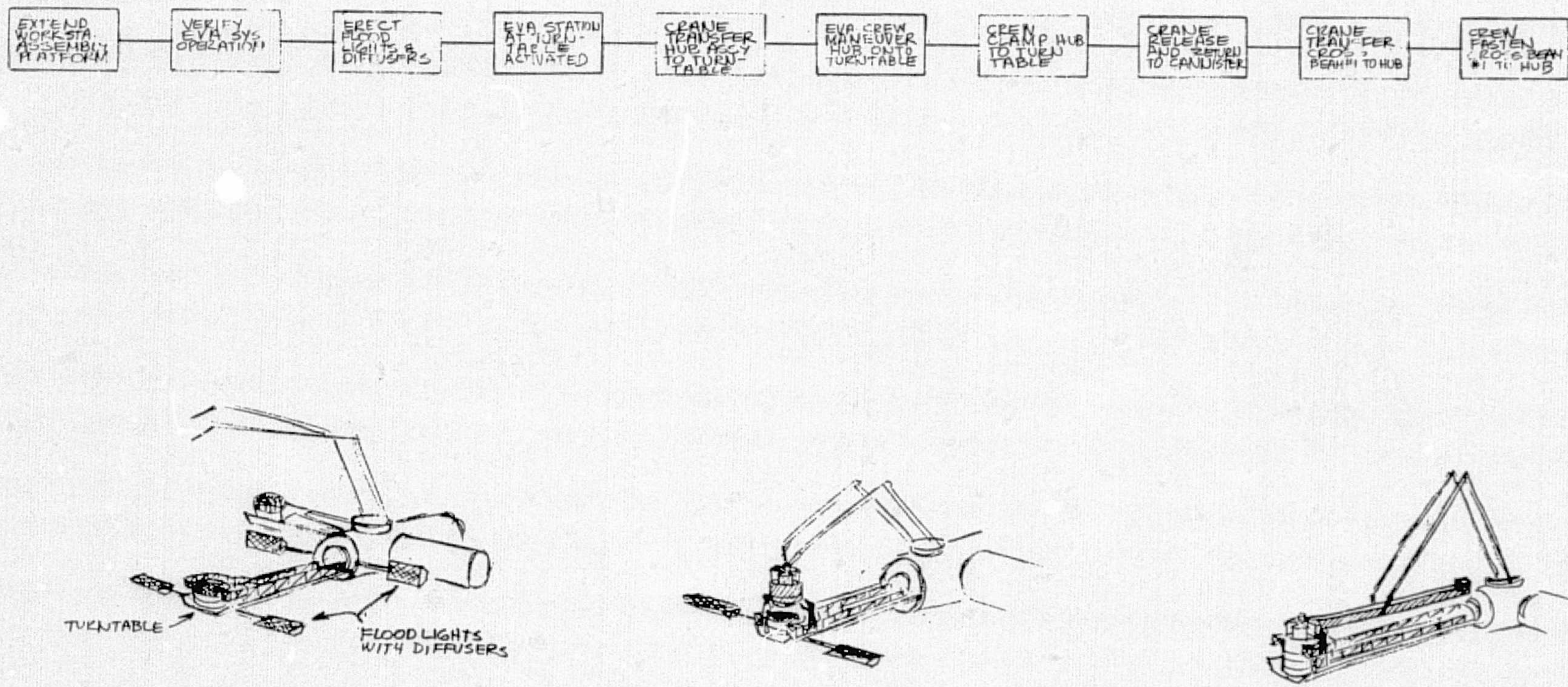


Figure 8. 3M Radiometer Assembly (1 of 4)

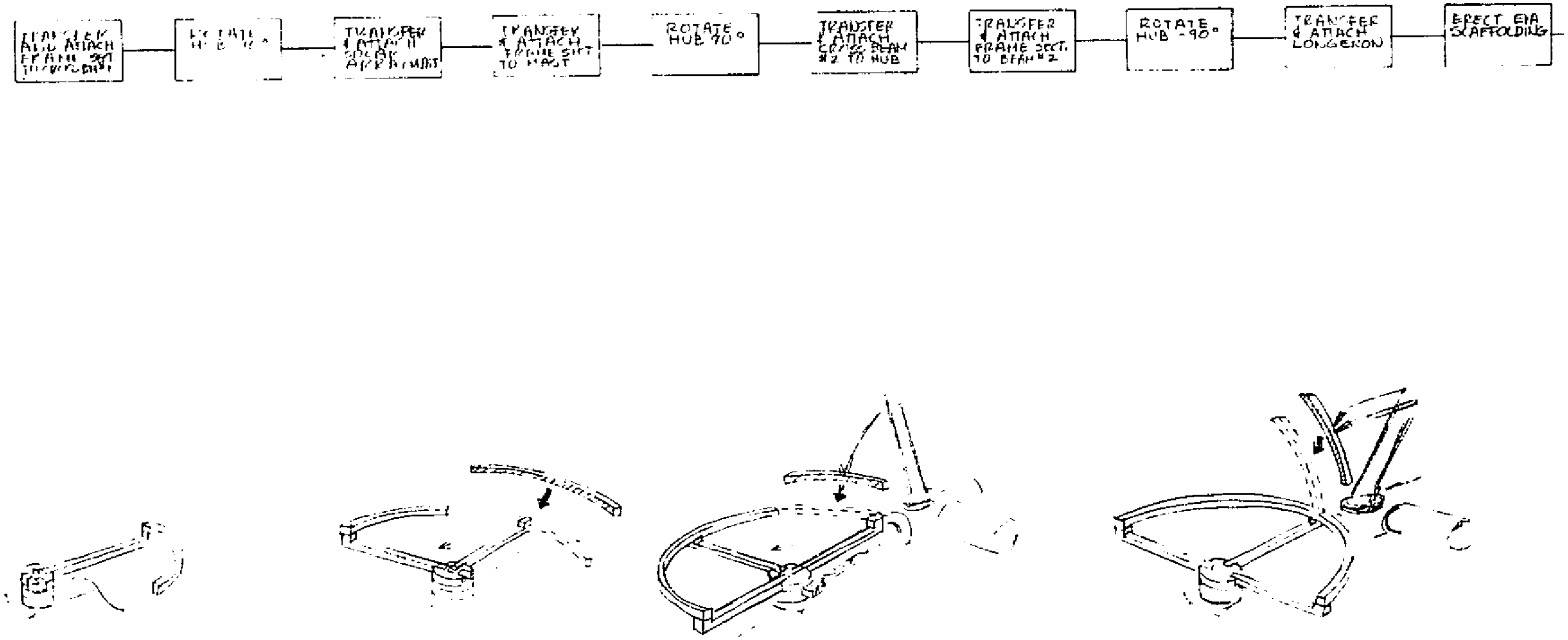


Figure 8. 3M Radiometer Assembly (2 of 4)

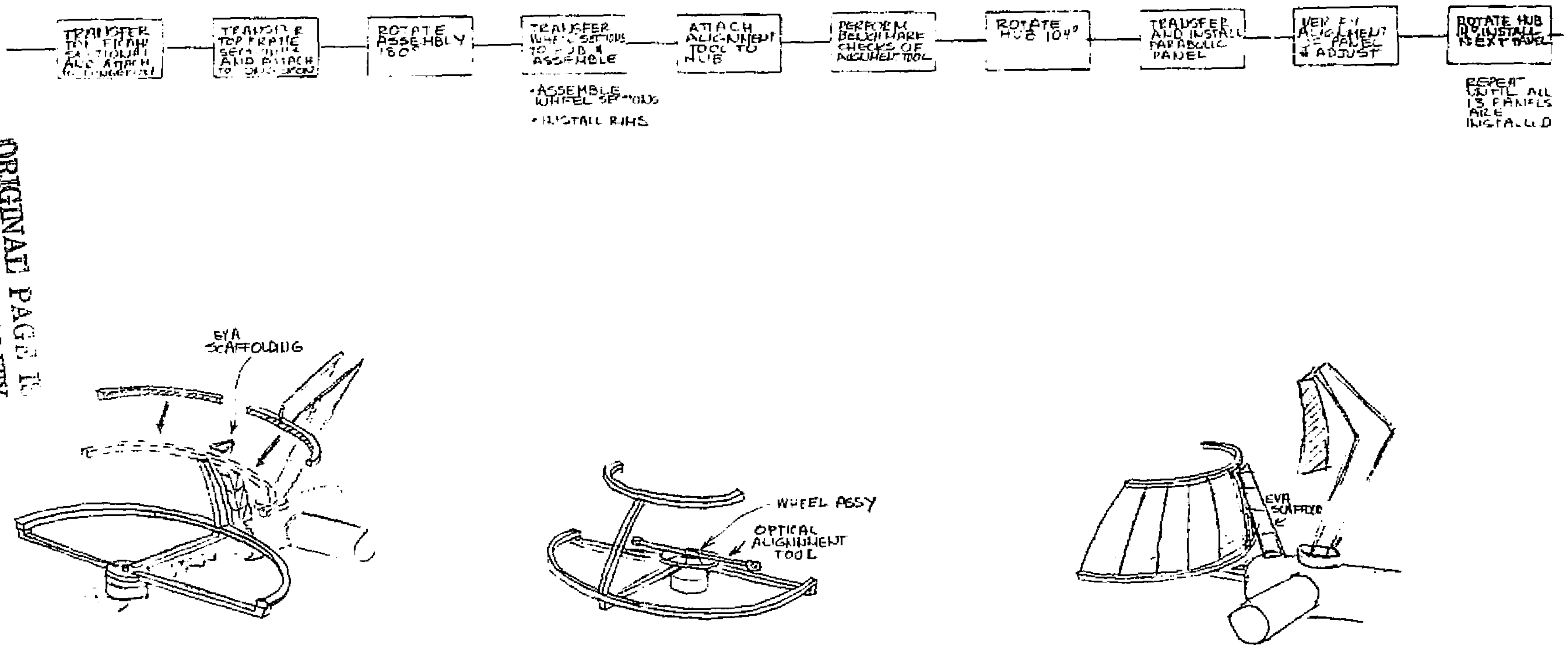


Figure 8. 3M Radiometer Assembly (3 of 4)

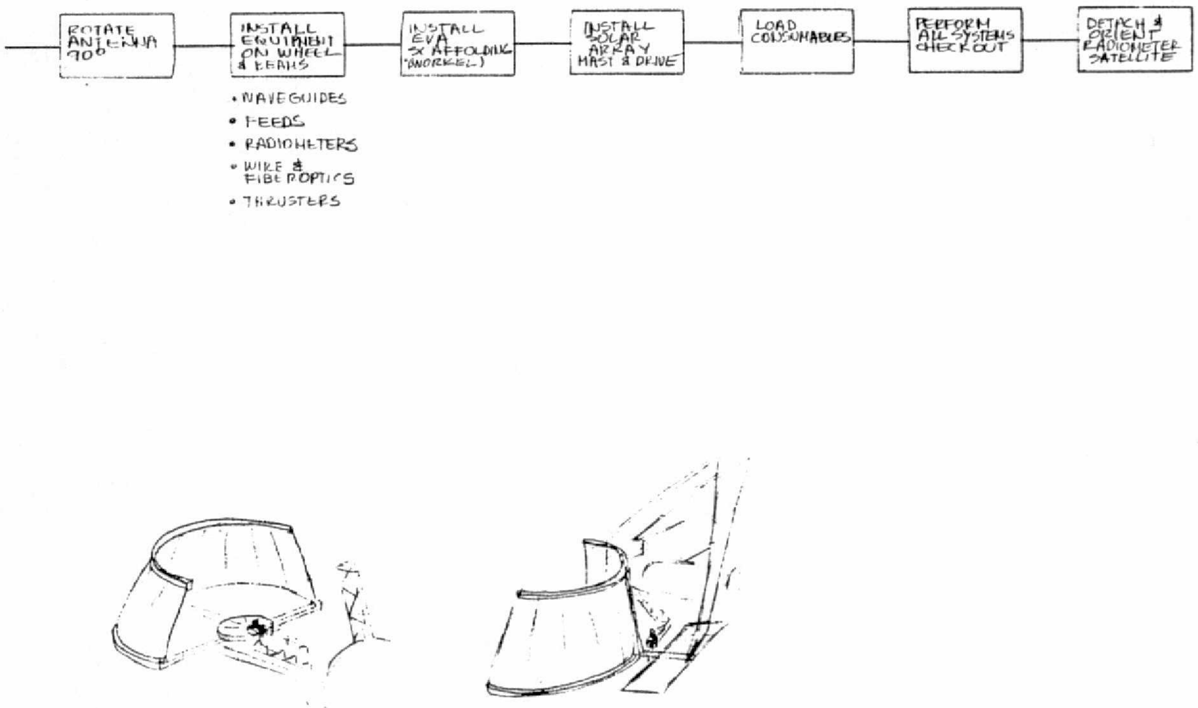


Figure 8. 3M Radiometer Assembly (4 of 4)

Shifts	Task
2	DELIVER CARGO MODULE 1 (SATELLITE BODY, TURNTABLE, ELECTRONICS)
2	DELIVER CARGO MODULE 2 (ANTENNA PANELS, STRUCTURE)
3	INSTALL WORK STATIONS, LIGHTS; POSITION CRANE
0.5	INSTALL SATELLITE BODY ON TURNTABLE
6	INSTALL FRAMES AND BEAMS AND ALIGN
6	INSTALL PANELS AND ALIGN
3	INSTALL/ASSEMBLE ROTATING TABLE
6	INSTALL SECONDARY MIRRORS AND FEEDS
12	ALIGN MIRRORS AND FEEDS
12	INSTALL RADIOMETERS/ELECTRONIC EQUIPMENT
2	INSTALL SOLAR ARRAY MAST AND PANELS
5	INSTALL THRUSTERS COMM ANTENNA, ETC.
3	CHECKOUT SATELLITE SYSTEMS
12	CHECKOUT/CALIBRATE RADIOMETERS
2	ALL SYSTEMS CHECKOUT
0.5	LOAD CONSUMABLES
0.5	DETACH AND ORIENT SATELLITE

71.5 SHIFTS
TOTAL TIME

Figure 9. Radiometer Assembly Sequence

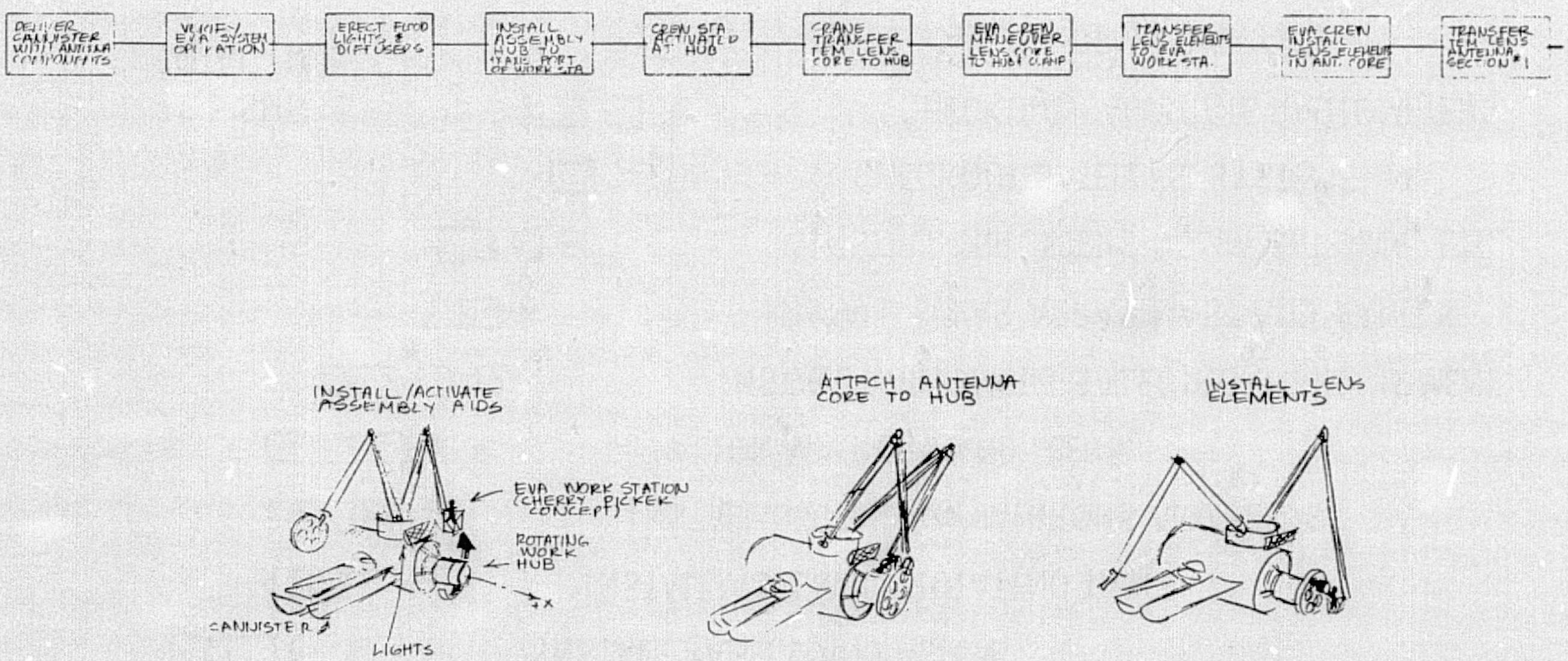


Figure 10. MBL Assembly (1 of 5)

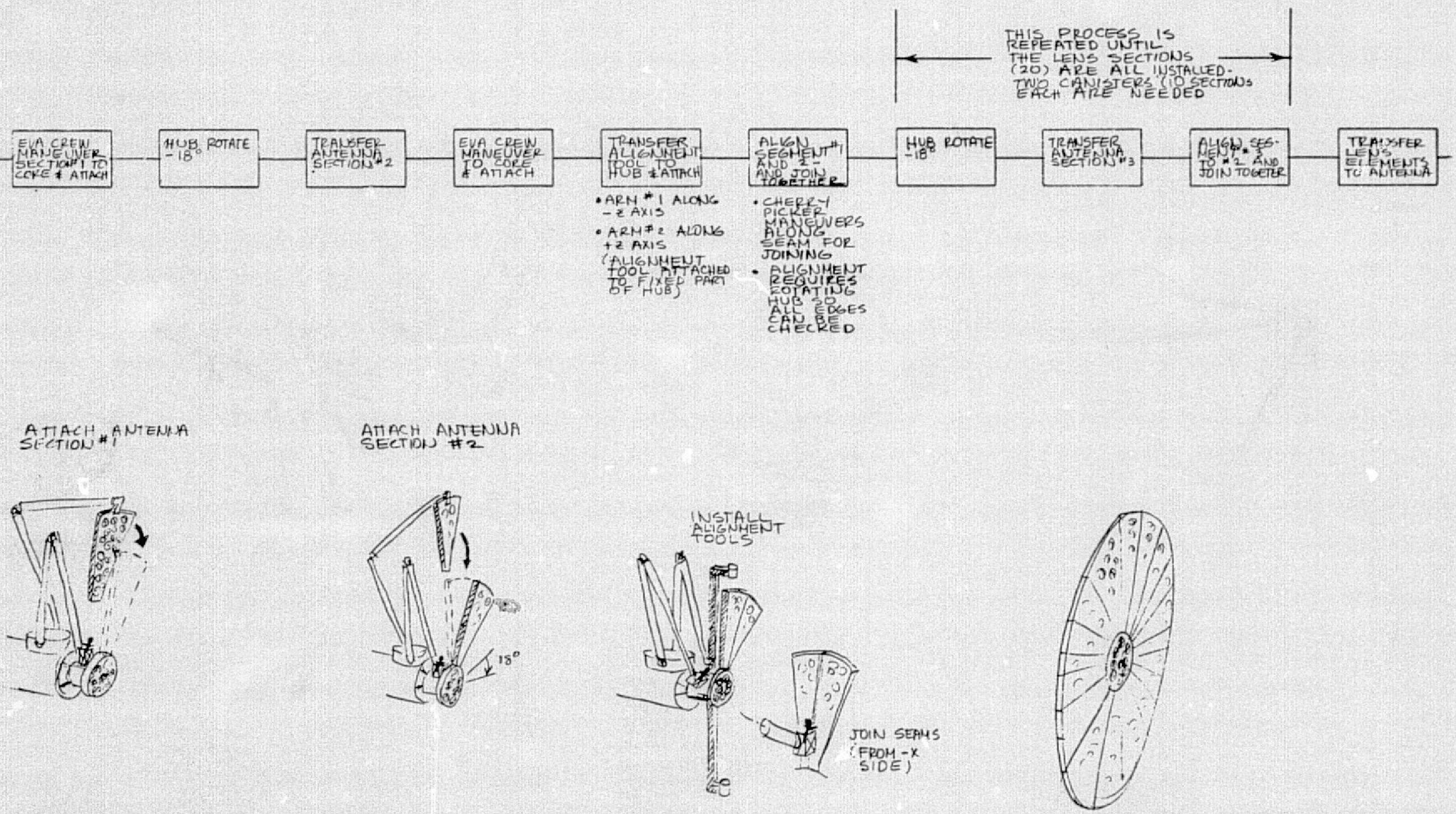


Figure 10. MBL Assembly (2 of 5)

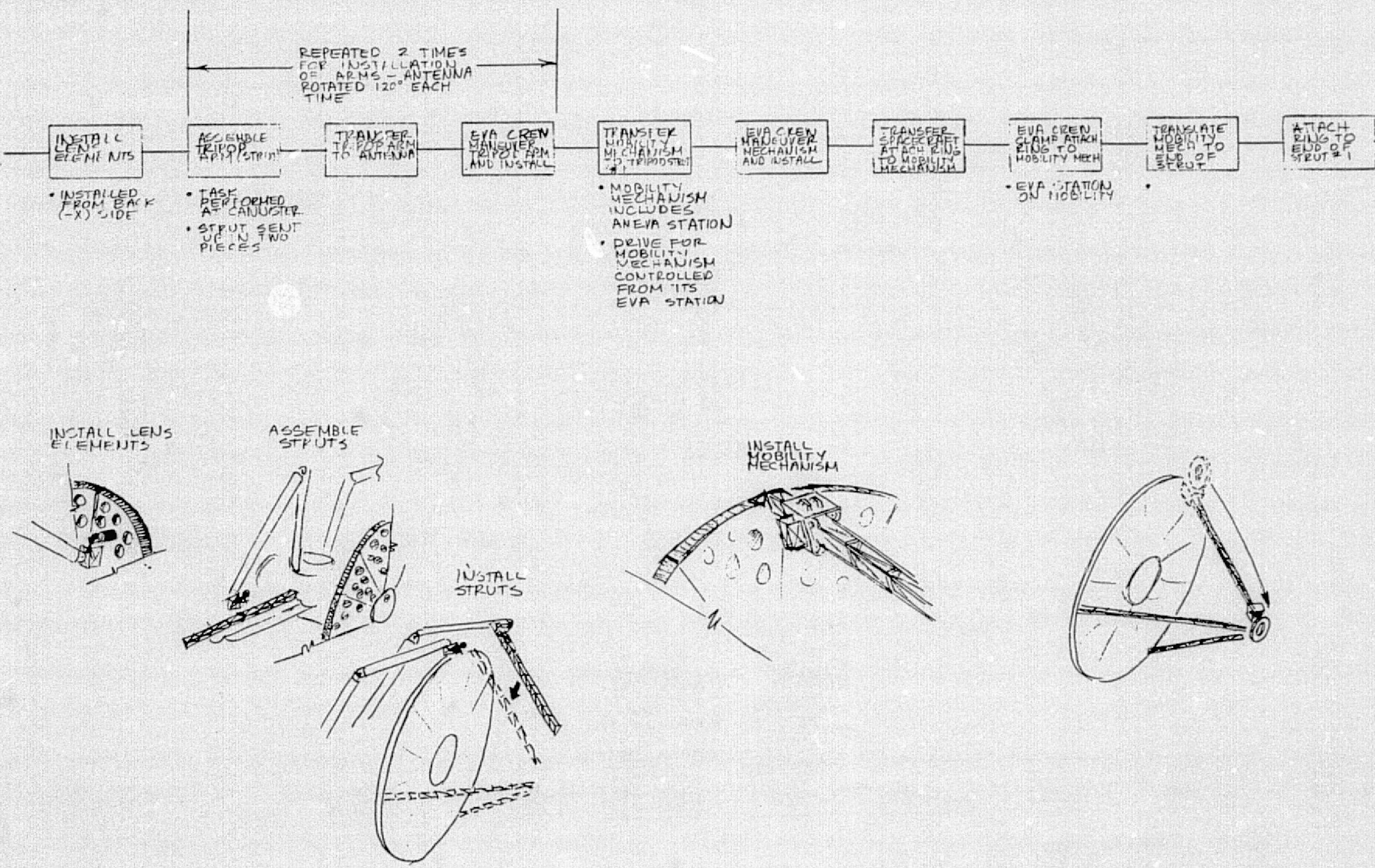


Figure 10. MBL Assembly (3 of 5)

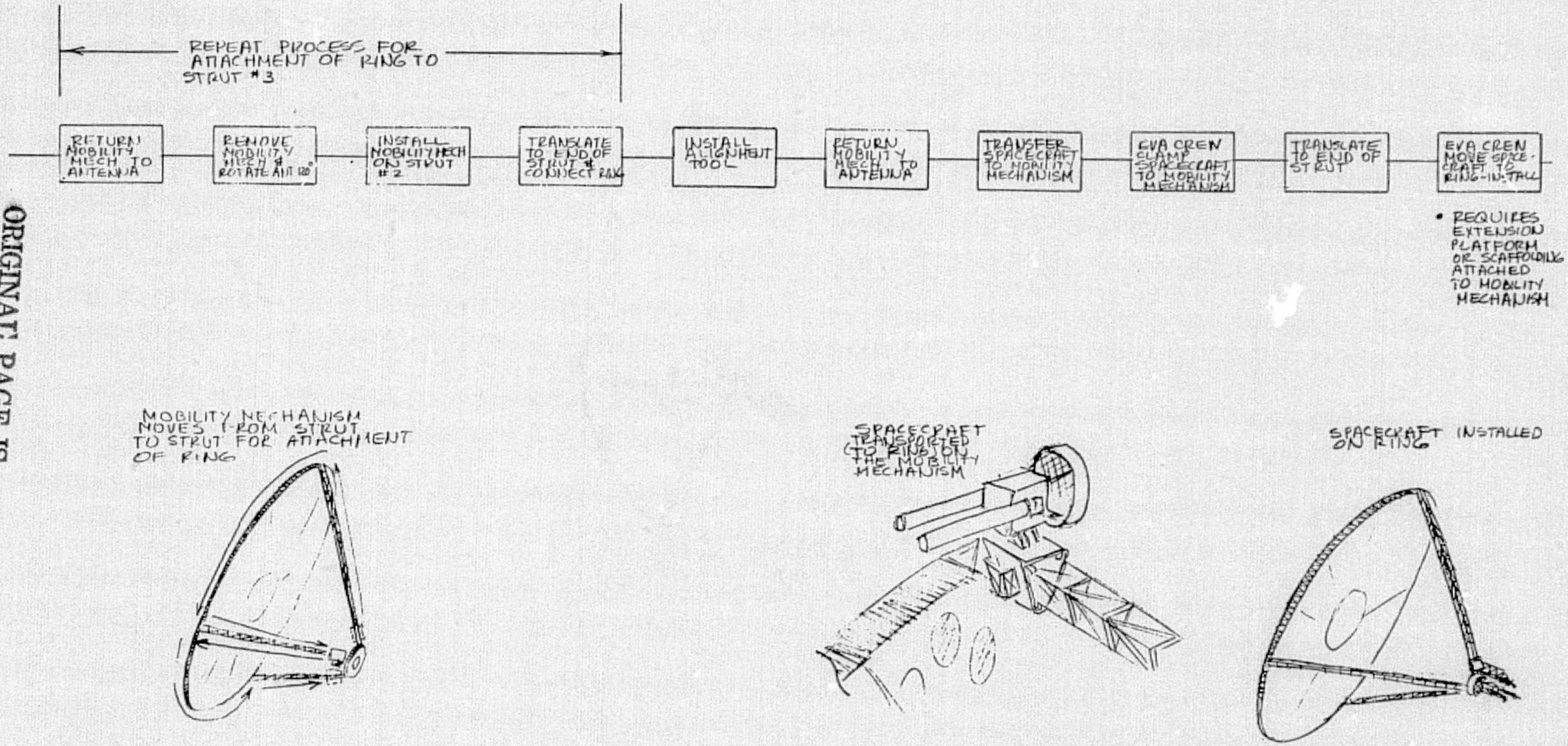


Figure 10. MBL Assembly (4 of 5)

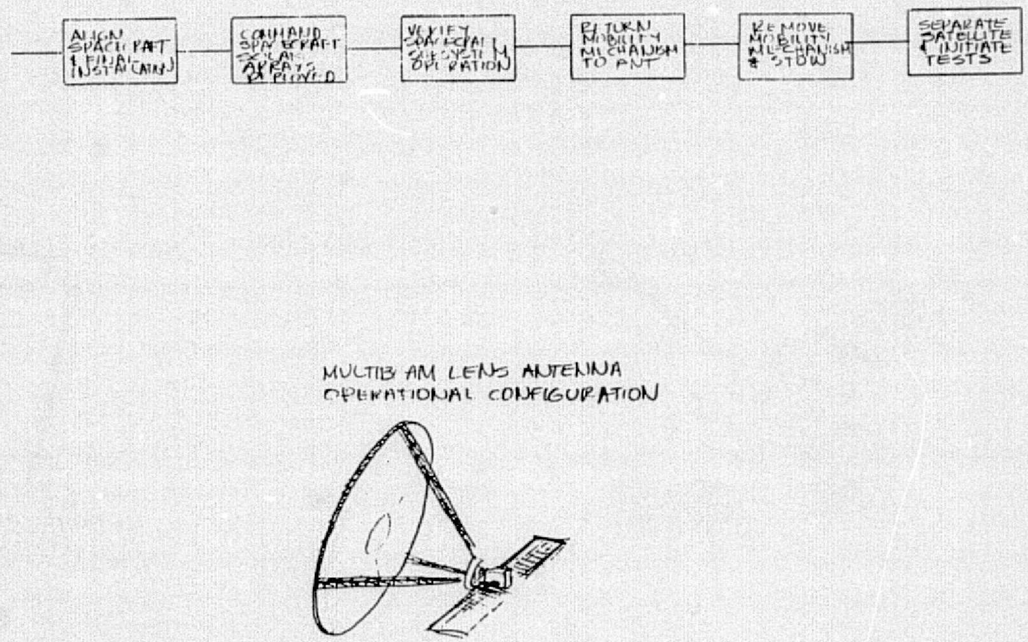


Figure 10. MBL Assembly (5 of 5)

Part 11

OTV CONCEPT DEFINITION

Section 1
INTRODUCTION AND SUMMARY

The Space Station Systems Analysis Study effort included the area of transportation, in particular the definition of an Orbit Transfer Vehicle (OTV). A number of program options were evaluated in Part 2 and transportation requirements calculated for each, both in terms of requirements to low earth orbit (LEO) and requirements to geosynchronous earth orbit (GEO). Results of these analyses indicated that very large amounts of mass must be transported from LEO to GEO; thus the need for an OTV. This in turn requires even larger quantities of mass to be transported to LEO, i.e., the necessary OTV propellant. It is therefore important that the OTV be a high-performance, lightweight, reusable system.

Early trade studies considered single stage, two-stage optimum or two-stage common OTV concepts. Clearly, a two-stage system is more efficient, requiring significantly lesser amounts of propellant, and therefore fewer supporting Shuttle flights. The optimum two-stage system is a smaller system than the two-stage common (in which the two stages are identical), but the amount of savings is not so significant as to overcome the advantage of stage commonality. Further, the common stage design has more potential payload capability. The common stage OTV concept was selected as shown on Figure 1-1.

As mentioned previously, the goal of OTV design was lightweight, or high-mass fraction (λ'). A number of groundrules were put forth towards achieving this goal. First of all, it appeared that space-basing would be highly desirable, i.e., boost the OTV to LEO and conduct all subsequent operations from there. In this manner, the empty OTV would be carried up in the Shuttle, thus avoiding high loads from the tanks full of propellants. Design loads would also be minimized during powered flight by keeping the accelerations down to about one-tenth g. A high-expansion-ratio, extendible-nozzle engine would be used, incorporating a zero-NPSH feature. Thus, tank pressures would

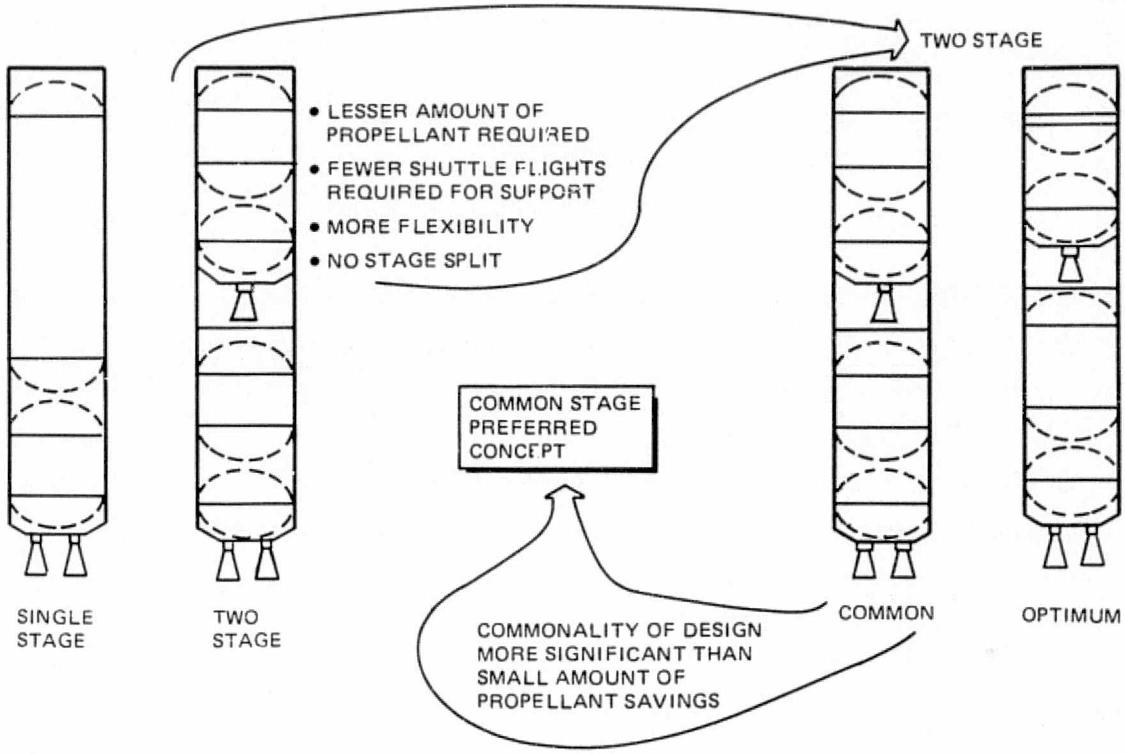


Figure 1-1. OTV Concept Selection

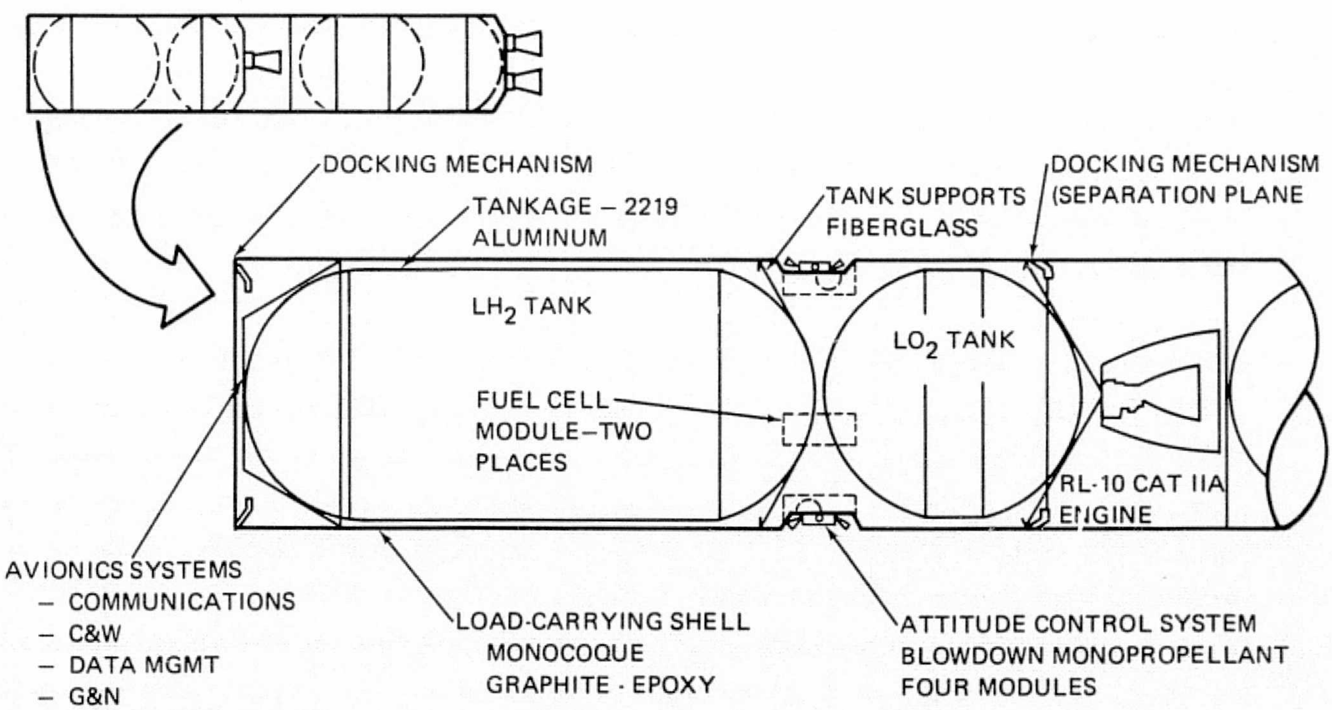


Figure 1-2. Baseline OTV Design

be at a minimum, saving considerable weight. Finally, extensive use of composite structure was outlined to minimize non-tank structure weight. The MDAC cryogenic Tug study results were used as a base for the efforts described in the following sections.

The OTV two-stage (common) concept selected is shown on Figure 1-2 and a summary of the major stage characteristics and capabilities is shown in Table 1-1.

Table 1-1 (Sheet 1 of 2)

OTV CHARACTERISTICS

Physical	OTV-1	OTV-2
Length - m (ft)	17.069 (56)	17.069 (56)
Diameter - m (ft)		
Shell	4.42 (14.50)	4.42 (14.50)
Tank	4.10 (13.45)	4.10 (13.45)
Mass - kg (lb)		
Dry	4260 (9392)	3737 (8239)
Burnout	5041 (11,113)	4462 (14,639)
Ignition	63,424 (139,824)	62,845 (138,548)
Propellant - kg (lb)		
Loaded	58,550 (120,079)	58,550 (120,079)
Usable	57,206 (126,116)	57,206 (126,116)
Mass Fraction (λ')	.9205	.9290

Performance - LEO to GEO, kg (lb)

Mission -	Delivery	49,858 (109,917)
	Round Trip	13,300 (29,321)
	Retrieval	17,535 (38,658)
	Expendable	64,000 (141,094)

Subsystems

- Propulsion - Category IIA RL-10 engines (one on OTV-2, two on OTV-1)
  - I_{sp} = 459 sec at 6:1 MR (mission effective = 455.6 sec)
  - Zero NPSH, tank head idle mode
  - Extendible nozzle, $\epsilon = 66.2/262$
  - Blowdown monopropellant ACS

Table 1-1 (Sheet 2 of 2)

OTV CHARACTERISTICS

•	Structures	- Graphite-epoxy monocoque load-carrying shell - 2219 aluminum tankage - Fiberglass tank supports - Square frame, four latch, extendable docking mechanisms
•	Avionics	- Shuttle-derived fuel cells (two), replaceable modules - Upper stage LADAR for automatic docking (uncooperative target) - S-band communications Orbiter compatible; NASA standard computer - Forward skirt mounting on aluminum isogrid structure

Section 2
OTV SIZING SUMMARY

2.1 SIZING FOR PROGRAM REQUIREMENTS

The OTV was sized in response to the LEO-to-GEO transport requirements. These consisted of objective elements, Space Station modules, crew modules, and logistics as needed for each program option.

The numerical distribution of delivery and round-trip payloads for a typical option (LG1) is shown in Figure 2-1. As seen, most of the payloads are under 20,000 kg for the delivery mission and 7,000 kg or under for the round-trip mission. These data suggest that the OTV design capability should be 20,000 kg for delivery and 7,000 kg for round-trip. These requirements were tabulated for each GEO program option. Parametric OTV capabilities were then compared to the mission requirements to determine the sizes needed. Delivery and round-trip payload capabilities are overlaid on the mission requirements for Option LG1 in Figure 2-1. Performance capabilities include single- and two-stage OTV's with the latter considered in both optimum and common stage configurations. The optimum consists of sizing the two stages for maximum performance, which is a propellant loading split between Stages 1 and 2 of about 2/1 for delivery missions and 55/45 for round-trip missions. For the common stage design, both stages are the same size. All the stages are reused in the primary mission mode; however, the capabilities for delivery in an expendable mode were also calculated to extend the capability for outsized payloads. The tic marks on each performance line indicate the transition points from integral stages to separate LO_2/LH_2 tank designs. The center ordinate of the chart is the total OTV propellant loading common to both the delivery and round-trip performance lines.

The bulk of the delivery missions (15 of 17) require less than 20,000 kg capability. This could be accomplished by both single- and two-stage OTV's, the single stage requiring 65,000 kg of propellant, and the two-stage requiring

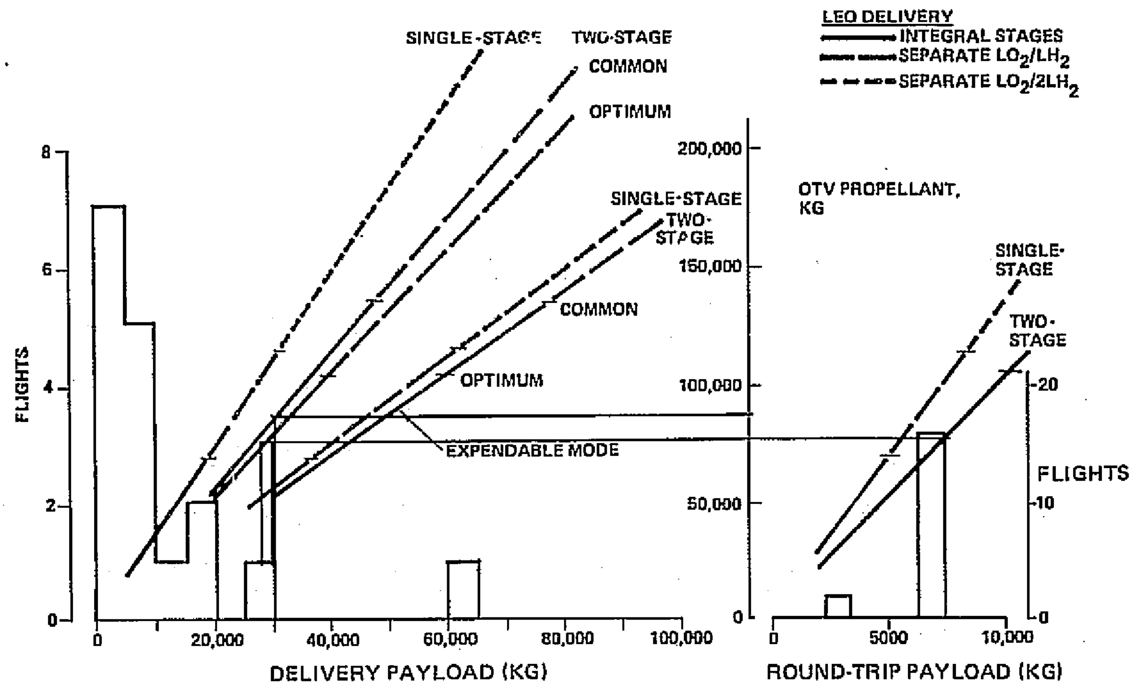


Figure 2-1. OTV Requirements/Capabilities (LG1)

about 50,000 kg. When the round-trip requirements (7,000 kg) are considered, a propellant loading of 100,000 and 80,000 kg would be required for the single- and two-stage OTV's respectively. Note that the single-stage version would have to be launched in two pieces (LH_2 tank and LO_2 tank/engine) and assembled in orbit. Also note that the 80,000-kg, two-stage OTV could accommodate the 28,000-kg delivery mission. Clearly, the 64,000 kg payload would size an OTV beyond that which could be used efficiently for 34 of the 35 LG1 flights. This mission would be accomplished by special means, probably multiple OTV elements used in an expendable mode. The propellant savings and flexibility of the two-stage OTV over that of the single stage resulted in the two-stage selection for Option LG1. The reduced OTV propellant alone would result in a $320 million saving due to decreased Shuttle flights (17 x $18.9 million). The common stage design was chosen over the optimum concept for commonality reasons, the performance difference being small; thus, an 80,000-kg propellant, common two-stage OTV (two 40,000 kg stages) was selected for LG1.

This analysis and selection process for sizing an OTV was done for all four program options; the individual results are shown in Figures 2-2, 2-3 and 2-4, for Options LG2, G and G'. The types selected, sizes, and major influences for each option are shown in Table 2-1.

The OTV size selected for LG2 was 55,000 kg of propellant per stage. The basic requirement of 53,000 kg to meet the 10,000 kg round-trip requirement was raised to 55,000 kg to accommodate the delivery of the 64,000 kg cross-phased array. The OTV would be expended for this mission.

Option G analysis resulted in 53,000 kg propellant per stage OTV to meet the 10,000-kg round-trip requirement. For Option G', a 55,000-kg OTV stage was selected. With this size, a two-stage OTV would be used to satisfy the round-trip mission requirement of 11,000-kg and one of the two common stages would be used for the 15,000-kg delivery mission.

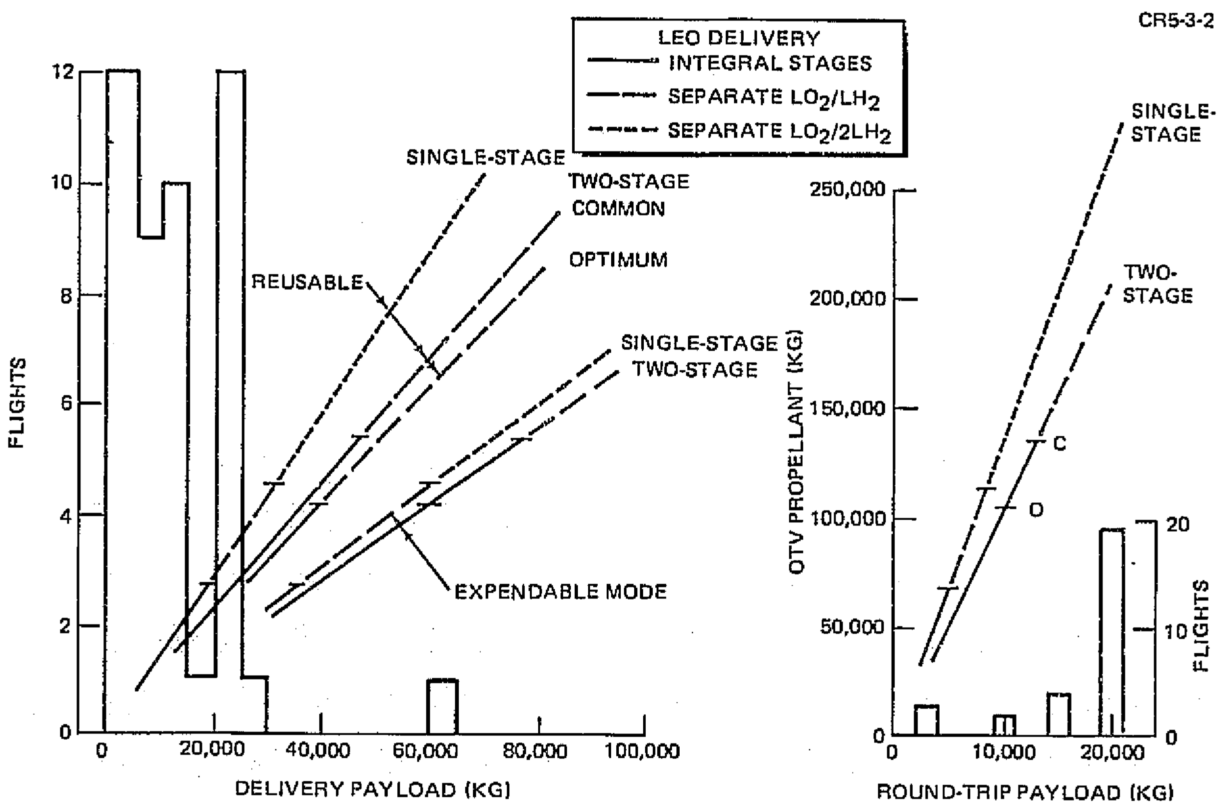


Figure 2-2. OTV Requirements/Capabilities (LG^2)

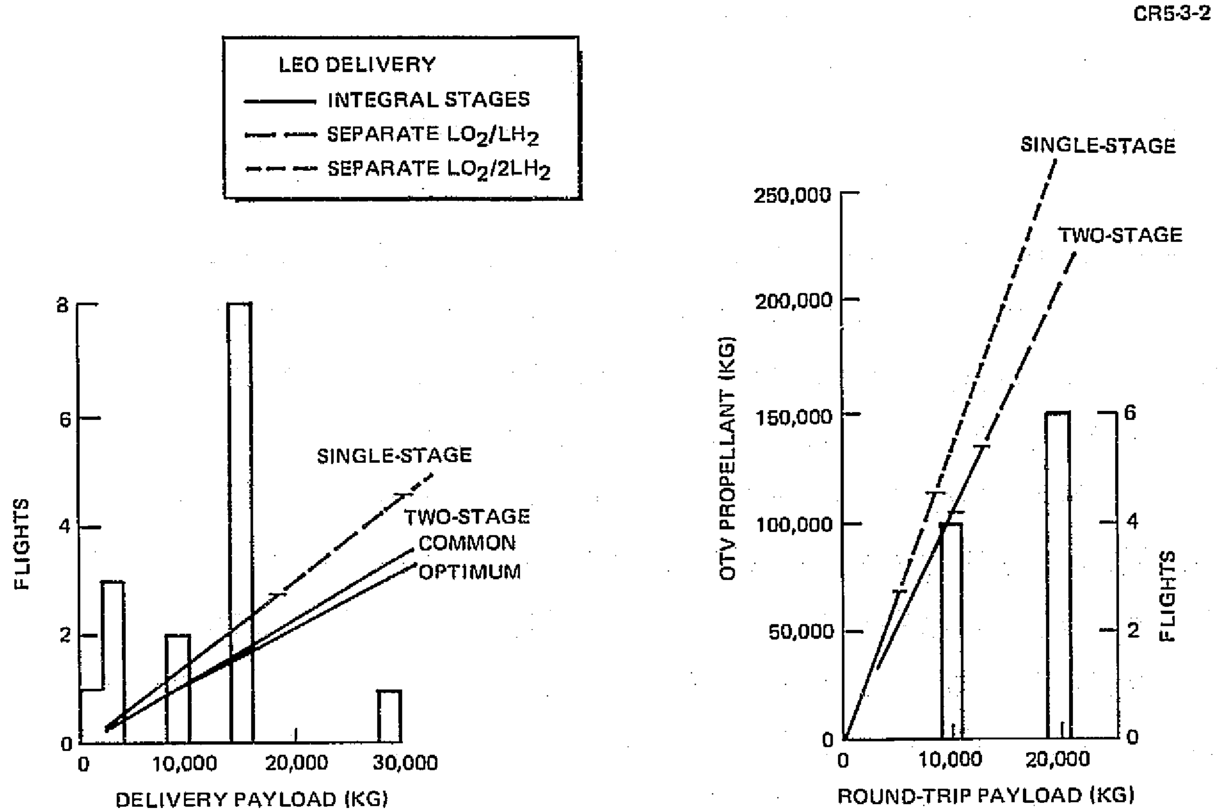


Figure 2-3. OTV Requirements/Capabilities (G)

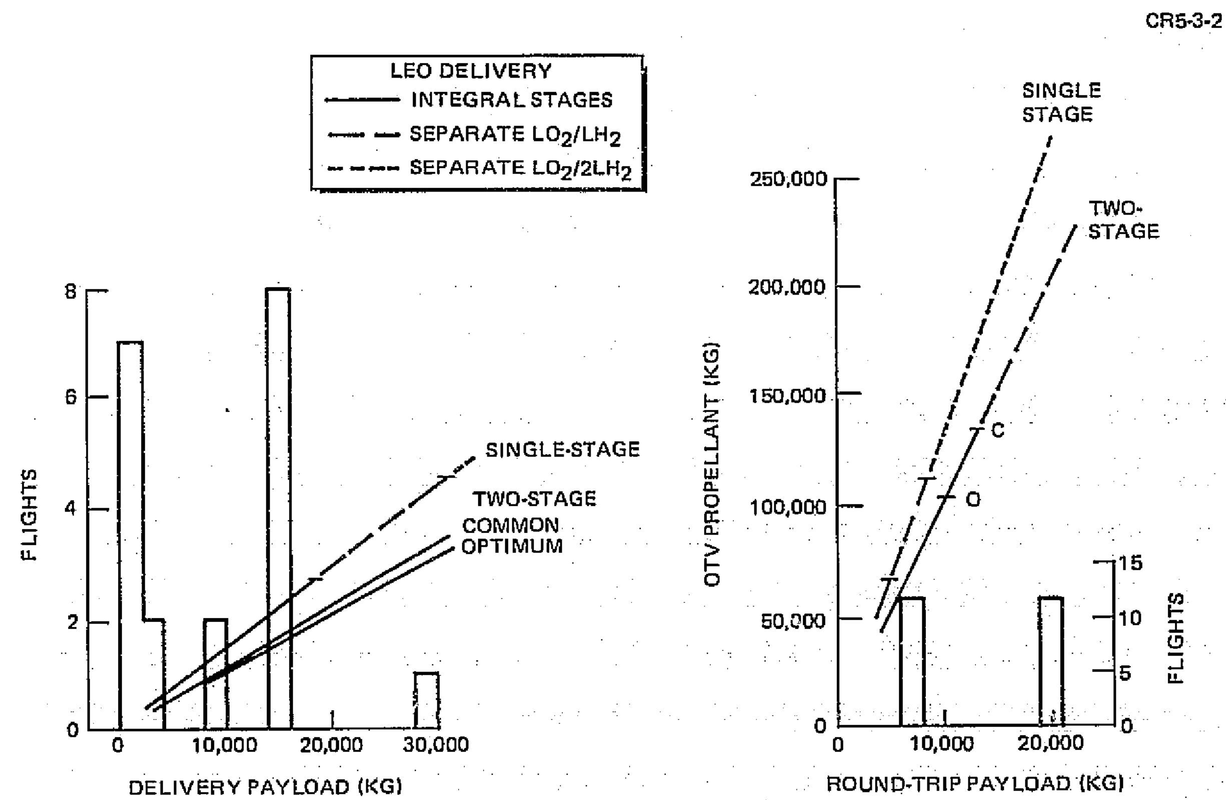


Figure 2-4. OTV Requirements/Capabilities (G')

Table 2-1

INITIAL OTV SELECTIONS

Option	Type	Propellant/ Stage (kg)	Payload (kg) Delivered	Round-Trip	Expendable	Major Influence
LG1	2-C	40,000	28,000	7,500	46,000	Delivery payload
LG2	2-C	55,000	39,000	11,000	64,000	Expendable payload
G	2-C	53,000	37,000	10,000	60,000	Round-trip payload
G	2-C	55,000	39,000 15,000	11,000	60,000	Round-trip payload and delivery (1 stage)

The two-stage common design OTV was selected for all four options based on the reduced logistics costs for propellant delivery and the commonality of design. The logistics cost savings of the two-stage OTV over the single stage OTV, due to reduced Shuttle flights at \$19.1 million, are shown in Figure 2-5. These cost savings, as a function of program, are: LG1-\$340 million; LG2-\$1.6 billion; G-\$560 million; and G'-\$880 million.

The OTV concept selected for development in the study was a two-stage common space-based reusable OTV with each stage sized to the maximum that could be launched on a single Shuttle flight.

2.2 ENGINE SELECTION

Based on early sizing values, an investigation of axial acceleration values and structural loading was undertaken to establish engine quantities desired. The OTV values used were as follows:

Dry Weight kg (lb)	7,120	(15,700)
Propellant Weight kg (lb)	59,870	(132,000)
	66,990	(147,700)
Payload Weight		
Delivery kg (lb)	39,370	(86,800)
Round Trip kg (lb)	10,160	(22,400)

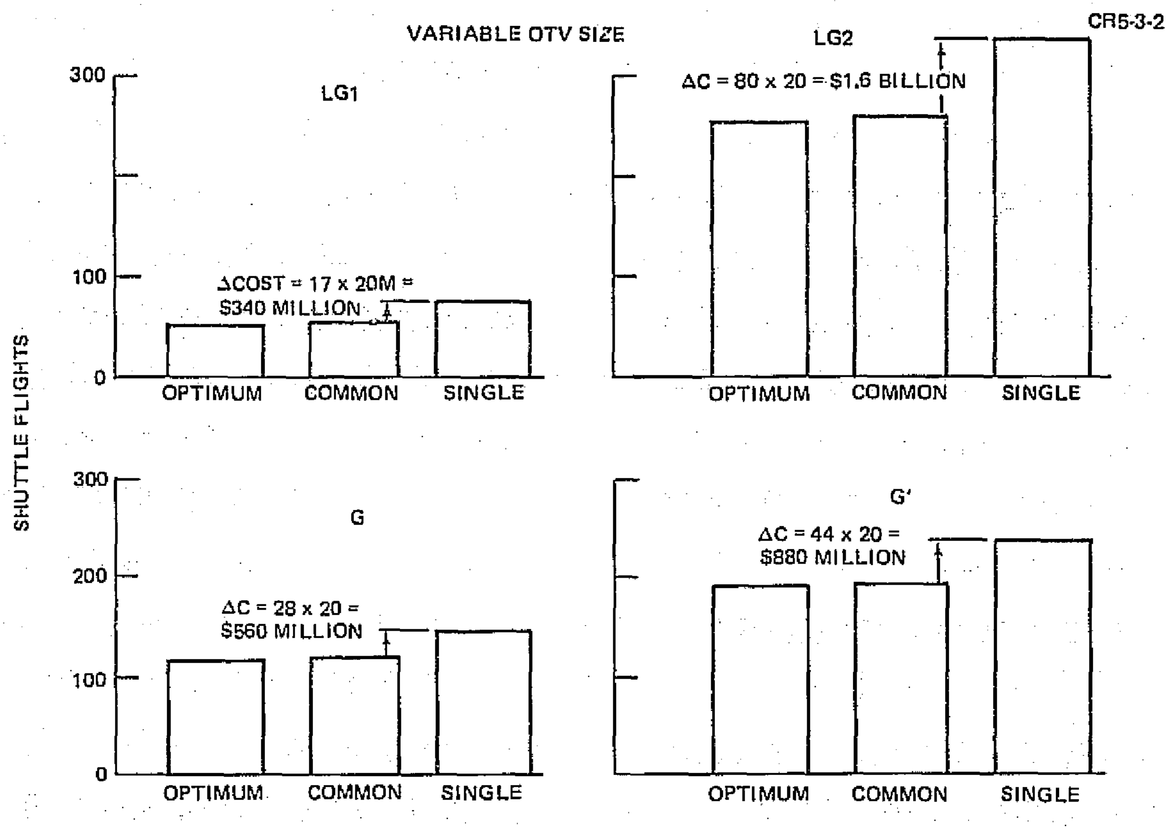


Figure 2-5. Shuttle OTV Propellant Flights Required

Using these data, tables of vehicle weight distribution were prepared for each mission phase, i.e., vehicle ignition, OTV-1 burnout, OTV-2 ignition, OTV-2 first burnout, etc. Considering first the delivery mission, the following values were determined for the OTV burns.

	OTV-1		OTV-2	
	kg	(lb)	kg	(lb)
First burn – Ignition	173,360	(382,200)	106,370	(234,500)
– Burnout	116,890	(257,700)	58,560	(129,100)
Second burn – Ignition	10,520	(23,200)	19,190	(42,300)
– Burnout	7,120	(15,700)	7,120	(15,700)

In order to hold the vehicle acceleration level to about 0.10g with payload aboard, the thrust level would have to be around 114.6 kN (25,770 lb), or the equivalent of about two RL-10-derivative engines. Assuming the OTV-1 had two of these engines, acceleration levels would be 0.079g at liftoff and 0.116g at first burnout. Values for second burn, i.e., the return trip, no payload, were 1.29g at ignition and 1.91 g at burnout. Using these values and the aforementioned weight distribution, structural loads were determined for all elements of the OTV-1 both at first burn ignition and burnout. Then, in order to determine if the second burn conditions were more severe, these loads were compared against the weight distribution to determine allowable acceleration levels. For OTV-1 second burn, the allowable g's, for the most critical structural components, were 1.35 at ignition and 2.05 at burnout, greater than the 1.29 and 1.91 previously noted. Thus the selection of two engines for OTV-1 seemed appropriate.

The same process was repeated for OTV-2. In this case, a single RL-10 engine was designated, resulting in the following g levels:

First Burn (with payload)

Ignition	0.064
Burnout	0.116

Second Burn (no payload)

Ignition	0.335
Burnout	0.955

As before, the second burn condition was found to be less critical than the first burn; thus the one-engine selection seemed satisfactory.

The entire process just described was repeated for the round-trip mission. In this case, however, the loads were also compared to those determined for the delivery mission, and they were found to be less critical. Accelerations were somewhat higher, though. For OTV-1 burn, with a payload aboard, g levels ranged from 0.094 to 0.153, and with no payload, from 1.179 to 1.915. For the second stage OTV, which always has payload aboard, accelerations ranged from 0.088 to 0.394. Acceleration histories for all cases are illustrated on Figure 2-6. The various data generated in the course of this analysis were also used to determine the loads presented in Section 4.1.1.

2.3 OTV FINAL SIZING

Final OTV size was based largely on the available volume of the Shuttle cargo bay, i.e., it was decided to make the OTV as large as possible since preliminary investigations indicated that such a sizing would be compatible with program requirements. This final OTV configuration is shown on Figure 2-7.

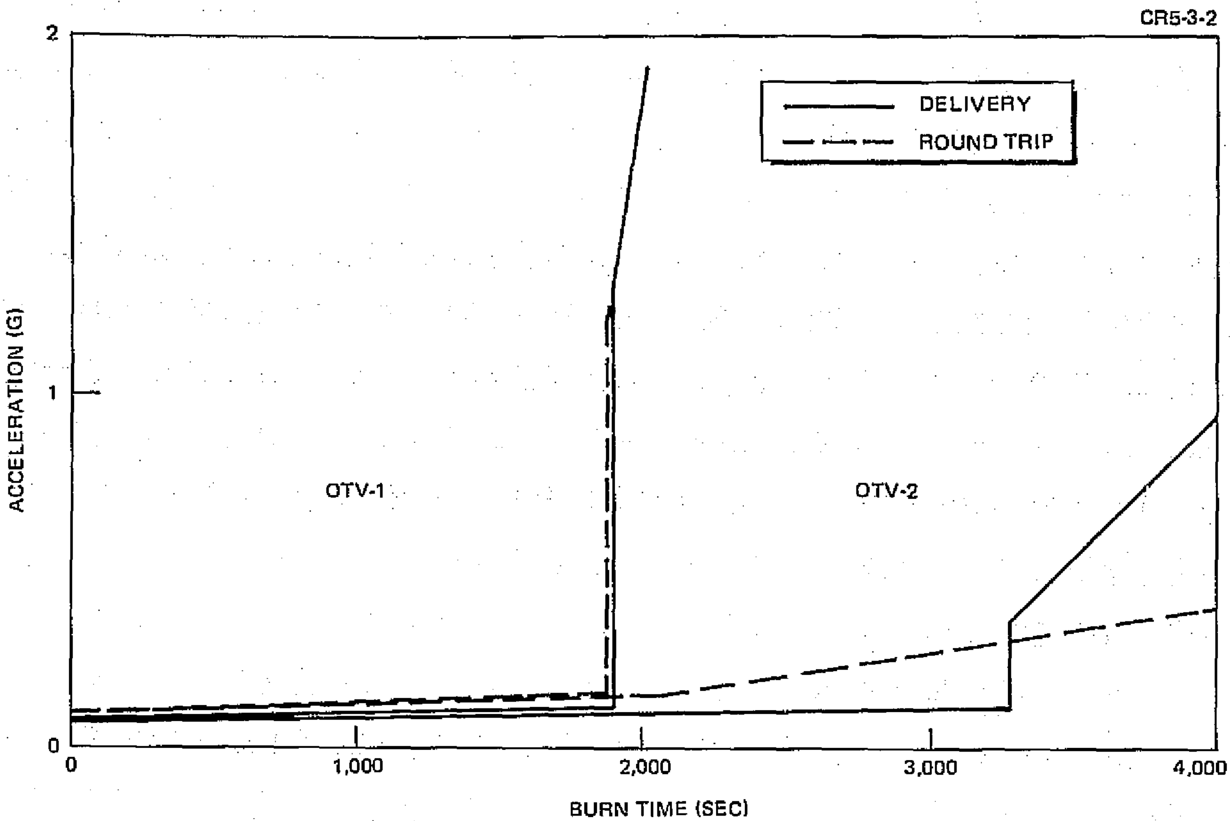
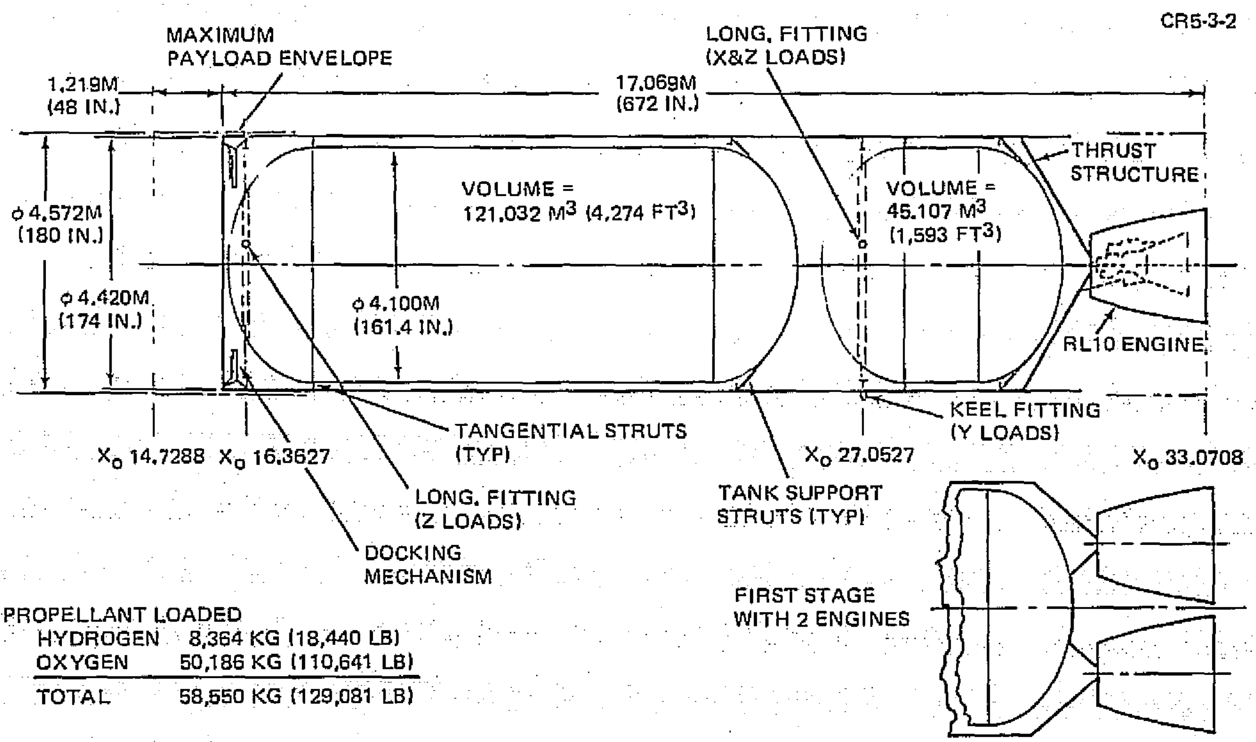


Figure 2-6. OTV Acceleration Histories



PROPELLANT LOADED

HYDROGEN	8,364 KG (18,440 LB)
OXYGEN	50,186 KG (110,641 LB)
TOTAL	58,550 KG (129,081 LB)

Figure 2-7. Orbit Transfer Vehicle

Space construction base guidelines were followed in the sizing exercise. The maximum external dimensions of the OTV were taken as 4.42m (14.5 ft) in diameter and 17.07m (56 ft) in length. This latter dimension provides room for planned EVA. The engines were assumed to be stored in a retracted position to save length. Also, a 25-cm (10 in) gap was left between tanks.

Maximum tank wall diameter was based on a combination of factors — accommodation of the multilayer insulation (MLI) and allowance of space for a hydrogen feed line to pass around the oxygen tank. Anticipated thickness of the MLI was about 6 cm (2.36 in), and for the vacuum-jacketed feed line about 10 cm (3.93 in). Hence, a diameter of 4.1m (13.45 ft) was selected, leaving a space of 16 cm (6.3 in) between the tanks and the outer shell. Although the space requirements around the hydrogen tank were not as severe, that tank was configured at the same diameter in order to have common dome and cylinder tooling.

The resulting configuration, as shown in Figure 2-7, has a hydrogen tank of 121.032 m^3 (4,274 ft^3) and an oxygen tank of 45.107 m^3 (1,593 ft^3). Resulting capacities, allowing 5% for ullage volume, are 8,364 kg (18,440 lb) of hydrogen and 50,186 kg (110,641 lb) of oxygen. Thus the final propellant load is 58,550 kg (129,081 lb), which is quite compatible with the desired propellant capacity determined from a review of program requirements.

Section 3
OTV PERFORMANCE

3.1 OTV MISSION PROFILE

The basic mission profile for the OTV is shown on Figure 3-1. The reusable OTV will be space-based in LEO, and will be used either to deliver payloads to GEO or to carry payloads on a round trip from LEO to GEO. Propellants will be delivered to the OTV via a Shuttle tanker; the OTV will be carried to LEO empty.

The configuration as pictured is a two-stage, common design, i.e., both stages are the same size, each containing 57,206 kg (126,118 lb) of liquid oxygen/liquid hydrogen usable propellants. The engines are Category IIA RL-10 derivatives, with two on the first stage and one on the second stage. Stage diameter is 4.4m, and overall length (without payload) is 34m.

The first-stage OTV provides the initial boost to the second-stage OTV and payload for the orbit transfer. After shutdown and separation, it then coasts back to LEO, circularizes, and awaits return of the second stage. Meanwhile, the second stage completes the transfer, and circularizes at GEO. After mission objectives are met, the second-stage OTV deorbits and transfers back to LEO, where it circularizes and rendezvous with the first stage.

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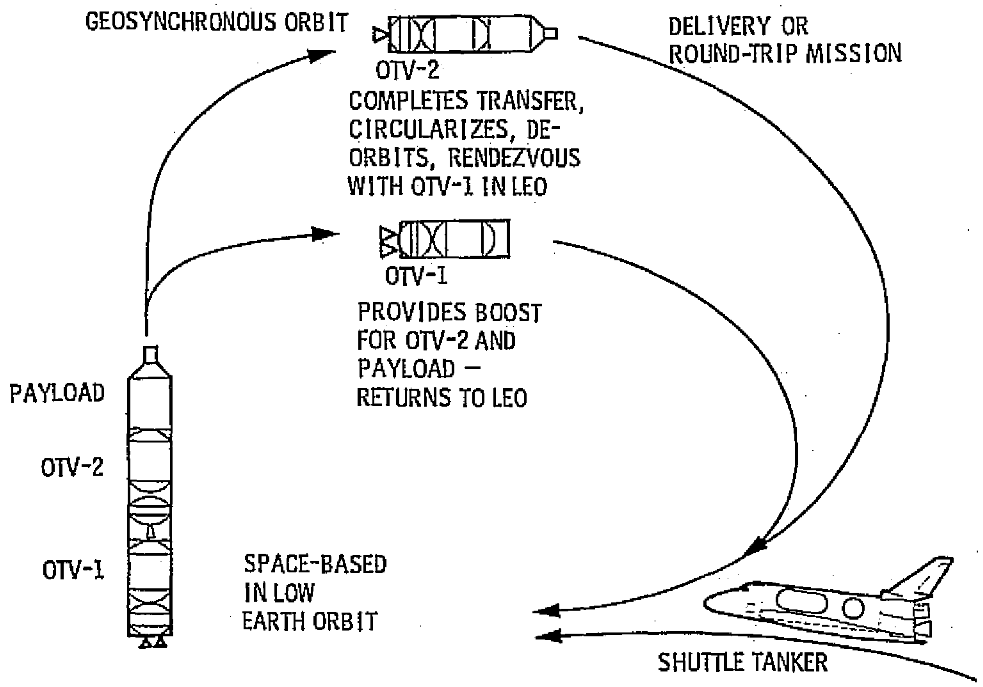

Figure 3-1. OTV Mission Profile

3.2 BASELINE PERFORMANCE

Basic payload capability for the two-stage (common) OTV as defined in the paragraph on final sizing is as follows:

Delivery (to GEO)	49,858 kg (109, 916 lb)
Round Trip (LEO-GEO)	13,300 kg (29,320 lb)
Retrieval (from GEO)	17,535 kg (38,658 lb)
Expendable (to GEO)	64,000 kg (141,097 lb)

The performance is based on each stage expending 57,769 kg (127,360 lb) of LO_2/LH_2 propellants at an oxidizer/fuel weight mixture ratio of 6:1. Reference mission velocity (one way) was assumed to be 4,320 mps (14,173 fps). Vacuum specific impulse delivered by the Category II RL-10A engines is 459 sec, which is reduced to an effective value of 455.6 sec considering the propellant used for tank head idle (THI) and that vented. The stage mass fractions (λ's) used for these performance calculations were 0.9197 for the first stage (OTV-1) and 0.9283 for the second stage (OTV-2). These λ's were calculated as

$$\frac{\text{Expended Propellant}}{\text{Expended propellant} + \text{Burnout Weight}}$$

and are based on weights found in Section 5 of this report (Appendix).

3.3 PAYLOAD SENSITIVITIES

The payload sensitivity to a number of OTV parameters was investigated. These parameters included specific impulse, mass fraction, mission velocity and propellant weight.

3.3.1 Specific Impulse Effects

The effects of changes in vaccum specific impulse of either or both OTV stages are shown in Figure 3-2 for the delivery mission. Specific impulse was varied from 440 to 480 sec, while other parameters were held constant, i.e., stage propellant weight 57, 769 kg (127, 360 lb), OTV-1 mass fraction 0.9197, OTV-2 mass fraction 0.9283, and one-way velocity 4, 320 mps (14, 173 fps). The following partials were determined:

Delivered Payload/OTV-1 Impulse	=	87 kg/sec (192 lb/sec)
Delivered Payload/OTV-2 Impulse	=	182 kg/sec (402 lb/sec)
Delivered Payload/OTV-1 and 2 Impulse	=	269 kg/sec (594 lb/sec)

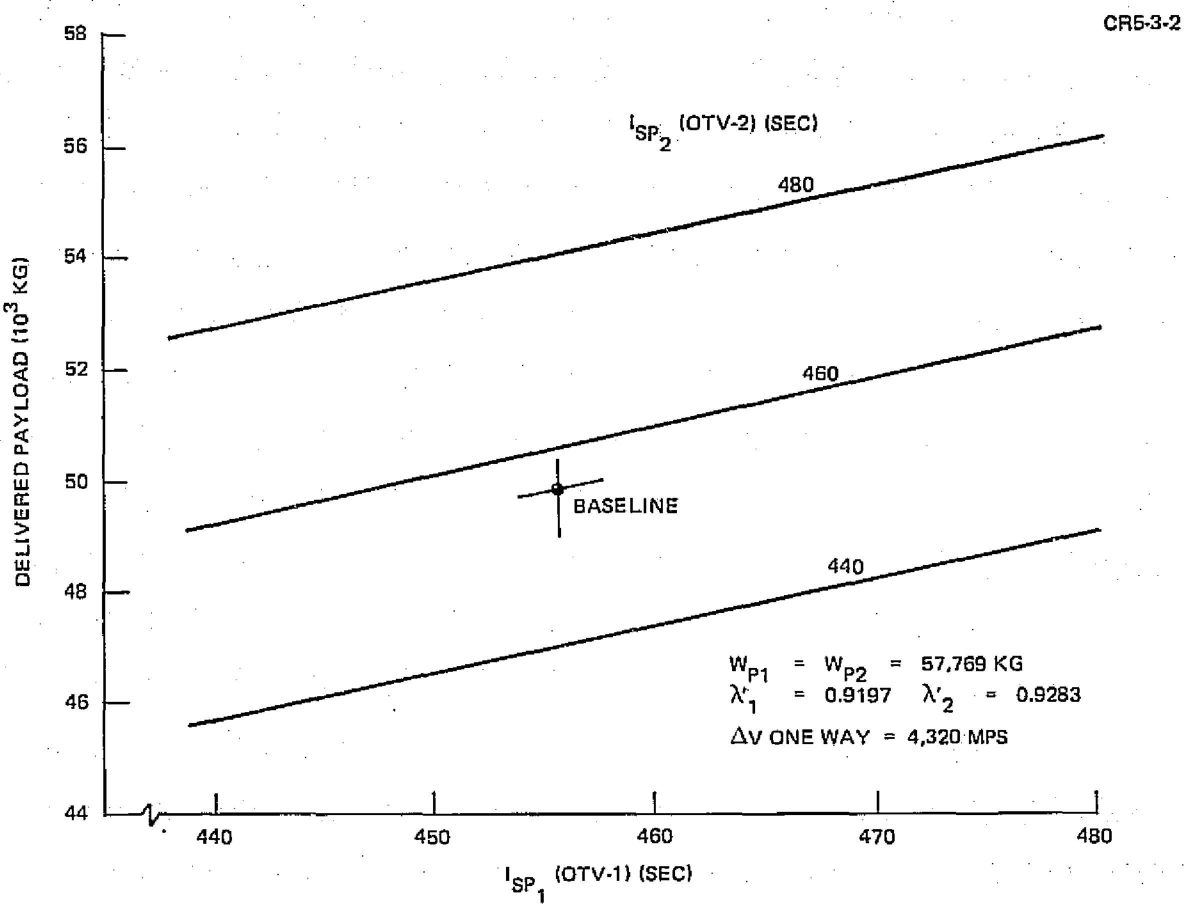

Figure 3-2. Effects of Specific Impulse

3.3.2 Mass Fraction Effects

The effects of mass fraction variations in either or both OTV stages are shown in Figure 3-3 for the delivery mission. Mass fraction (λ') was varied from 0.90 to 0.94, while the following were held constant: stage propellant weight, 57,769 kg (127,360 lb); vaccum specific impulse (both), 455.6 sec; one-way velocity 4,320 mps (14,173 fps). The following partials were determined:

Delivered Payload/OTV-1 λ'	=	2,050 kg/0.01 fraction (4,520 lb/0.01 fraction)
Delivered Payload/OTV-2 λ'	=	910 kg/0.01 fraction (2,010 lb/0.01 fraction)
Delivered Payload/OTV-1 & -2 λ'	=	2,960 kg/0.01 fraction (6,530 lb/0.01 fraction.

3.3.3 Mission Velocity Effects

The effects of increasing mission velocity by 30.5 mps (100 fps) were as follows:

Delivered Payload - 842 kg (1,856 lb), or 27.62 kg/mps (18.56 lb/fps)

Round-Trip Payload - 292 kg (643 lb), or 9.6 kg/mps (6.4 lb/fps)

3.3.4 Propellant Weight Effects

The effects on delivered payload of changes in propellant weight of either or both OTV stages are shown in Figure 3-4. Propellant weight was varied from 40,000 kg (88,185 lb) to 70,000 kg (154,324 lb), while specific impulse was held at 455.6 sec, stage mass fractions were held at 0.9197 and 0.9283 for stages 1 and 2, respectively, and one way velocity was kept at 4320 mps (14,173 fps). The following partials were determined from these data:

Delivered payload/OTV-1 propellant	=	0.497 kg/kg (lb/lb)
Delivered payload/OTV-2 propellant	=	0.342 kg/kg (lb/lb)
Delivered payload/OTV-1 and OTV-2 propellant	=	0.864 kg/kg (lb/lb)

3.4 PROPELLANT OFF-LOAD EFFECTS

The effects of two different types of propellant off-load were investigated, that of a direct-percent off-load in either or both stages, and that of a change in mixture ratio in either or both stages.

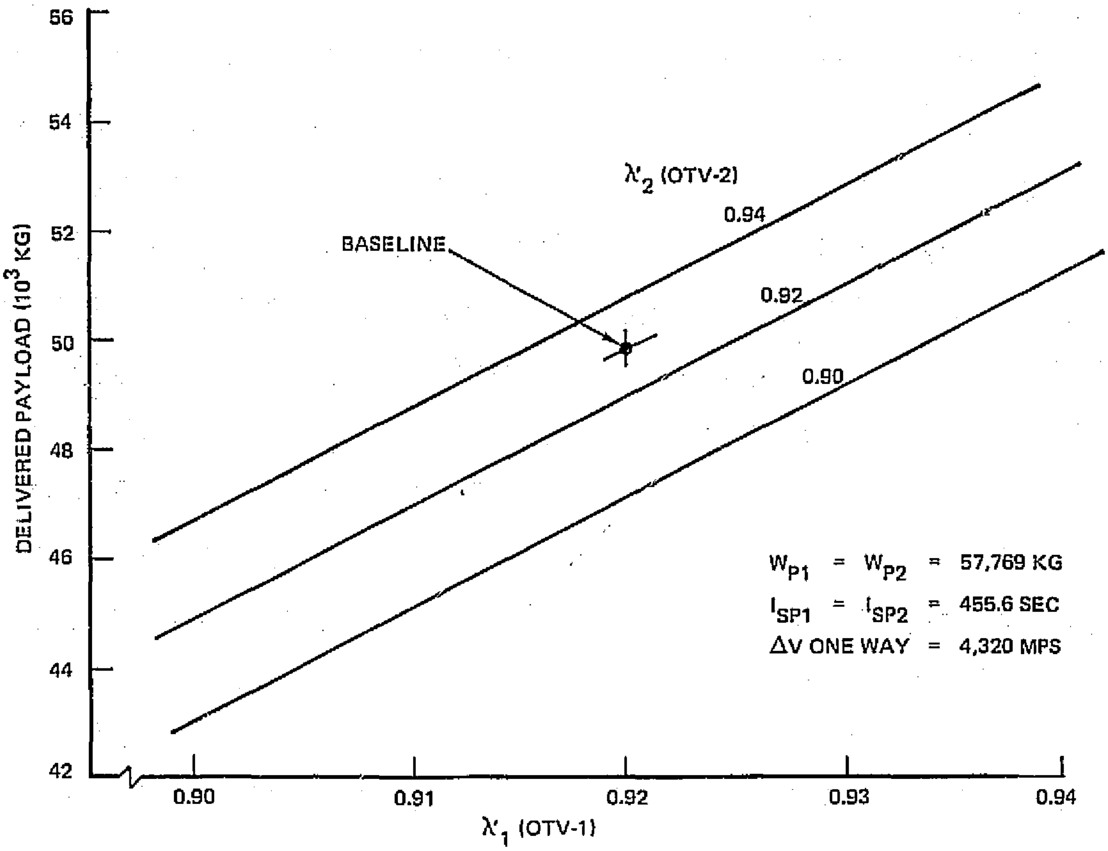

Figure 3-3. Effects of Stage λ'

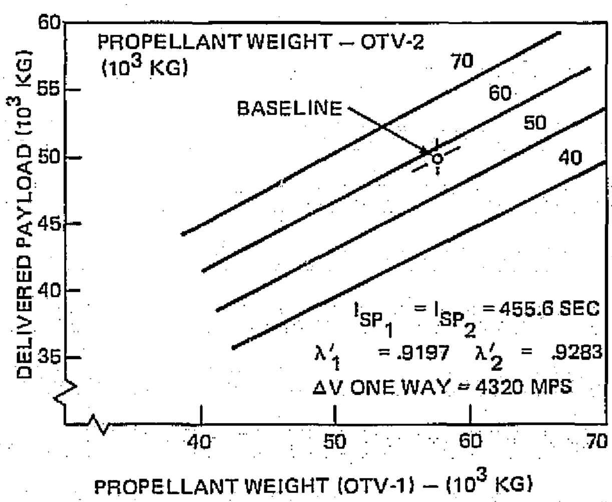

Figure 3-4. Effects of Propellant Weight

3.4.1 Direct Percent Off-Load

The payload effects due to propellant off-loading were determined for up to a 30% off-load in either or both stages. In each case, the stage weight was held constant, so a new mass fraction was calculated. In addition, the propellant vented and that used for THI were held constant; thus, the effective specific impulse was changed for each case of off-load.

The results of this investigation are shown on Figure 3-5 for both the delivery mission and the round-trip mission. Shown are effects of off-load in either stage while the other is at 100% capacity (57, 769 kg), and effects of an equal percent of off-load in both stages.

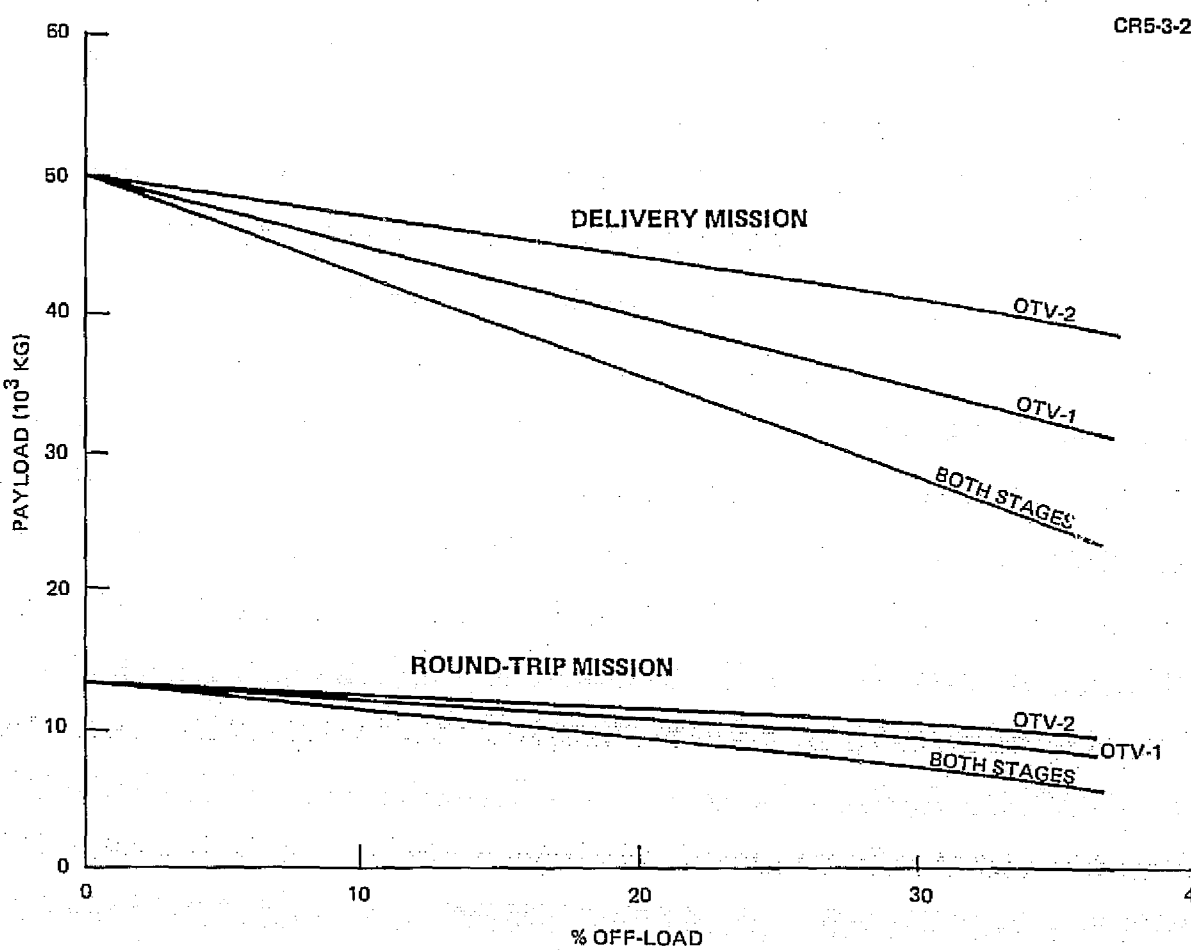

Figure 3-5. Effects of Stage Off-Loading

3.4.2 Mixture Ratio Effects

Payload effects due to changing the propellant mixture ratio in either or both stages were determined. Changing mixture ratio results in one of the propellants being off-loaded. As the mixture ratio goes down from 6:1, the LH_2 is held constant and the LO_2 is off-loaded an appropriate amount. If the mixture ratio goes up , (greater than 6:1), the LO_2 is held constant and LH_2 is off-loaded. Stage weights were held constant; hence, revised mass fractions were calculated for each case. Also, effective specific impulse had to be recalculated for each case. Nominal impulse was taken from basic engine data, shown on Figure 3-6. Using a constant propellant for THI and venting, appropriate effective impulses were then determined.

The results of these calculations are shown on Figure 3-7 for one delivery mission and the round-trip mission. Shown are effects of mixture ratio shift in either stage while the other is at 6:1, and effects of changing both stages the same amount.

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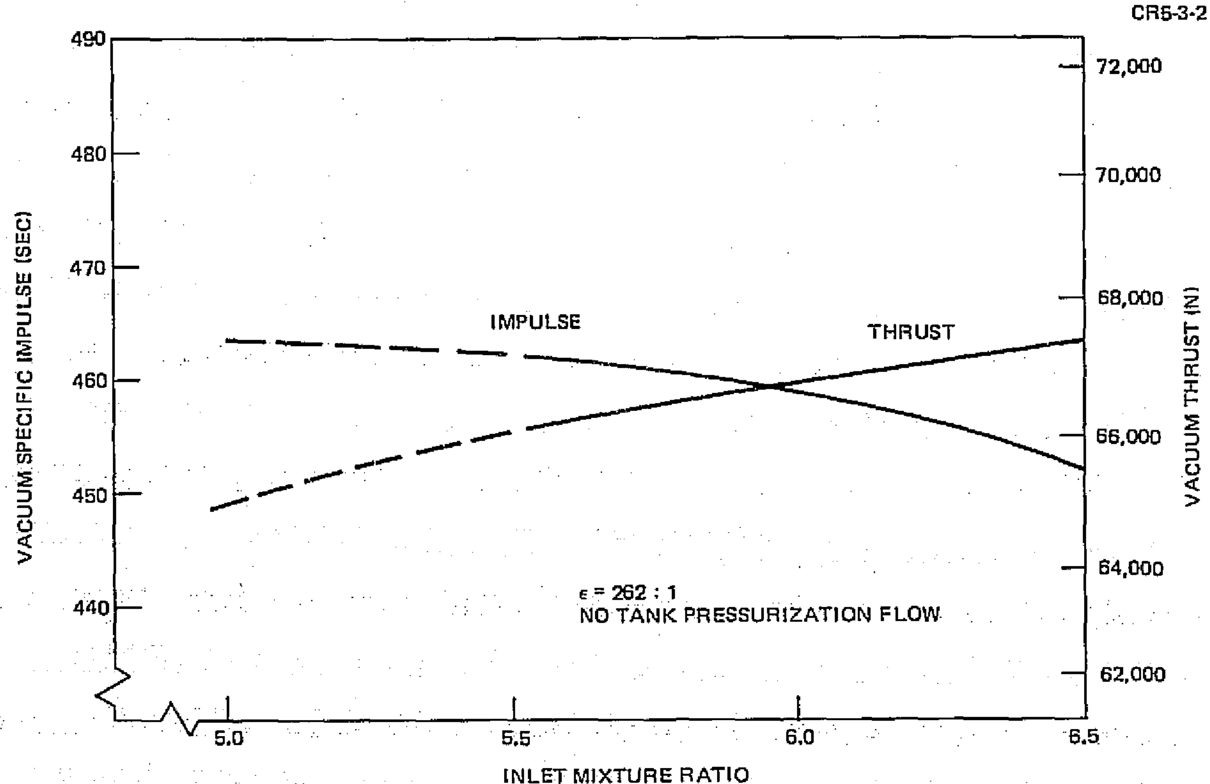

Figure 3-6 Estimated Effects of Inlet Mixture Ratio on Vacuum Specific Impulse and Thrust Derivative IIA and IIB Engines, Full Thrust

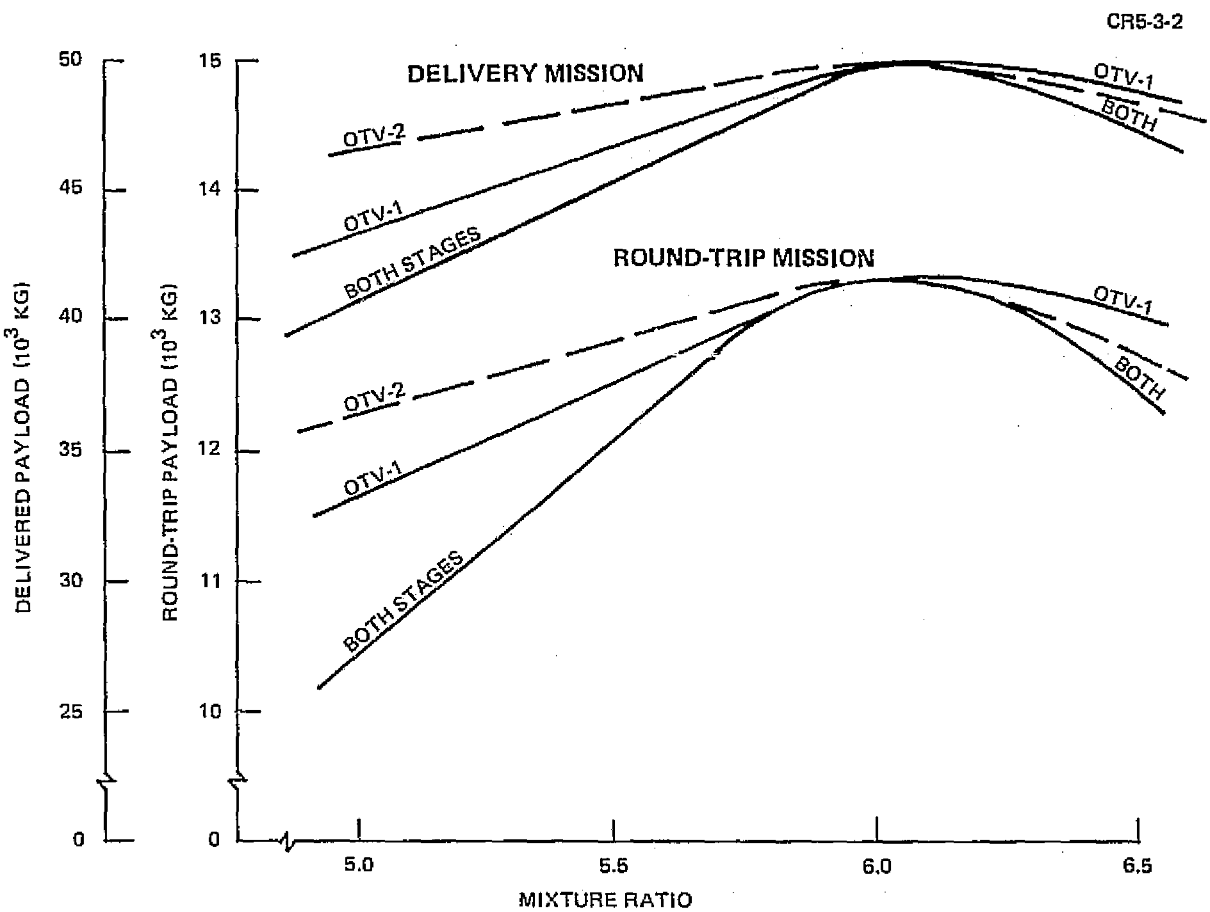

Figure 3-7. Effects of Mixture Ratio

Section 4
SUBSYSTEM DESCRIPTIONS

4.1 STRUCTURES

The general approach to structural design is based on previous in-house studies. The diameter of the stage (4.420m) was assumed to be the same as the maximum module diameter allowed by the JSC Space Construction Base Guidelines and Criteria, dated January 1977. The total propellant was taken from the results of initial performance studies similar to those described in Section 3. However, the stage proved to be too long. The stage was resized to the maximum length with the same mixture ratio and a lower propellant mass. The current length provides room for planned EVA. The structural arrangement is shown in Figures 4-1 and 4-2 (OTV 770216). An ullage volume of 5% is provided in each main propellant tank. The outer shell is the main body load-carrying member and the tanks are suspended within the shell.

4.1.1 Structural Loads

Three types of loads were considered: ground handling, Orbiter payload bay flight loads, and spaceflight loads. The first two were based on accelerations found in JSC 07700 and a no-propellant condition. Five Orbiter flight conditions were evaluated. Loading due to axial load plus bending moment was determined.

Spaceflight loads were based on maximum one-way payload delivery and maximum round-trip payload carry. Ground handling accelerations were determined to be smaller than Orbiter flight accelerations and were thus not critical. A summary of the critical loads for each body section is shown in Table 4-1.

4.1.2 Shell Structure

Two options were considered: a load-carrying shell with suspended tanks, and a load-carrying tank with an attached shell. In considering thermal

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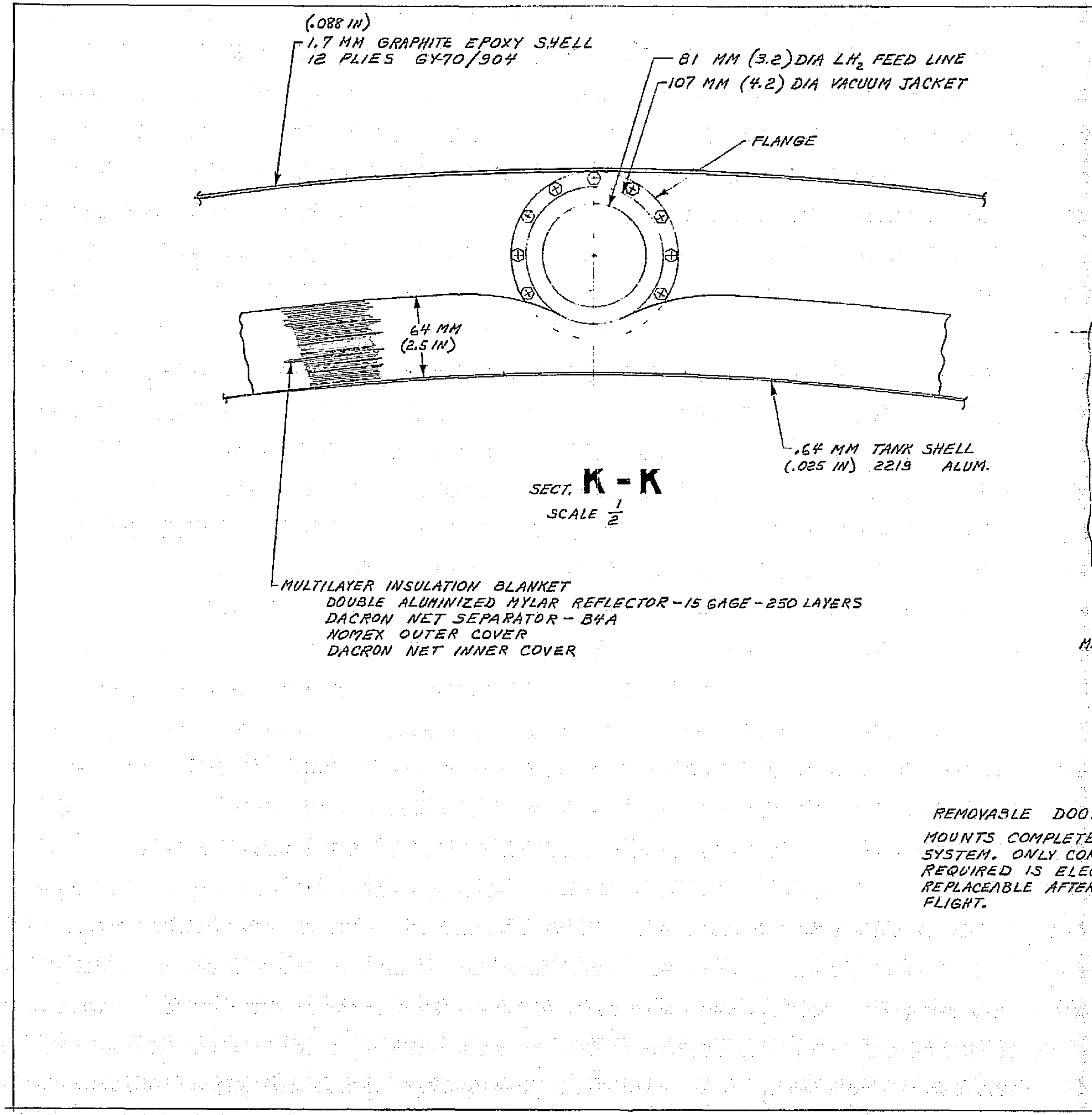

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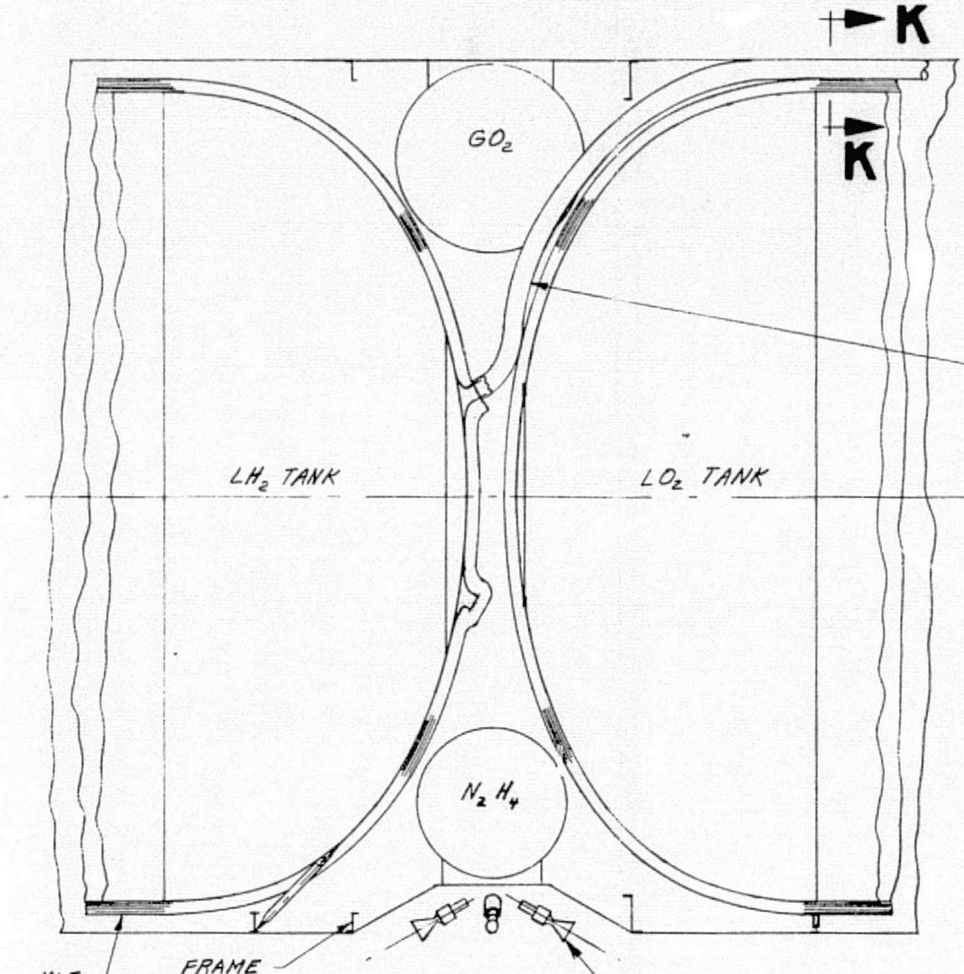

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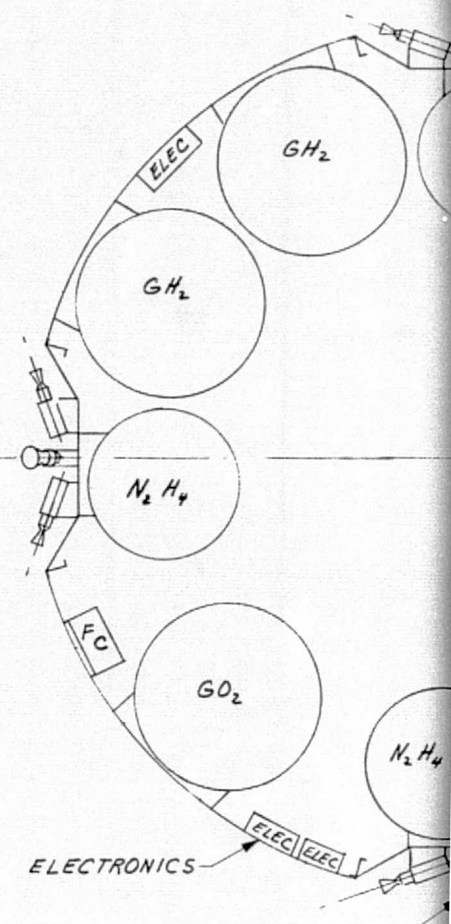

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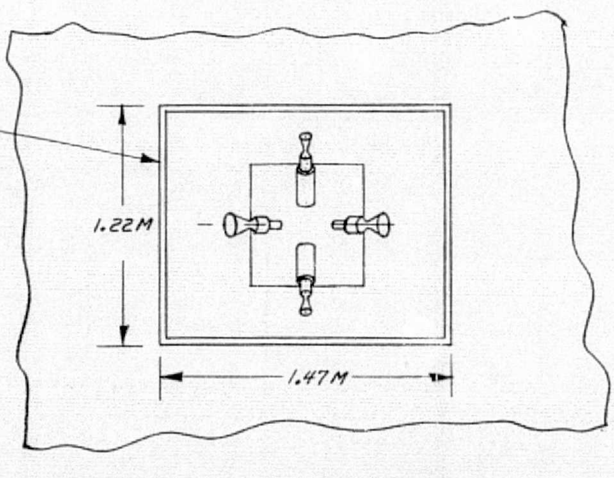

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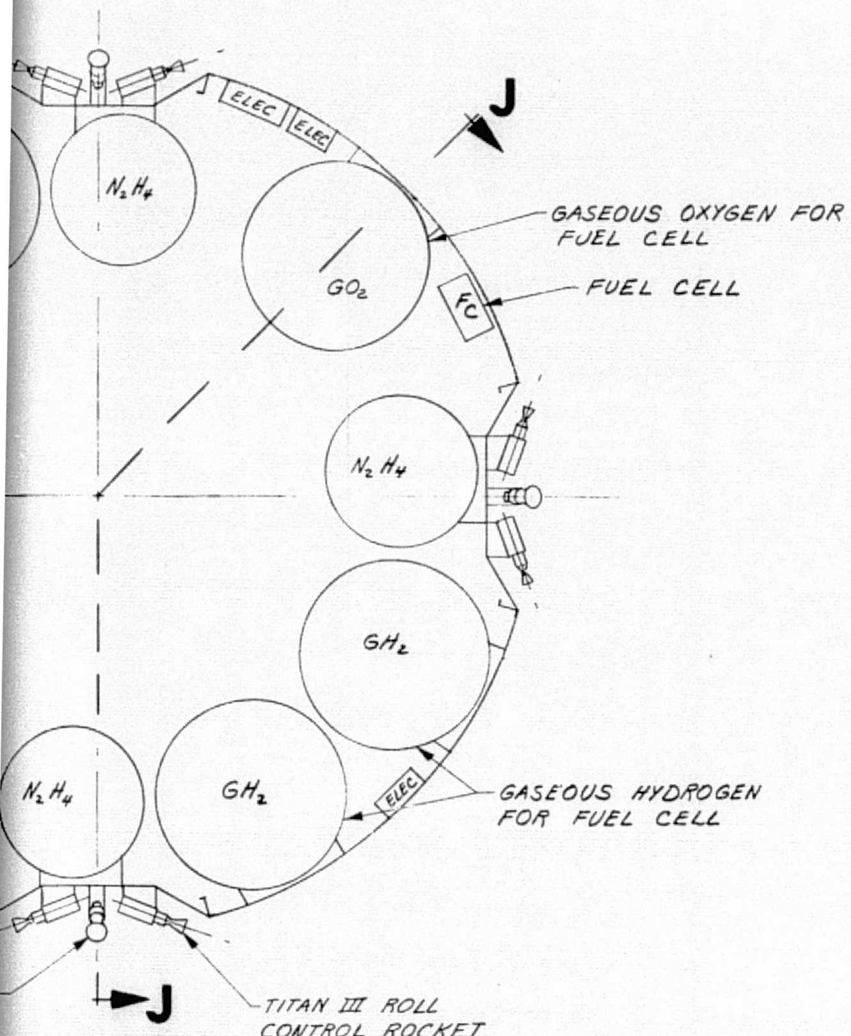

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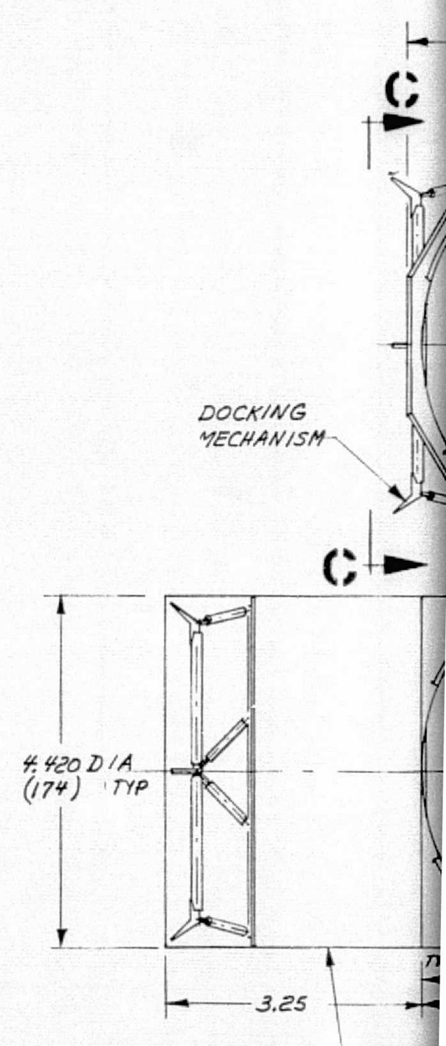

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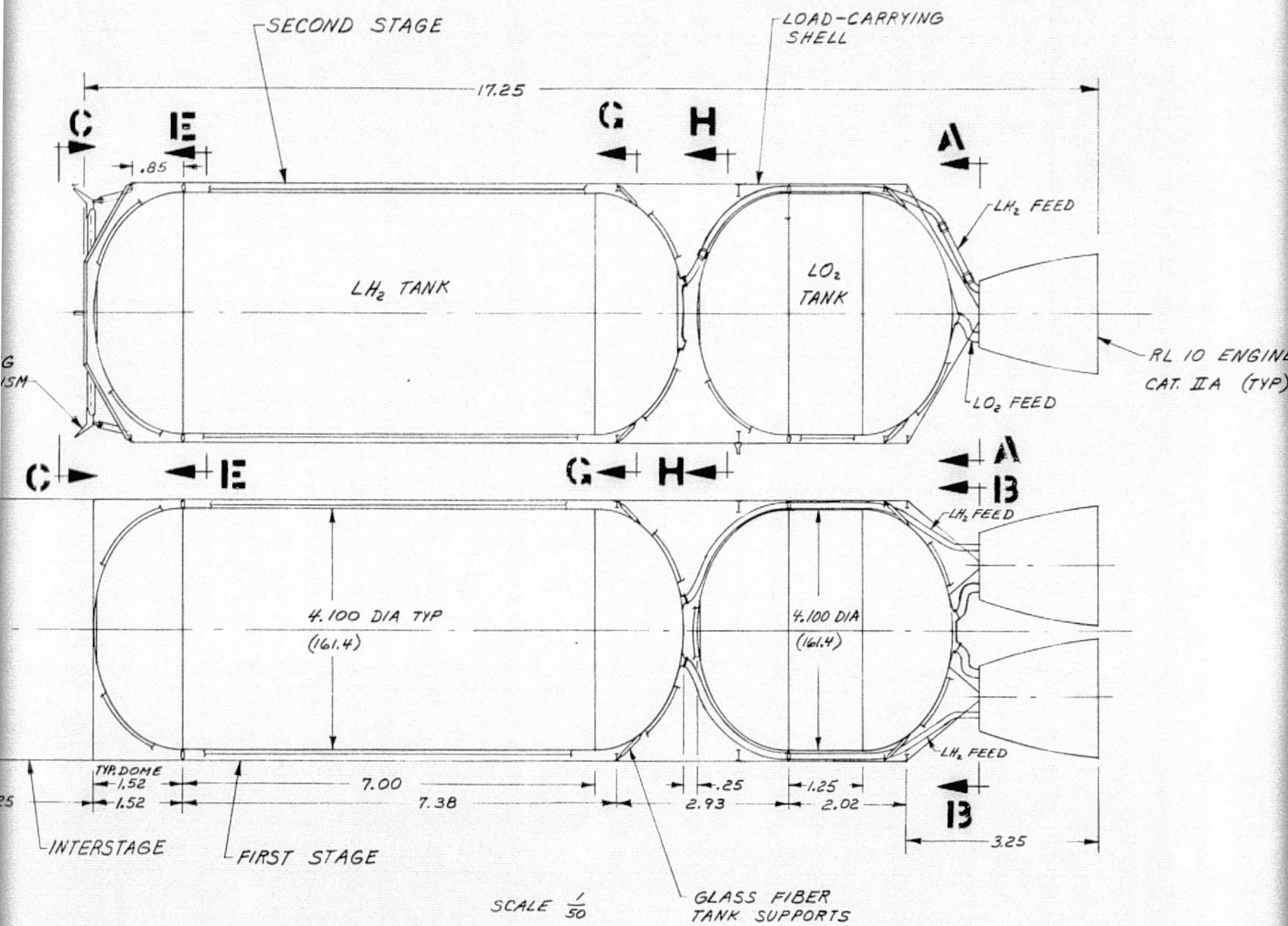

Figure 4-1. Layout – Orbital Transfer Ve

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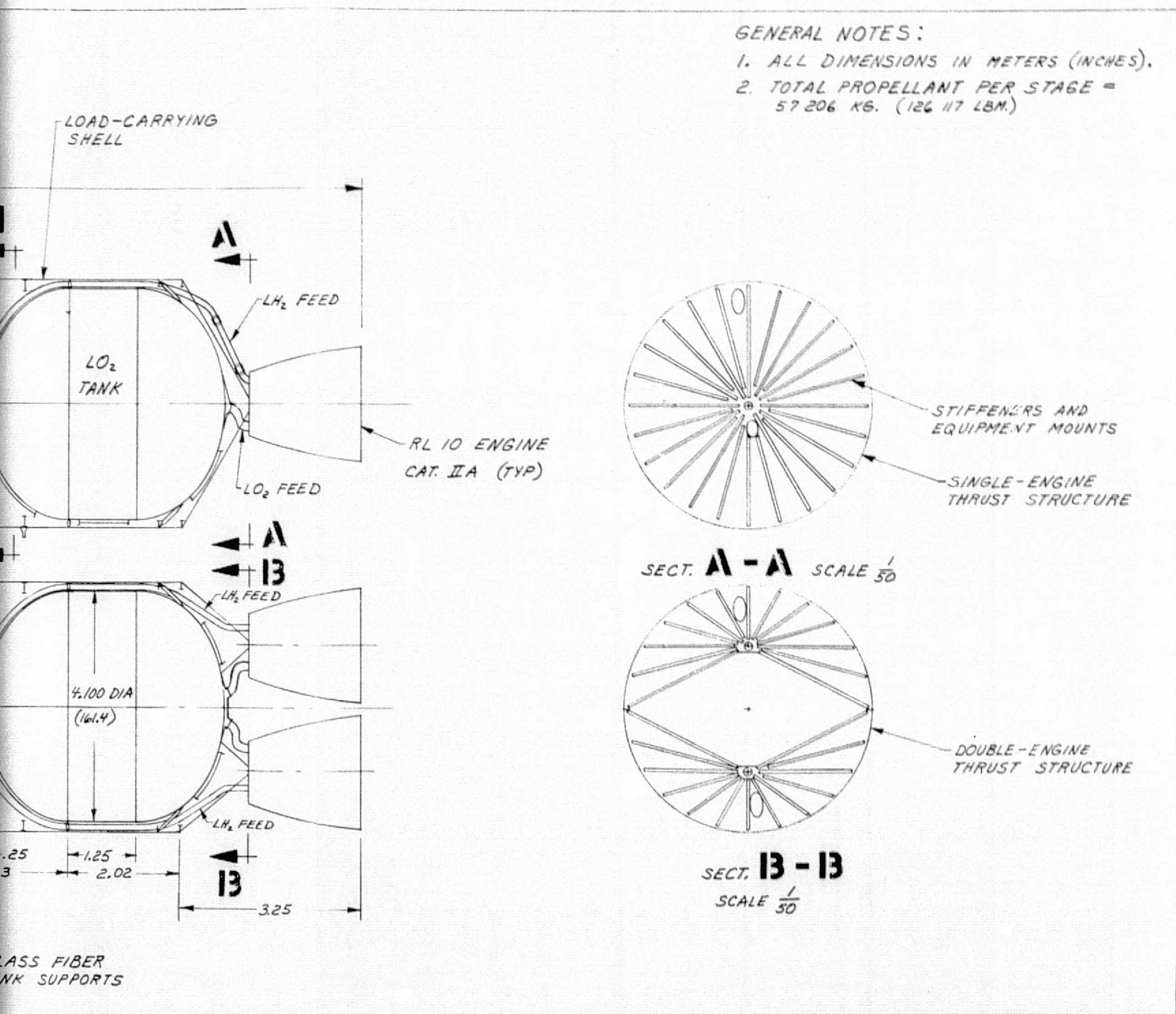

CONTRACT NO.			MCDONNELL DOUGLAS ASTRONAUTICS CO. WESTERN DIVISION HUNTINGTON BEACH, CALIFORNIA
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PREPARED	H. ANDERSON	77-2-16	
CHECKED			
ENGINEER	H. ANDERSON		
DESIGN ACTIVITY APPROVAL			SIZE: / CODE IDENT NO. 18355 / DRAWING NO. OTV-770216
CUSTOMER APPROVAL			SCALE NOTED / SHEET 1 OF 2

Figure 4-1. Layout – Orbital Transfer Vehicle

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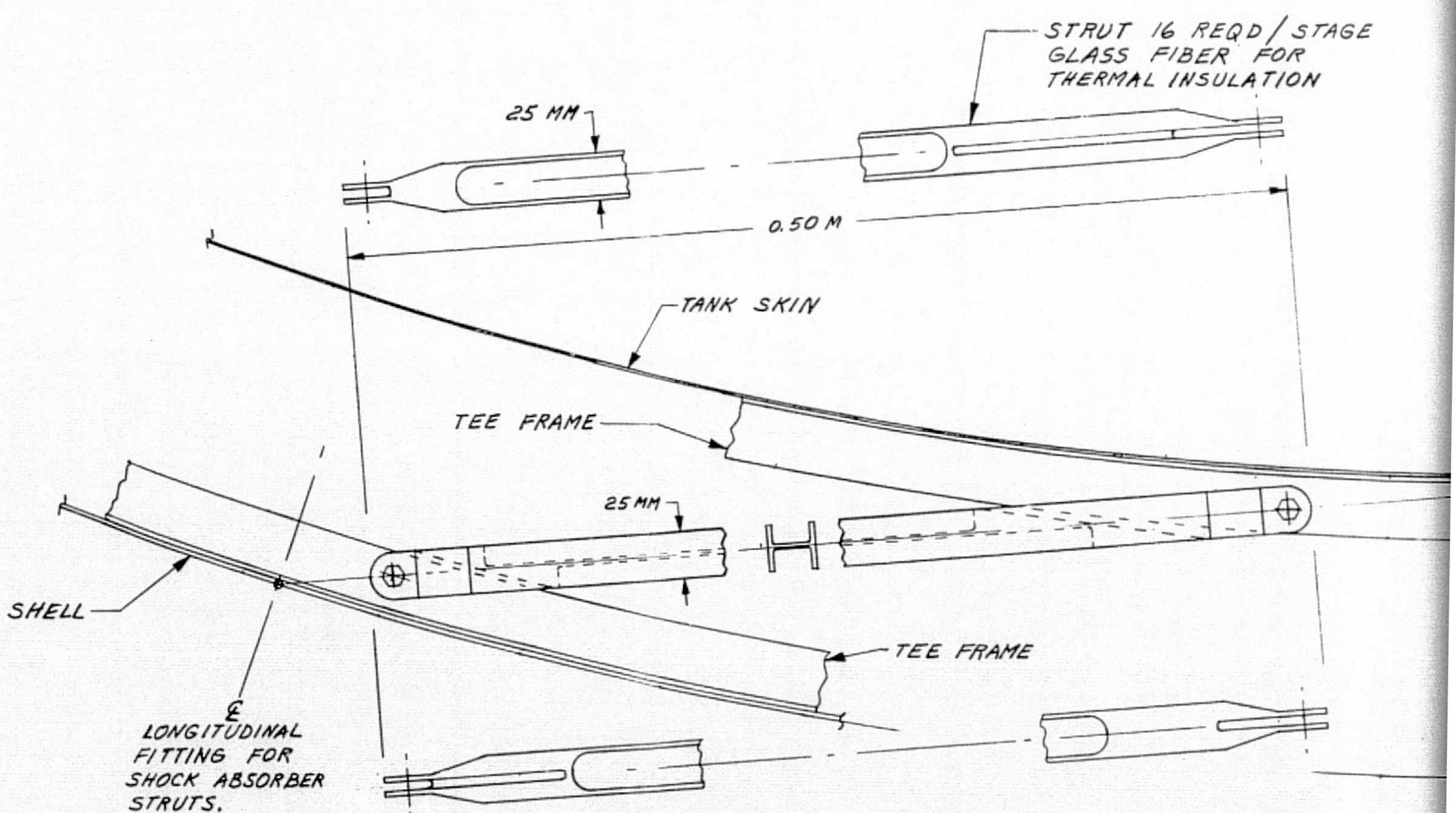

VIEW F
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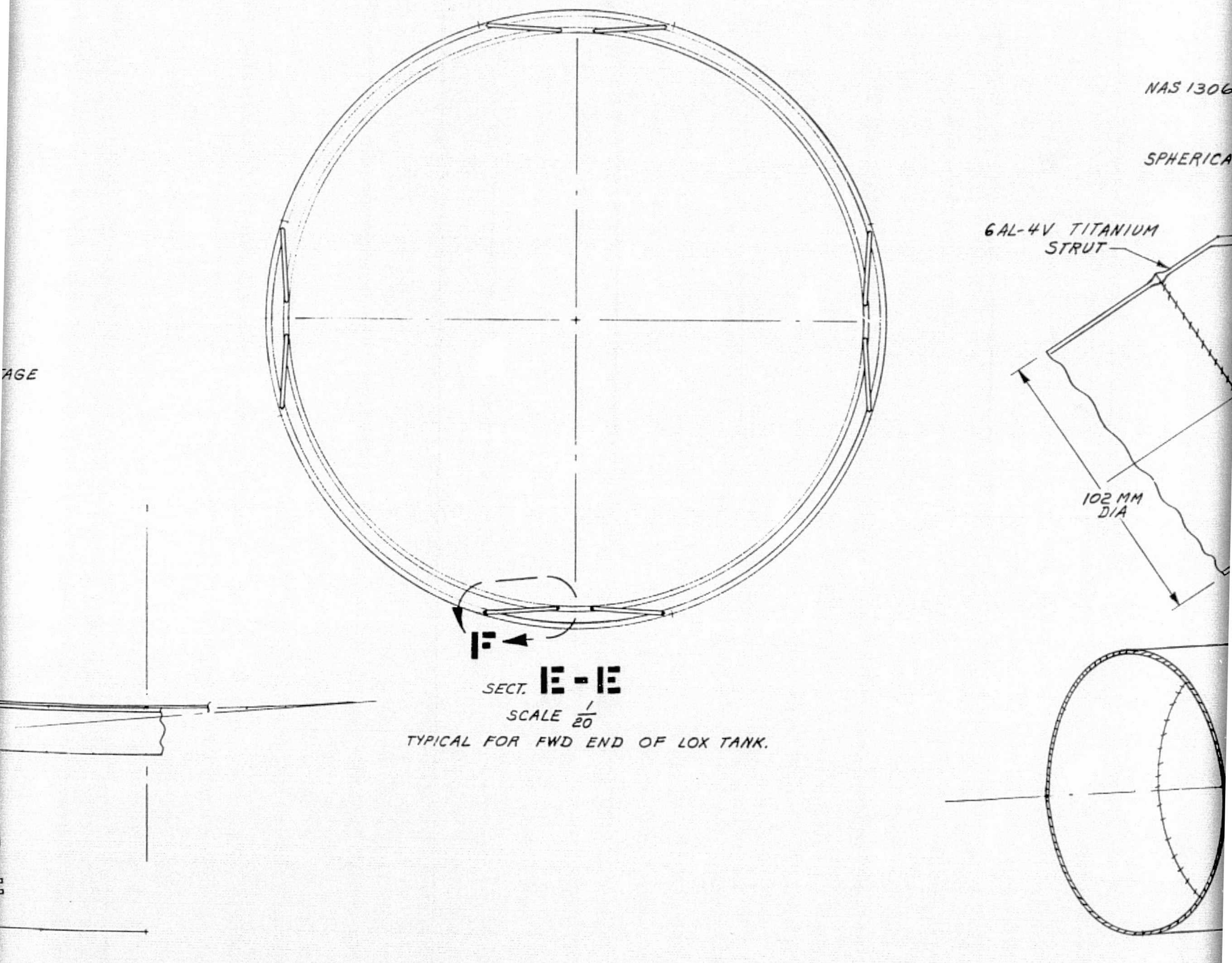

SECT. E-E

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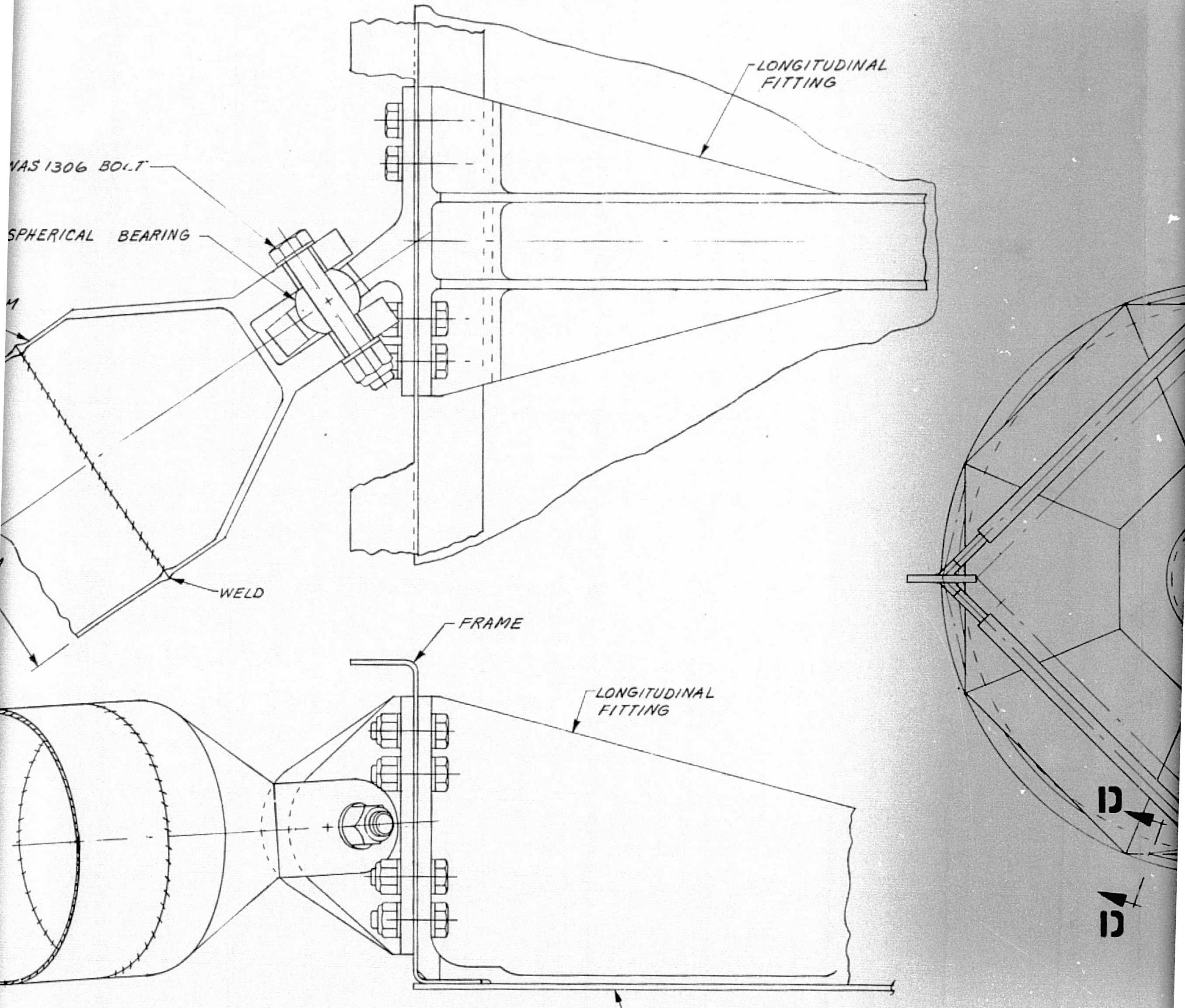
TYPICAL FOR FWD END OF LOX TANK.



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SHOWING ATTACHMENT OF SHOCK ABSORBER TO FRAME.

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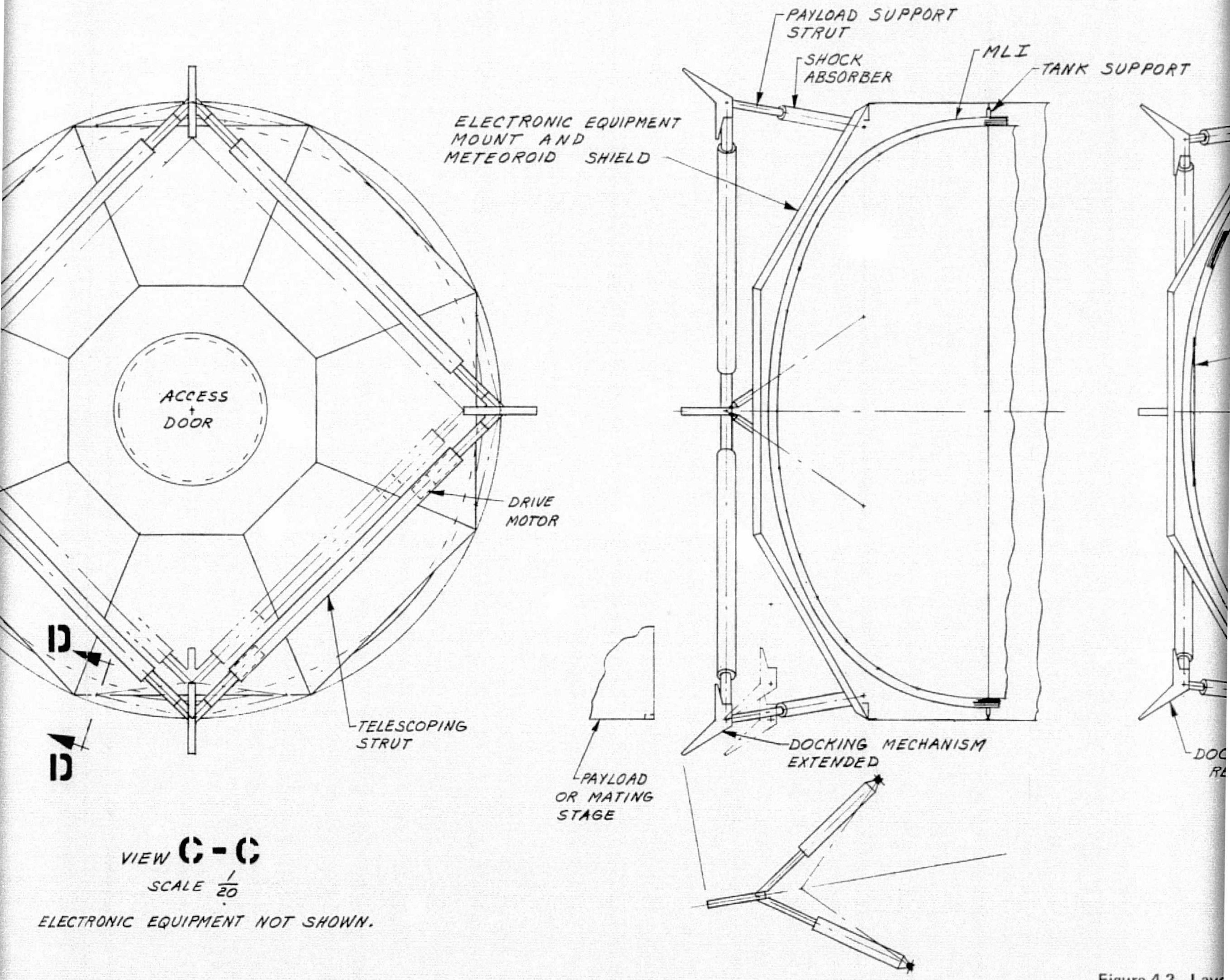

Figure 4-2. Layo

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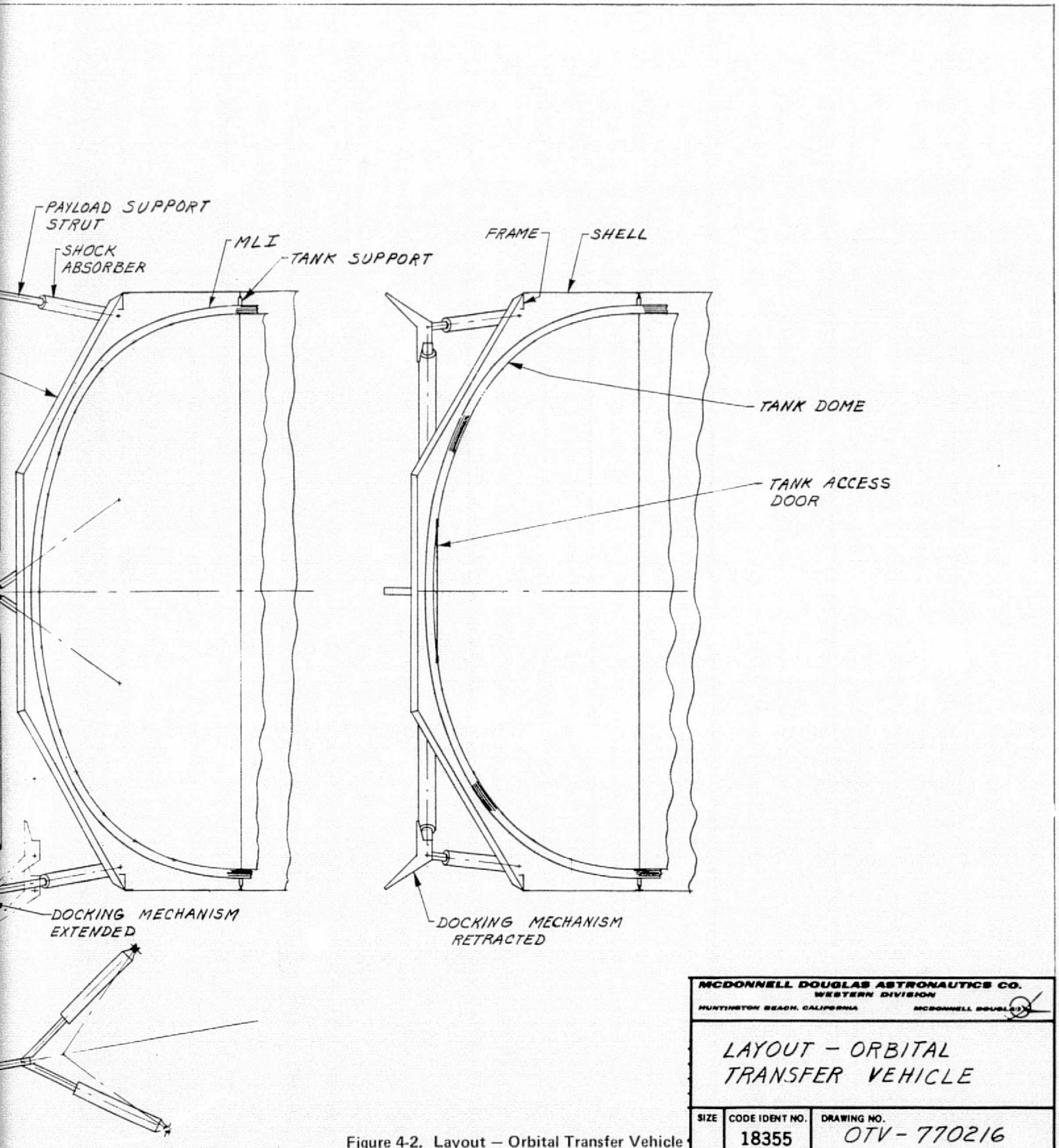

Figure 4-2. Layout – Orbital Transfer Vehicle

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4.1.5.2 Mechanism Description

To reduce costs, the docking mechanisms were assumed to be identical on both stages. They are similar to those on a previous Space Tug Study (Contract NAS8-29677). The system is shown in View C-C of Figure 4-2. During launch in the Orbiter, the mechanism is stowed aft and inboard so that it does not protrude beyond the stage. When docking is desired, the tubular square frame holding the four latches is extended forward beyond the stage by extending the shock absorber struts by compressed air. Each side of the square frame is made up of three sections. A fixed section with a centralizing acme thread is bolted to each side of the guide arm with two through-bolts. A left hand thread is used on the section at one end of each side and a right hand thread on the section at the other end. The threaded sections are joined by an extruded section into which a threaded machined fitting has been bolted at each end. The threads are lubricated with vacuum grease to provide a long service life. The length of a side of the square frame is increased or decreased by rotating the center section of extruded tubing on that side, like a turnbuckle. To expand or contract the frame, the four center sections must be rotated synchronously.

The drive motor and the flex spline of a harmonic drive are bolted to a flange on the end of one of the fixed threaded sections. A planetary wave generator is used. Each of the other three sides has an identical harmonic drive arrangement but without the drive motor. For radial motion of the latches the drive motor is energized, and the square frame is expanded or contracted until the latches are at a diameter compatible with the mating surface.

The maximum mass of a payload was assumed to be less than the mass of a loaded OTV, i.e., 66,971 kg (147,646 lbm). The closing velocity was assumed to be 0.305 mps (1 ft/sec). The total energy to be absorbed is 1,555.5 Joules (1,147 ft-lb). Due to slight misalignment of the mating surface at impact, two-thirds of the kinetic energy was assumed to be taken at one latch. The shock absorbers were designed with a maximal stroke of 30 cm (11.8 in.). Helium gas was assumed because of its high specific heat ratio of 1.66. Reversible abiabatic compression of the gas was assumed and

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the initial gas pressure was calculated to provide the necessary work of compression. This pressure was found to be 34.5 N/cm^2 (50.0 psi).

The shock absorber strut incorporates an antirebound feature by displacing oil while being compressed, but not allowing the oil to flow back into the cylinder when the compressed air attempts to extend the cylinder after the cylinder has been compressed.

4.1.6 Meteoroid Protection

The duration of exposure to meteoroid flux is undetermined, but is probably on the order of 30 days maximum per mission. This is considerably longer than the 1 to 6 days defined in the previous Space Tug Study. The previous Space Tug relied on the shell and MLI blankets for protection of the cylindrical sections of the main tank and various pressure vessels, and on the purge bags and MLI blankets for protection of the main tank domes. The current configuration has no purge bags since the stage is not loaded with propellant on earth. However, the current MLI is 250 layers of reflectors on the LO_2 tank and 180 layers on the LH_2 tank, vs 45-50 layers on the previous study. Also there is a vapor shield of 0.41 mm (0.016 in.) aluminum between the MLI and the tank wall. In addition, the shell thickness of the current configuration varies from 1.68 to 2.79 mm (0.66-0.110 in.) compared to the previous configuration of honeycomb sandwich with 0.25 mm (0.010 in.) faces. It is felt that the current configuration is probably adequate for a 30-day mission. With more information on the mission duration, a precise meteoroid protection analysis may be made.

4.1.7 Avionics Support

The avionics support structure consists of an eight-sided "conic" structure, which extends over the forward dome of the LH_2 tank and is attached to the frame, which also supports the docking mechanism struts. This support structure also acts as a meteoroid barrier for the dome. A circular door in the center provides access to the tank access door and the LH_2 vent and relief valve cluster.

The support structure is composed of a framework of 7075 aluminum beams, with 7075 aluminum isogrid panels attached to the framework to provide a closed meteoroid barrier. Holes in the nodes of the isogrid provide attachment of avionics components. If necessary, thermal control devices may be connected to the support structure to provide heating or cooling of the avionics components.

Table 4-1

SUMMARY OF ULTIMATE FLIGHT LOADS IN ORBITER AND SPACE FLIGHT

	N_c Ultimate N/m (lb/in)					
	Boost Maximum Load Factor Orb Alone	Liftoff	Landing A	Landing B	Orbical Delivery	Orbital Round Trip
Forward Skirt	11.3	29.8	39.2	32.1	12,259 (70.0)	1,564
	2,573.8	4,175.7	5,591.6	4,888.3		
LH_2 Tank	17,511.2	32,953.0 (188.2)	14,299.6	18,725.6	12,609	12,080
Inter-Tank	21,687.9	44,168.8 (252.2)	25,425.2	28,476.4	12,784	12,340
	-7,524.5	5,832.5	23,277.6	13,824.0		
LO_2 Tank	-7,524.5	-4,908.4	14,076.3 (80.4)	6,295.7	12,784	12,340
Inter-Stage	-809.2	517.4	2,892.7	1,626.2	12,259 (70.0)	11,539

transfer to the main propellant tanks, which causes boiloff, it was felt that tanks suspended by low-thermal-conductance struts would better minimize thermal transfer, especially from the intertank structure. The meteoroid protection problem appeared to impose about the same mass penalty for either option. Therefore, a load-carrying shell was chosen as the baseline configuration.

For the shell surrounding the hydrogen tank (one of the two most highly loaded), the following material and construction configurations were designed: (1) GY-70/904 graphite epoxy monocoque; (2) 7075-T6 aluminum monocoque; (3) aluminum isogrid; and (4) sandwich consisting of graphite epoxy faces with aluminum honeycomb core. A summary of the masses per unit area for these configurations is shown in Table 4-2. From this trade study, graphite epoxy monocoque was chosen for its minimal mass and its lower construction costs compared to graphite epoxy sandwich.

Table 4-2

LH_2 TANK SHELL COMPARISON OF MASSES

Config. No.	Configuration	Mass kg/m^2 (lb/ft^2)
1	Graphite epoxy monocoque	3.576 (0.732)
4	Graphite epoxy sandwich (2 ply faces)	~3.641 (0.746)
4	Graphite epoxy sandwich	4.535 (0.929)
3	Aluminum isogrid	<4.709 (0.964)
2	Aluminum monocoque	7.101 (1.454)

Assuming that graphite epoxy monocoque would be optimum or near optimum for the other sections of the shell, these sections were designed with this material and construction without repeating the trade study. A summary of shell thicknesses is shown in Table 4-3.

Table 4-3

SUMMARY OF SHELL THICKNESSES

Section	mm	in.
Intertank	2.79	0.110
LH_2 tank	2.24	0.088
LO_2 tank	1.68	0.066
Forward skirt	1.68	0.066
Interstage	1.68	0.066

In all designs using graphite epoxy monocoque, thicknesses were assumed to be in multiples of four plies--0 degree, ±45 degrees, 90 degrees--so that bending stiffness would be nearly isotropic.

In the intertank area there are many pressure vessels which must be easily removable for recharging. Most of the shell is cut away for removable doors. The carry-through structure then becomes primarily eight longerons. The door housing the ACS rockets also supports the hydrazine tank to fuel them. The entire unit is removed and replaced after each flight. Only an electrical connection must be made to make the system operable.

At each of the docking-mechanism support-strut attachment points there is a longitudinal fitting (shown in Section D-D of Figure 4-2) to distribute the concentrated load. This is typical for the second stage forward skirt and the interstage structure.

4.1.3 Tankage and Supports

The LO_2 tank diameter (4.10m) was chosen to provide room on opposite sides of the tank for the 10 cm diameter vacuum jacketed LH_2 feed lines and the multilayer insulation. The LH_2 tank diameter was chosen to be identical to the LO_2 tank diameter so that common tooling could be utilized in making the domes and cylinders.

Cassinian domes of $n = 2$ and $k = 0.40$ were chosen because that combination is the flattest that can be obtained without tensile buckling of the dome due to pressure. Spherical domes were considered but rejected because the stage is length-limited and minimal gages may be used with Cassinian domes anyway.

Tank wall thicknesses were based on a pressure of 13.8 N/cm^2 (20 psia) which is necessary for the vapor shield/venting system of the tank. This pressure is adequate for the RL-10 engine inlet pressure requirements.

2219 aluminum was selected as the tank wall material because of good cryogenic properties, good weldability, extensive experience, and abundance of test data.

The aft supports of each tank are 16 pairs of tubular laced glass fiber epoxy hinged struts. The choice of material was based on thermal conductivity and economy of fabrication. The forward struts of each tank are tangential supports which allow for radial and longitudinal expansion and contraction of the tank due to pressure and temperature changes while restraining the tank from sideways motion.

4.1.4 Thrust Structure

The first and second stage thrust structures have different geometry but are of the same general construction. The thrust of the RL-10 engine is approximately 66,700 N (15,000 lbf limit). A 1.68 mm (0.066 in.) conical skin of graphite epoxy is stiffened with graphite epoxy or aluminum stiffeners. The stiffeners are also used to mount the multitude of pressure vessels, valves, and lines usually associated with liquid rocket engines. Since the shell is the primary load-carrying structure, the thrust structure is attached directly to the shell rather than the LO_2 tank aft dome. This eliminates the need to penetrate the LO_2 tank with Huck lockbolts or similar attachments.

4.1.5 Docking Mechanism

4.1.5.1 Basic Concept Selection

Once both OTV stages have been delivered to LEO and have been fueled, they will have to be docked together for flight. Likewise, since the stages separate during the course of the mission, they will again have to dock together prior to initiating the next mission. The means for docking the two stages will be integral with the structure that joins the two, that is, the interstage structure.

The interstage will most probably have to be transported to orbit with the upper stage OTV (OTV-2). The OTVs are sized so that they take up all the usable length of the Orbiter cargo bay. Hence, the only room for the interstage is at the stage aft end, around the engine. Since there is no need for a docking mechanism at the aft end of OTV-1, it is presumed that the interstage will be transported with OTV-2. This is shown on Figure 4-3.

A number of options are available regarding both interstage and docking interface location. First of all, the interstage could remain attached to OTV-2, and with its docking interface (passive) on the aft end, mate with the OTV-1 with an active docking interface at its forward end. In this case the front ends of both OTVs would be the same, as the upper stage would have provisions to dock with payloads. A performance penalty would have to be paid, however, as the interstage would then be second-stage weight.

In order to avoid the performance penalty, it is preferred to maintain the interstage with the first stage OTV; thus, after delivery to orbit, the interstage would have to be attached to OTV-1. The normal stage interface, with attendant docking mechanism, would be at Location A, shown on Figure 4-3.

There are a couple of options available for attaching the interstage to the lower stage, plane B on the figure. The simplest might be to provide another docking mechanism at that interface, and pay an appropriate weight penalty. In this case, on the initial trip to LEO, the OTVs would dock as is, and that interface (at B) would then remain intact until such time as a return trip to earth is necessary. The alternate to that would be to provide a field joint at

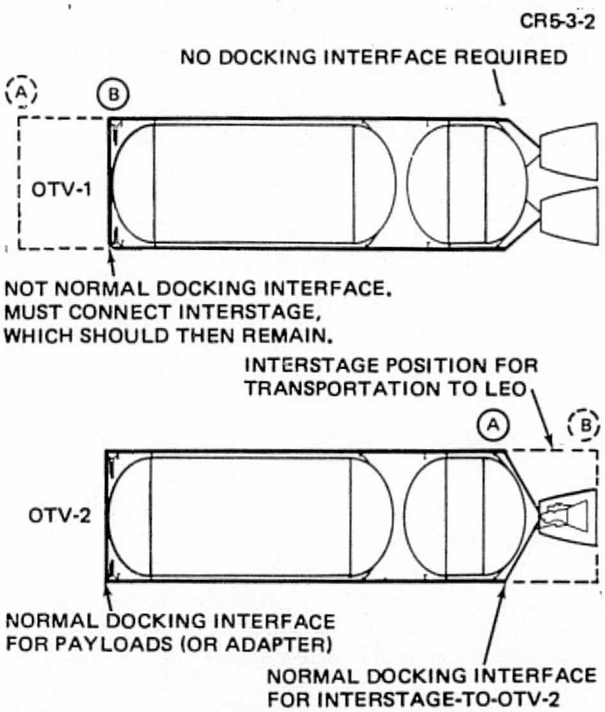
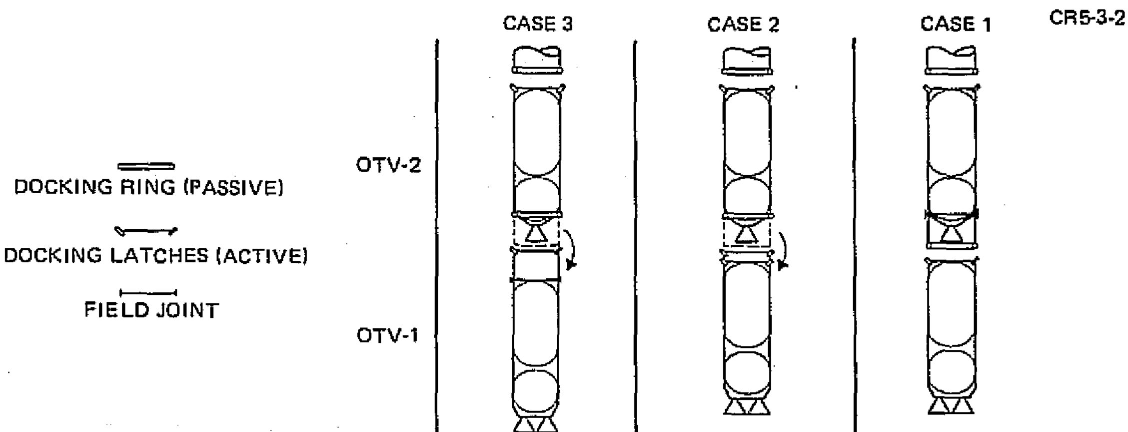


Figure 4-3. Docking Concepts

Location B. The the interstage would have to be removed from OTV-2 and bolted to OTV-1, which would most likely be an EVA operation. The reverse would then have to take place to provide for the return trip to earth.

The three cases discussed are summarized on Figure 4-4. Case 1 is that where the interstage remains with the upper stage. Case 2 calls for the double docking interface, and Case 3, the field joint and attendant EVA operations. Case 2 is the preferred approach. Although there is some performance penalty with the additional docking interface, it is not nearly as much as that of Case 3. By having the additional mechanism, there is no need for EVA. Also, the OTVs would have a common forward end, and each would be available to accommodate a single-stage mission.



	CASE 3	CASE 2	CASE 1
INTERSTAGE FOR TRANSPORTATION TO LEO	OTV-2	OTV-2	OTV-2
INTERSTAGE FOR MISSIONS	OTV-1	OTV-1	OTV-2
EVA REQUIRED (INTERSTAGE TRANSFER)	YES	NO	NO
FIELD JOINT	OTV-1	NONE	OTV-2
DOCKING LATCHES	OTV-1	OTV-1 (TWO)	OTV-1
DOCKING RING	OTV-2	OTV-2	OTV-2
OTV-1/OTV-2 COMMON FRONT END	NO	YES	YES
DOCKING INTERFACES	ONE	TWO	ONE
WEIGHT ON OTV-1 KG (LB)	353 (778)	439 (968)	153 (338)
WEIGHT ON OTV-2 KG (LB)	92 (202)	92 (902)	273 (602)

Figure 4-4. Docking Concept Evaluation

4.2 PROPULSION

The OTV main propulsion system is based on the use of the oxygen/hydrogen propellant system which has been well demonstrated on the Apollo program and Centaur vehicle. Selections of the Pratt & Whitney (P&W) RL-10 derivative main engine, hydrazine attitude control subsystem, and other support subsystems will be discussed in the following paragraphs.

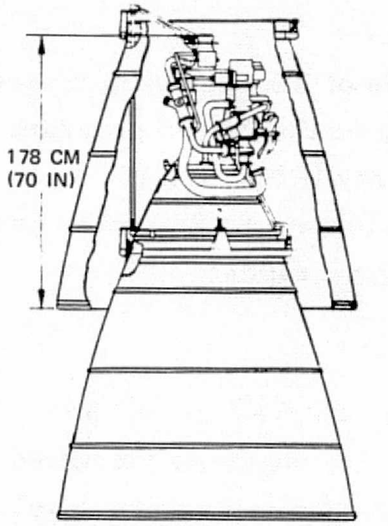
4.2.1 Main Engine and TVC

A P&W derivative of the RL-10A-3-3 engine (used on the Centaur vehicle) was selected for the OTV main engine. This derivative engine is identified as the Category IIA RL-10, and design and cost characteristics were developed by P&W during a NASA/MSFC-funded study titled Design Study of RL-10 Derivatives. This engine selection was based primarily on cost considerations as discussed in Section 4.2.4.

The Category IIA main engine configuration is shown in Figure 4-5 and 4-6.

The Derivative IIA engine is derived from the RL-10A-3-3 engine, with increased performance and operating flexibility. With a nominal full thrust level of 66,723N (15,000 lb) (in vacuum) at a mixture ratio of 6.0:1, the Derivative IIA engine is defined as the RL10A-3-3 engine with the following changes:

- Add two-position nozzle and recontour primary section to give a large increase in specific impulse with no increase in engine installed length. Engine installed length is therefore, limited to 178cm (70 inches). With a truncated two-position nozzle installed, this engine has to be able to be installed and tested in the existing test facilities at FRDC.
- Reoptimize RL10A-3-3 injector for operation at a full thrust mixture ratio of 6.0:1.
- Add tank head idle mode (THI) of operation. THI is a pressure fed mode without turbopump rotation. Propellants are supplied from the vehicle tanks at saturation pressure. Propellant conditions at the engine inlets can vary from superheated vapor, through mixed phase, to liquid. The objectives are to supply a low thrust for



THRUST	: 66,723 N (15,000 LB)
CHAMBER PRESSURE	: 276 N/CM2 (400 PSIA)
AREA RATIO	: 66.2/262
I_{SP}	: 459 SEC AT 6.0 MR
OPERATION	: FULL THRUST (SATURATED PROPELLANTS)
	: MANEUVER THRUST (SATURATED PROPELLANTS)
CONDITIONING	: TANK HEAD IDLE
WEIGHT	: 233 KG (513 LB)
LIFE	: 190 FIRINGS/5 HOURS
ENVELOPE	: 178/323 CM LENGTH (70/127 IN)
	: NOZZLE EXIT DIAMETER (40/79.6 IN) 102/202 CM

Figure 4-5. Derivative IIA Description

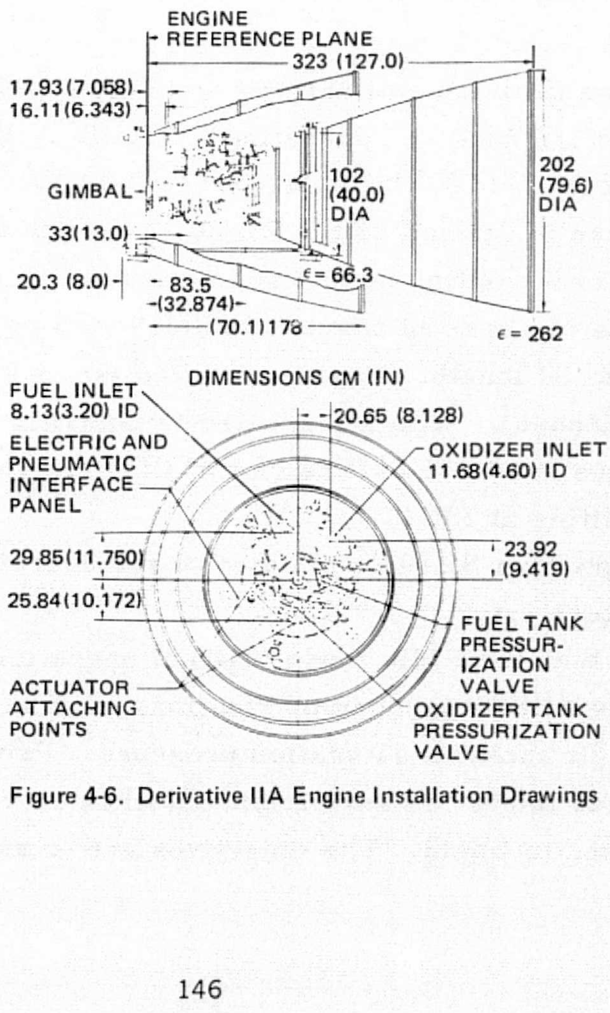


Figure 4-6. Derivative IIA Engine Installation Drawings

propellant settling and also obtain useful impulse from the propellants used to condition the engine and vehicle feed systems.

- Add maneuver thrust (MT) mode of operation. MT provides low thrust in pumped mode, without significant impact on the engine's design.
- Add two-phase pumping capability. Allows operation at both full and maneuver thrust levels with saturated propellants in the vehicle tanks and without tank pressurization system or vehicle mounted boost pumps.
- Add capability for both H_2 and O_2 autogenous pressurization. May be required on very long burn planetary missions in order to avoid excessively low propellant vapor pressure.

Although maneuver thrust is noted as an added feature on the derivative IIA engine, a need has not been identified for this capability on the OTV. Therefore, if a cost savings could be realized, this feature could be deleted as a requirement for the Category IIA engine. The desirability of the H_2 and O_2 autogenous pressurization capability will be discussed in Section 4.2.2.

A schematic of the engine full-thrust fluid flow path with key pressure and temperature values is shown in Figure 4-7. This schematic shows the requirement for $11 N/cm^2$ (16 psia) propellant at the engine interface. It also shows that the chamber is cooled by the hydrogen which is vaporized, passed through the turbine for pump driving power, and then dumped into the combustion chamber, where it is mixed with the oxygen and burned. This is a very efficient cycle, since there is no requirement for an external power source to drive the turbopump.

A schematic of the engine tank-head-idle (THI) flow is shown in Figure 4-8. This schematic shows information similar to that of the full-thrust operation. In this operating mode the turbopumps are not rotating, and propellant feed is strictly by tank pressure. Both propellants are in a gaseous state when they enter the combustion chamber. The hydrogen is heated and vaporized in the chamber cooling passages and the oxygen is vaporized in a heat exchanger in the hydrogen feed line.

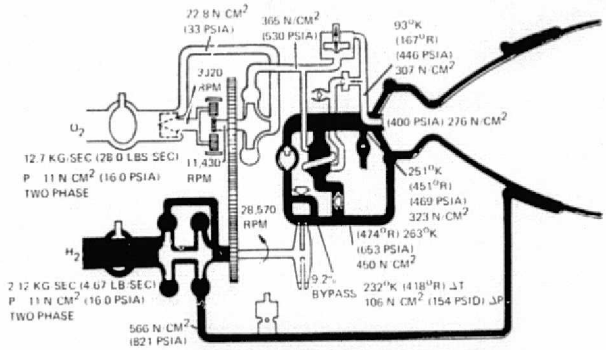


Figure 4-7. Derivative IIA Propellant Flow Schematic, Full Thrust, MR = 6.0

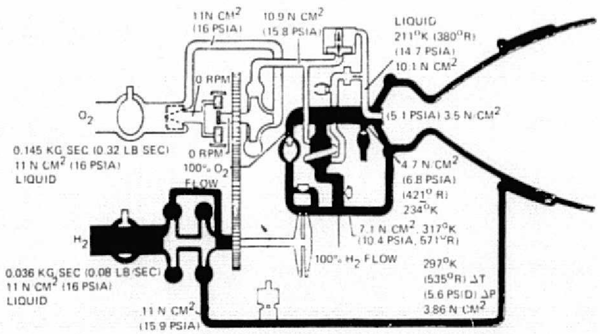


Figure 4-8. Tank Head Idle Propellant Flow Schematic Derivative IIA Engine

The extendible nozzle shown in Figure 4-6 provides increased specific impulse with no increase in installed engine length. The dump cooled extendible nozzle is formed by a smooth outer skin and a corrugated inner skin. The corrugations form coolant passages for hydrogen which enters at an inlet manifold located at an engine area ratio of 66 and discharges to the atmosphere after passing through exit nozzles formed by dimples in the corrugations at an overall engine area ratio of 262. The extendible nozzle coolant supply originates at the turbomachinery gearbox and is supplied to the inlet manifold of the extendible nozzle by a quick disconnect feed value.

The nozzle is translated by means of a jackscrew actuation system, which consists of three ballscrew jackshafts which are attached on the aft end of the primary nozzle by individual drive gearboxes and bearing assemblies, and supported at their forward end by an adjustable link. The nozzle drive/synchronization is provided by two redundant electric motors and three interconnecting flexible cables which transmit motor torque to three gear transmissions which drive the ballscrew shafts. The interface between the primary primary nozzle and extendible nozzle is sealed by the use of finger leaf seals.

The steady-state and transient performance characterisitcs of the RL10-Derivative IIA engine are summarized in Table 4-4.

The TVC actuators selected for the OTV are the Apollo SPS electromechanical actuators which were designed and fabricated by Cadillac Controls Co.

The Apollo SPS Gimbal Actuator is a linear-stroke electromechanical servo actuator. It provides a force output proportional to control current input.

Internal position and velocity feedback devices provide electrical outputs which are summed in an external circuit. The closed loop thus formed makes the actuator a stable-position control servo.

Each actuator contains a compound-wound dc motor with an RFI filter. The motor drives a pair of contrarotating magnetic particle clutches through

Table 4-4

STEADY-STATE AND TRANSIENT PERFORMANCE SUMMARY
(DERIVATIVE IIA ENGINE)

Full Thrust Performance	
Thrust, N (lb_f) Vac	66,723 (15,000)
Mixture Ratio	6.0
Chamber Pressure, N/cm^2 (psia)	276 (400)
Specific Impulse, sec	459.2
Required Inlet Condition	
Fuel	<40% vapor
Oxidizer	<40% vapor
Tank Head Idle Performance	
Thrust, N (lb_f) Vac	698 (157)
Mixture Ratio	4.0
Specific Impulse, sec	387
Typical Tank Head Idle Transient	
Initial Thrust, N (lb)	409 (92)
Final Thrust, N (lb)	698 (157)
Cooldown Time in sec[4]	89/90
Start Transient	
Tank Head to Maneuver Thrust Time, sec[1]	1.56 ±0.30
Impulse, N-sec (lb-sec)[2]	17,260 ± 5340 (3,880 ±1200)
Maneuver Thrust to Full Thrust Time, sec[1]	1.31 ±0.12
Impulse, N-sec (lb-sec)[2]	91,670 ± 6670 (20,608 ±1500)
Deceleration Transient	
Full Thrust to Maneuver Thrust Time, sec[1]	0.4 ±0.11
Impulse, N-sec (lb-sec)[3]	31,110 ± 4890 (6,994 ±1100)
Maneuver Thrust to Tank Head Idle Time, sec[1]	1.0 ±0.10
Impulse, N-sec (lb-sec)[3]	7406 ± 1156 (1,665 ±260)
Shutdown Transient	
From Full Thrust Time, sec[5]	0.12 ±0.03
Impulse, N-sec (lb-sec)	7264 ± 667 (1633 ±150)
Propellants Discharged, kg (lb)	9.1 (20)
From Pumped Idle Time, sec[5]	0.11 ±0.03

(1) To 90% of Thrust Change
(2) 2.0 seconds duration
(3) 1.4 seconds duration
(4) Tank Pressure = 11 N/cm^2 (16 psia), initial Engine Temperature = 278°K (500°R), Cold Inlet Lines
(5) To 5% of Initial Thrust Level

Table 4-4

STEADY-STATE AND TRANSIENT PERFORMANCE SUMMARY
(DERIVATIVE IIA ENGINE) (Continued)

Impulse, N-sec (lb-sec)	3250 $\pm$ 222 (731 $\pm$50)
Propellant Discharged, kg (lb)	6.8 (15)
From Tank Head Idle Time, sec[5]	0.08 $\pm$0.02
Impulse, N-sec (lb-sec)[6]	$\leq$1486 ($\leq$334)
Propellants Discharged, kg (lb)	5 (11)

[5] To 5% of Initial Thrust Level

[6] Shutdown impulse from THI varies with initial conditions and operating time in THI.

spur gears (Figure 4-9). The clutches are excited by a control current, which reaches rotating coils through brushes and slip rings. As the excitation current increases, a proportional torque is produced at the output pinion. This pinion drives a gear which is integral with a recirculating ball screw nut. The ball screw translates applied torque to output force.

The output force will act to extend or retract the actuator depending on which clutch is excited. The ball screw is guided and aligned by means of a recirculating ball nut and spline, which also transmits screw reaction torque to the structure. The actuator is connected to the engine by self-aligning spherical rod ends which permit small angular excursions for engine gimbaling. Velocity generators are provided for rate feedback and are driven from the ball screw nut by antibacklash spur gears. Position transducers provide position indication and position feedback, and are driven linearly by attachment to the ball screws. Motors, clutch pairs, velocity generators, and position transducers are duplicated to provide redundancy for reliability purposes. Snubbers are provided at extend and retract travel limits. The snubbers consists of multiple belleville springs and serve to reduce impact loads in case of overtravel. Actuator components are supported on a cast, machined, aluminum load-carrying structure. The entire actuator is enclosed in a welded stainless steel cover with welded metal bellows for

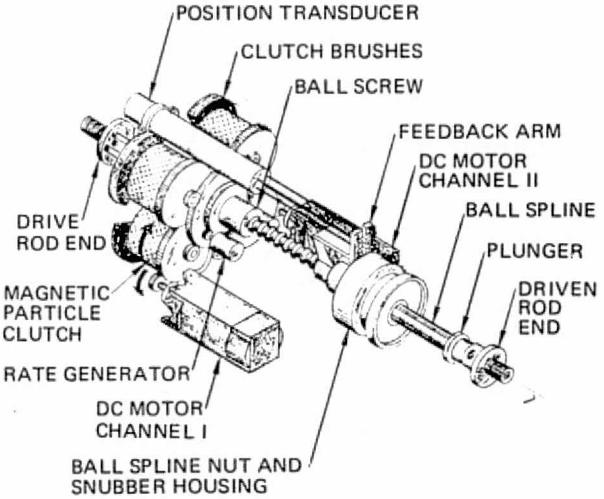


Figure 4-9. Actuator Schematic

angular and linear travel. A visual pressure indicator is provided for external monitoring of internal pressure. Each actuator weighs approximately 11.8 kg (26 lb), has a null length of 56 cm (22 in) and a stroke of 5.08 cm (2 in).

4.2.2 Support Subsystems

This section contains a brief description of the major support subsystems, i.e., pressurization, feed, fill and drain, propellant utilization, and pneumatic. The vent subsystem characteristics are discussed in Section 4.3.

4.2.2.1 Pressurization

The Category IIA RL10 has zero NPSH start capability, provided that the vapor pressure of the incoming propellants is between 11 and 13.6 N/cm^2 (16 and 20 psia). There is some question at this time whether this condition can be satisfied without pressurization when the proposed thermal control system is employed in combination with long engine burn times. Therefore, it is recommended that the autogenous bleed capability of the engine be employed until test and/or flight data establish the thermodynamic characteristics of the propellants during orbital mission operations.

Autogenous pressurization is only available when the engine is operating. During this period of time, warm hydrogen and oxygen gases are supplied from engine interfaces and directed into the tank ullage volumes. Even with autogenous pressurization it is possible, under adverse engine burn/thermodynamic conditions, that a separate prepressurization subsystem could be required to meet the 11 N/cm^2 (16 psia) pressure limit prior to engine start.

4.2.2.2 Main Engine Feed

The main engine feed assemblies for OTV use both developed and new prevalves. The LH_2 feedline has a 7.6 cm (3.0 in) MLI wrapped duct from the tank outlet to a 7.6 cm (3 in) prevalve. The ducting from the prevalve to the engine interface is the same diameter, is insulated, and has a transition section to the 8.1 cm (3.2 in) engine interface. The LO_2 feedline is insulated 10.2 cm (4 in) ducting from the tank outlet to the engine interface and also contains a 10.2 cm (4 in) prevalve. This feedline has a 10.2 cm to 11.7 cm (4.6 in) transition section at the engine interface. Both prevalves are pneumatically actuated. The 10.2 cm (4 in) LO_2 valve is a Parker ball valve which was used on the Saturn I-C stage. The 7.6 cm (3 in) LH_2 valve is a new valve, but could be similar in design to the other Parker valves.

Feedline thermal conditioning is accomplished during main engine THI operation. Liquid propellants are maintained at the feedline inlets by acceleration force provided by the main engine idle-mode thrust.

4.2.2.3 Fill and Drain

The LH_2 and LO_2 fill and drain lines interface with the tanker vehicle through a docking ring located at the forward end of the OTV. Both fill and drain lines are 2.54 cm (1.0 in) in diameter for compatibility with the tanker, and are insulated with multilayer insulation from the docking interface to the tank interface. Self-sealing disconnects can be used at the docking ring interface to close the fill and drain lines when the tanker vehicle is disconnected from the OTV. This design eliminates the need for active shutoff valves on the OTV.

The fill and drain system will probably require diffusers and/or baffles in the propellant tanks to meet the vent requirements during low-g propellant resupply from the tanker.

4.2.2.4 Propellant Utilization (PU)

The selected mode of propellant utilization is closed loop with continuous sensing capacitance probes. The probes are existing design (e.g., Transonics) concentric-tube configurations with an expected outage accuracy of ±1/4%. The control loop operates as follows: (1) the probe outputs enter a PU electronics assembly (signal conditioner), (2) the conditioned signal goes to a module interface unit (MIU), (3) then to a digital computer which determines the proper engine PU valve command, (4) to an MIU, and (5) to the engine PU valve. A PU valve mixture ratio control range of ±0.5 is considered to be adequate.

4.2.2.5 Pneumatics

The pneumatics assembly provides regulated helium (324 ± 8 N/cm^2 primary - 470 ± 12 psia) for the main engine, Tug valve actuation and docking system supply. With the exception of one component, the assembly is composed of developed hardware. The components are tabulated below:

Component	Quantity	Previous Use	Manufacturer	Remarks
• 1.27 cm (1/2 in) disconnect	1			New development
• 1.27 cm (1/2 in) check valve	1	S-IVB	Carter	
• 1.27 cm (1/2 in) burst disc/relief valve	1	S-IVB	Calmec	
• 0-028 m^3 (1 ft^3) bottle	1	PT-4	Pressure Systems Inc.	
• 1.27 cm (1/2 in) dual regulator	2	S-IVB	Fairchild	
• 1.27 cm (1/2 in) solenoid	2	S-IVB	Calmec	
• 1638 cm^3 (100 in^3) plenum	1	S-IVB	Airtec	

4.2.3 Attitude Control System

The OTV attitude control system (ACS) is described in this section. The baseline system selected is a blowdown monopropellant hydrazine system based on previously developed and qualified hardware. A moudlar concept is employed whereby each of four independent modules is replaceable in earth orbit. Each module contains a blowdown propellant tank, four thrusters and an electrical interface. The ACS impulse requirements for the OTV were scaled from results obtained during the Space Tug Systems Study (Cryogenic) for NASA/MSFC.

4.2.3.1 Impulse Requi ments

The total impulse determined during the above referenced Tug Study was 235,755 N-sec (53,000 lbf-sec) for a 30-day round-trip mission. Two-thirds of this impulse was required for translational maneuvers associated with rendezvous, docking, etc; one-third was required for attitude stabilization. Assuming that this same fractional distribution applies to the OTV, and that the translational and stabilization impulses are proportional to mass and moment of inertia, respectively, the OTV impulse was determined as follows:

$$I\ (\text{Total}) = 53,000\ \left(\frac{132,000}{55,000}\right) \times \frac{2}{3} + \frac{147,000}{19,800} \times \frac{1}{3}\ )$$

$$\cong 960,811\ \text{N-sec}\ (216,000\ \text{lbf-sec})$$

Therefore, the OTV requires 240,203 N-sec (54,000 lbf-sec) impulse per module, assuming uniform propellant use from the modules.

4.2.3.2 Systems Comparisons and Selection

Three ACS systems were considered for OTV application: cryogenic, storable bipropellant (N_2O_4/MMH), and blowdown monopropellant hydrazine. The cryogenic system was assumed to be integrated with the main propellant system, with resupply propellants provided from the main tanks. Therefore, this system was not modularly replaceable. However, the storable systems were assumed to be replaceable modules with no orbital transfer of propellant and gases. These systems are shown schematically in Figures 4-10, 4-11, and 4-12.

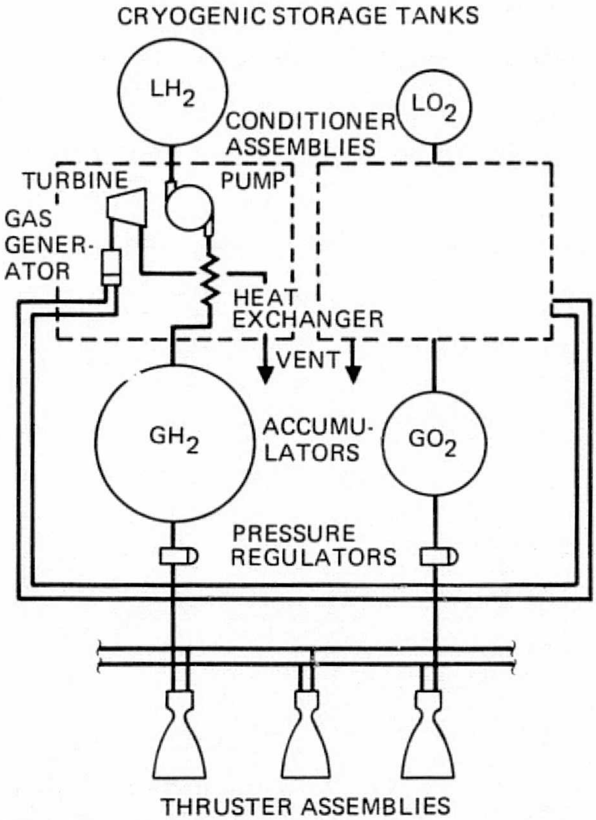


Figure 4-10. Cryogenic ACS System

CR5-3-2

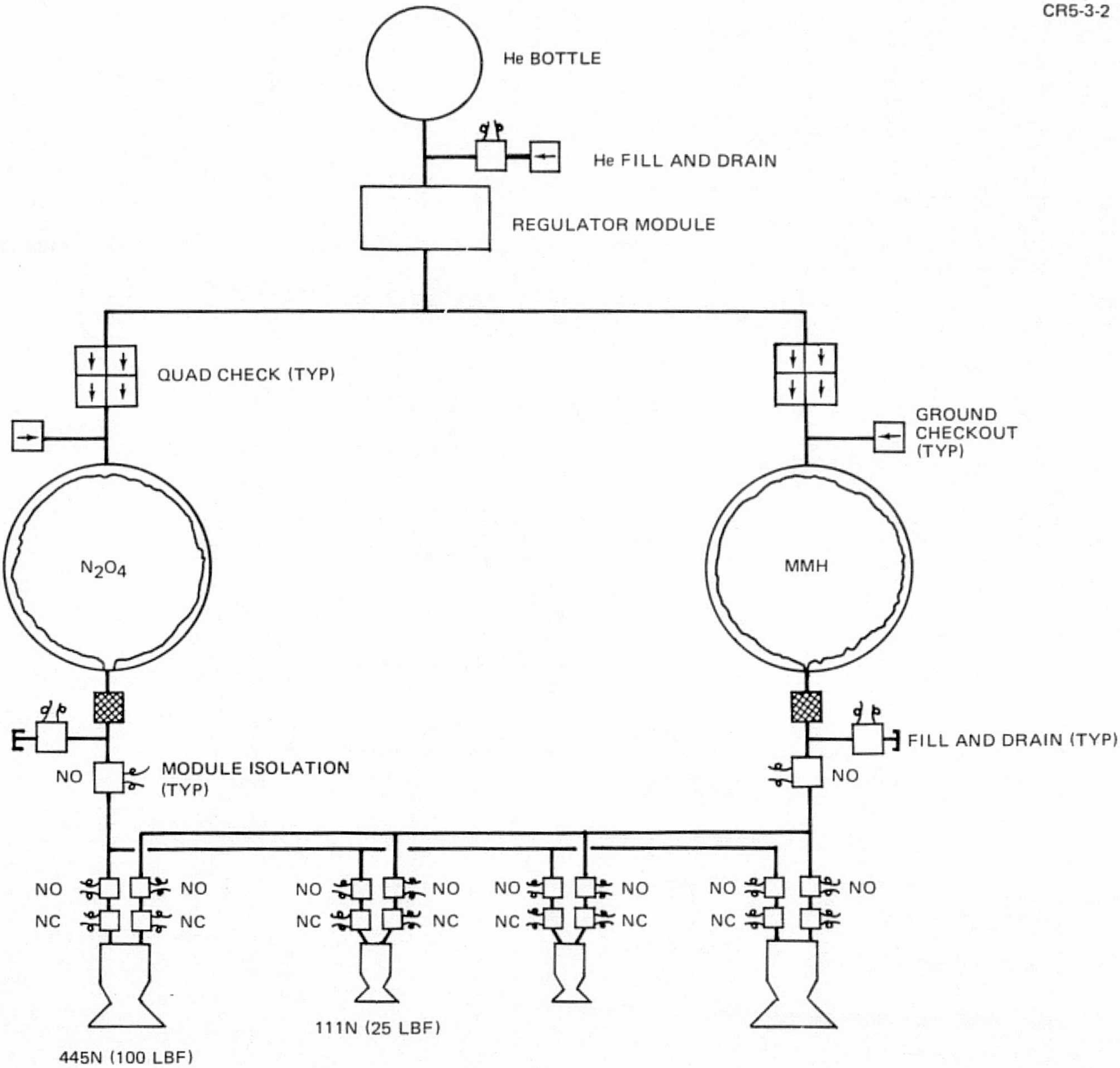


Figure 4-11. Bipropellant ACS Module

CR5-3-2

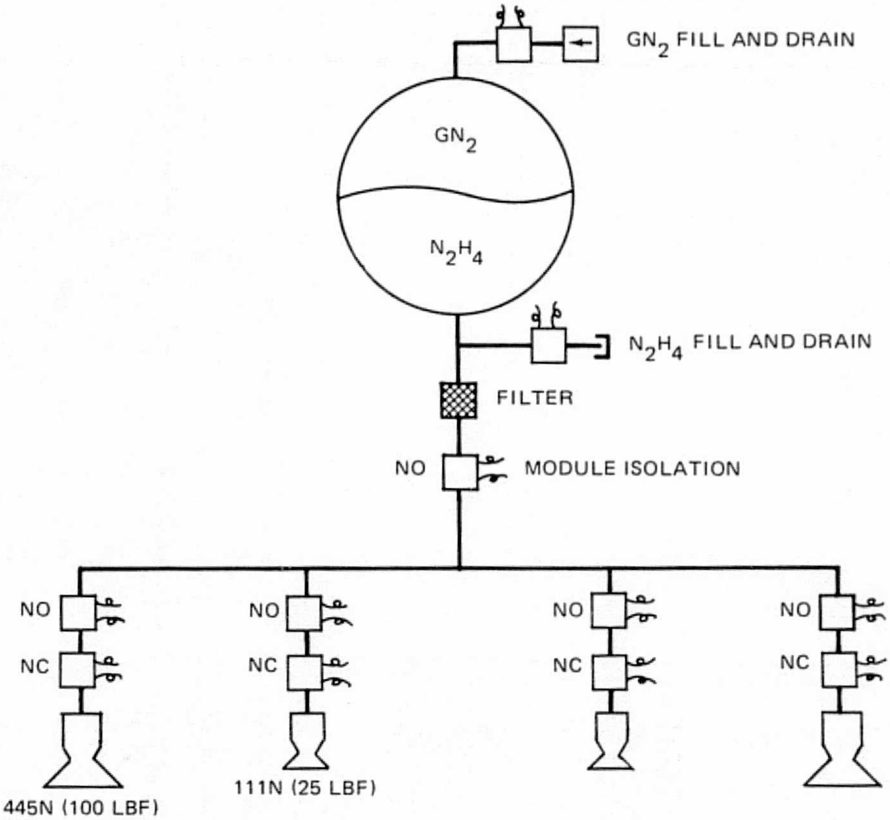


Figure 4-12. Blowdown Monopropellant ACS Module

The cryogenic system was eliminated early in the comparisons because the development cost estimate of $40 million (1973 dollars) was three to four times higher than storable propellant systems. In addition, there was no significant weight advantage at the impulse levels being considered.

The bipropellant and blowdown monopropellant systems were then compared on a weight basis considering the components shown in Figures 4-11 and 4-12. Thrust levels of 445 N (100 lbf) and 111 N (25 lbf) were considered to be adequate for pitch/yaw control and roll control, respectively. The weights are based on previously developed thrusters and components, and new propellant tanks and pressurant bottles (bipropellant only). Other data which affect the weight estimates are tabulated below:

	Bipropellant	Monopropellant
Average Isp	2667 N-sec/kg (272 sec)	2108 N-sec/kg (215 sec)
Propellant tank pressure	1.586 MN/m^2 (230 psia)	2.62 MN/m^2 (380 psia)
Blowdown ratio	N/A	2:1
Pressurant bottle pressure	27.58 MN/m^2 (4000 psia)	N/A
Propellant tank material	Titanium 6AL 4V	Titanium 6AL 4V
Pressurant bottle material	Titanium 6AL 4V	Titanium 6AL 4V
Tank safety	2.0	2.0

The module weights were then calculated as a function of total impulse; the results are shown in Figure 4-13. For the previously determined module impulse of 240,203 N-sec (54,000 lbf-sec) it can be seen that each bipropellant and monopropellant module weighs 125 kg (275 lbm) and 147 kg (325 lbm), respectively. Therefore four modules weigh 500 kg (1,100 lbm) and 588 kg (1,300 lbm), respectively.

The monopropellant system was selected even though it is 88 kg (200 lbm) heavier than the bipropellant system, because the development cost in approximately one-half the bipropellant system cost, and the system is inherently more reliable, since it requires less than one-half as many components. One possible disadvantage of the blowdown system is the decrease in thrust level as the propellants are used. If further study indicates that this is an unacceptable characteristics, a pressurization subsystem can be added to each module with little or no weight increase. The system reliability will decrease, but will still be higher than the bipropellant system.

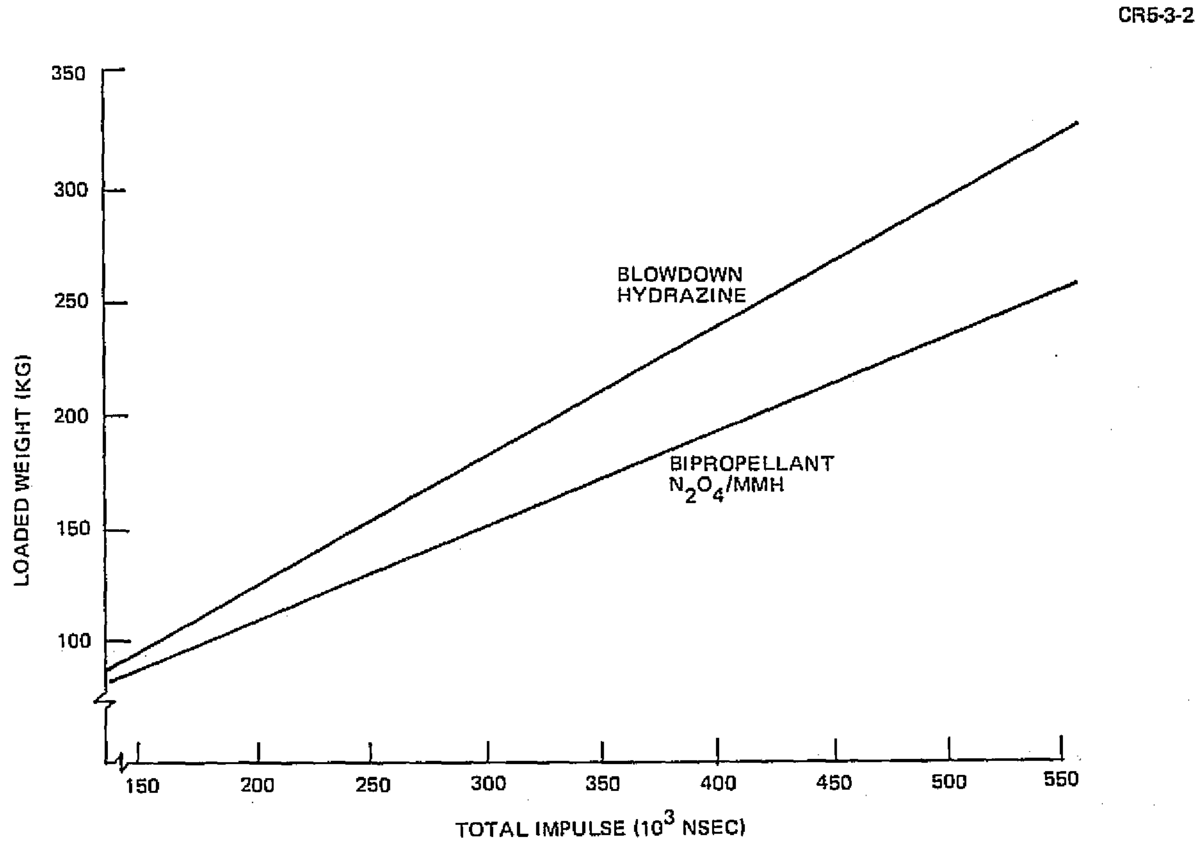


Figure 4-13. ACS Module Weight

4.2.4 Advanced Engines

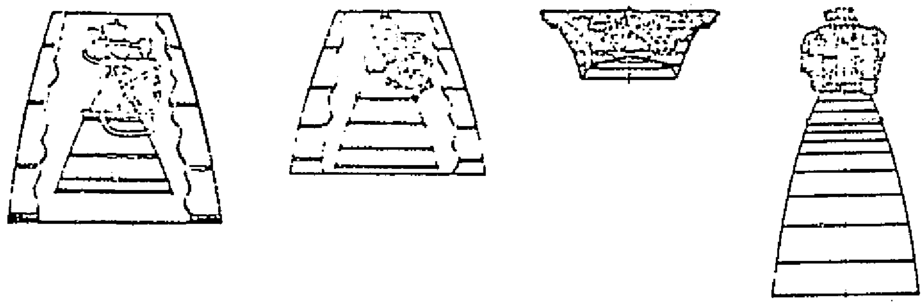
Higher performance engines than the Category IIA RL-10 derivative selected would be available if additional development funds were available. Three advanced cryogenic rocket engines have received significant R&D funding to date, the Category IV RL-10 (NASA/Lewis), the Aerospike (NASA/MSFC), and the Advanced Space Engine - ASE (USAF). Characteristics of these engines are shown in Figure 4-14. As indicated, performance increases in terms of higher specific impulse and lower weight are possible compared to the Category IIA RL-10. In addition, the Aerospike engine is substantially shorter than the other engines, which could be of significant advantage for length-constrained vehicle applications.

If any of these three engines were developed and available for the OTV, the improved performance characteristics could be used beneficially. However, significant development costs are involved to get a qualified engine. When addressed in the MDAC Cryogenic Tug study, the performance increases did not justify the increased RDT&E costs. Further investigation of these engines for OTV application is necessary.

4.2.5 Off-Loading/Mixture Ratio

The OTV was sized at a 6:1 propellant mixture ratio (weight ratio of oxidizer to fuel). Nominal OTV performance was based on engine characteristics at that ratio. In the event off-loading were required, e.g., for less energetic missions, it would be generally more advantageous to off-load oxidizer prior to off-loading fuel, depending on the magnitude of the velocity decrease. Oxidizer off-loading decreases the mixture ratio, and will result in increased specific impulse with a decrease in engine thrust. Fuel off-loading, or raising the mixture ratio, has the opposite effect.

These effects are shown on Figure 4-15, which was extracted from Pratt & Whitney documentation. The data were extrapolated slightly (as indicated by the dashed lines) to cover a wider range of mixture ratio than was presented.



	CAT. IIA RL 10	CAT. IV RL 10	AEROSPIKE	ADVANCE SPACE ENGINE (ASE)
SPECIFIC IMPULSE (SEC)	459	470	470	471
MIXTURE RATIO	6.0	6.0	5.0	6.0
THRUST (N)	66,723	66,723	66,723	66,723
LENGTH (M)(1)	1.78	1.68	0.56	1.28
WEIGHT (KG)	233	192	145	166
RDT&E (1973 $M)	57	119	140	154
DEVELOPMENT TIME (MONTHS)	48	60	60	66

(1) RETRACTED LENGTH – TWO-POSITION NOZZLE

Figure 4-14. Advanced Engine Characteristics

CR5-3-2

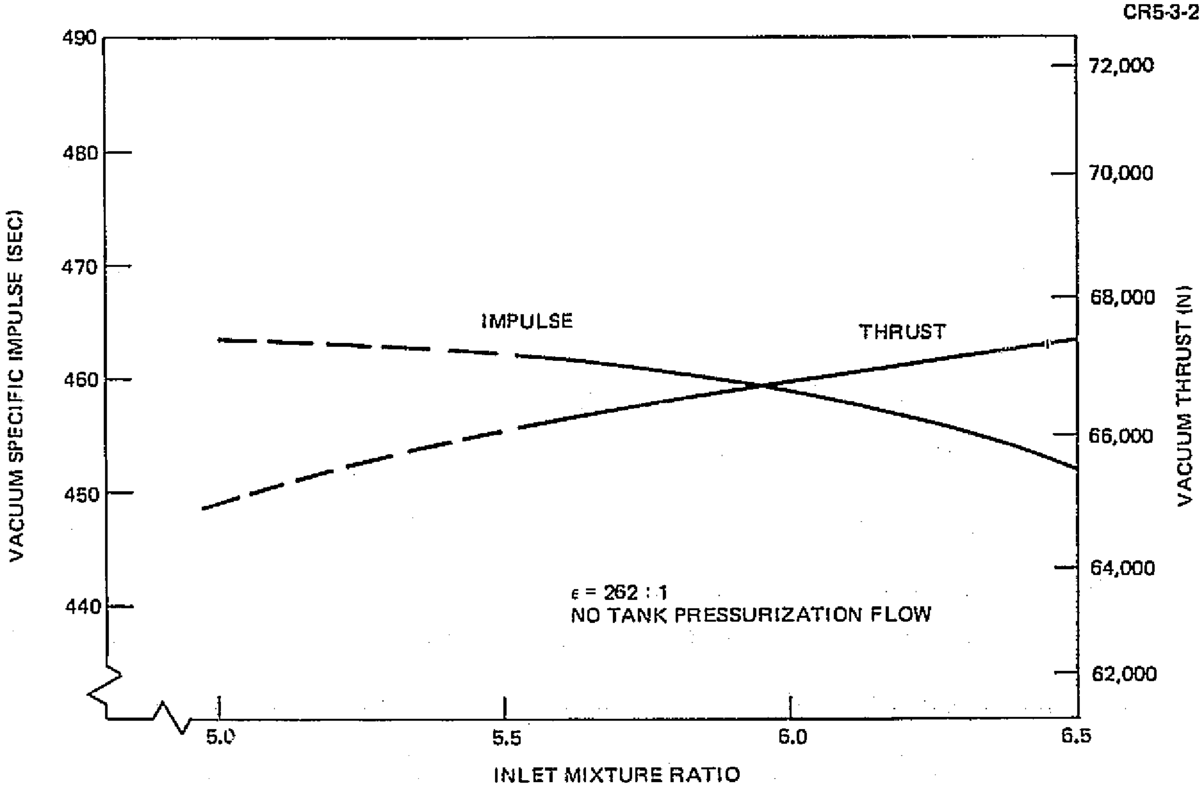


Figure 4-15. Estimated Effects of Inlet Mixture Ratio on Vacuum Specific Impulse and Thrust Derivative IIA and IIB Engines, Full Thrust

4.2.6 Engine Life

The current specification engine life for the Category IIA RL-10 is 5 hours. As was pointed out in Section 2.2, engine burn times per mission for the OTV's are somewhat lengthy, being slightly over 1 hour for the single-engine OTV-2 and just over 1/2 hour for the dual-engine OTV-1. Thus, the number of stage reuses, especially for the upper stage, would be limited based on the 5-hour engine life.

The 5-hour life of the Categroy IIA RL-10 is based on Pratt & Whitney accumulated failure-free operating time on RL-10 hardware. Therefore, it is assumed that the engine life could be extended up to a maximum of 20 hours if a test program were accomplished to demonstrate this capability. It is also assumed that the basic cost involved is that of the test program itself.

Pratt & Whitney has published program costs (1973 dollars) for four RL-10 derivative engines. The test program costs for the 5-hour life Category-IIA engine are $24.3 million and the test program for a 10-hour life Category-IV engine is $29.9 million. Assuming this difference is due to additional tests required to demonstrate the additional 5-hour life, it will cost $5.6 million (1973 dollars) to demonstrate each 5-hour life increment. Therefore, increasing Category-IIA engine specification life to 20 hours (assuming this is possible) would cost 3 X $5.6 million or $16.8 million. The Pratt & Whitney program costs did not include propellant costs because they considered the propellant to be GFE. The propellant quantities are significant since 5 hours of engine firing requires 38117 kg (84,034 lb) of LH_2 and 228,700 kg (504,200 lb) of LO_2 (mixture ratio of 6:1 and Isp of 459).

4.3 THERMAL CONTROL

A high-performance thermal control system is required for efficient LH_2 and LO_2 storage during a 60-day OTV mission, and while the OTV is being fueled in orbit. Evacuated MLI, consisting of multiple radiation barriers, has been shown to give very low, effective thermal conductivities (Reference 1) and is proposed as the basic thermal protection mechanism. Because the OTV tanks only contain cryogens in orbit, the use of a vacuum jacket or purge blanket around the MLI to allow ground-hold capability is not required. External support of the MLI is provided by a heavier face-sheet, as described in detail below.

While in orbit, the OTV tanks must be vented to prevent pressure buildup resulting from heat leak through the MLI. Because the liquid-gas interface position in the tanks is not precisely known, ordinary tank venting is not reliable, since liquid, rather than vapor, may be wastefully vented. To circumvent this problem, reliable low-g venting can be achieved by use of a thermodynamic vent system (Reference 2). This system expands vent fluid (liquid) to a lower pressure and temperature, exchanges heat with the warmer tank fluid (or intercepts the incipient heat flux) and boils the vent fluid so as to always vent vapor. This is the thermodynamic equivalent of oriented (or settled) vapor venting. The thermodynamic vent system proposed for the OTV uses a vapor-cooled shield, external to the tanks and integrated with the MLI blanket, to intercept the heat flux through the MLI. The shield is constructed of 0.40 mm (0.016-in.) thick high-conductivity aluminum sheet to which a vent flow tube is thermally connected. The shield is colder than the tank and sustains a thermal gradient to transfer the intercepted MLI heat flux to the vent fluid. This kind of vent system has been fully developed and ground tested for large LH_2 and LO_2 tanks (References 3 and 4), and its performance will be evaluated in low-g in a proposed Spacelab experiment (Reference 5).

The vent fluid is liquid, which is reliably supplied from a capillary acquisition device inside the tank, and which is expanded through a static orifice to a lower pressure and temperature. This liquid is then boiled in the shield at constant temperature, utilizing the high latent heat of vaporization of the vented liquid. Studies have shown (Reference 3) that somewhat less vent weight penalty is required if gas is vented; however, gas is not readily suppliable in low-g (while liquid is), and indeterminate gas or liquid venting would require a variable orifice (or cryogenic regulator) because of the large difference in orifice flow rate between liquid and gas. This cryogenic regulator is a potential source of unreliability and is rejected in favor of more reliable liquid venting through a static orifice. The vent rate is controlled by a regulator system which senses tank pressure, but at warmer temperatures of the vent fluid, where more reliable and accurate regulation is possible.

The vapor-cooled shield (VCS) provides a convenient support for the MLI blanket, as shown in Figure 4-16. The MLI material assumed is 0.15-mil double-aluminized mylar (although the sturdier 0.25-mil mylar could be used with about a 20% increase in MLI blanket weight) with dacron B4A net spacers, which are formed in gore sections and laid up on the VCS (supported by tooling) with the edges overlapped and taped. The heavier face sheets used top and bottom (Figure 4-16) provide support for the blanket. There are perforations in both the MLI and the VCS for depressurization of the MLI during evacuation. A heavy dacron net is placed next to the VCS to provide an outflow path during depressurization. The blankets are held to the VCS with nylon thread/buttons at the edges, and a lap joint is provided at these edges and laced up after the VCS halves are mated together. Any access openings at the top and bottom of the shield are filled with lap-joint plugs taped in place similar to the method shown in Figure 4-16. This kind of MLI system has been completely developed by MDAC (Reference 1) and has demonstrated an effective thermal conductivity of 3.507×10^{-5} W/m-°K (2.027×10^{-5} Btu/hr-ft-°R) at LH_2 temperatures at a layer density of 100 layer-pairs per inch.

For any given mission duration, the vent rate (and total vent weight penalty) decreases with thicker MLI, while the MLI weight increases with thickness. Clearly there is an optimum MLI thickness which minimizes the sum of the vent loss weight and MLI weight. For the OTV LH_2 tank, for a 60-day mission (following complete filling of the tank), the optimum MLI thickness is 4.52 cm (1.78 in.) (or 178 layer-pairs) resulting in a H_2 vent loss of 297 kg

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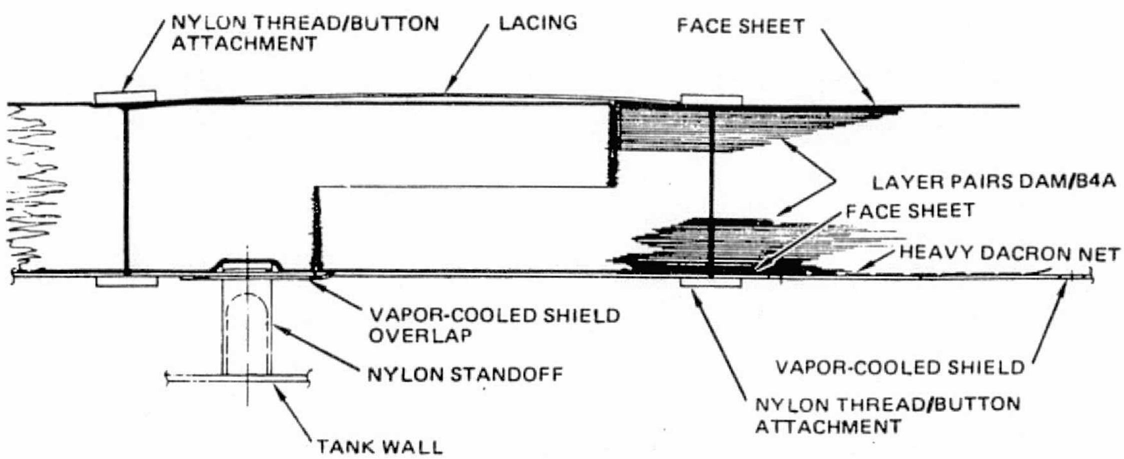

Figure 4-16. MLI Blanket and Lap Joint Construction

(654 lb) and an MLI weight of 325 kg (717 lb). Similarly, the optimum O_2 tank MLI thickness is 6.17 cm (2.43 in.) (or 243 layers-pairs) resulting in an O_2 vent loss of 180 kg (397 lb) and an MLI weight of 194 kg (428 lb). The VCS and supports weigh 172 kg (380 lb) for the LH_2 tank, and 82 kg (180 lb) for the LO_2 tank.

It must be emphasized that careful attention should be paid to the minimization of heat leak to the tanks through other sources, such as tank supports and plumbing, to achieve these optimum vent losses. The remaining heat capacity of the vent fluid may be used to cool the plumbing and supports to achieve very low conductive heat leak through these sources.

A more advanced thermal control system operational design which reduces weight penalties but adds system control complexity is to use the vented H_2 gas in the VCS around the LO_2 tank. The H_2 gas has sufficient sensible heat capacity to intercept the heat flux through the LO_2 tank MLI, thus requiring <u>no</u> venting from the LO_2 tank, and also reducing the LO_2 MLI thickness required. The H_2 vent flow, after it leaves the H_2 tank VCS, is warmed up to about 56K (100°R) (above the LO_2 freezing point), enters the LO_2 VCS and warms up from 56K to about 97K (175°R) while intercepting heat flux through the O_2 tank MLI. The O_2 MLI thickness needs to be only 1.57 cm (0.62 in.) which would weigh only 60 kg (132 lb), thus saving 180 kg (397 lb) of O_2 vent loss and 134 kg (296 lb) of MLI weight for a total weight savings of 314 kg (693 lb).

Control of this system would be more complex since two tank pressures would have to be monitored and used to adjust the vent flow. One method of control would be to bypass (as required for LO_2 tank pressure control) some of the H_2 vent flow before it enters the LO_2 tank VCS. Development of this kind of vent system should be pursued to achieve substantial OTV performance benefits.

4.4 AVIONICS

The OTV missions include transfer of both manned crew modules and unmanned experimental modules. Both stages must be able to fly autonomously; the lower stage returns to LEO following upper-stage separation, the upper stage continues to GEO, rendezvous and docks to a space base and returns. Therefore, avionics must be provided for two active stages.

Some simple guidelines for design of the OTV avionics are listed in Table 4-5. The provision for crew control of the stages plus the need to provide emergency communications in case of upper-stage abort result in a much more complex system than would be necessary for unmanned autonomous stages. The development cost of the manned capability can, of course, be reduced to a minimum by the use of Orbiter components. The use of Orbiter data bus equipment would also aid in reducing hardware modifications and would simplify mission module stage interfaces.

4.4.1 OTV Avionics Description

The stage electronics system is illustrated in Figure 4-17. Both stages require the same complement of equipment with the exception of the laser detection and ranging system (LADAR), which is only employed on the upper stage. The only differences between the stages occur in the main engine electronics (due to two engines being controlled on the lower stage and one on the upper) and in the software.

Dual computers (one on standby) transfer and receive data through multiplexer/demultiplexer units (MDM) via the input/output (I/O) unit to equipment located in the forward, intertank, and aft portions of the vehicle. Uplink and downlink data transfer is directly between the I/O and the communications system which consists of the signal processor, transponder, and RF equipment. An interface is also provided for data transfer between stage I/O's and the mission module I/O's.

Two MDM's in the forward section of the stage interface with a power control and distributor (which have been packaged as an assembly to reduce maintenance) the LADAR, and the guidance and control equipment. The intertank

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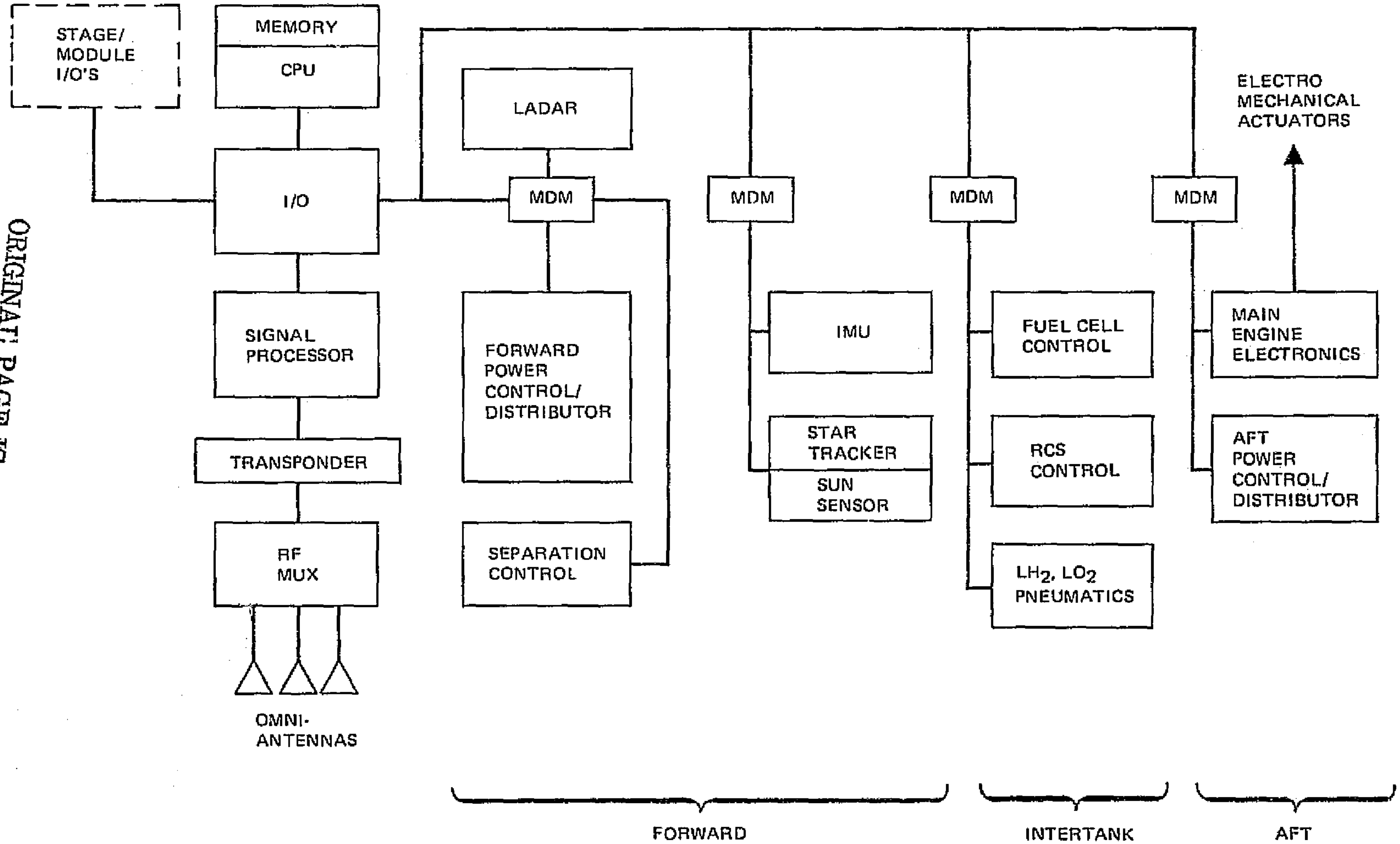


Figure 4-17. OTV Avionics Block Diagram

Table 4-5

OTV AVIONICS GUIDELINES

- Upper stage critical electronics fail operational, fail-safe as a minimum.
- Provide for manual control of upper stage and lower-stage abort.
- Provide for manual/automated control of upper stage allowing return to LEO.
- Provide for docking to uncooperative target.
- Consider on-orbit maintainability requirements and task minimization.
- Minimize DDT&E costs.
- Maximum acceleration of upper stage with payload is 0.955 g, 1.9 g without payload.
- Both stages must return to LEO by individual guidance capability.
- Provide for communications between the mission module and the tracking and data relay satellite (TDRSS) in case of upper-stage abort.

area contains redundant fuel cell systems plus control units (all of which contain multiplex interface adapters, MIA's, for interfacing with the MDM's) for reaction jet drivers and pneumatic controllers. The aft MDM interfaces the main engine electronics and aft power distributer. The MDM's also interface with signal conditioning and instrumentation electronics (not shown) in all sections of the stage.

4.4.2 Avionics Equipment Requirements

Equipment quantities, weight and power requirements are listed by subsystem in Table 4-6. In many cases, redundancy is accomplished internal to the units. This is reflected in lower number of required units with somewhat increased weight and power requirements over individual units.

4.4.3 Subsystem Tradeoffs/Recommendations

Equipment types and sources were evaluated for the various OTV avionics subsystems. Table 4-7 summarizes the candidates, trade considerations, and resulting recommendations. The following sections describe the considerations involved for each subsystem in more detail.

Table 4-6 (Page 1 of 3)

EQUIPMENT LIST

Subsystem	Quantity Upper	Quantity Lower	W/Unit (Kg)	Power/Unit (W) Stdby	Power/Unit W (Op)	Weight (Kg) Upper	Weight (Kg) Lower
Data Management							
Central Computer	2	2	22	8.6	42	44	44
System Control and Computer Interface Unit	1	1	25	50	50	25	25
Multiplexer/Demultiplexer	4	4	4	50	50	16	16
Wire Harnesses and Connectors	X	X	20	--	--	20	20
Guidance and Navigation							
IMU	2	2	25	75	180	50	50
Acceleromters	2	2	2	8	8	4	4
Star Tracker/Sun Shield	2	2	16	--	23	32	32
Rate Gyro Assembly	2	2	4	25	25	8	8
LADAR and Electronics	1	-	18	5	40	18	--
RCS Jet Driver	4	4	7	--	10	28	28
Communications							
Omni Antenna	3	3	2	--	--	6	6
RF Mux	1	1	1	--	--	1	1
Transponder	2	2	13	35	65	26	26
Network Processor	1	1	8	42	42	8	8
Microwave Equipment	X	X	3	--	--	3	3
Audio Control Unit	-	-	5	34	34	--	--
Audio Terminal Unit	-	-	2	--	2.7	--	--

Table 4-6 (Page 2 of 3)

EQUIPMENT LIST

Subsystem	Quantity Upper	Quantity Lower	W/Unit (Kg)	Power/Unit (W) Stdby	Power/Unit W (Op)	Weight (Kg) Upper	Weight (Kg) Lower
Control and Display							
Engine Electronics	1	1	13	--	50	13	13
Rotation/Translation Electronics	-	-	10	--	40	--	--
Attitude Direction Electronics	-	-	20	--	22	--	--
Keyboard/CRT	-	-	12	20	90	--	--
Display Electronics	-	-	16	207	207	--	--
Flight Control System Unit	-	-	20	40	40	--	--
Hand Controllers	-	-	10	--	5	--	--
C&W Annunciator Panel	-	-	2	10	60	--	--
C&W Electronics Unit	-	-	6	45	45	--	--
Mission Timer	-	-	1	4	4	--	--
Event Timer	-	-	1	4	4	--	--
Master Timing Unit	-	-	6	31	31	--	--
Instrumentation							
Transducer	120	120	0.5	--	--	60	60
Power Supply/Signal Conditioners	3	3	8	22	22	24	24
Power							
Fuel Cell Power Plant	2	2	52	--	--	104	104
Power Distributor/Controllers	3	3	5	30	30	15	15
Wire Harnesses	X	X	30	--	--	30	30

Table 4-6 (Page 3 of 3)

EQUIPMENT LIST

Subsystem	Quantity Upper	Quantity Lower	W/Unit (Kg)	Power/Unit (W) Stdby	Power/Unit W (Op)	Weight (Kg) Upper	Weight (Kg) Lower
System Battery	1	1	30	--	--	30	30
Accessory Weight	2	2	35	--	--	70	70
Reactant Tanks and Line Set	1	1	40	--	--	40	40
Emergency Battery	1	1	34	--	--	--	--
TOTALS						675	657

4.4.3.1 Data Management

The control computer requires a minimum 16-bit word length, a 32,000-word memory and operation rates consistent with the state of the art. The NASA standard computer meets these requirements with seven additional 4k memory units. Two computers would be used in the upper stage, two in the lower. A system control unit would monitor the output of the computers with selection of the controlling unit on the basis of error count. Manual override when carrying a manned module or remote command control in the unmanned case would also be possible with this unit.

The standard computer was selected on the basis of cost savings resulting from an expected high production rate, its compatibility with space (it is not convectively cooled, requiring an atmosphere), and its ease of growth should additional capacity be required. The Space Ultrareliable Modular Computer (SUMC) was rejected due to its development status. The AP-101 Orbiter computer's capacity was considered excessive, and it is convection cooled. The Spacelab unit was considered and rejected due to obsolescence (it is a modification of a missile computer) and the fact it is of foreign manufacture with attendant logistics and spares problems.

The computers would transfer data via a computer input/output unit incorporating the required redundancy. To reduce the number of units requiring maintenance and since off-the-shelf units cannot be employed, the data control and I/O units would be combined. Although a development unit, it would incorporate many of the elements (such as the bus controllers) avilable from the Orbiter's I/O.

The interface between the I/O and Tug systems for command and control (data bus system) would consist of Orbiter multiplexer/demultiplexer units (MDM) providing serial data and command channels as well as discrete inputs and outputs. The bus would operate at the standard 1 Mbps rate. The units would be located in forward, mid, and aft sections of each stage and the manned module. Orbiter multiplex interface adapters (MIA) would be incorporated in all interfacing systems for compatibility.

Table 4-7 (Page 1 of 3)

AVIONIC SYSTEMS RECOMMENDATIONS

Subsystem	Candidates	Trade Considerations
Data Management	SUMC Derivative	• 16-bit word computer required.
• Computers	*NASA standard spacecraft computer	• Excessive capability in Orbiter AP 101
• Interface units	*Orbiter units	• Major Spacelab system redesign; foreign manufacture.
• Bus interface units	Modified Spacelab system	• SUMC in development.
		• Orbiter MDM and bus controller elements of I/O applicable.
Guidance, Navigation, and Control	NASA standard intertial reference unit	• Subsystem used in upper stage must be failop, fail safe.
• Inertial measurement unit	*Orbiter IMU, rate gyro assembly, star trackers	• Orbiter system would require no modification
• Rate gyro set	Sperry ASLG-15 laser gyro (IMU) system	• Many components/systems available off-the-shelf.
• Star tracker/sun sensor		• System selection may be made on minimum cost.
• Control electronics unit		
Rendezvous and Docking	*Scanning LADAR	• Automatic system required for upper stage in unmanned mode; backup TV guidance.
• Rendezvous and docking	LLTV system	• Passive lower stage; Orbiter active rendezvous in LEO.

*Recommendations

Table 4-7 (Page 2 of 3)

AVIONIC SYSTEMS RECOMMENDATIONS

Subsystem	Candidates	Trade Considerations
	Orbiter rendezvous radar	• Manual alignment aids with crew module.
	*Alignment aids	• Non-cooperative target.
Communications	• *Off-shelf antennas	• Transponder must be compatible with Orbiter S-band interrogator.
• Antennas	• Phased arrays	• System must be TV bandwidth compatible.
• Transponders	• NASA standard S/C transponder	
• Power amplifiers	• *Orbiter components	
• Signal processors		
Power	• Develop solar array/battery system	• Solar arrays require retraction during burns; 2 g acceleration max at burnout.
• Power source	• *Modified Shuttle fuel cells	• Array/communications orientation conflicts.
• Controllers/distributors	• New technology fuel cells	• RTG's pose radiation hazard.
	• *Separate reactant tanks	• Fuel cell poisoning a problem using fuel tank reactants.
	• Inert flush and dilute	• Separate tanks an operations problem.
		• Battery system size prohibitive due to mission durations and manned requirements.

*Recommendations

Table 4-7 (Page 3 of 3)

AVIONIC SYSTEMS RECOMMENDATIONS

Subsystem	Candidates	Trade Considerations
Displays, Controls, Caution and Warning • Keyboard, display electronics, CRT • FCS panel, controls • Rotation/translation control electronics	• *Modified Orbiter equipment • New design	• Low cost • Mission compatible

*Recommendations

The resulting subsystem is considered to be the lowest-cost system which can be configured which meets the reliability goals of a manned spacecraft. At the same time, it uses the most advanced electronics designs which are available off-the-shelf.

4.4.3.2 Guidance, Navigation, and Control

Components of the guidance, navigation, and control system consist of the inertial measurement unit, rate gyro assembly, accelerometer assembly, star tracker, and a control electronics unit. The major requirements imposed on the system are that it meet the reliability goals of a manned flight vehicle and be capable of integration with the data management display and controls subsystems with a minimum of development effort. For this reason, the NASA standard inertial system was rejected, as were laser IMU's, although the latter might provide some reduction in maintenance requirements. The Orbiter equipment is recommended since it is compatible with the data bus system previously selected and meets all other requirements as well. Some modification to the rotation and translation control electronics unit is expected.

4.4.3.3 Rendezvous and Docking

The OTV upper stage must be capable of unmanned automatic docking; it is assumed that the lower stage will return to programmed LEO space coordinates and remain there in a passive status with active rendezvous performed by the retrieving vehicle. The only equipment known capable of automatic acquisition, tracking, alignment, closure and docking is the prototype LADAR system developed for NASA/MSFC by ITT. Since this unit has been in development for many years, DDT&E costs should be negligible. In conjunction with the LADAR, a TV system of low light level has been required in previous studies for inspection and axis alignment, and as an initial backup system until the LADAR is proven. Due to the requirements for additional television, signal processing, and transmission equipment on the OTV, plus the remote control facilities on the ground or in orbit, it is hoped that this system might be eliminated. This would only be achieved through an augmented test program.

4.4.3.4 Communications

If the TV link can be eliminated and data rates per stage held to 16 Kbps or less, link margins should be sufficient to allow the use of off-the-shelf omni-antennas, associated microwave hardware, and transponders. The crew module would, of course, require a capability for digital voice at 32 kbps in the manned mode in addition to command and telemetry channels. Although no off-the-shelf assemblies are available for performing the transponder and signal processor functions, standard components and circuit elements may be obtained from numerous sources. In addition, such equipment may be obtainable from other programs prior to OTV start since units developed for free-flying payloads and compatible with the Orbiter payload interrogator should be usable for the crew module application.

4.4.3.5 Displays, Control, Caution and Warning

In the manned mode, the support of a crew module by the OTV will entail providing the crew a manual control capability for the upper stage and an abort capability for both stages. The possibility of an upper-stage abort requires that elements of the data management and communications system sufficient to sustain the crew until rescue also be provided.

Equipment for display and control, which would include a keyboard, display electronics, CRT, and a flight control panel, are all available from the Orbiter. Modification of the latter and rotation/translation electronics for the hand controllers would probably be extensive.

4.4.3.6 Electrical Power Subsystem

Of the existing systems available to provide power, RTG's have been ruled out due to small power output and radiation hazards. Solar arrays were eliminated due to maximum acceleration g loads approaching 2 which would require retraction during burns, with subsequent extension. This would impose development costs which might become large to produce the required reliability and lifetimes with arrays capable of repeated cycling.

Remaining conventional systems include pure battery and fuel cell derivatives. Battery weight is considered to be prohibitive as a result of power levels and mission duration. The candidates for fuel cell systems include a modification of units designed for the Orbiter and a new design which will operate at low pressure 10.3 N/cm^2 (15 psia), and use fuel tank reactants directly. The latter approach is not recommended at this time. A low-pressure system would be a new development, and would require an accumulator, pumps, etc. Further, propellant purity might well be a problem.

Therefore, the recommended system would employ Shuttle-derived fuel cells and use sets of Shuttle tanks. Each fuel cell module could then be designed as a replaceable module, with only electrical connections, and no fluid or gas line connections.

Based upon the power profile shown in Figure 4-18, a system capable of supplying an average power of 770W and peak power of 1,200W is required for each stage.

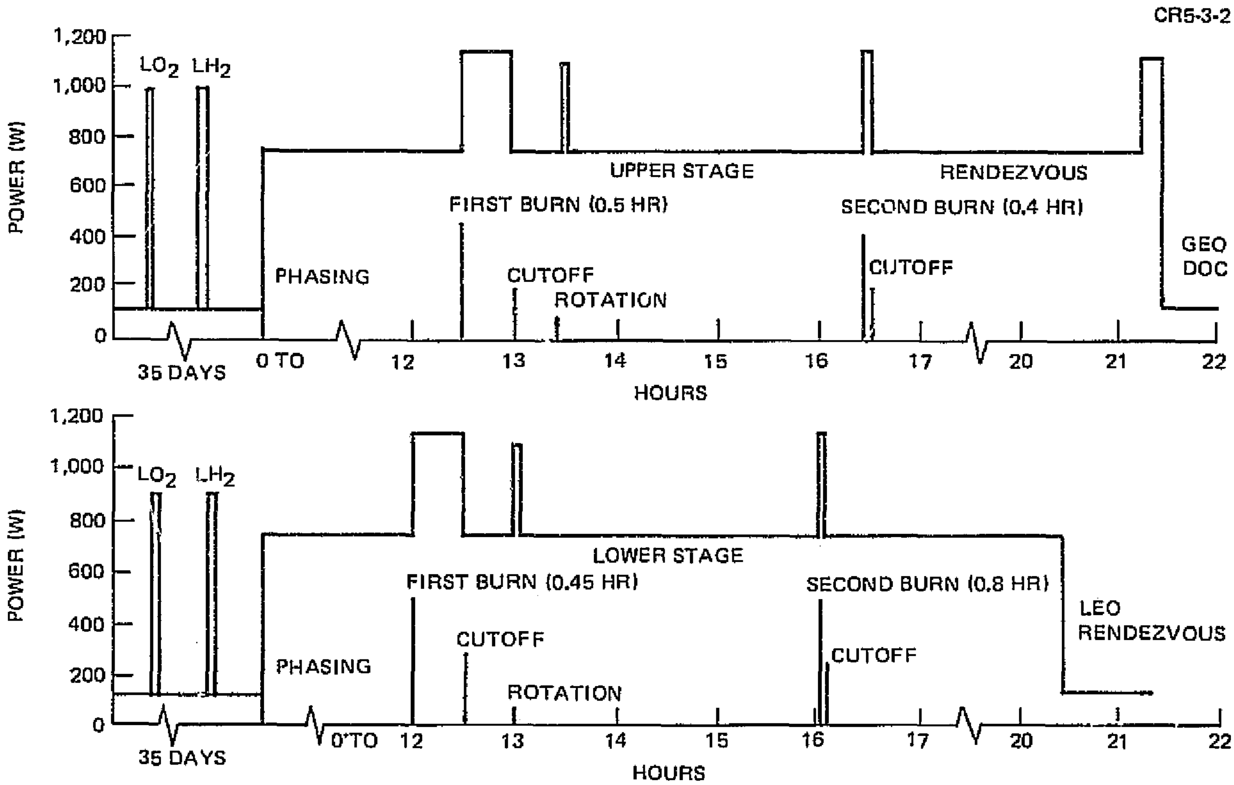


Figure 4-18. OTV Power Profile

Section 5
MASS PROPERTIES

The OTV weights were generated using an MDAC computer program called DAKTUG. The program uses an external data file for geometry and mission constraints and a series of operator-prompted options for subsystem selections. The program sizes tankage and support subsystems based on input options, and then integrates various subroutines to define the resulting geometry, areas, volumes, and detail weights.

The summary of the detail printout from DAKTUG for the booster option is contained in Mass Properties, Part 6 of this Volume. Table 5-1 is a summary of the OTV booster mass plus the upper stage as comparison. The primary assumptions are a 30-day mission, 770W for OTV plus 300W for payload, total APS impulse of 960,811 N-sec (216,000 lb-sec) and a useable propellant of 57,206 kg (126,118 lb).

The primary difference between the two stages in Table 5-1 is in the propulsion section - quantity of engines, lines, pneumatics, umbilicals, actuators, etc. The basic structure was assumed to be the same for this iteration with the exception of the thrust structures. In the avionics the difference is less instrumentation/wiring with less engines and only one TVC Battery with the single engine. Trapped propellants differences result from the use of two sets of lines for the booster.

The majority of subsystem weights and interrelationships between subsystems were developed during the Phase-B Cryogenic Tug Study, resulting in more detail than the current level of definition. A limit of 10% was used for contingency. In Figure 5-1, preliminary estimates of the fully loaded OTV booster are presented. The λ' for each stage is based on total expendables and is 0.9205 for the booster and 0.9290 for the upper stage.

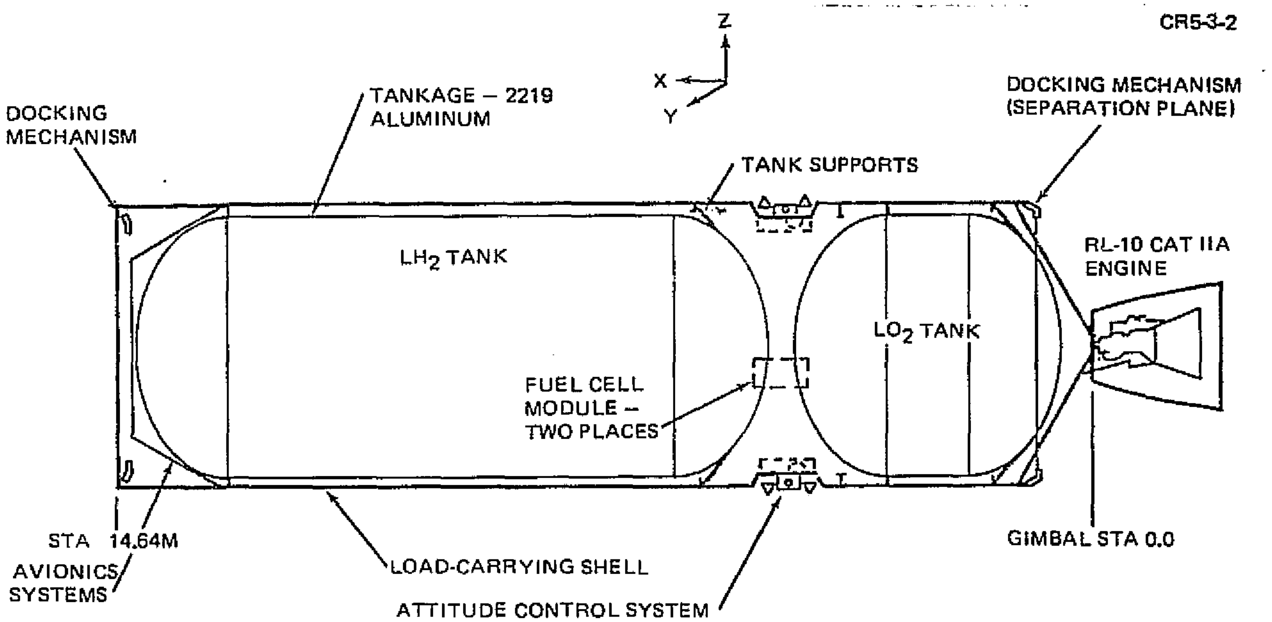
Table 5-1 (Page 1 of 2)

OTV MASS SUMMARY

Description	Stage Mass (Kg) Booster		Upper	
Structure	1,662		1,587	
Fuel Tank and Supports		438		438
Lox Tank and Supports		205		205
Body Structure		744		744
Thrust Structure		152		77
Meteoroid Shield		20		20
Payload Interface		103		103
Thermal Control	478		478	
Avionics	692		677	
Data Management		113		113
GNC		36		36
Communication		69		69
Instrumentation		125		122
Electrical Power Source		225		215
Power Distribution and Control		51		50
Equipment Thermal Control		73		72
Propulsion	1,041		655	
Engines		432		216
Support		518		348
ALPS		91		91
Dry Weight	(3,873) (8538 lb)		(3,397) (7489 lb)	
Contingency	387		340	
Total Dry Weight	(4,260) (9392 lb)		(3,737) (8239 lb)	
Residuals	781		725	
FPR		173		173
PU		145		145
Pressurization (GO_2/GH_2)		329		329
Trapped		134		78

Table 5-1 (Page 2 of 2)
OTV MASS SUMMARY

Description	Stage Mass (Kg) Booster	Upper
Burnout	(5,041) (11,113 lb)	(4,462) (9837 lb)
Inflight Losses	58,383	58,383
APS Maximum Capacity	359	359
Vent Propellant	409	409
Idle Propellant	154	154
Fuel Cell Reactant	255	255
Usable	57,206	57,206
Ignition	(63,424) (139,825 lb)	(62,845) (138,548 lb)



	CENTER OF GRAVITY			MOMENT OF INERTIA		
MASS	X	Y	Z	ROLL	YAW	PITCH
63,424 KG (139,822 LBM)	6.6M	0.0	0.0	10.6	147.0	147.4

$KG \cdot M^2 \cdot 10^3$

Figure 5-1. Fueled OTV Mass Properties

Section 6
SPECIAL STUDIES

6.1 TANKER STUDIES

6.1.1 Basic Tanker Concept

Since the OTV is a space-based concept, fueling and refueling will take place in low earth orbit. A rather substantial quantity of propellants will be required to load both stages, and a number of Shuttle flights will be necessary to complete the task (assuming, of course, that the Shuttle vehicle will serve as the tanker).

For nominal size OTV with a capacity of 58,550 kg (129,081 lb) perstage, a total of 100,372 kg (221,282 lb) of liquid oxygen and 16,728 kg (36,880 lb) of liquid hydrogen must be transported to orbit. These numbers include usable propellant, losses, boiloff, residuals, etc. Further, initial chilldown of the tanks may very well require up to a week or more, and substantial propellant losses. Therefore, it is important that the tanks be kept chilled once thoroughly chilled so that these losses are not incurred when refueling for subsequent missions.

Two basic tanker concepts may be considered, one which carries the propellants separately and one that carries both at the same time in the nominal 6:1 mixture ratio. The number of flights necessary to transport the propellants is the same in either case. For separate tankers, assuming a Shuttle capacity of 27,215 kg (60,000 lb), it would take about 3.5 loads of LO_2 and less than a full-capacity load of LH_2, or a total of five flights. For combined propellant transfer, i.e., a Shuttle tanker load of 3,888 kg (8,571 lb) of LH_2 and 23,327 kg (51,429 lb) of LO_2, it would take over four tankers full, or again, five flights. Tank volume required for each of these two cases, assuming a 2% ullage for each tank, is shown on Figure 6-1. As can be seen from this figure, a separate LO_2 tanker makes very poor use of the cargo bay volume, while take up very nearly all usable bay volume. On the other

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hand, the combined propellant tanker mades good use of the bay volume and would easily accommodate whatever docking means may be necessary to allow OTV dock to the Orbiter for propellant transfer. In addition, the combined propellant tanker permits a single tanker configuration, rather than two, which would be simpler, since the Shuttle would not have to be reconfigured during OTV loading.

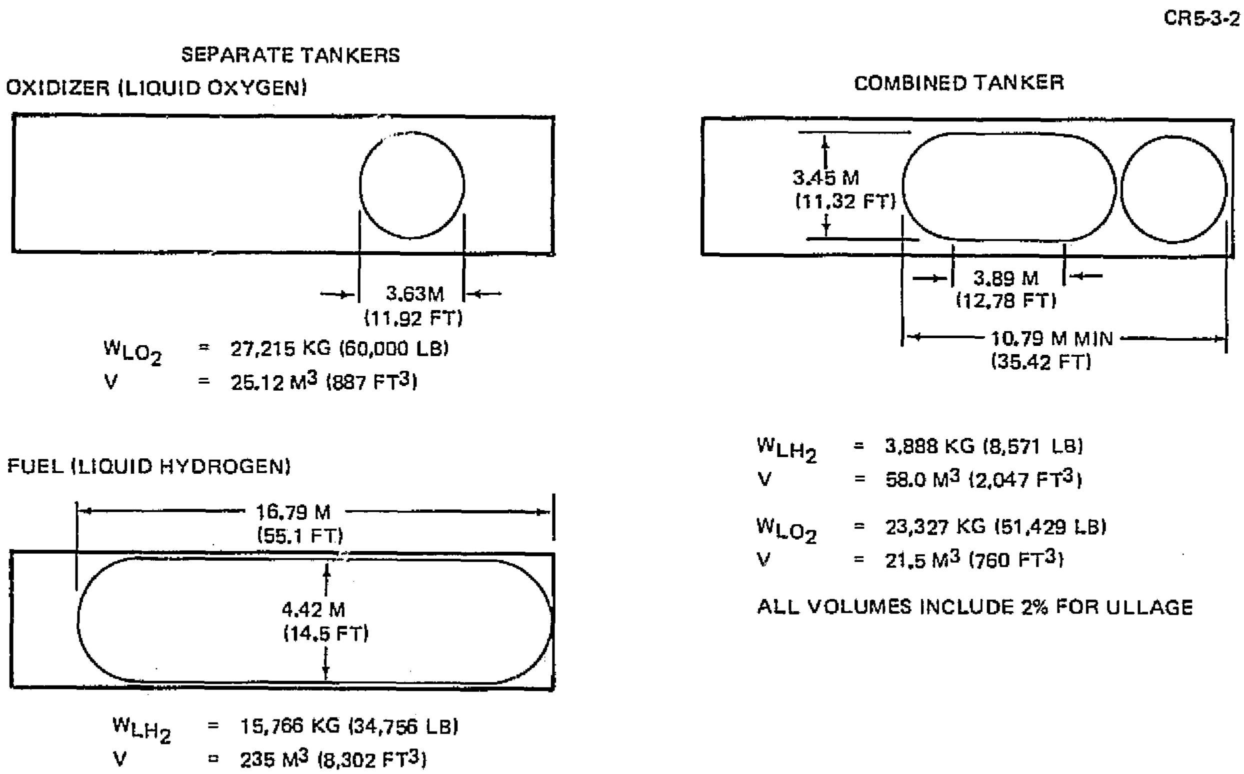

Figure 6-1. Shuttle Tanker Concepts

Other considerations favor the combined tanker, such as simultaneous chilldown. Initial chilldown will take some time since the OTV is a high-performance system. The first tanker may be lost for all practical purposes due to the large amounts of boiloff. Hence, one trip would conceivably be saved. It is also possible that there is some structural advantage to simultaneous tank chilldown, i.e., considering contractions, deflections, etc., of both tanks. Finally, the two-propellant tanker permits replenishment, or top-off, of both tanks, if necessary. In the event there was some lengthy mission delay, and replenishment should become necessary, tank top-off could be achieved with a single tanker.

In summary, the two-propellant tanker was selected for OTV fueling for the following reasons:

- Better cargo bay use.
- Single tanker configuration.
- Simultaneous tank chilldown.
- Simultaneous tank replenishment.

6.1.2 Tanker-OTV Positioning

Assuming a two-propellant (combined) tanker, the various possible tanker-OTV positions for propellant transfer had to be addressed. Some of these arrangements are shown in Figure 6-2.

The simplest, most straightforward approach seems to be that shown by Part A of the figure. In this arrangement, the tanker is stationary in the bay, and each OTV docks to some mechanism in the tanker for the transfer. If the mechanism were to be a full 4.57m (15 ft) in diameter, then it would probably be extendable in order to move the interface out of the cargo bay interior to a plane even with the Shuttle exterior surface as shown. It is also assumed that docking will be at the forward end of the OTV, as there will be docking mechanisms located there. In addition, there would be no structure at the aft end of the stage around in engine or engines. In this case, there would be a common front interface on the two OTV stages.

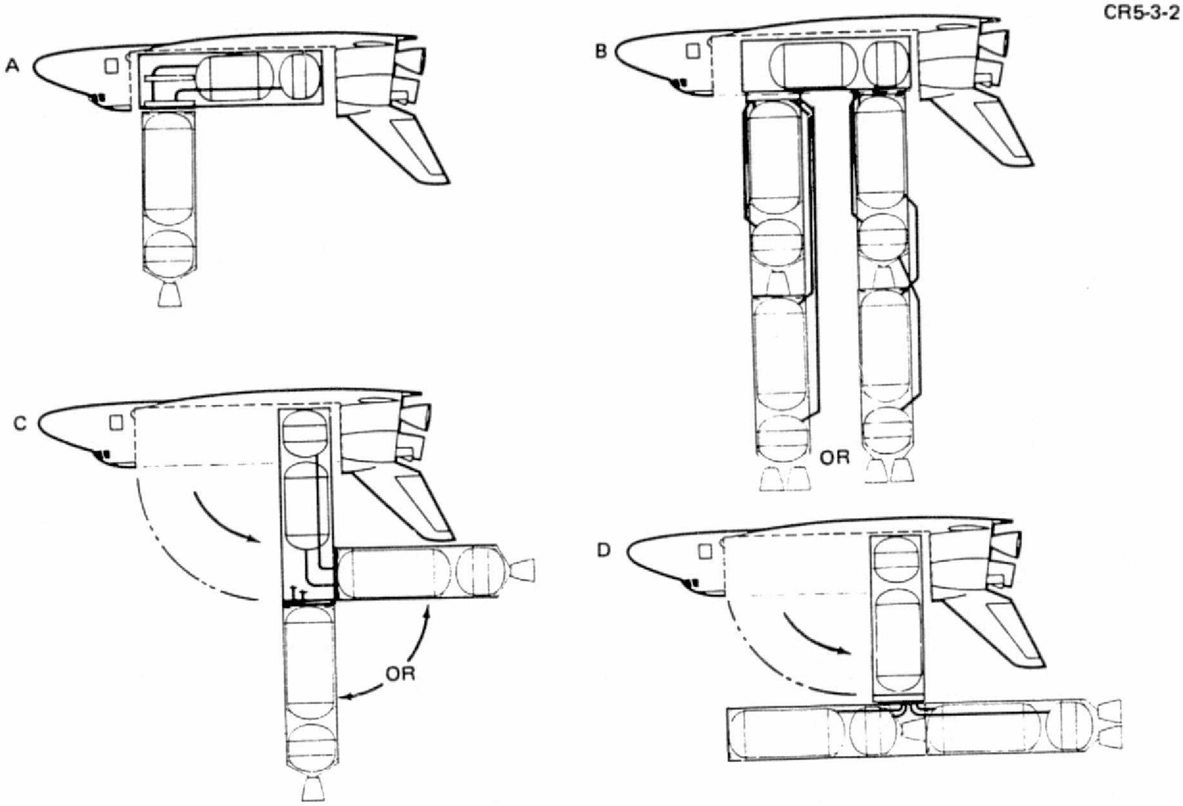

Figure 6-2. Tanker-OTV Positions

The tanker could possibly be a swingout arrangement as shown in (C) or (D), but that would add complexity. There would appear to be no advantages to (C); the arrangement in (D) would provide for simultaneous chilldown, fueling and topping, but would require double propellant transfer arrangements in the interstage and a quick disconnect/remateable set of propellant lines for subsequent stage separation.

It would also be possible to fuel the OTV in the assembled configuration, as shown in Section (B). In this case, in order to get propellant to the lower stage (OTV-1), there would have to be propellant lines running all the way down the outside, or lines between tanks, such that all propellants passed through the forward tanks. In either case, a lot of extra plumbing would be required, and the separation plane would have to provide for disconnect and remating of propellant lines.

Thus, it was concluded that the single-OTV loading arrangement was most advantageous. The two stages will be docked together following loading, and prepared for the mission. Stage capability, in terms of rendezvous and docking and attitude control will be available for the vehicle assembly.

6.1.3 Tanker Design

6.1.3.1 Acquisition System

The basic propellant transfer system concept is a passive system using distributed screen acquisition channels with gaseous helium pressurization. The performance and design of this kind of system has been studied previously in Reference 2. The overall layout of the channel system is shown in Figure 6-3 for the LO_2 tank. The LH_2 system is similar in concept. The acquisition system is arranged in four arms distributed to be in contact with the bulk of the liquid in the tank. The channels are against the wall because cryogenic propellants are wetting liquids and are wall-bound in low gravity.

The basic transfer time, transferring both LH_2 and LO_2 simultaneously, is 20 hours. The channel arms are rectangular in cross-section and are oversized to provide conservatively low pressure drop during transfer. The channels are 15.3 x 2.5 cm (6 x 1 in.) for the LO_2 tank and 15.3 x 5.1 cm (6 x 2 in.) for the LH_2 tank (which provides a retention margin during outflow of at least a factor of 10.) The residual fluid trapped in the channels at the end of transfer is less than 0.5%. There is a low probability that a puddle of liquid could be trapped between the channel arms due to random accelerations near the end of tank draining. For this reason, the channels are aligned with the axes of the Shuttle/OTV so that RCS accelerations would tend to position the puddle over the channels. In the worst case, the puddle would only amount to about 1.8%. Both the channels and the tank are made from aluminum, to be compatible with the aluminum vapor-cooled shield (described below) and for high strength with light weight. The channels are held snugly against the tank wall by epoxy-fiberglass compression supports, as shown in Figure 6-3. To accommodate the differential contraction during chilldown between the fiberglass supports and the aluminum tank/channels, belleville springs are used at the channel (see Figure 6-4). When the tank is pressurized, the tank strain overcomes nearly all of the contraction effect, and this is also accommodated by the belleville springs. These supports are adjusted during assembly of the tank-screen assembly to provide a good fit. The central support members also support the

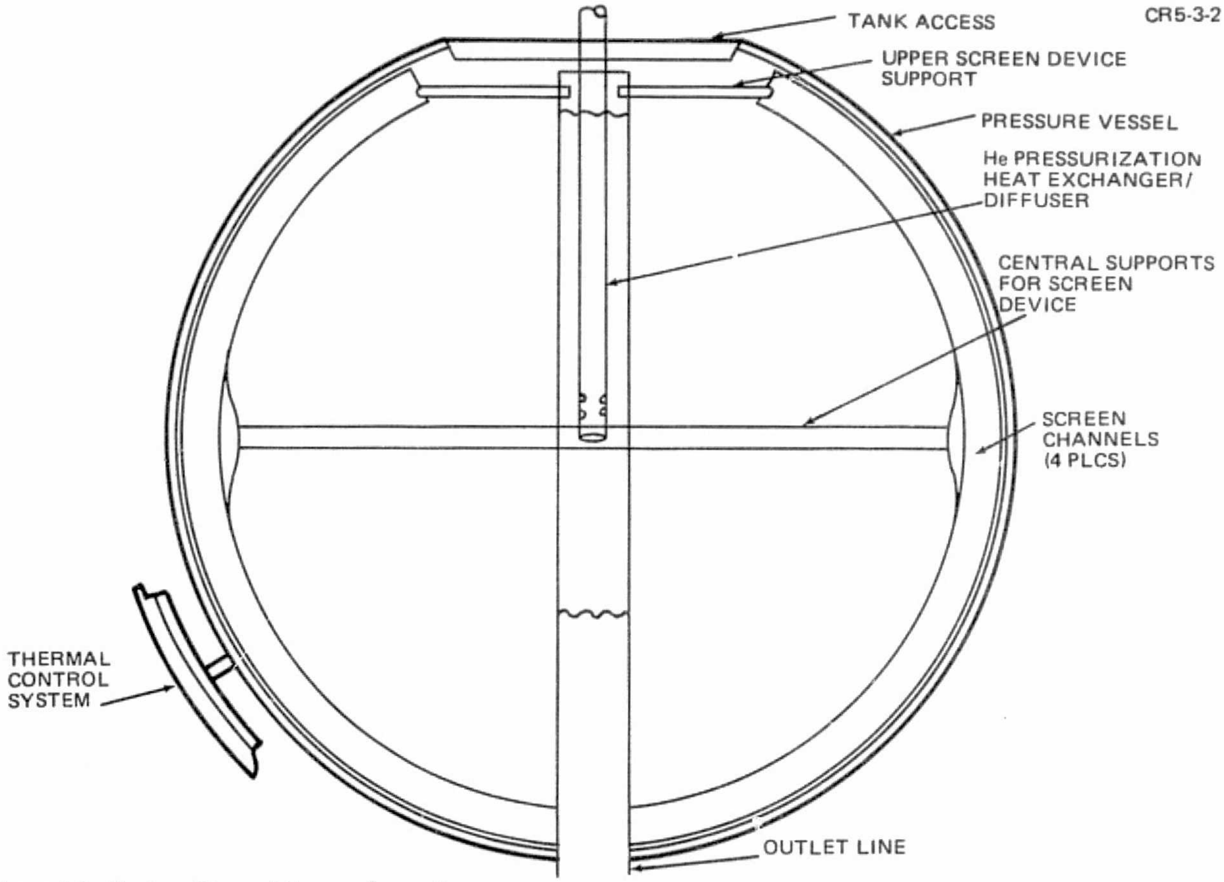

Figure 6-3. Tanker Channel System Layout

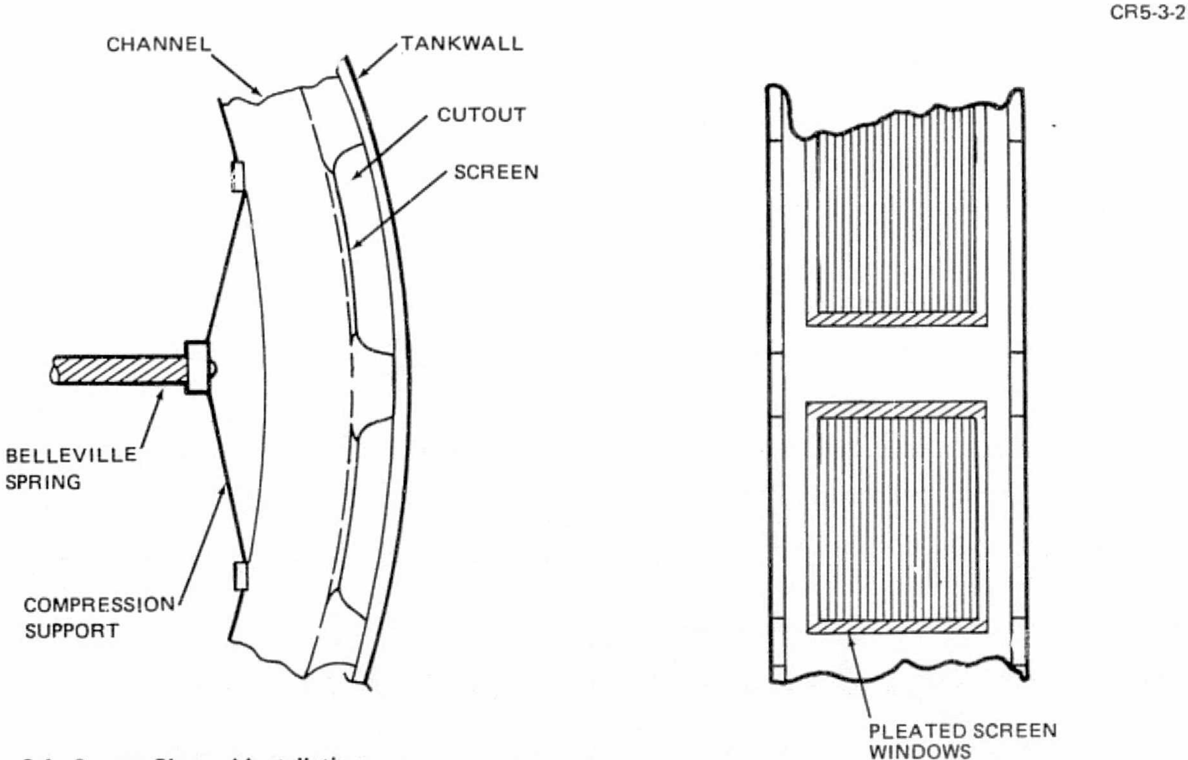

Figure 6-4. Screen Channel Installation

pressurization heat exchanger/diffuser pipe (described below) and provide a poor heat conduction path from the diffuser pipe to the channels. The top support members do not contact the diffuser pipe.

As shown in Figure 6-4, the screen is held off the tank wall and cutouts provide a flow path for the wall-bound liquid. The screen material should be the finest mesh conveniently available to provide maximum retention capability against docking and RCS accelerations. The finest aluminum screen available is 200 x 1,400 mesh, which is very costly and difficult to obtain, but which could be seam-welded to the aluminum channels with no concern for adverse differential contraction effects during chilldown. On the other-hand, stainless steel screen is readily available down to a mesh of 325 x 2,300 (which has about 50% more retention capability than the 200 x 1,400 mesh). In order to attach stainless steel screen to the aluminum channel, the screen is first seam-welded to an aluminum foil window frame, which in turn is TIG-welded to the channel. This fabrication method is used in the Shuttle OMS screen device construction where stainless steel screen is welded to titanium channels (Reference 6). The small differential contraction (~0.013 cm/0.005 in.), which occurs between the aluminum channel and stainless steel screen during chilldown, would tend to loosen the screen. In order to accommodate this contraction, provide rigidity to the screen system, and provide extra flow area, the screen is fabricated in 12.7 x 12.7 cm (5 x 5 in.) windows, and pleated as shown in Figure 6-4. This is the general fabrication method used for the OMS screen device, and it has proved to be resistant to vibration, transient flow surges, and sloshing.

6.1.3.2 Pressurization System

The pressurization system to be used to transfer the LH_2 and LO_2 to the OTV uses gaseous helium, stored at high pressure at ambient conditions (~250 K), but uses cold in the LH_2 and LO_2 tanks by being chilled through an in-tank heat exchanger/diffuser (see Figure 6-3). This method uses the helium sensible heat to vaporize H_2 and O_2 in the tanks, which contributes to tank pressurization and reduces helium requirements. The helium requirements are 42.2 kg (93 lb) for the LH_2 tank and 13.6 kg (30 lb) for the LO_2 tank, which, along with 5.4 kg (12 lb) residual, will be stored at

C-3

2,760 N/cm^2 (4,000 psi) in a titanium-lined, fiberglass-wound storage sphere (similar to those being developed for the Shuttle RCS) weighing 295 kg (650 lb). The major uncertainty with this system is the low-gravity behavior of the cryogen surrounding the helium heat exchanger/diffuser. However, this type of pressurization system will be flight-verified with a Spacelab experiment (Reference 5).

A more advanced type of pressurization system, which would eliminate most of the helium and the heavy storage sphere, is to use the OTV tank vapor return to the tanker tank with a vapor pump to provide pressurization as described in Reference 7. The disadvantage of this method is that it depends on the currently unknown chilldown and vapor generation characteristics of the OTV (especially for the initial filling), and thus the tanker is not independently operable, which could impose mission limitations. However, the concept has the advantages of lower weight, minimal helium solubility concerns, and requires no large quantities of high-pressure helium in the Shuttle bay, and may be worth further investigation.

6.1.3.3 Transfer System

Because of the relatively long transfer times available, the volumetric flow requirements are low and the transfer lines can be quite small, on the order of 2.5 cm (1.0 in.) diameter for both the LH_2 and LO_2 systems. It is most important that zero heat leak be transmitted back through the transfer lines to the screen channels, since vapor generation inside the channels could lead to retention loss. It is recommended that the transfer lines be vacuum-jacketed and kept wet up to the transfer valves by active cooling of the transfer lines using the H_2 (and O_2 if applicable) vent fluid. Preliminary analysis indicates that the vent fluid should have sufficient heat capacity to accomplish this, as described below in detail.

6.1.3.4 Tanks and Support Structure

The tanks used for storage of the LH_2 and LO_2 on the tanker are fabricated of high-strength 2219 aluminum with a wall thickness of 0.089 cm (0.035 in.) for the spherical portions of the tanks, and 0.178 cm (0.070 in.) for the cylindrical section of the LH_2 tank, based on a maximum design tank pressure of 17.2 N/cm^2 (25 psia) for both tanks. The LH_2 tank weighs about 330 kg (730 lb) and the LO_2 tank about 100 kg (225 lb). The tank and structure arrangement is shown in Figure 6-5. The tanks are supported from a shroud with high-strength, low-conductivity supports made of S-glass-filament-wound composite tubes, and assumed to be 1.27 cm (0.5 in.) diameter by 0.05 cm (0.02 in.) wall for the LH_2 tank, and 1.27 cm (0.5 in.) by 0.1 cm (0.04 in.) wall for the LO_2 tank. There are 24 supports 1.22m (48 in.) long and 8 supports 0.81m (32 in.) long for the LH_2 tank, and 24 supports 1.07m (42 in.) long for the LO_2 tank. The shroud structure supports the tanks and thermal control system and provides support for the OTV mating adapter/docking ring and attachment of the tanker to the Shuttle bay. The shroud structure and docking ring are similar to those studied previously in Reference 7, and weigh 640 kg (1,412 lb) and 354 kg (780 lb), respectively.

6.1.3.5 Thermal Control System

The thermal control system design utilized for the tanker is identical to that proposed for the OTV, except that the MLI system is optimized for a 7-day mission and the MLI is enclosed in purge bags and purged with helium to allow for ground loading and hold time of the LH_2 and LO_2. The MLI thickness for the LH_2 tank is 1.55 cm (0.61 in.) resulting in a vent loss of 58 kg (129 lb) and an MLI blanket weight of 78 kg (171 lb). For the LO_2 tank, the optimum MLI thickness is 2.11 cm (0.83 in.), resulting in a vent loss of 38 kg (84 lb) and an MLI weight of 47 kg (103 lb). The LH_2 and LO_2 VCS's weigh 100 kg (220 lb) and 47 kg (104 lb), respectively. Again, as with the OTV, the H_2 vent gas could be used in the LO_2 tank VCS to keep the LO_2 tank vent-free and reduce the MLI requirements. However, the weight savings for this short mission are so minimal that they may not be worth the added complexity. If the H_2 vent gas were used in this capacity, it could be warmed up to 56K (100°R) by cooling the H_2 transfer line before entering the LO_2 VCS, and then used to cool the O_2 transfer line after it left the shield.

While in the atmosphere, the MLI blankets must be purged with helium to prevent cryopumping of air and moisture into the MLI. While on the launch pad, the MLI would be purged with GSE-supplied helium, but during launch and reentry, the purge gas must be supplied from onboard. It is estimated that 6 kg (13 lb) of helium, stored in a 28 kg (62 lb) storage sphere would be required. The purge gas would be introduced between the VCS's and the tanks, and both the shields and the MLI are perforated to allow permeation of the purge gas and depressurization of the system during evacuation in space.

6.1.3.6 Overall Tanker Configuration

The overall arrangement of the tanker was shown in Figure 6-5. The heavy LO_2 tank and helium spheres are situated aft, and the docking ring and LH_2 tank forward, in order to ensure that the tanker C.G. falls within the required envelope for the maximum Shuttle payload of 29,484 kg (65,000 lb). The weight breakdown for the tanker components is shown in Table 6-1.

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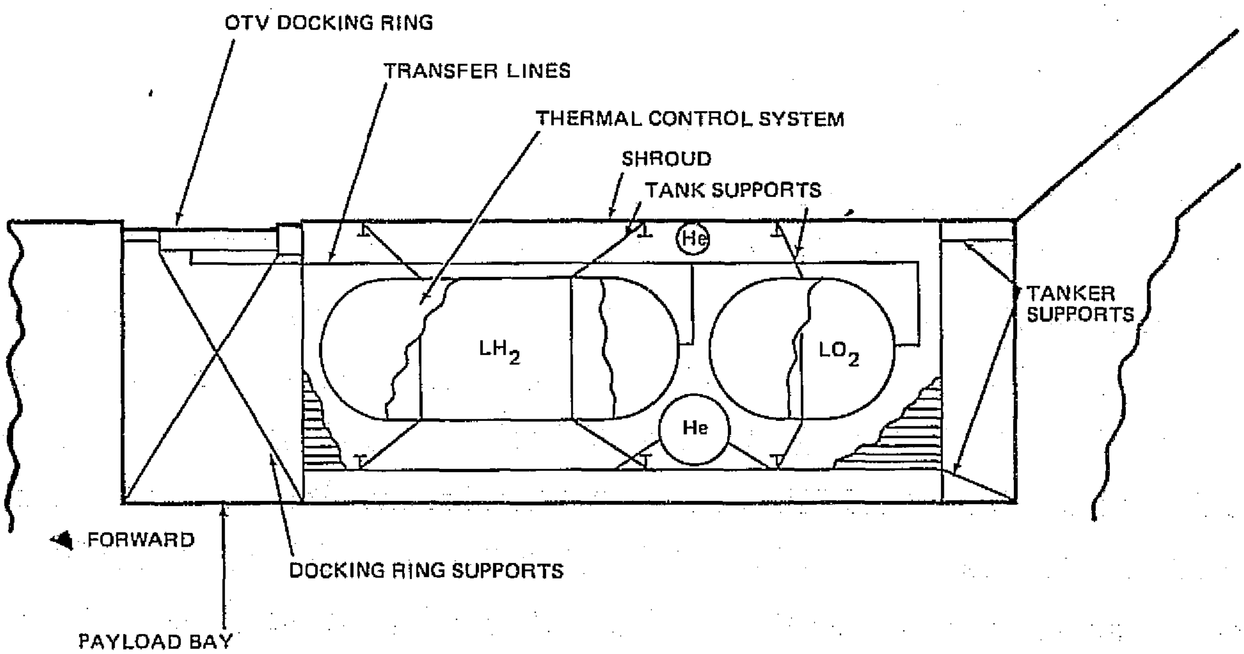

Figure 6-5. Overall Tanker Arrangement

Table 6-1

TANKER WEIGHT ESTIMATE

Item	kg		lb	
Structural Shroud	640		1,412	
Docking Ring	354		780	
Tanks	444		979	
LH_2	331		730	
LO_2	102		225	
Supports	11		24	
Pressurization System	356		785	
He Sphere	295		650	
He	61		135	
Screen Channels	48		106	
LH_2	31		68	
LO_2	15		34	
Supports	2		4	
Transfer/Fill Lines, Components	92		203	
Thermal Control System	334		735	
LH_2 MLI	78		171	
LH_2 Vapor-Cooled Shield	100		220	
LO_2 MLI	47		103	
LO_2 Vapor-Cooled Shield	47		104	
Purge System				
He Sphere	28		62	
He	6		13	
Bags, Components	28		62	
Dry Weight		(2,268)		(5,000)
Propellant		(3888)		(8571)
Delivered LH_2	3,709		8,176	
Vented H_2 (7 days)	58		129	
Residual GH_2	102		224	
Residual LH_2	19		42	
		(23,328)		(51,429)
Delivered LO_2	23,143		51,020	
Vented O_2 (7 days)	38		84	
Residual GO_2	44		97	
Residual LO_2	103		228	
TOTAL		29,484		65,000

6.2 MISSION TIMELINE

A typical OTV mission timeline is shown in Figure 6-6. It is assumed that the Space Shuttle is the supporting vehicle, and that two are available. Turnaround time is assumed to be 14 days, and the two vehicles can be launched 2 days apart. Shuttle flights are represented by the triangles on the figure, with the apex indicating flight direction (up or down) and the day of the flight shown by the number by the apex.

The establishment of the OTV in LEO (space-based) is shown on the left of the chart. Two Shuttle flights, one each for the first and second stage OTV's (OTV 1 and OTV 2), will be required. This is a one-time operation, and need not be repeated until such time as the OTV is returned to earth.

The mission proper then starts at Day 16 with the first Shuttle tanker flight. As was pointed out in Section 6.1, a total of five Shuttle tanker flights will be required to complete OTV fueling. The sixth Shuttle flight is designated to bring payload, or cargo. This may be a manned crew module for a GEO sortie mission, or merely some cargo – a lab, structure, etc. – for pure delivery to GEO. The OTV flight to GEO is shown starting on Day 50.

For a pure delivery mission, the OTV could return on the next day. However, there would be no rush, since the next Shuttle tanker would not be available until Day 61 (shown by the dashed lines in Figure 6-6). Therefore, the total minimum mission time, from start of fueling to start of fueling, would be 45 days.

In the case of a sortie mission, the OTV may remain in GEO for 30 days, not initiating return flight until Day 80. At that time, a Shuttle flight would also have to be launched in order to meet the OTV upon return to LEO and subsequently transport the manned crew module back to earth. In that event, fueling for the next mission would have to start on Day 82 with a second Shuttle flight. The mission time, then, for the sortie mode, would be a minimum of 66 days. If a useful cargo could have been brought up on the Day 80 flight, the mission cycle of 66 days could be maintained. If, however, the full six flights (five tankers, one payload) are necessary, starting with Shuttle No. 2 on Day 82, then the mission time would be increased another 12 days due to the final Shuttle flight for payload.

The number of OTV flights per year follows based on these data. The 45-day mission could be repeated eight times in a year, while only five full sortie missions could be accomplished. If more flights than that are required, there would have to be an additional OTV in space. In that event, more fuel in a shorter time would be required, necessitating either additional Shuttle vehicles and launch pads or greater capacity delivery vehicles.

Also to be considered is engine burn time. For eight flights per year, and mission burn times of 0.56 hr for OTV-1 and 1.12 hr for OTV-2 (Section 2.2), the total engine burn time in a year would be 4.5 hr for the OTV-1 engines and 9 hr for the OTV-2 engine. As pointed out in Section 4.2.6, the current specification engine life for the category IIA RL-10 is 5 hr. Clearly, this engine life should be increased if a frequent and costly return trip to earth for engine replacement were to be avoided.

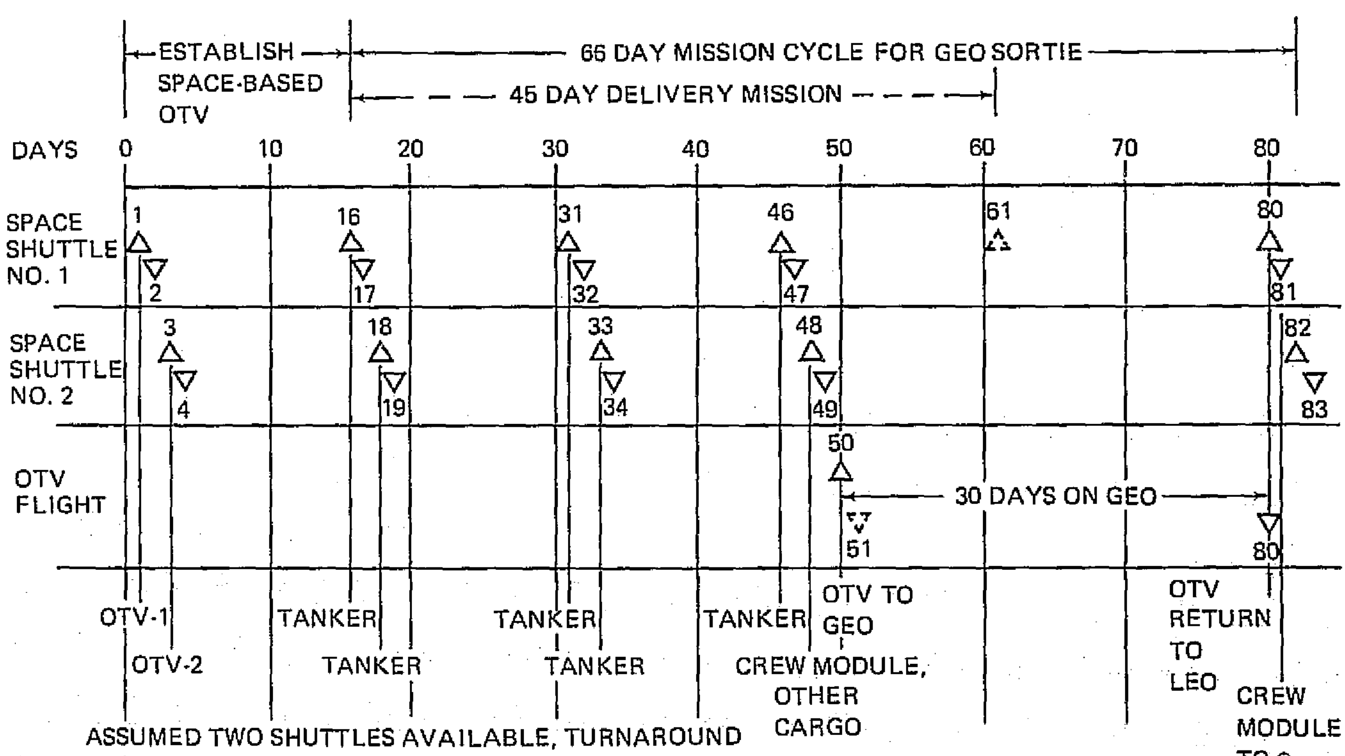

Figure 6-6. OTV Mission Timeline

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Section 7
OTV COST DATA

7.1 TOTAL PROGRAM COSTS

DDT&E costs, production costs, and an average operational cost per flight have been determined for the two-stage orbit transfer vehicle (OTV), the tanker vehicle, and all other supporting effort normally required to complete a total propulsive stage program. Total program DDT&E is $378.7M and total program production is $154.9M, resulting in a total program cost of $533.6M. The DDT&E cost includes $213.2M for OTV design and development, $37.1M for tanker vehicle design and development, and $128.4M to provide for project management, system engineering and integration, system test, logistics, facilities, and ground support equipment. The production cost includes $95.7M for 3 two-stage OTV's, $9.6M for 2 tanker vehicles, and $49.6M to provide for project management, sustaining engineering, and initial spares during a 4-year production time span. These costs are summarized by major system element in Table 7-1.

7.2 OTV COSTS

The OTV costs of $213.2M DDT&E, $95.7M production, and $308.9M total are summarized by subsystem in Table 7-2. The average operational cost per flight at a rate of 8 per year is $98.1M, of which $95.5M is for 5 shuttle flights required to transport propellant to LEO with the remaining $2.6M for ground operations, replacement parts, and propellants. It is assumed that the activities involved in ground operations and replacement of parts would be comparable to similar activities defined in the 1973 Space Tug System Study. All costs are expressed in mid-fiscal year 1977 dollars excluding prime contractor fee.

7.3 COSTING APPROACH

Reference cost data for the orbital vehicle subsystems, the tanker vehicle and all supporting cost elements were obtained from the 1973 Space Tug Systems Study (Cryogenic), Volume 8 Programmatics and Cost, Book 2

Table 7-1

OTV TOTAL PROGRAM COST SUMMARY
(Millions of 1977 Dollars)

	DDT&E	Production	Total
Project Management	12.2	4.8	17.0
System Engineering and Integration	31.5	40.2	71.7
Orbital Vehicle	213.2	95.7	308.9
Tanker Vehicle	37.1	9.6	46.7
Initial Spares	-	4.6	4.6
System Test and Evaluation	37.5	-	37.5
Logistics	3.9	-	3.9
Facilities	5.9	-	5.9
Ground Support Equipment	37.4	-	37.4
Total Program	378.7	154.9	533.6

Table 7-2

OTV COSTS BY SUBSYSTEM FOR BOTH STAGES
(Millions of 1977 Dollars)

	DDT&E	Production*	Total
Structure	28.1	11.2	39.3
Thermal Control	3.4	4.5	7.9
Avionics	25.4	38.8	64.2
Propulsion	136.1	26.9	163.0
Final Assembly and Checkout	20.2	14.3	34.5
Total	213.2	95.7	308.9

*Total for 3 vehicles

Option 2. Additions, modifications and deletions to the reference cost data were performed as necessary to determine cost estimates which reflect the current OTV design. The costing approach used for major items involved iterative interaction with the engineers assigned to the OTV task and is explained in the following paragraphs.

7.3.1 Structure

Costs for the tanks, tank supports, and payload interface assemblies are based on methodology used in the Space Tug study adjusted to reflect increases in tank sizes. The outer shell, interstage, and thrust structure assemblies are all defined for the OTV as monocoque graphite epoxy which is different than the Space Tug design and therefore the costs for these assemblies are based on recent aircraft cost experience for graphite epoxy structure.

7.3.2 Thermal Control

The OTV thermal control design is similar to that used in the Space Tug study. Therefore, the thermal control subsystem costs are directly related to Space Tug costs adjusted as necessary to account for greater area requiring thermal protection.

7.3.3 Avionics

The avionics subsystem is divided into the five major categories of data management, guidance and navigation, communications, instrumentation, and electrical power. Data management DDT&E costs are reduced from the Space Tug estimate to reflect use of the standard NASA computer and the Shuttle Orbiter multiplexer-demultiplexer unit. Costs for the remaining data management components are based on Space Tug costs for similar items. OTV software is estimated at one-half the amount defined for the Space Tug. This reduction is achievable through the use of some existing software, more efficient programming techniques, and more advanced hardware. Guidance and navigation DDT&E costs are also reduced to reflect the use of the Shuttle Orbiter IMU and Star Tracker. Costs for the laser radar and supporting electronics are based on Space Tug costs for similar items. Specifications for communications equipment assumed that most of the items would be developed on other space programs so that only minimum DDT&E effort would be required. The production costs for these items are obtained from corresponding Space Tug items adjusted for variations in quantity requirements. Instrumentation equipment is similar to that defined for the Space Tug, therefore, Space Tug costs are used for this equipment with only minor revisions. Electrical power equipment consists primarily of Shuttle Orbiter-developed fuel cells and reactant tanks. The DDT&E cost for this equipment is for adaptation to the OTV and production costs reflect data obtained from Orbiter subcontractors. The cost of the power distribution system is based on Space Tug estimates for similar equipment.

7.3.4 Propulsion

The propulsion subsystem consists of the main engine, main engine support plumbing, and the reaction control system. The main engine is an upgraded Category IIA RL-10 with engine life extended from 5 to 20 hr. The DDT&E cost of $91M to accomplish this effort includes the Space Tug Pratt & Whitney estimate for the basic upgrading plus an additional amount for testing to achieve the 20-hr engine life. The engine unit cost is obtained from the earlier Pratt & Whitney cost data. The main engine support plumbing consists of items similar to those employed in the Space Tug but must be designed to support two engines in the first stage and one engine in the second stage. Engineering judgment related to the cost of this equipment assumed that the total DDT&E cost for both stages would equal 1.5 times the Space Tug single-stage estimate and that the total production cost for both stages would be two times the Space Tug single-stage estimate. The reaction control system is defined as a blowdown monopropellant system previously analyzed as an alternate for the Space Tug. The costs for this system are based on cost data developed for subsystem tradeoffs conducted during the Space Tug study.

7.3.5 Final Assembly and Checkout

This effort includes final assembly tooling, installation and assembly design, and physical assembly and checkout of the subsystem hardware into the total stage. The final assembly tooling estimate is the same as that determined for the Space Tug. The remaining effort is estimated as a percentage of subsystem DDT&E and production first unit costs using factors developed for the Space Tug study.

7.3.6 Tanker Vehicle

The tanker vehicle consists of structure, propellant transfer, and thermal control subsystems similar to those contained in the primary OTV. Estimates for these subsystems along with final assembly and checkout of the tanker reflect application of the same approach and methodology used for the OTV as described above. The tanker production cost is for two vehicles, plus initial spares. The total tanker DDT&E cost includes the vehicle development, plus the additional supporting effort for system test, logistics, facilities, and GSE.

7.3.7 Project Management

The project management effort provides cost/performance management, project direction, and configuration management during the DDT&E and production phases of the program. This effort is estimated as a percentage of vehicle, logistics, facilities, and GSE costs for the DDT&E phase and as a percentage of vehicle costs for the production phase using factors developed during the Space Tug study.

7.3.8 System Engineering and Integration

This major category includes system specifications, interface definitions, safety, reliability, human factors and other related tasks during both the DDT&E and production phases as well as sustaining engineering during the production phase. The basic system engineering effort is estimated as a percentage of vehicle DDT&E and production costs using Space Tug factors. The sustaining engineering effort is estimated as a level of support during four years of vehicle production using previously developed methodology which relates sustaining engineering to vehicle unit cost, production rate, and stage size.

7.3.9 System Test and Evaluation

This category includes the test hardware and test operations necessary to perform total system tests on the orbital vehicle and evaluate its performance prior to flight. The test hardware is estimated as a percentage of vehicle first-unit production cost using the Space Tug relationship. The test operations cost is the same as that used in the Space Tug study.

7.3.10 Logistics, Facilities and GSE

The remaining supporting elements comprising the total OTV program include training, inventory control, manufacturing facilities, test facilities, and items of ground support equipment required for transportation, handling, and checkout. The costs for these three elements are all estimated as a percentage of vehicle DDT&E cost using factors developed during the Space Tug study.

Part 12

SUMMARY OF RESULTS AND FINDINGS OF SPACE PROCESSING WORKING REVIEW

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SPACE PROCESSING WORKING REVIEW
Summary of Results and Findings

INTRODUCTION AND SUMMARY

McDonnell Douglas Astronautics Company (MDAC) sponsored and hosted a Space processing Working Review at the Space Systems Center, Huntington Beach, California. The one-day session 27 October 1976 was held to critique development plans for products to be manufactured in space. The results of the review, which also included an assessment of the system requirements were applied to the MDAC Space Station Systems Analysis Study.

Attendees, representing constituencies from private industry and the aerospace complex, collectively reviewed and commented upon material prepared by the MDAC Study Team. In addition to MDAC, the following organizations were invited to attend the workshop.

Mr. Merton A. Robinson
Mr. John M. Walsh
Dr. Allen A. Strickler
Beckman Instruments, Inc.
Anaheim, California

Mr. Howard Klink
Motorola Semiconductor
Products Group
Phoenix, Arizona

Dr. George F. Neilson, Jr.
Owens - Illinois
Toledo, Ohio

Dr. Carl D. Graves
Mr. Robert L. Hammel
Mr. Paul R. Mock
Mr. Donald M. Waltz
TRW Systems, Inc.
Redondo Beach, California

Dr. Waldo Rall
United States Steel Corporation
Pittsburgh, Pennsylvania

The MDAC Study Team attendees included the following individuals:

Mr. G. V. Butler
Mr. J. L. Cobb
Mr. R. J. Gunkel
Mr. S. J. Harris
Mr. R. E. Holmen
Dr. H. B. Kelly
Mr. W. R. Marx
Dr. G. L. Murphy
Mr. D. W. Richman
Mr. F. J. Sanders
Mr. L. O. Schulte
Mr. F. H. Shepphird
Mr. R. J. Thiele
Dr. R. Weiss
Dr. H. L. Wolbers

It was the express purpose of the working meeting to identify the design and test requirements necessary to establish an evolutionary development program

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that includes precursor ground tests, space proof-of-concept demonstrations, pilot-plant operations, and finally, a commercial manufacturing facility. The conference also sought to identify potential problem areas, of either a technical or business concern, for which solutions must be worked out in the future. Each of the attendees was provided a data package, the content of which is described elsewhere in this report.

The agenda for the day included the following subjects:

1. Overall review of Space Station systems concepts
2. Space processing background and pertinent related experience
3. Introduction to prototype product form case studies:
   - Bioprocessing
   - Ultrapure glasses
   - Shaped crystals
4. Instructions for splinter sessions
5. Splinter sessions for each case study
6. General review session and synthesis of commentary

The emphasis placed by members of the study team in attendance was upon the identification of Space Station design requirements evident from early space processing activities beginning in the 1983-84 time period. Specific comments from the experts in attendance included the following:

1. The three case studies appear to adequately describe the procedures, process steps, and equipment necessary to transition a product from R&D to pilot plant demonstration and initial commercial production.
2. The requirements, insofar as equipment characteristics are concerned in general, would remain the same during the R&D process development, and process optimization steps; a 7 to 10 scale-up in physical and operational requirements could be expected during the transition to pilot plant activities.
3. The protection of proprietary rights and confidentiality of data are of paramount importance to potential industrial users of space facilities; the impacts on design features for this form of protection should be assessed.

4. Government sponsorship of R&D activities up to the point of proof-of-concept, where profitability and probability can be adequately assessed, are likely prerequisities to private capital committments from industry.

All the participants in the working session were of the opinion that the meeting was informative, meaningful, and productive and that a worthwhile exchange of views and facts was accomplished.

MATERIALS REVIEWED DURING THE WORKSHOP

The MDAC Space Station Systems Analysis Study is charged with evaluating space processing as a major emphasis activity of the future. In particular, definition of the processing steps necessary to transition a commercially attractive product form from process development, through process optimization to the point of pilot plant demonstration is being examined. The purpose of this study task is to define the requirements that space processing with a commercial emphasis would impose on an early (circa 1985) Space Station.

Three cases are being studied during the course of the system study. The cases have been selected to describe individually and collectively the range and extent of early Space Station requirements (i. e., the resources and services required to support space processing activities) and to give focus to design features especially important to future commercial users of the facilities. From many candidates examined, the three cases selected as representative design drivers are: (1) biologicals processing using the enzyme urokinase as the example, (2) ultrapure glasses using a fiber optics application as the example, and (3) semiconductor grade silicon producted in space in ribbon form. For each case the following information updated as of the workshop period was provided:

1. The steps and elements of a development plan to carry the product area from R&D through process development, process optimization, and demonstration of pilot plant operations, to commercial production.
2. Identification of the specific role of the Space Station system in the development plan, up to the point of pilot plant demonstration.

3. Definition of the process flows, equipment, resources, crew support (man-machine interface), time spans, control parameters, functional operations, and identification of the critical process steps in the terms of a process/resource timeline for the product.
4. Summaries of equipment requirements, functional requirements, and operational requirements, and suggested facility layouts for process.
5. Assessments of the equipment and system costs and development schedules.

This information is illustrated in Figures 1 through 13.

During the course of the conference, many technical subjects as well as business-oriented discussion topics were covered. It was not the intent of the forum to reflect current corporate policy of the constituencies represented; rather, it was planned that the conference project dynamically to future potential problem areas which must be solved as they are encountered. Future technical policy directions, evolutionary patterns of government, industry methods of transacting business and innovative approaches to problem solutions were sought in terms of educated opinions as to the future course of the industrialization of space.

The opinions of the participants from private industry were solicited in answer to issues suggested by a set of some 26 questions. The questions, which appear in the next two sections of this document covered both technical and business oriented topics. The term "industry" within context of the question refers to the sector of the economy occupied and served by the corporations represented, rather than the specific company stated policies per se.

TECHNICAL FACTORS DISCUSSED DURING THE SESSIONS

A presentation was made to the group as a whole of Space Station system concepts, past and present. Primary emphasis was placed on the more recent concepts, among these the emerging space construction base configurations. One point which was emphasized was the difference between the current Space Station Systems Analysis Systems Analysis

Study and other related studies of the past; i. e., the current approach stresses the industrialization of space as contrasted to the "laboratory in the sky" approach of prior studies.

A presentation was made subsequently to those in attendance at the space processing background and related projects. The essentials of this briefing were the projected evolution of space processing, beginning with basic investigations and phenomena-oriented research, and leading toward achieving the ultimate goal of factories in space. The review focused on the requirement already identified for Spacelab mission hardware applicable to the Space Processing Program Activity (SPA) which might serve as a point of departure for the identification of Space Station space processing mission hardware.

During the course of the working session, splinter groups met to discuss each of the three individual cases. These groups included those most expert in the disciplines involved in the cases. The system requirements reflected by each case of the Space Station were also reviewed. The subjects discussed during the splinter sessions included the following topics:

1. Case study review and assessment
2. Process research and development activities
3. Equipment requirements
4. Support system requirements
5. Technical and business issues
6. Schedule and logistics
7. Proprietary rights

The data matrix produced during the working review is shown in Figures 14, 15, and 16.

The 15 questions, which were of a technical nature, are listed below. A consensus of the answers and comments returned, without identifying the specific individuals who provided answers, are also included.

T1. Do the prototype products selected for case study adequately describe the range and extent of foreseeable space processing activities? Are there major gaps, such as important manufacturing steps, not considered that would impact on support requirements?

Cases appear adequate. Allow for flexibility for new products and uses.

T2. Do the physical and operational scale-ups from early laboratory R&D to proof-of-concept and pilot plant development activities appear realistic in the time period?

Transistion appears realistic assuming success oriented schedule—slip of 5 to 10 years would be less optimistic.

T3. Are the projections for technology advance adequate for supporting a Space Station buildup in space processing activities in the 1984-85 time frame?

Projections are optimistic and presuppose R&D funding to the point-of-concept validation.

T4. Are there pivotal advancements which can be wholly or partially accomplished by space production? For example, what material characteristics need to be improved?

Early demonstrations in space will identify winners which are yet not clear.

T5. Assuming these technology advancements lead to eventual production of commercial products, can their future market value be quantified?

Cannot predict with sufficient confidence to provide an answer.

T6. What percentage of material will require 50% higher quality in the 1984-85 time frame? 100% higher quality? 200% higher quality?

Should emphasize new materials rather than improved existing material quality.

T7. What is the probable product life cycle for typical space-produced products?

Same as ground experience; 5 - 10 years in duration.

T8. For a typical product, what resupply frequency might be required to support production? For how many products?

Four to 12 missions per year; up to five products in development.

T9. What procedures are necessary for isolation and protection of the final or purified product?

Use same as with ground experience.

T10. What measures must be employed to prevent and /or correct product carry-over when subsequent batches are run or different materials are processed?

Consider the use of disposable liners.

T11. What are the cleanliness requirements and what related federal specifications are pertinent?

Same as industry standards; Federal Drug Administration (FDA) regulations apply to bioprocessing.

T12. What personnel protection procedures must be observed and what federal regulations are pertinent?

Same as industry standards including Operational Safety and Health Administration (OSHA) requirements.

T13. What quality assurance provisions are required to verify product, container, and facility purity and sterility?

Same as industry standards, FDA for bioprocessing.

T14. What types of data and records need to be kept and safeguarded?

Process and evaluation data protection is essential.

T15. What methods and procedures for on-site inspection are necessary to be able to meet eigher industry or federal government requirements?

Only applicable to bioprocessing.

Taken as a whole, the industry participants view space processing with some reserve. Sufficient orbital experience has not been compiled at this point in time to establish a solid basis for a commercial commitment to space manufacturing. This, in essence, leaves the initial space-work to be funded almost solely by the government. Their projections did provide a basis for identifying design drivers. For example, the use of potential dangerous substances (e.g., flammables, toxics, corrosives, biohazardous materials) would be required. These requirements have impacts on safety features of the Space Station.

BUSINESS FACTORS DISCUSSED DURING THE SESSIONS

It was the primary purpose of the workshop to concentrate on technical issues. However, since commercial and industrial constituencies were represented, the opportunity was capitalized upon to present to the forum some of the business-related issues that must be faced in the future.

Eleven of the 26 questions relate more closely to business issues than to technical requirements. These questions and a consensus of the answers provided are as follows:

B1. How far would the government-sponsored space R&D activities have to progress prior to private capital commitments from industry?

All R&D complete to the point of proof-of-concept and product characterization, i.e., evidence of profitability and probability of commercial success.

B2. How quickly would industry take on new ventures in space processing following early successes in space R&D?

Space ventures would be speculative, having to compete with other ground products, hence, a follow-the- leader phenomenon is expected.

B3. How responsive would industry be to developing new markets for unique products manufactured in space?

The market is the driver, not the product. Merely the uniqueness of space does not offer an advantage over ground-produced products.

B4. What factors, which influence new venture marketing risk and timely market penetration, need to be considered and programmed into future government policy?

Assurance to industry of continued profit and availability of space facility, proprietary position also important.

B5. What fraction of a projected 1984-85 market would you allocate to a space-manufactured product (in a current product line)?

None currently foreseen.

B6. What amount of venture capital from industry would be available between now and the 1984-85 time period; (1) based upon what is known today, and (2) based upon what can be expected from the SPAR and Sortie Flight Program.

Very little.

B7. What expected return on investment would be required to attract private capital investment to space manufacturing?

Discounted cash flow return on investment--20 to 40%.

B8. What assurances and guarantees on the part of the government are required to control cost schedules of STS payload charge allocations?

Need long-term commitment by the government with predictable charges keyed to cost-of-living index -- stringent requirement.

B9. What sureties are required to protect rights in data and other proprietary positions of the venture project?

Absolute sureties with 3 - 5 years protection.

B10. What waivers to existing government regulations or changes to the law are required to protect the proprietary and patent rights to the ventures?

Whatever required to protect proprietary rights.

B11. What policies and guarantees are required regarding government control of the space facilities and their operation?

Noninterference in processes and procedures, guarantee of continuity at indexed dollars.

The general tenor of the responses received from industry to the business-oriented topics is one of caution and a conservative approach. The particular area of most concern is rights in data and maintenance of a proprietary position. It is likely for the industries participating, that important proof-of-concept demonstrations and clear-cut product advantages would be required, and probably financed by the government, as an inducement to private investment in space processing. Therefore, commercial space processing offers promise for the future; a well-founded R&D program on the many facets of the unknown remains the immediate goal.

CASE I

BIOPROCESSING USING THE PROTOTYPE PRODUCT FORM UROKINASE

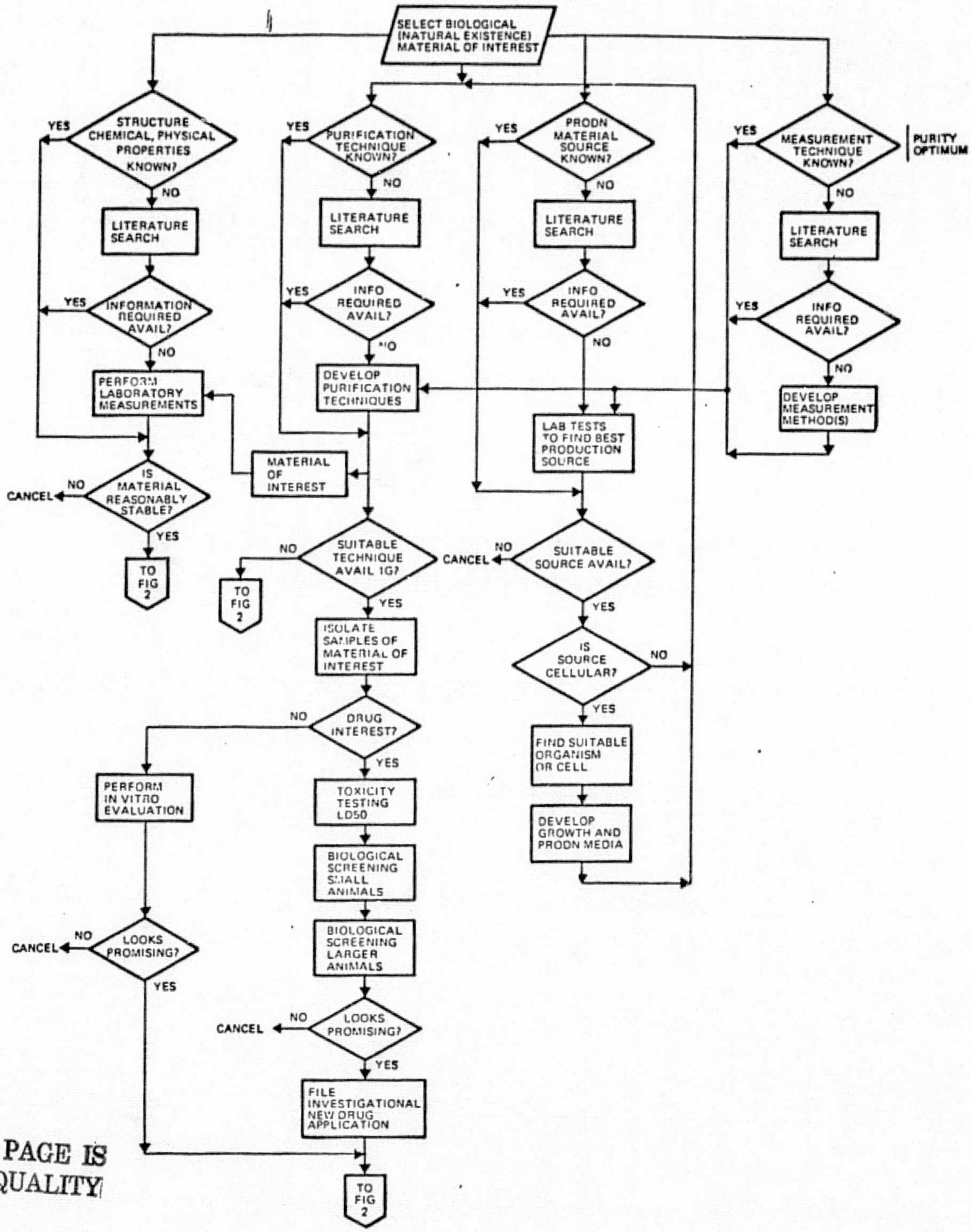

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Figure 1. Research and Development Phase

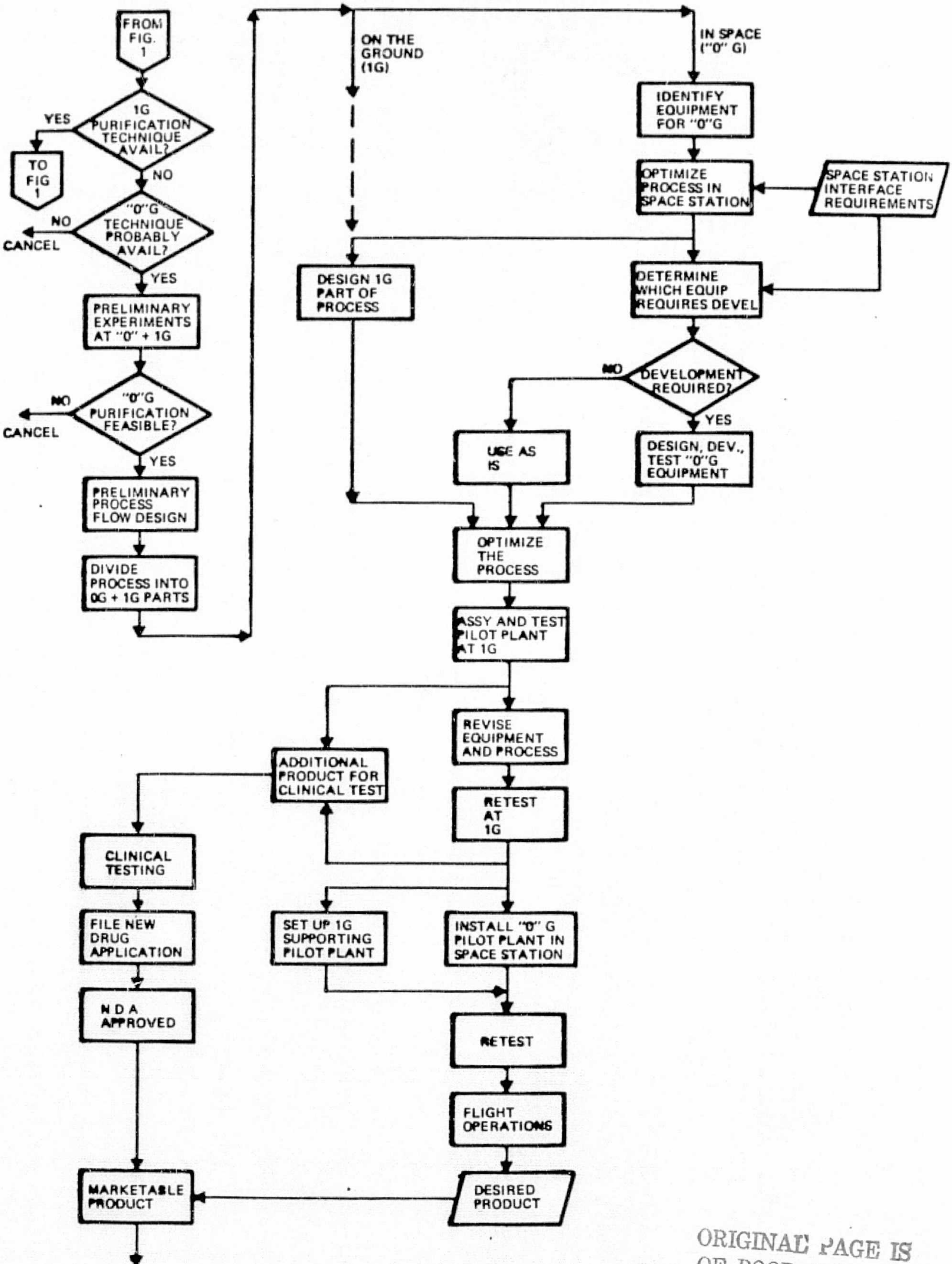
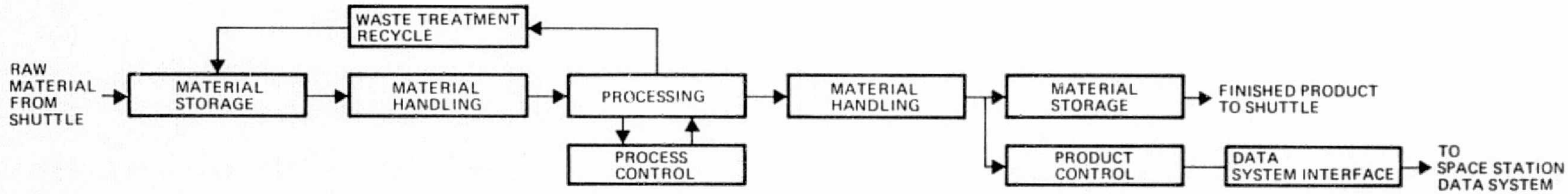


Figure 2. Pilot Plant Development for Space Processing of Biological Material



MATERIAL STORAGE
- LIQUID STORAGE
- GAS STORAGE
- REAGENT STORAGE
- REFRIGERATED STORAGE (~4°C)
- CRYOGENIC STORAGE (~ -70°C)
- CONTAINERS + LINERS

MATERIAL HANDLING/TRANSPORT
- PLUMBING INTERCONNECT
- LIQUID PUMPS
- GAS PUMPS
- MANUAL BATCH TRANSFER (IN ZERO G CONTAINER)

PROCESSING (TYPICAL
- SOLUTION PREPARATION
- INCUBATION
- STERILIZATION
  - STEAM
  - DRY HEAT, CHEMICAL
- MIXING
  - STIRRER
  - SPARGER
- HOLDING
  - PLAIN TANKS
  - TEMPERATURE CONTROLLED
- RATE FREEZER
- SOLVENT EXTRACTION
  - COUNTER CURRENT DISTRIB.
- DISTILLATION
  - SOLVENT EVAPORATION
- ION EXCHANGE
- LIQUID CHROMATOGRAPHY
- FILTRATION
- CENTRIFUGATION
  - ULTRA CENTRIFUGATION
- ADSORPTION, BATCH
- ELECTROPHORESIS, CONTINUOUS
- DIALYSIS
- DIFFUSION
- CHEMICAL REACTORS
- DRYING
  - LYOPHYLIZATION
  - OVEN
- AIR/LIQUID SEPARATION
- SAMPLING
- PACKAGING
  - VIALS
  - BAGS
  - FILL EQUIPMENT
  - SEALING EQUIPMENT
- WASTE RECYCLE
- DISPOSABLE LINERS
- EQUIPMENT CLEANING
- TOOLS

PROCESS MEASUREMENT AND CONTROL
- LIQUID FLOW RATE
- GAS FLOW RATE
- WEIGHING (MASS MEASUREMENT)
- TEMPERATURE
- PRESSURE
- PH
- LIGHT SCATTERING
- VISCOSITY
- CELL COUNTING
- LIQUID VOLUME
- SAMPLE CONTAINERS
- SAMPLING EQUIPMENT
- CONCENTRATION

END PRODUCT CONTROLS
- MICROSCOPE
- SLIDE STAINER
- CELL COUNTER
- SPECTROPHOTOMETER
- ANALYTICAL ELECTROPHORESIS
- IMMUNOCHEMISTRY
- WET CHEMISTRY
- BACTERIOLOGY
- PYROGEN TESTING

SUPPORTING FACILITIES
- ELECTRICAL POWER
- HEATING
- COOLING - COLD ROOM
- WASTE DISPOSAL
  - DRY
  - LIQUID
- HEAT REJECTION
- CREW SUPPORT
- DATA SYSTEM INTERFACE
- VACUUM

Figure 3. Generalized Biological Process for Space

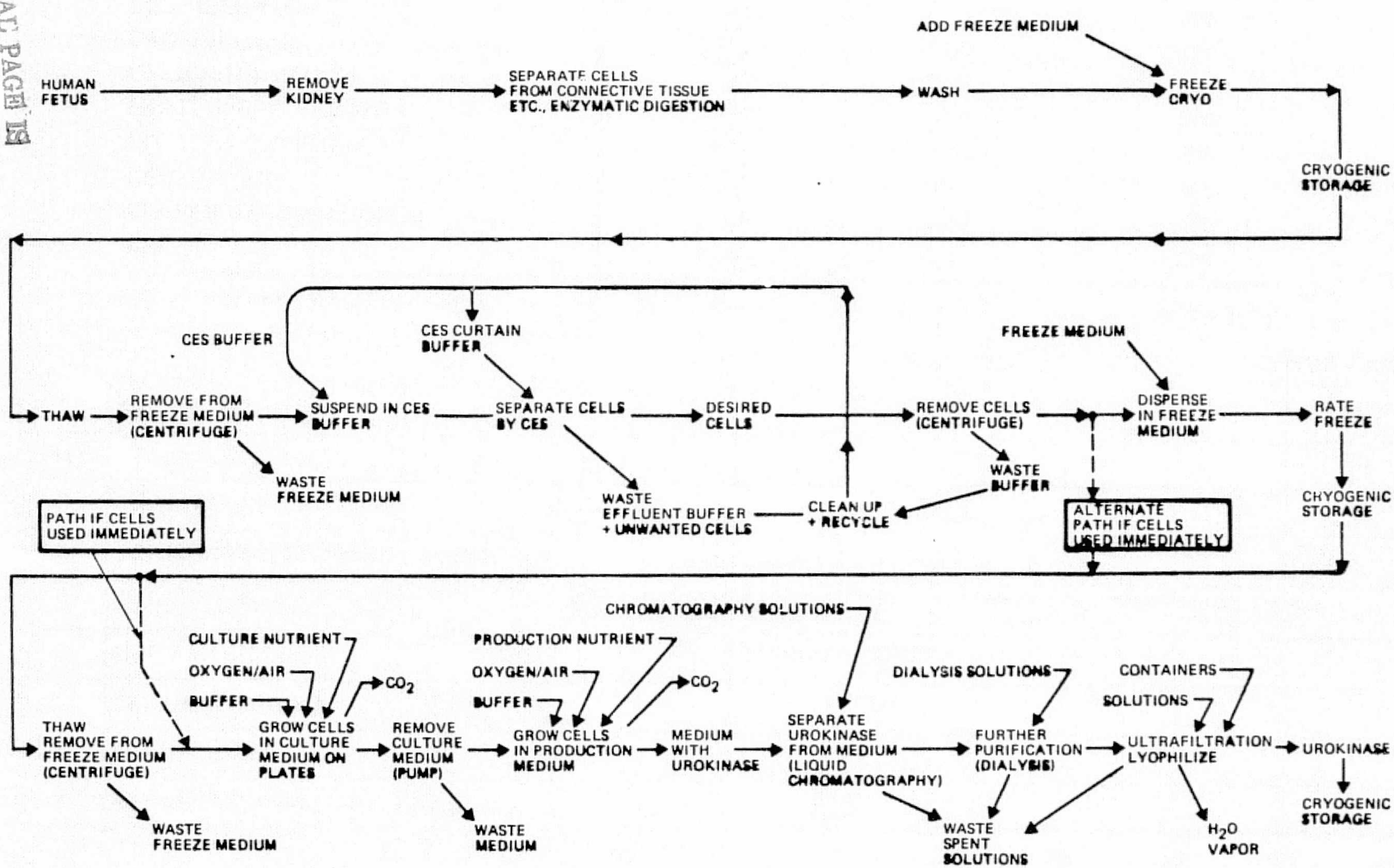


Figure 4. Urokinase Process

Figure 5

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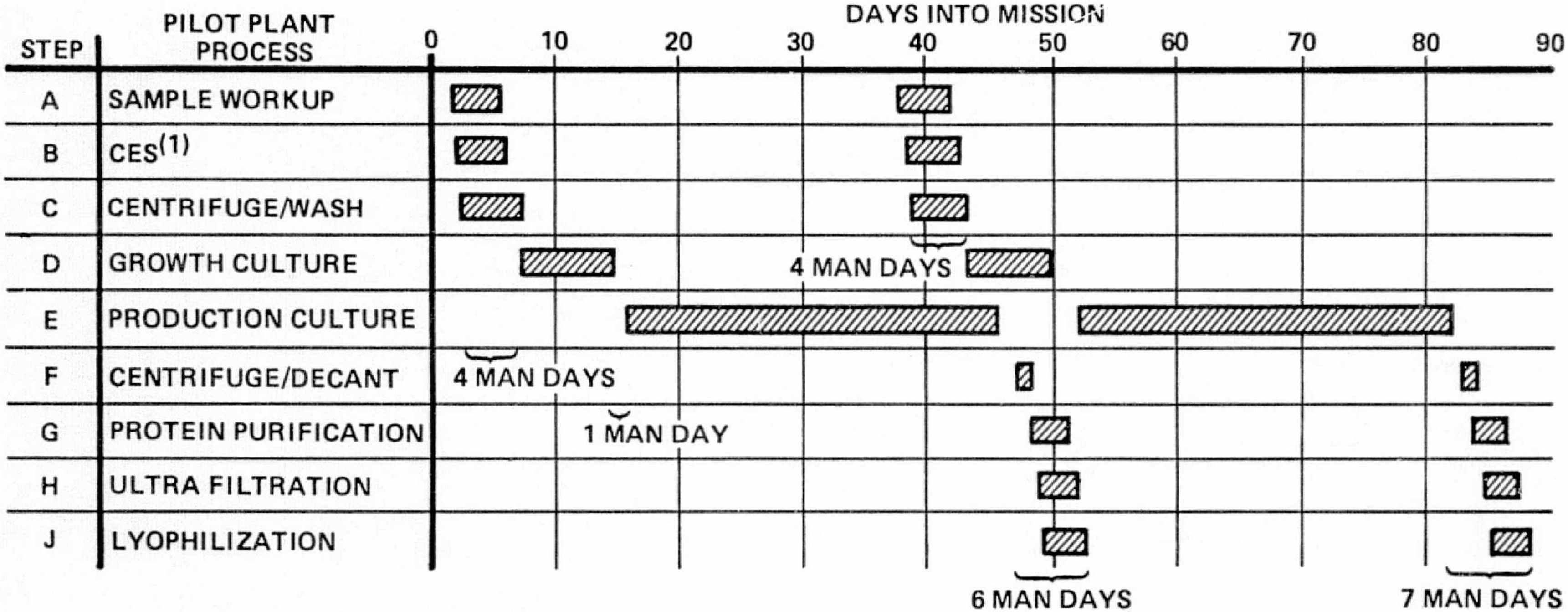
PROCESSING SCHEDULE CASE EXAMPLE

(IKG UROKINASE PRODUCED DURING MISSION)

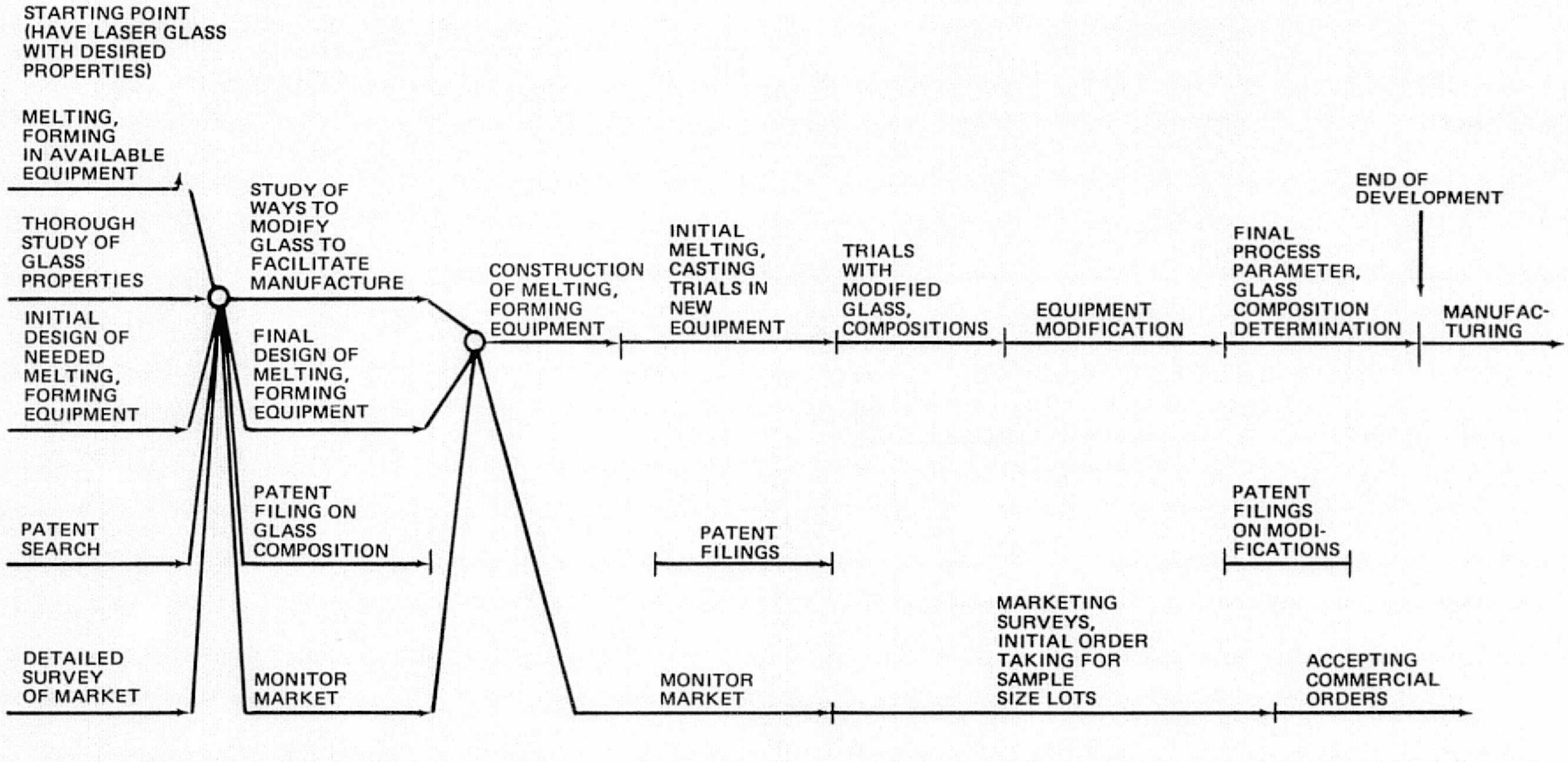


EQUIPMENT REQUIRED	WEIGHT (KG)	VOL (M^3)	POWER (W)
CES	272	0.44	2,500
BUFFER RECONDITIONER	45	0.04	100
CENTRIFUGE	275	0.65	2,000
CULTURE CHAMBER	115	0.45	500
PROTEIN PURIFICATION	150	0.70	200
ULTRAFILTRATION	15	0.15	200
LYOPHILIZER	400	0.70	3,500
REFRIGERATOR	70	0.12	350
TOTALS	1,342	3.25	5,350 PEAK

(1) CES = CONTINUOUS ELECTROPHORESIS SYSTEM
1KG UROKINASE PRODUCED IN SPACE = 10,000 TREATMENT REGIMENS

CASE 2

ULTRAPURE GLASSES PROCESSING USING THE PROTOTYPE PRODUCT FORM FIBER OPTICS



O – DECISION POINTS (COST ESTIMATES FOR ALL FOLLOWING PHASES NEEDED AT EACH DECISION POINT)

Figure 6. Ultrapure Glasses Products Flow (Process R&D and Optimization)

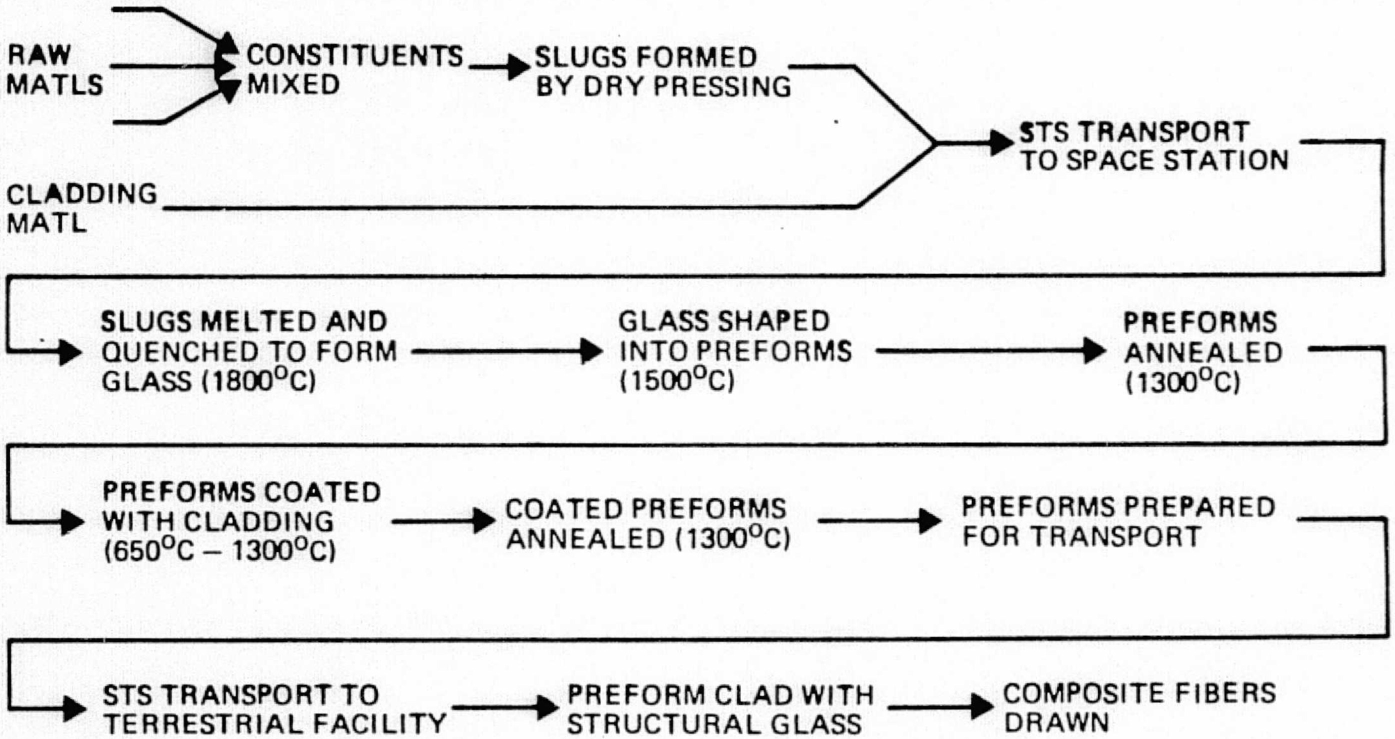


Figure 7. Fiber Optics Preform Production

CASE 3

CRYSTALS/METALS PROCESSING USING THE PROTOTYPE PRODUCT FORM SEMICONDUCTOR GRADE SILICON PRODUCED IN RIBBON FORM

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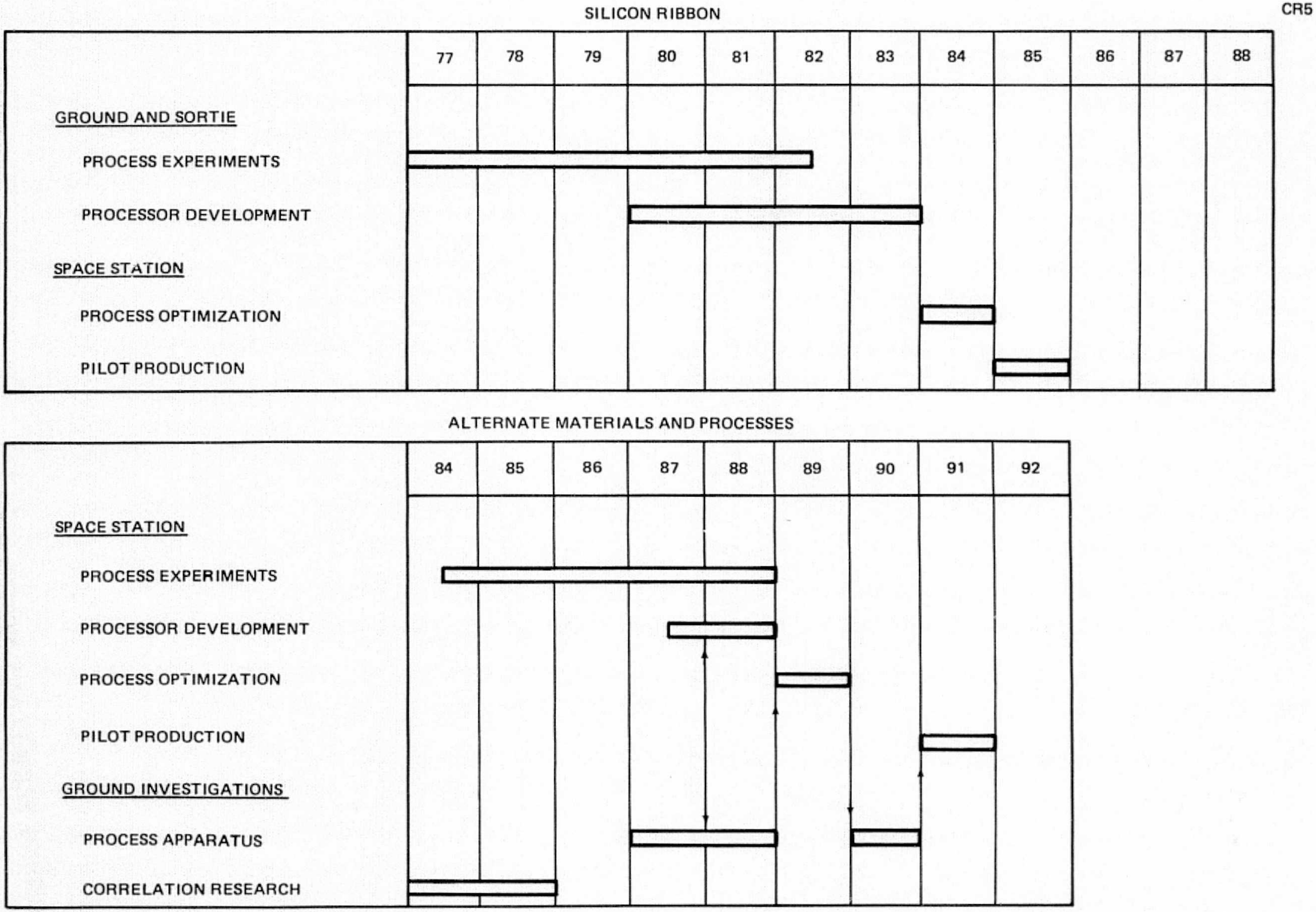

Figure 8. Shaped Crystal R & D Schedule

CR5

	82	83	84	85	86	87	88
MODIFY APPARATUS							

TEST	OBJECTIVE	EQUIPMENT
PROCESS SYSTEM EVALUATION	OPERATIONAL TEST OF PROCESS APPARATUS	PROTOTYPE PROCESS APPARATUS
OPERATIONAL SHOCK AND ACCELERATION EFFECTS	DETERMINE EFFECTS OF TRANSIENTS ON PROCESSOR OPERATION	PROTOTYPE PROCESSOR ACCELEROMETERS, RIBBON CHARACTERIZATION EQUIPMENT
END PRODUCT CONFIGURATION CONTROL	EVALUATE METHODS OF CONTROLLING SHAPE, ORIENTATION, DOPANT, ETC. DURING PROCESSING	PROTOTYPE PROCESSOR, RIBBON CHARACTERIZATION EQUIPMENT
ALTERNATE MATERIALS	EVALUATE ALTERNATE MATERIALS PROCESSING ON PROTOTYPE PROCESSOR	PROTOTYPE PROCESSOR, RIBBON CHARACTERIZATION EQUIPMENT
SERVICING PROCEDURES	DEVELOP SERVICING AND MAINTENANCE PROCEDURES	PROTOTYPE PROCESSOR, SERVICING EQUIPMENT
		TETHERED PRODUCTION PLANT, SERVICING EQUIPMENT
EXPAND CAPABILITIES	DEVELOP METHODS AND APPARATUS FOR PROCESSING SOLAR CELLS	PROTOTYPE PROCESSING APPARATUS, ION-IMPLANTATION EQUIPMENT, VACUUM DEPOSITION CHAMBER, RIBBON CHARACTERIZATION EQUIPMENT

Figure 9. Space Station Tests

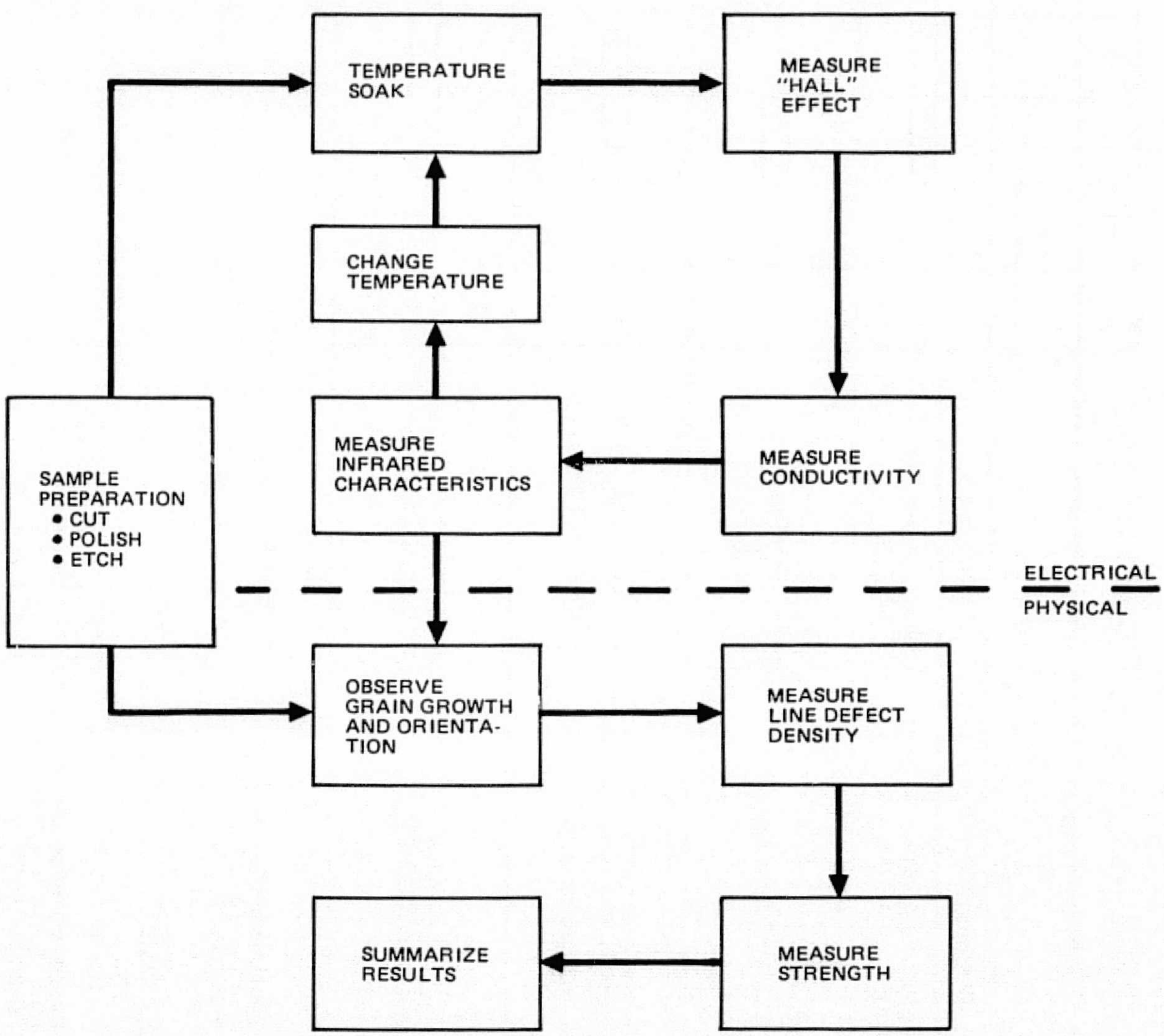

Figure 10. Product Analysis

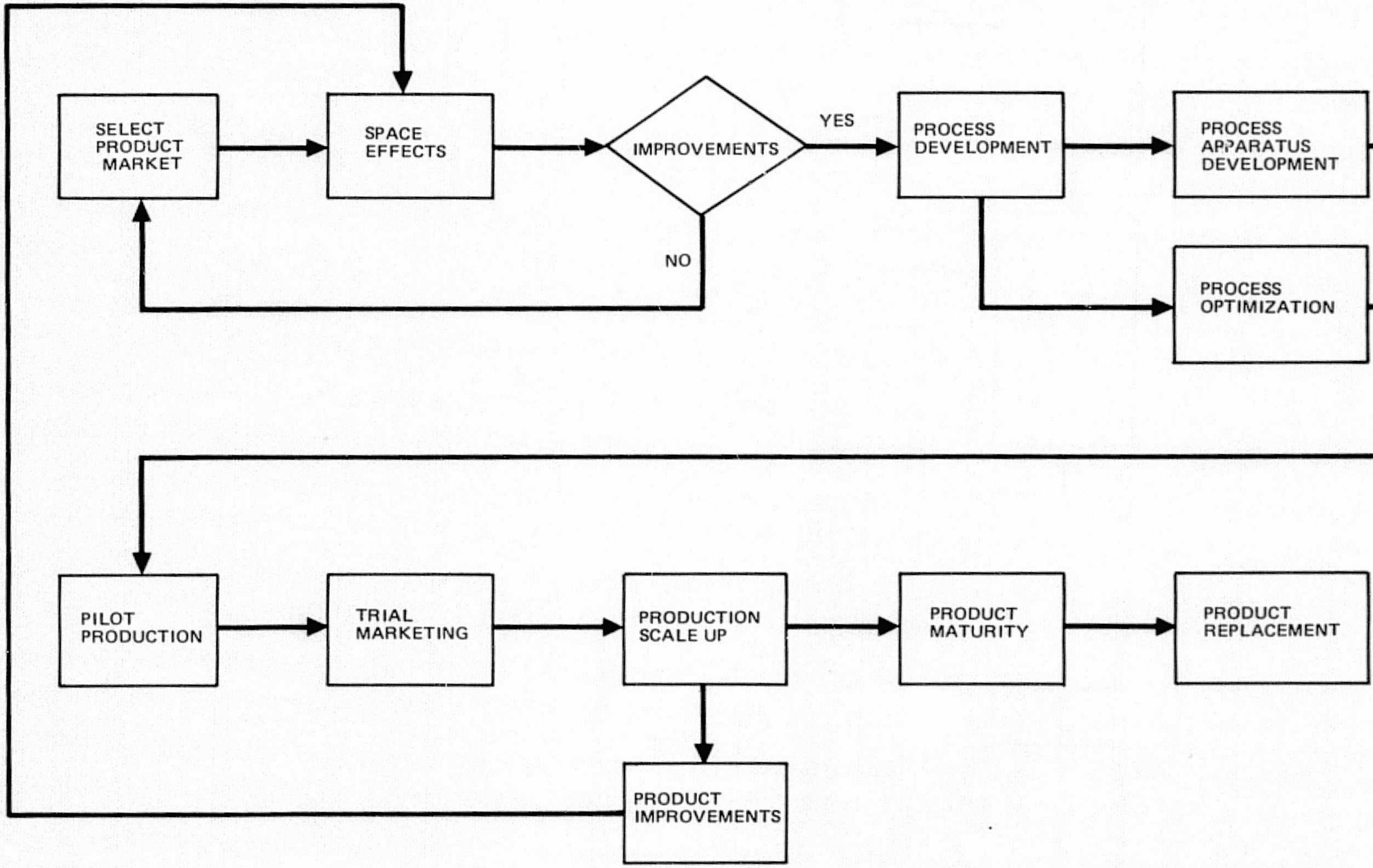

Figure 11. Space Product Cycle

CR5

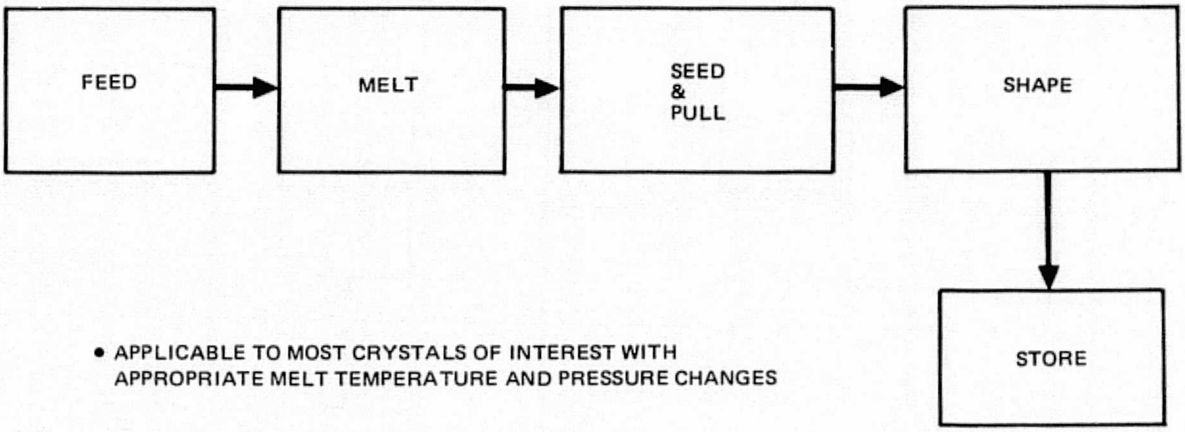

Figure 12. Shaped-Crystal Process

CR5

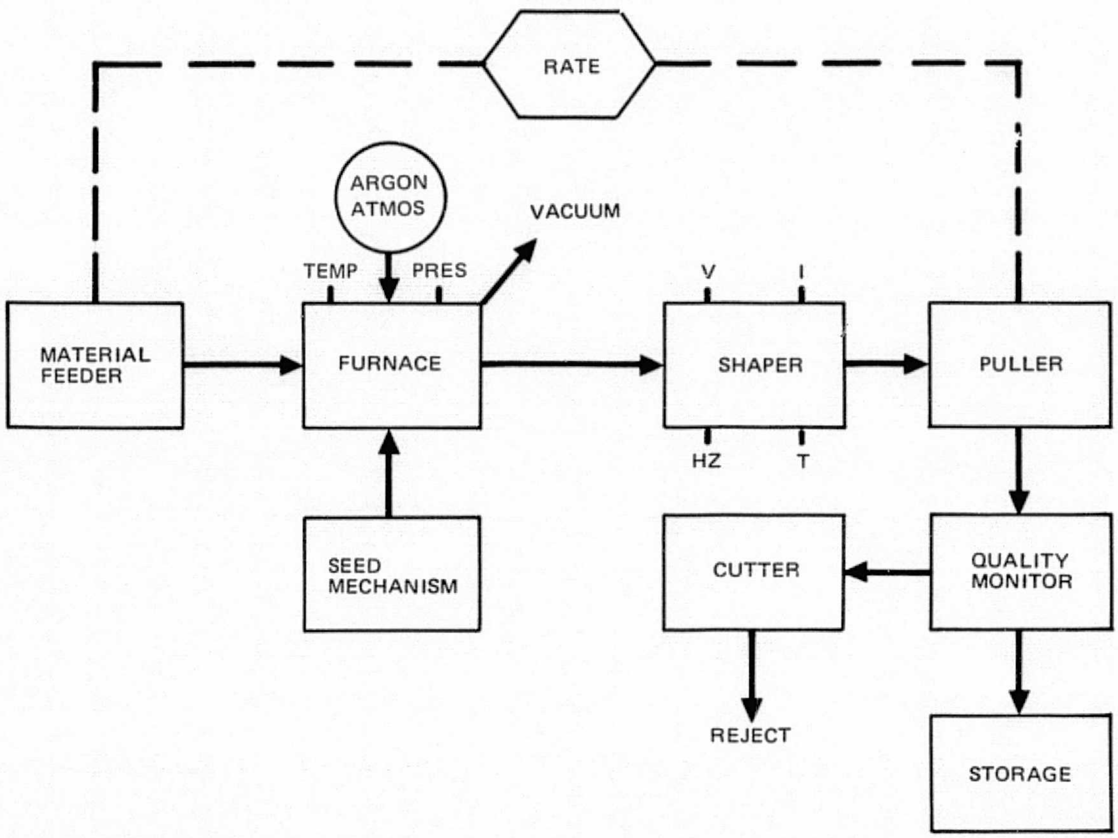

Figure 13. Shaped-Crystal Processor

CASE DATA SHEETS

Requirements/Characteristics (Units)	R/D Laboratory	Process Development	Process Optimization	Pilot Plant
CIRCA	1986	1986	1986	1990
	Program Phase			
1. Physical Accommodations				
o Equipment volume (m^3)	3.3	3.3	3.3	33
o Equipment weight (kg) (itemize)	1342	1342	1342	13,000
o Consumables weight (kg/day) (specify)	3	3	3	30-50
o Consumables volume (m^3/day)	—	—	—	—
2. Crew				
o Number of personnel	3	3	3	12
o Skills		BIOSCIENTIST/TECHNICIAN		
o Number of shifts (continuous activity required)	3 DURING CRITICAL PERIODS CONTINUOUS			
3. Environmental Conditions				
o Temperature - Max (°C) - Min (°C)	NOMINAL; DEPENDING ON SPECIFIC DESIGN APPROACH MAY REQUIRE 4° AMBIENT			
o Humidity (%)	70	70	70	70
o Cleanliness class	100,000	100,000	100,000	100,000
o Acoustic limit (db)	70	70	70	70
o Acceleration limit (g)	$\sim 10^{-3}$	$\sim 10^{-3}$	$\sim 10^{-3}$	$\sim 10^{-3}$
4. Power				
o Average power (w)	1,600	1,600	1,600	10,000
o Peak power (w)/ - Duration (hours)	5,350/72	5,350/72	5,350/72	25,000/72
o Illumination - Type - Brightness (Lumens/m^2)	NOMINAL	NOMINAL	NOMINAL	NOMINAL
5. Data Management/Communications				
o Computation/program storage	NONE	NONE	NONE	NONE
o Digital rate/link (real time) (kBPS) - Duration (min) - Source	10 KB/DAY FOR PROCESS MONITORING			TBD
o Digital storage (MB)	NONE	NONE	NONE	TBD
o Video bandwidth/link (real time) - Duration (min) - Source	TV MONITOR TO GROUND WITH REMOTE SCAN AND ZOOM CAPABILITY			
o Video storage (min)	—	—	—	TBD
o Voice link	NOMINAL	NOMINAL	NOMINAL	NOMINAL
o Remote satellites	NONE	NONE	NONE	NONE
6. Waste Management (specify toxic/benign)(1)				
o Weight (kg)	225	225	225	TBD
o Volume (m^3)	0.5	0.5	0.5	TBD
o Special precaution/hazards(2)	PROTECTION OF RAW MATERIALS AND FINAL PRODUCT			
7. Logistics				
o Mission resupply period (days)	90	90	90	TBD
o Material and supplies delivered - Weight (kg)	225	225	225	TBD
- Volume (m^3)	0.5	0.5	0.5	
o Product returned - Weight (kg)	MINIMAL	MINIMAL	1(3)	10 (4)
- Volume (m^3)			< 1	1
o Equipment/parts - Weight (kg)	150	150	150	1500
- Volume (m^3)	0.3	0.3	0.3	3

(1) REQUIRES CRYOGENIC STORAGE TO PREVENT BACTERIA GROWTH
(2) PROCESSING AND PRODUCT ANALYSIS ACTIVITIES COULD INVOLVE USE OF FLAMMABLES, CORROSIVES, TOXICS, RADIOISOTOPES AND BIOHAZARDOUS MATERIALS
(3) EQUIVALENT TO 72 X 10^6 INTERNATIONAL UNITS
(4) EQUIVALENT TO 720 X 10^6 INTERNATIONAL UNITS

Figure 14. Space Processing System Characteristics and Support Requirements for Bioprocessing Case

Requirements/Characteristics (Units)	R/D Laboratory	Process Development	Process Optimization	Pilot Plant
CIRCA	1984	1984	1985	1987
	Program Phase			
1. Physical Accommodations INCLUDING WORK AREA				
o Equipment volume (m^3)	42	42	42	
o Equipment weight (kg) (itemize)	1,725	1,725	1,725	X7-10
o Consumables weight (kg/day) (specify)	0.4	0.4	0.4	
o Consumables volume (m^3/day)	—	—	—	
2. Crew				
o Number of personnel	4-6	4-6	4-6	9
o Skills	MATERIAL SCIENTIST, TECHNICIAN			
o Number of shifts (continuous activity required)	2-3	2-3	2-3	3
3. Environmental Conditions				
o Temperature				
- Max (°C)	NOMINAL	NOMINAL	NOMINAL	NOMINAL
- Min (°C)				
o Humidity (%)	NOMINAL	NOMINAL	NOMINAL	NOMINAL
o Cleanliness class	10,000	10,000	10,000	10,000
o Acoustic limit (db)	70	70	70	70
o Acceleration limit (g)	10^{-3}	10^{-3}	10^{-3}	10^{-3}
4. Power				
o Average power (w)	17,000	17,000	17,000	
o Peak power (w)	26,000	26,000	26,000	X7-10
- Duration (hours)	12	12	12	
o Illumination				
- Type	NOMINAL	NOMINAL	NOMINAL	NOMINAL
- Brightness (Lumens/m^2)				
5. Data Management/Communications				
o Computation/program storage				
o Digital rate/link (real time) (KBPS)				
- Duration (min)				
- Source				
o Digital storage (MB)	NOMINAL REQUIREMENTS			
o Video bandwidth/link (real time)				
- Duration (min)				
- Source				
o Video storage (min)				
o Voice link				
o Remote satellites				
6. Waste Management (specify toxic/benign)(1)				
o Weight (kg)	NOMINAL	NOMINAL	NOMINAL	TBD
o Volume (m^3)	—	—	—	TBD
o Special precaution/hazards	TOXIC METAL OXIDES POWDERS, FLUORIDES			
7. Logistics				
o Mission resupply period (days)	90	90	90	90
o Material and supplies delivered (2)				
- Weight (kg)	100	100	100	1,000
- Volume (m^3)	—	—	—	2
o Product returned (2)				
- Weight (kg)	100	100	100	1,000
- Volume (m^3)	—	—	—	2
o Equipment/parts (2)				
- Weight (kg)	10	20	40	200
- Volume (m^3)	—	—	—	4

(1) – FURNACE PRODUCES TOXIC AND CORROSIVE VAPORS
(2) – INCLUDE PACKAGING AND SHIPPING CONTAINERS

Figure 15. Space Processing System Characteristics and Support Requirements for Fiber Optics Glasses Case

CIRCA	1984	1984	1985	1986
	Program Phase			
Requirements/Characteristics (Units)	R/D Laboratory	Process Development	Process Optimization	Pilot Plant
1. Physical Accommodations				
o Equipment volume (m^3)	52	52	52	X10
o Equipment weight (kg) (itemize)	7,200	7,200	7,200	
o Consumables weight (kg/day) (specify)	10	10	10	
o Consumables volume (m^3/day)	0.01	0.01	0.01	
2. Crew				
o Number of personnel	3	3	3	X2
o Skills		2 SCIENTISTS, 1 TECHNICIAN		
o Number of shifts (continuous activity required)	2 AT 10 HOURS CONTINUOUS OPERATIONS			3
3. Environmental Conditions				
o Temperature				
- Max (°C)				
- Min (°C)	NOMINAL	NOMINAL	NOMINAL	NOMINAL
o Humidity (%)	NOMINAL	NOMINAL	NOMINAL	NOMINAL
o Cleanliness class	10,000	10,000	10,000	10,000
o Acoustic limit (db)	70	70	70	70
o Acceleration limit (g)	$< 10^{-3}$	$< 10^{-3}$	$< 10^{-3}$	$< 10^{-3}$
4. Power				
o Average power (w)	9,700	9,700	9,700	X10
o Peak power (w)	16,500	16,500	16,500	
- Duration (hours)	(1/2 HR)			
o Illumination				
- Type	NOMINAL	NOMINAL	NOMINAL	
- Brightness (Lumens/m^2)				
5. Data Management/Communications				
o Computation/program storage	NONE IDENTIFIED AT CURRENT LEVEL OF DEFINITION			2×10^8 BITS
o Digital rate/link (real time) (kBPS)	40 kBS DATA AND VOICE	10 MINUTES/ DAY	DATA FORMATTER	12 kBS CONTINUOUS
- Duration (min)				
- Source				
o Digital storage (MB)	–	–	–	CASSETTES
o Video bandwidth/link (real time)				
- Duration (min)				
- Source	–	–	–	–
o Video storage (min)	–	–	–	–
o Voice link	–	–	–	–
o Remote satellites	–	–	–	–
6. Waste Management (specify toxic/benign)				
o Weight (kg)	MINIMAL	MINIMAL	MINIMAL	TBD
o Volume (m^3)	–	–	–	TBD
o Special precaution/hazards	PHOSPHINE GAS, RADIOISOTOPES, A_2O_3			
7. Logistics				
o Mission resupply period (days)	90	90	90	90 OR TBD
o Material and supplies delivered				
- Weight (kg)	1,000	1,000	1,000	TBD
- Volume (m^3)	2	2	2	
o Product returned				
- Weight (kg)	800	800	800	TBD
- Volume (m^3)	2	2	2	
o Equipment/parts				
- Weight (kg)	200	800	800	TBD
- Volume (m^3)	4	15	15	

Figure 16. Space Processing System Characteristics and Support Requirements for Shaped Crystal Case

ADVANTAGES OF SPACE CONSTRUCTION BASE IN THE EVOLUTIONARY PATH OF COMMERCIAL SPACE PROCESSING

(Action Item 21)

OBJECTIVE

The objective of this analysis is to compare the space processing requirements with presently planned Spacelab and Space Construction Base (SCB) capabilities. The primary ground rules and assumptions used in the analysis are as follows:

1. Precursor R&D activities through 1983 will be actively supported by the NASA Space Processing Program (SPA) and will include sounding-rocket, Orbiter, Spacelab, and free-flyer missions. By 1983, products with commercial value to be produced in space will have been identified.

2. Spacelab is as defined by the ERNO document* RF-ER-0005:

   Mission duration - 7 days
   Allowable payload weight - 3,800 kg (8,360 lb)
   Average electrical power for payload - 3 kW
   IOC - 1980

   *Assumes Orbiter landing with weight of payload in bay not to exceed 11,364 kg (25,000 lb)

3. The first SCB module will be launched in 1984. The SCB will provide required resources such as processing times (uninterrupted periods of operation in excess of 90 days, duration), adequate levels of micro-gravity and disturbance-free environment, electrical power, heat rejection, pressurized and unpressurized work space, crew skills and services, and data management functions.

4. The space processing program objective and requirements reflect a long-range evolutionary development of a commercially oriented, privately sponsored industrial operation in space as the ultimate goal to be realized. The advantages that the SCB can provide over the Spacelab in supporting the space processing program objectives

will focus on the operational and physical capabilities of the facility. Cost advantages such as the reduction of transportation expenses have not been considered.

ANALYSIS

For each of the three case studies, a typical research plan and development schedule as shown in Figure 1 was prepared. These plans depict the time-phased steps necessary to carry the prototype product from basic research through process development and optimization to the ultimate goal of commercial production. As shown on the chart, there is an evolution of activities leading to production. With the exception of those activities which must be done on the ground, there are four modes of space flight that could be utilized: (1) sounding rocket flights, (2) STS/sortie flights including early Spacelab missions, (3) STS/Spacelab flights, and (4) space construction base (SCB) missions. Each class of activity, as evidenced by the case studies, follows a progression of more complex operations involving larger complements of equipment, longer mission durations and extended capabilities in space.

For the bioprocessing case the plan for development progress follows an increase in required capabilities, as shown in Table 1. It is evident, when the time frame and availability of the space research facilities are examined, that the sounding rockets will play an important early role in basic research oriented missions (i. e., Steps 2 and 3). These missions will establish important directions for later applied research activities suitable for programming into early Spacelab missions (i. e., Steps 4, 5, and 6). However, the role of rocket flights will diminish as the extended capabilities of the more advanced facilities become available.

Step 6 of the development evolution represents the transition from research-oriented activities to commercial production. This transition is characterized by (1) a change in emphasis from investigative procedures toward increase in efficiency in operations, yield improvement, and quality assurance, and (2) a change in motivation from scientific pursuit to profit-oriented production.

Figure 1

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DEVELOPMENT SCHEDULE-BIOPROCESSING CASE

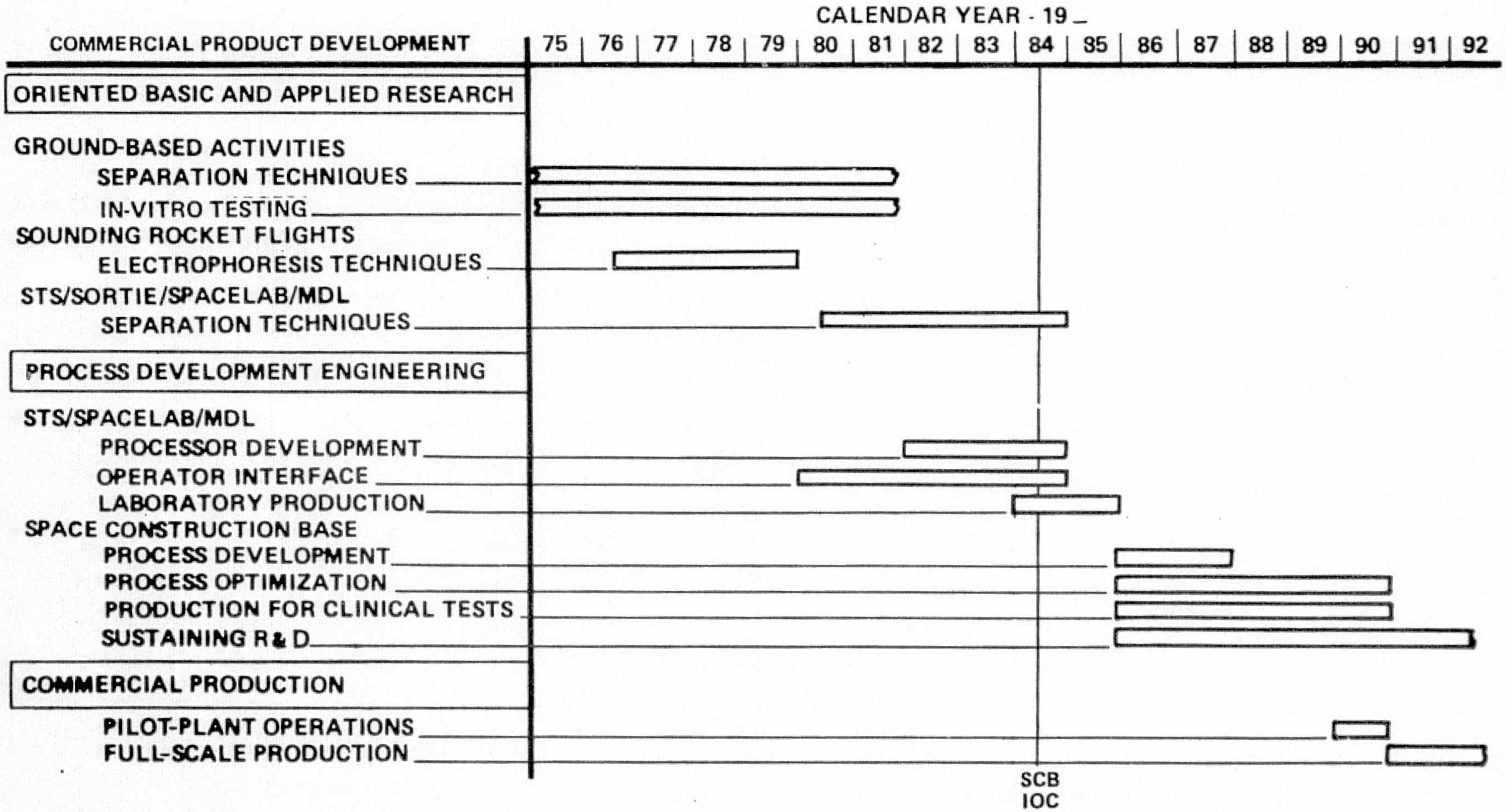


Table 1

DEVELOPMENT OF COMMERCIAL PRODUCTION OF BIOMATERIALS IN SPACE

Basic Steps in Evolutionary Development	Minimum Required Capability of Mission Hardware			
	Weight kg	Avg Power kW	Crew Size	Duration (Days)
1. Ground-based research	(Not Done in Space)			
2. Basic R&D on separation techniques	25	0.5	0 (1)	0.02
3. Process-methods-oriented R&D	100	1	0 (1)	0.02
4. Processor development and engineering	500	2	1	5
5. Limited laboratory production	1,000	3	2	5
6. Production process development	1,500	3	3	7
7. Process optimization	5,000 (2)	4	3	45
8. Production for clinical tests	12,000 (2)	4	3	60
9. Pilot plant production	12,000 (2)	6	4	90
10. Full-scale production	25,000 (2)	10	6	>90

(1) Can be automated payloads

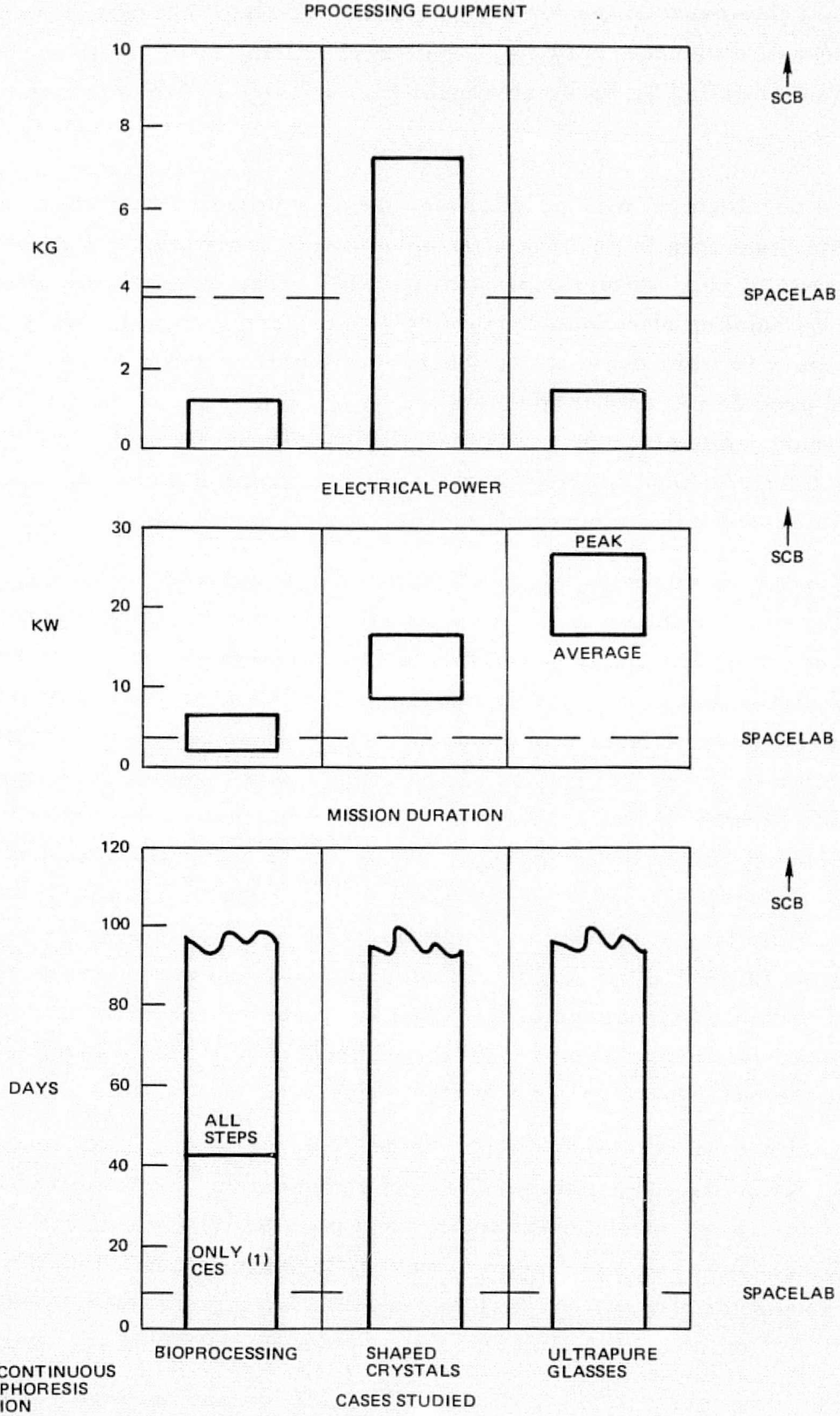
(2) Requires dedicated module(s)

The average power requirements at Step 6 of the development plan exceed the Spacelab capabilities for the ultrapure and shaped crystal processes as defined in the case studies. The bioprocessing requirements, however, could be satisfied by Spacelab capabilities at Step 6 of the development plan.

As the development plan progresses, the SCB/mission hardware will have the dominant role in process development, optimization, and support of commercial production (Steps 6 through 10). The longer-term missions, where a number of repeated trials could be accommodated, would be necessary to work out optimal process parameters. Successive trials would provide the data to evaluate and determine the best set points for individual temperatures, pressures, and flow rates at each step of a particular process. During these activities, complete access to all equipment for adjustments and reconfiguration would be essential.

The transition to commercial operations demands order-of-magnitude increases in available process times with commensurate power and energy services over what is required for research and development. Thus, a clear distinction in the facility capability that SCB offers commercial space processing over Spacelab is apparent. At least an order-of-magnitude reduction in the redistribution of recurring costs together with extended periods of time without interruption can be expected at this transition. Cost reduction can be achieved in part by the fact that learning and increased efficiency attendant upon repeated operations increases the productivity of labor. The biomaterials study describes the possibility of a sixty-fold increase in production, as the mission period is increased from 90 to 360 days, without an increase in the processing equipment complement. It is this quantum-jump class of increase in productivity that is required to make space processing commercially attractive.

A direct comparison of Spacelab, as defined in Figure 2, and space construction base for commercial space processing is noteworthy. Production process development and optimization activities will require a significant on-orbit capability. The requirement delineated on the chart represents the transitional step in the evolution from research and development and optimization. As can be seen from the requirements described for the three case studies, the mission duration and average power



(1) CES = CONTINUOUS ELECTROPHORESIS SEPARATION

Figure 2. Summary of SCB/Spacelab Comparison Data

requirements exceed Spacelab capabilities. Therefore, it would not be possible to mechanize the entire complement of mission hardware necessary to pursue product-oriented process development and optimization activities within the confines of Spacelab. However, certain individual steps could be investigated during Spacelab missions. For example, the continuous electrophoresis system, which is a crucial part of the bioprocessing flow, could be evaluated in part by means of Spacelab missions; the overall process could not be so accommodated, however. For commercially significant processes to be developed, the space construction base is a necessity.

SPACE PROCESSING CASE STUDY BACKGROUND

In Part 1 of the Space Station System Analysis Study, three Space Processing product development cases were examined where Space Station support would be justified to conduct R&D activities. The R&D activities would then lead to the establishment of a Space Station based pilot plant followed by full scale space production operations. These cases were documented in the Space Station Systems Analysis Study, Part 1 Final Report, Volume 3, Appendixes, Book 1, MDC G6508.

The cases studied in Part 1 were:

- Case 1 - Production of Silicon in Ribbon Form
- Case 2 - Production of Biological Materials (Urokinase Example)
- Case 3 - Inorganics Processing (Fiber Optics Applications).

At the writing of the Part 1 case studies report, and indeed at the present time, (Part 2 report), the lack of hard factual data is cause to state that none of the three studied cases is assured of technical or economic success. However, it is believed that each case has appreciable promise, and, more importantly, each case is typical of a distinctive space processing commercial venture to serve as a good basis for determining Space Station requirements.

For convenience, the Part 1 conclusions are repeated:

- As space processing cycles from the initiation phase to the full-scale commercial production phase, the role of the Space Station will be essential during the transition from early R&D to pilot-plant demonstration.

- As such, Space Station activities will experience a growth and buildup in capabilities with increasing crew size, power requirements, thermal and environmental conditioning, and complexity of operations.

- Space Station activities will include establishment and control of routine operations, optimization versus experimentation, and changeover from laboratory to production facilities.

- When space processing activities mature to where expansion of production is justified, the Space Station will provide the necessary supporting environment to expedite the transition to commercial ventures.

These conclusions come from the conviction that there is a general fit and compatibility between the goals of space processing (eventual manufacturing in space) and the projected resources of a Space Station.

The Part 2 Case Studies for Biological Materials (Urokinase example) and Fiber Optics follow as separate stand-alone reports within this document.

This report was prepared by the TRW Defense and Space System Group for the McDonnell Douglas Astronautics Company in support of the Space Processing objective of the NASA/JSC Space Station Systems Analysis Study. TRW was assisted during these studies by subcontract support from Beckman Instruments, Owens-Illinois, and U.S. Steel Corp.

CASE STUDY

PRODUCTION OF BIOLOGICAL MATERIALS

UROKINASE EXAMPLE

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1.0 INTRODUCTION

The Space Station Study Part 1 report identified three case studies that would be used to derive Space Station subsystem requirements relative to space processing [1]. The biological case study results are the subject of this report.

The purpose of the biological case study is to provide to the Space Station subsystems design activity requirements which are representative of a biological processing facility. To this aim, an example material (urokinase) was chosen for process analysis. It is emphasized that the resulting equipment inventory can be used to process many materials and is not unique to the case example.

1.1 CASE STUDY BACKGROUND

The Space Station Study, Part 1, resulted in the identification of several areas of endeavor that should benefit from the resources potentially available via the Space Station [1]. Among the several candidates is the field of space processing. Space processing encompasses the area of research into, and development of, materials which uniquely benefit from processing in space; the ultimate objective being products of significant economical value.

The biological material chosen for study is the enzyme urokinase. This material was chosen principally because earlier experimentation indicated some benefits from space processing might occur [2]. In addition, ground-based work being conducted by Abbott Laboratories provided a source for general information regarding processing protocol requirements.

1.2 CASE STUDY OBJECTIVE

The objective of the biological processing case study is to develop requirements for the Space Station subsystems design. Generation of requirements that are both reasonable and representative of biological processing is important and, therefore, the case example selection must be realistic. However, the actual material selected (in this case, urokinase)

[X] denotes references listed in Section 6.0.

is not the important feature of the case example. Rather, it is the fact that urokinase represents a semi-continuous flow process whose equipment complement could be representative of many biological material production processes.

The case study results will provide requirements for the Space Station power generation and distribution subsystem, the thermal/environmental control subsystem, the habitability subsystem and the control subsystem. The requirements will derive from the power, energy; waste heat, volume and weight parameters of the urokinase pilot plant equipment and production process.

1.3 CASE STUDY APPROACH

The approach taken during the biological processing case study involved separation of the overall study into several elements. The first element consisted of an examination of the new product development process in the pharmaceutical industry. The results of the examination highlighted the role Space Station could play in a new product development activity.

The second element involved the development of the urokinase production process. The individual unit processes were established in terms of equipment and equipment requirements.

The third element defined Space Station requirements by developing both an R&D Laboratory and a pilot plant processing example. A pilot plant process timeline and an example of an optimum equipment complement was developed.

2.0 BIOLOGICAL PROCESSING

The activities involved in developing a new biological product from the research and development phase to the production phase are generally similar for most biological products. While the specifics vary from material to material, the general structure of the activities, the decision points and the qualification procedures are similar.

This section of the report will discuss the general procedures followed in terrestrial laboratories. In addition, the general process which might be carried on in the Space Station will be identified.

2.1 NEW PRODUCT DEVELOPMENT

The process steps to carry a biological product form from the research and development stage through the steps of laboratory development, process optimization, space processing and pilot plant operations are discussed in this section.

2.1.1 Research and Development Phase

During this phase, the activities shown in Figure 2-1 are usually carried out. In many cases extensive research and development work has already been performed before the material of interest has been selected. In these cases, the effort required begins at the points where information is missing. Activities performed in the research and development phase include:

(a) A literature search is performed to gather all pertinent material on the biological to be made.

(b) The product and its related compounds are characterized chemically (if necessary) and its stability in various environments is measured. In particular, the material's stability is evaluated in the conditions encountered in processing.

(c) A search is made (often involving laboratory investigations) to find the best source of the desired material. The source may be animal, vegetable or microbial in origin. If the source is microbial, the bacterial strain to produce the product is usually isolated or developed.

(d) In the case of a microbial or cellular process, the medium for growing the organism or for producing the desired product is developed and is optimized for its relationship to the organism to produce the best yield.

(e) The necessary techniques are developed for measuring the amount of the substance of interest.

(f) Techniques for isolation and purification of the product are developed.

(g) If the material is of clinical interest, a series of animal tests are performed. These tests include toxicity testing, biological screening in first small animals then more definitive

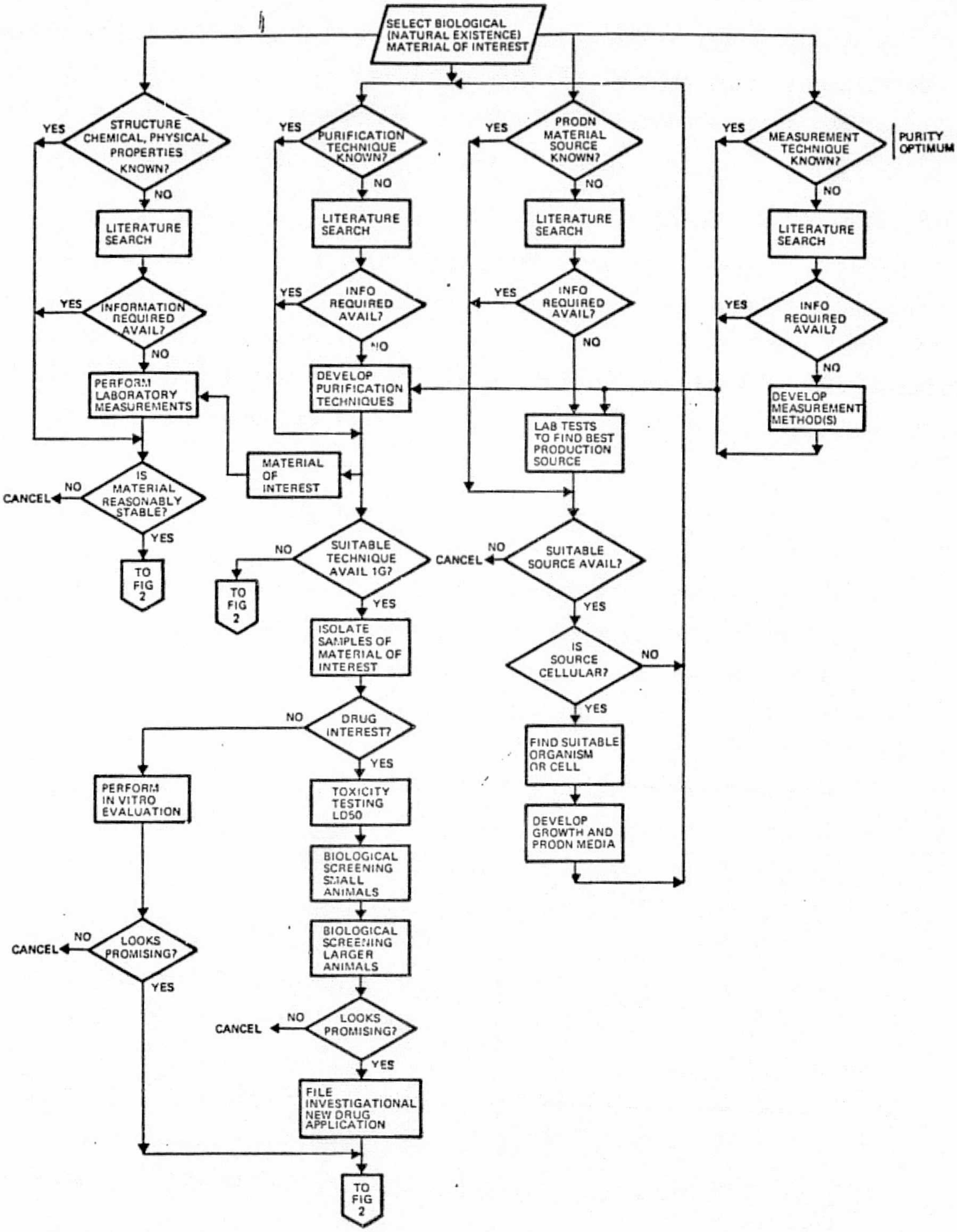


Figure 2-1. Biological Products — Research and Development Phase

testing in larger animals. If these results are positive, an Investigational New Drug Application is filed and development is continued.

At several points in Figure 2-1 the term "cancel" is encountered. This indicates that the outcome of the activity is reason to terminate the R&D activity.

After successful completion of the Research and Development Phase, the effort will proceed to the Pilot Plant Phase.

2.1.2 Pilot Plant

If no suitable technique is available for the isolation and purification of the material of interest at 1-g, then the possibility of employing a zero-g separation technique will be investigated.

Some of the techniques which have been considered for the separation of biological materials under zero-g are continuous electrophoresis, counter current distribution of particulates and freezing front techniques. The selection of an appropriate separation technique is based on (1) the separation requirements such as purity required and the nature of the impurities, and (2) the properties of the material to be separated.

When a candidate zero-g separation technique has been selected, it will be evaluated (to the maximum extent possible) at one-g. It is essential that the candidate separation technique be tried zero-g. These trials may be conducted in either the Spacelab or the Space Station. These trials may include the design of special equipment to perform the experiment. If data from these experiments show that it is feasible to perform the desired separation, then design of the separation equipment for the pilot plant can proceed on a firm foundation. Limited animal and clinical testing will be performed on the material isolated during the developmental work.

During this phase, the laboratory process is scaled up to the desired pilot plant levels. During pilot plant design, it is advantageous to consider using a continuous flow process wherever possible. New variables associated with pilot plant operation will surface at this stage of development. For example, an enzyme may be damaged by heating it excessively as it passes through a pump. Pumps may also cause mechanical damage to cells as they pass through the mechanism. At this point, it is common to find that one may have to add a heat exchanger at the pump to cool the material or may have to add a stabilizer to the media to preserve the material of interest.

It is often at this stage where processes which were carried on by centrifugation are changed over to filtration processes since filtration processes are generally preferable for larger scale operations.

Another example of a process which is somewhat different at the pilot plant level than it is at the laboratory level is liquid chromatography. It is much easier to perform batch adsorption at the pilot plant level rather than liquid chromatography which is the laboratory level.

Figure 2-2 shows the activities which are carried out during the Pilot Plant Phase.

After a complete process flow has been laid out, the operations to be conducted in space are selected. The equipment to perform these operations are then identified. The process flow for the space operations is then optimized to give the best possible product yield within the Space Station's constraints (i.e., volume, power, weight, crew availability, etc.).

The equipment selected is surveyed to determine which pieces may be used as is in zero-g and which require further development or adaption.

In parallel with this activity equipment for the balance of the process is selected to be compatible with the Space Station operations. The entire process stream is then optimized and its operation is confirmed by running as much as possible of the process at one-g. Based on the results of this test, the process or its equipment will be revised and retested.

At this point, the equipment for the zero-g portion of the operation will be installed in the Space Station and functionally tested. The balance of the Pilot Plant will be set up on the ground and will be used to support the Space Station Biological Process Pilot Plant during Flight Operations.

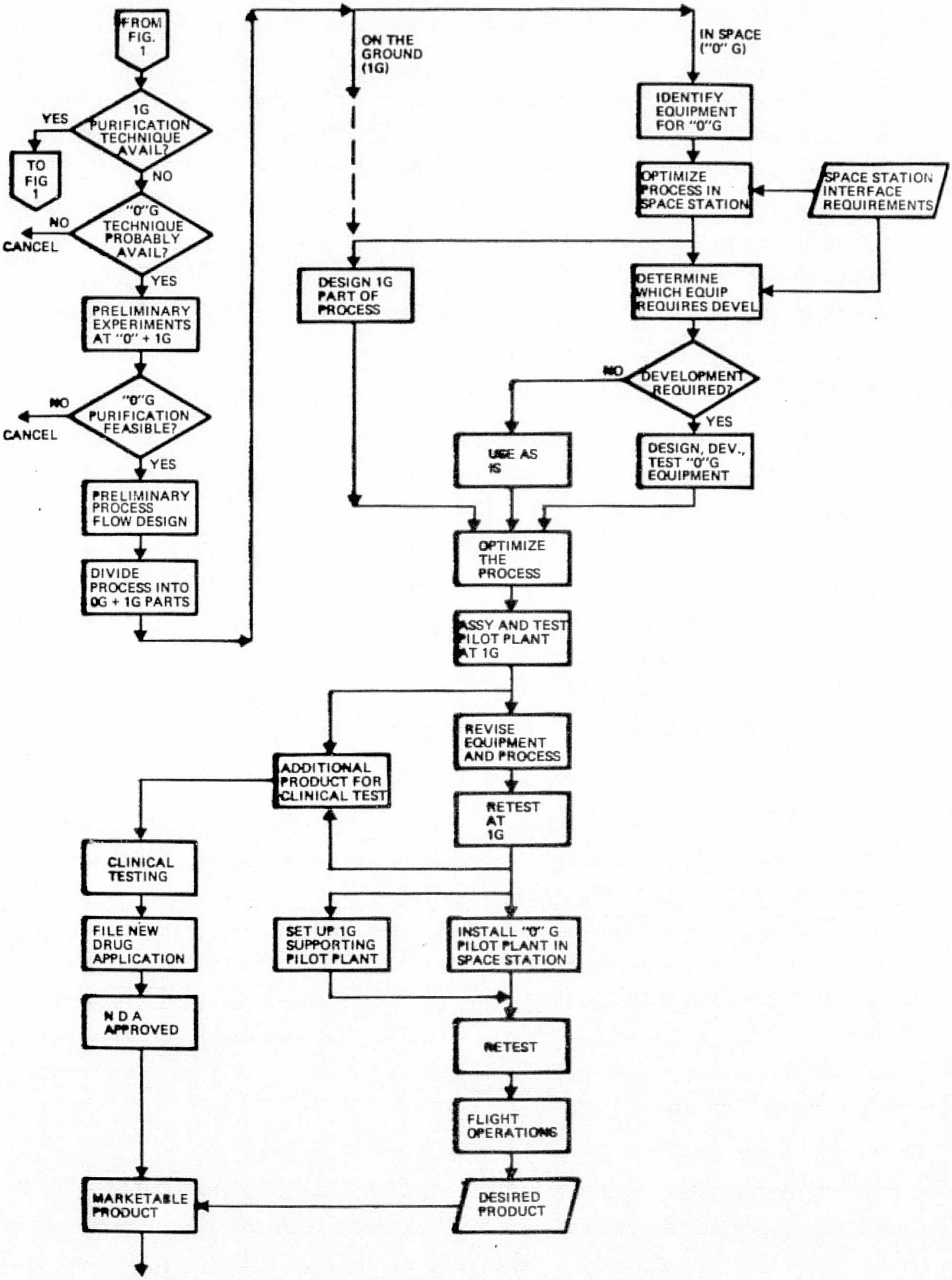


Figure 2-2. Pilot Plant Development for Space Processing of Biological Material

Larger quantities of the drug are required to support clinical testing than can be conveniently made in the laboratory. Therefore, some of the product produced in the Pilot Plant Development Phase of the program will be used to support testing of the biological material. For medicinal materials, human testing is conducted under the Investigational New Drug Application. Up to 20 kg of the material may be required since human tolerance and efficacy studies are conducted involving 500 to 1000 people. Data from this testing is required before a New Drug Application is granted permitting the drug to be marketed.

2.1.3 Scheduling

A typical new drug requires 3 to 6 months of work on the ground before the material is ready for initial toxicity and screening tests in animals. These initial toxicity and screening tests with live animals require at least 90 days to complete. Pilot Plant development (not involving spaceflight) usually requires 6 to 12 months. The longest time span in getting a new drug on the market is that required for clinical testing. The time span between the investigational Initial New Drug Application and the New Drug Application approval is typically 9 years.

2.2 ROLE OF THE SPACE STATION

The Space Station defines a set of limiting interfaces for the biological processing pilot plant. The Space Station sets the constraints of volume, weight, geometry, power, personnel availability, environment, etc. The impact of these interfaces can be examined by listing the equipment that is to be used in the Pilot Plant and then for each piece of equipment defining what Space Station resources are required to support that particular piece of equipment. The total of the pieces of equipment to operate a given process will then set the size or scale of the process which may be operated in the Space Station.

It is not possible to define quantitatively the various Space Station supporting functions required until a particular biological product, its associated production process and the production rate have been selected. This selection process will be discussed in Section 4.0, Space Station Requirements.

2.2.1 Biological Processes in Space

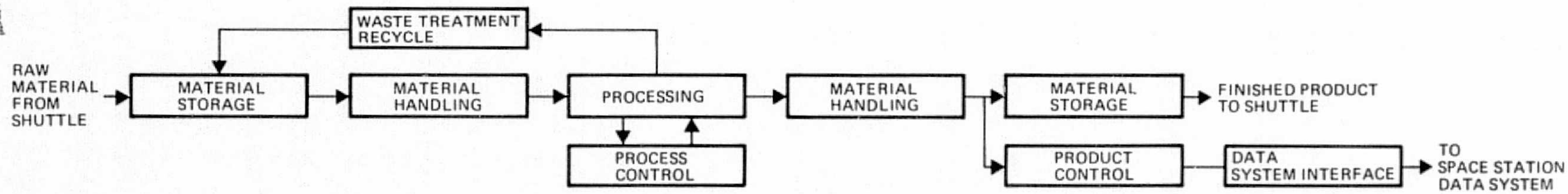
A Pilot Plant for processing biological materials in space is shown in Figure 2-3. This is a very generalized flow diagram which is applicable to almost any biological material.

In this diagram, we show raw materials coming from the Space Shuttle into a materials storage area on the Space Station. At the end of the product stream is the finished product going into material storage ready for return to the ground on the Space Shuttle.

The material storage block includes all forms of storage including cryogenic storage of very sensitive biological materials, large-scale storage of water which might be used in the process and storage of containers or liners which are associated with the process. Material storage is a more serious concern in the Space Station Pilot Plant than it is on the ground since the space involved for all activities is severely limited.

The block labeled "Material Handling" occurs at several places in the operation and may well occur internally in the processing operation. This block includes all manner of transporting materials or product between the various process steps and removing samples for test. In its simplest form, it can be a plumbing line connecting two different stages; in its more complex form, it can be manual batch transfer of material from one stage to another. In the case of liquids, manual batch transfer will involve handling the liquids in a container which confines them so that they do not escape into the operating environment of the Space Station.

The block labeled "Processing" probably includes more operations than any other. In this block the actual production of the material is carried out. In Figure 2-3 we have included a long list of possible processing operations. In some cases, these are operations which can be performed under zero-g without modification of the standard one-g equipment. For example, incubation is almost the same under zero-g as it is under one-g. Other operations become much more complex under zero-g and will involve developmental considerations before they can be applied. These include, for example, lyophilization. In this process the liquid must be confined in the container until it is frozen. It may be feasible to accomplish this by centrifugation during the freezing process.



MATERIAL STORAGE
- LIQUID STORAGE
- GAS STORAGE
- REAGENT STORAGE
- REFRIGERATED STORAGE (~4°C)
- CRYOGENIC STORAGE (~-70°C)
- CONTAINERS + LINERS

MATERIAL HANDLING/TRANSPORT
- PLUMBING INTERCONNECT
- LIQUID PUMPS
- GAS PUMPS
- MANUAL BATCH TRANSFER (IN ZERO G CONTAINER)

PROCESSING (TYPICAL
- SOLUTION PREPARATION
- INCUBATION
- STERILIZATION
  - STEAM
  - DRY HEAT, CHEMICAL
- MIXING
  - STIRRER
  - SPARGER
- HOLDING
  - PLAIN TANKS
  - TEMPERATURE CONTROLLED
- RATE FREEZER
- SOLVENT EXTRACTION
  - COUNTER CURRENT DISTRIB.
- DISTILLATION
  - SOLVENT EVAPORATION
- ION EXCHANGE
- LIQUID CHROMATOGRAPHY
- FILTRATION
- CENTRIFUGATION
  - ULTRA CENTRIFUGATION
- ADSORPTION, BATCH
- ELECTROPHORESIS, CONTINUOUS
- DIALYSIS
- DIFFUSION
- CHEMICAL REACTORS
- DRYING
  - LYOPHYLIZATION
  - OVEN
- AIR/LIQUID SEPARATION
- SAMPLING
- PACKAGING
  - VIALS
  - BAGS
  - FILL EQUIPMENT
  - SEALING EQUIPMENT
- WASTE RECYCLE
- DISPOSABLE LINERS
- EQUIPMENT CLEANING
- TOOLS

PROCESS MEASUREMENT AND CONTROL
- LIQUID FLOW RATE
- GAS FLOW RATE
- WEIGHING (MASS MEASUREMENT)
- TEMPERATURE
- PRESSURE
- PH
- LIGHT SCATTERING
- VISCOSITY
- CELL COUNTING
- LIQUID VOLUME
- SAMPLE CONTAINERS
- SAMPLING EQUIPMENT
- CONCENTRATION

END PRODUCT CONTROLS
- MICROSCOPE
- SLIDE STAINER
- CELL COUNTER
- SPECTROPHOTOMETER
- ANALYTICAL ELECTROPHORESIS
- IMMUNOCHEMISTRY
- WET CHEMISTRY
- BACTERIOLOGY
- PYROGEN TESTING

SUPPORTING FACILITIES
- ELECTRICAL POWER
- HEATING
- COOLING - COLD ROOM
- WASTE DISPOSAL
  - DRY
  - LIQUID
- HEAT REJECTION
- CREW SUPPORT
- DATA SYSTEM INTERFACE
- VACUUM

Figure 2-3. Generalized Biological Process for Space

The block labeled "Process Control" includes the measurements which are made on the process materials during the processing operation. They range from the very common measurements of temperature and pressure to more sophisticated cell counting.

The area labeled "Product Control" is intended to be the measurements that are made on the product as it emerges from the process stream. It is important to make some of these measurements in zero-g so that the appropriate process corrections can be made if there is something wrong with the product and that a lot of time will not be wasted.

In addition to the equipment associated directly with the product are the supporting facilities supplied by the Space Station. In general, these include electrical power, heating, cooling, waste disposal and other items not directly concerned with the process but essential in the support of it.

3.0 UROKINASE PROCESS

Figure 3-1 shows the process for making urokinase which serves an example process although it may not fly in itself. This is a process which is currently under development and which may be a candidate for use in space. The area of the process which is particularly suitable for use in space involves separating the urokinase-producing cells from other cells which may be present. This separation is performed by a Continuous Electrophoresis System. The performance of this system will probably be greatly improved by operating it in zero-g.

3.1 PROCESS DESCRIPTION

The entire process for making urokinase is shown in Figure 3-1. It starts with removing a kidney from a human fetus and freezing the cells until it is feasible to separate them in a Continuous Electrophoresis System. It is at this point that the operation could certainly move into space. The frozen cells are carried to the Space Station in the Space Shuttle. The cells are then thawed and removed from their freeze medium. They are resuspended in a buffer suitable for use in the Continuous Electrophoresis System.

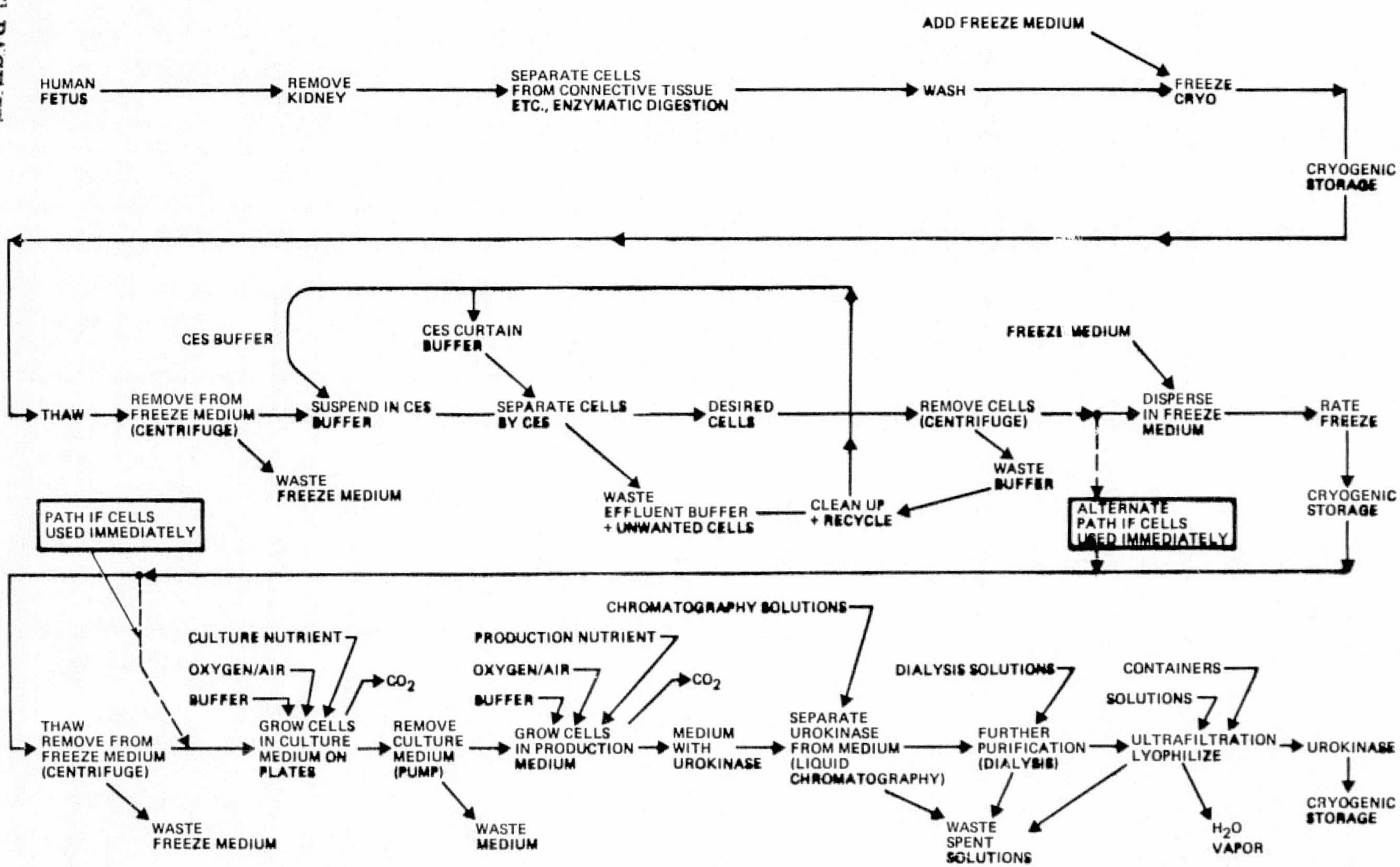


Figure 3-1. Urokinase Process

The desired cells are separated by continuous electrophoresis. The cells are removed from the electrophoresis buffer. If it is desired to return them to the ground at this point, they are frozen; if not, they are cultured to give a larger number of the desired type of cells. The culture media is removed and production medium is added. The urokinase is produced in the production medium. In further steps, the urokinase is separated by chromatography from the production medium. The urokinase is refined in further steps and eventually, after dialysis, is lyophilized in containers for transport to the ground by the Shuttle.

Referring to the urokinase process flow diagram (Figure 3-1), we note several options in the division of the total process into ground-based and space-based operations. Thus, the process in space may start with the cryogenically stored raw cell mixture, and be terminated at different stages, such that the product returned to earth is:

A. a CES-purified kidney cell fraction;
B. a CES cell fraction amplified by growth in culture; or
C. urokinase.

In Case A or B, the cell fraction is returned either frozen in a cryo-protective medium, or in a maintenance transport medium. In Case C, the urokinase is preferably returned in purified, dry form. In Case C, the operation in space includes a production culture step after the growth culture step, then protein separation, purification, concentration and lyophilization (freeze-drying).

Arguments favoring inclusion of the culturing steps in the space process are at least twofold:

1. Since the missions are of relatively long duration, the available time and equipment on board can be used for multiplying the valuable end product, and
2. Culturing in space may prove to be more efficient than on the ground, particularly for tissue cells (e.g., kidney cells) that are limited by the available area on which to grow. In space it may be possible to grow the cells on small suspended beads having a very large total surface area.

On the other hand, culturing and protein purification in zero-g demand added space, energy, weight and manpower that could be applied to other operations.

For our example, we have elected the case where the space process terminates in a dry urokinase product. This provides an opportunity to illustrate more completely the possible operations in space and the considerations that enter into the optimization process.

3.2 UNIT PROCESS OPERATIONS

In the discussion of unit process operations, it is convenient if the process flow of Figure 3-1 is characterized by discrete processing operations. Accordingly, the process flow is simplified into the following operations:

a. CES sample workup
b. CES operation
c. Centrifuge/wash
d. Growth culturing
e. Production culturing
f. Centrifuge/decant
g. Protein purification
h. Ultrafiltration
j. Lyophilization

CES = Continuous Electrophoresis System

It is to be noted that all the following unit operations must take place in a sterile environment.

3.2.1 Pre-CES Sample Workup

This comprises withdrawing successive small aliquots from the on-board store of frozen mixed-cell sample, thawing, centrifuging, washing by one or two centrifugation steps, then resuspending in CES buffer for introduction to the CES. Although the CES can run essentially without interruption for hours to days, the kidney cells are exposed to CES buffer only the minimum time necessary for CES processing. Hence only small portions of raw sample are worked up at a time, say for 30-minute increments of CES operation. To reduce manpower requirement, it may be possible to automate this frequent sampling and pre-processing step. It is assumed that the same centrifuge is used as in process steps 3.2.3 and 3.2.6.

3.2.2 Continuous Electrophoresis System (CES) Operation

The choice of design for this system and its processing capacity in space are not yet well defined. Values shown below are reasonable compromises within the performance ranges projected for two different approaches: the CPE (Continuous Particle Electrophoresis) system using a flowing buffer layer with field applied edge-to-edge, and the DLE (Deflected-Lamina Electrophoresis) with field applied normal to the cell faces.

Sample processing rate	5 ml/min
Sample solids content	2%
Desired cell fraction content	1.5% of sample solids
Delivery rate of desired cells	0.0075-g of cells/min
Concentration of delivered cells	0.2% solids
Volume delivery rate, desired cell suspension	10 ml/min
Weight, excluding cooling system	172 kg
Cooling system weight (mechanical refrigeration)	100 kg
Volume, less mechanical cooling system	0.22 m^3
Volume of mechanical cooling system	0.25 m^3
Buffer volume flow rate (buffer effluent filtered and recycled)	100 ml/min
Watts dissipation (field zone and electrodes)	400 W
Cooling power	600 W

3.2.3 Centrifuge/Wash

To reduce the exposure time of the kidney cells to the CES medium, the cells may be delivered from the CES into collector vessels containing protective additives. When a volume sufficient for a centrifuge run has accumulated, e.g., 2 to 3 liters, the cells are spun down, washed if necessary, resuspended and introduced into the growth culture. An alternative is to omit any additive from the collector vessels, but to spin down the delivered cells more frequently and in smaller volumes, say 300 ml each, adding each to the growth medium as it accumulates.

It is assumed that the centrifuge used here will be the same as that used in unit process 3.2.6. The specifications shown approximate those of a standard, high-speed refrigerated laboratory centrifuge[1]. It is recognized, however, that a space-adapted model is required and may vary from the specifications shown. In addition, the unit could use outer space as a heat sink to reduce cooling power requirement. The processing rate shown assumes a 1-hour run time. This may prove to be lower in practice.

Average processing rate	2-3 liters/hr
Power	2000 W
Weight	275 kg
Volume	0.65 m^3

3.2.4 Growth Culture

The kidney cells from the previous operation are introduced into specially designed culture chambers. Typical dimensions for commercial production[2] are 61 cm x 61 cm x 61 cm (0.226 m^3 or 226 ℓ). These contain arrays of glass plates upon which the cells will multiply up to the point of confluence, i.e., until the available plate surface is completely covered. The cells may not multiply in free suspension. As mentioned earlier, the culture medium is thermostatted and provided with gas exchange for supply of oxygen and removal of CO_2.

The generation time (or time for doubling of cell count) is two days. The limit on multiplication is 30 generations, at which time the cells begin to transform, show altered chromosome structure and lose their capacity to produce urokinase.

It is assumed that the chambers used for growth are the same that are subsequently used for the production medium. This would involve draining the chambers of growth medium and substituting production medium while leaving the cells in position on the plates.

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[1] International, Model PR-6 or Beckman/Spinco with J21 rotor, for example.

[2] Abbott Laboratories.

The required volume capacity of the growth/production chamber (or array of chambers) is variable, depending on the results of the optimization analysis in any case example.

3.2.5 Production Culture

By our previous assumptions (pending verification), the production culture unit will be the same as used for growth. Although the media and operating conditions may vary, there is the same need for oxygen supply and CO_2 removal. The cells do not significantly multiply during the production cycle. The medium is instead designed to maximize the urokinase production. The maximum useful life of the production culture is 40 days, at which time the accumulation of urokinase and/or other products results in a dropoff of urokinase production rate. The cells of the exhausted production batch cannot be again used. The supernatant is withdrawn for removal and purification of the urokinase. The culture chamber is cleaned for reuse.

In the subsequent case example calculations we assumed that the mass concentration of urokinase at maturity of the production culture is 1.5% and equal to the average mass concentration of cells in the culture.

3.2.6 Centrifuge/Decant

The liquor from the production culture is centrifuged in consecutive batches, at relatively high speed if necessary, to remove residual cells, debris and particulates of the culture broth from the supernatant. The centrifuge is that used for process steps 3.2.1 and 3.2.3.

3.2.7 Protein Purification

This comprises a selection of a set of established procedures, adapted by their sequence and the choice of operating conditions to isolate the protein of interest. This may include steps of concentration, precipitation by salt or solvent addition, adsorption, ion-exchange or other types of chromatography and electrophoresis. Substantial manual handling may be involved and a requirement for convenient transfer, mixing, filtration, etc., of relatively large volumes (of the order of tens of liters at a time) in zero-g. Relatively large solvent volumes may be required. An adequately sized work station must be provided, and a balance struck between overall equipment bulk on the one hand, and excessive time consumed on the other hand in working with repeated small batches.

The specific steps involved in urokinase purification are not defined at this time. Values shown below are rough estimates based on general laboratory practice. A more or less standard complement of devices for this purpose will be used, adapted as necessary for operation in space.

Equipment weight (including tanks, pumps, etc.)	100 kg
Weight of purification solvents, media, chemicals, etc.	1 kg/ℓ production liquor
Equipment volume (less volume of purification media)	0.7 m^3
Volume of purification media	0.001 m^3/ℓ production liquor
Power	200 W

3.2.8 Ultrafiltration

This operation desalts the purified protein solution and subjects it to concentration prior to lyophilization. The process is a form of reverse osmosis, the protein solution being applied under pressure to one side of a semipermeable membrane array, and a circulating wash solution applied to the other side. A representative apparatus applicable to the process is the Bio-Rad Model DC30 Hollow Fiber System.

Processing capacity	20 ℓ/day
Weight	15 kg
Volume	0.15 m^3
Power	200 W

3.2.9 Lyophilization

This step freeze-dries the protein concentrate in small vials. Provision for automatic capping is usually integral within the apparatus. The product is then ready for low temperature storage and return to earth. Lyophilizers of larger than laboratory scale carry a heavy burden of pumping and refrigeration equipment. A substantial saving in weight and power for a space-borne operation can be achieved if the external environment can be used as a heat and vapor sink. The following values apply to a system with a processing capacity of 3.5 liters per day:

	Weight (kg)	Vol. (m^3)	Power (kW)
System using space as vapor and heat sink	120	0.5	0.2
System using mechanical pump and refrigeration	400	0.7	3.5

4.0 SPACE STATION REQUIREMENTS

The Space Station requirements relative to biological processing evolve from an initial R&D Laboratory for general investigations to a pilot plant for a specific material (urokinase as the case example). This evolution represents a spectrum of requirements for Space Station subsystem design.

4.1 R&D LABORATORY

The R&D Laboratory provides equipment to conduct exploratory or diagnostic experimentation on biological materials. The processing equipment is similar to that found in the pilot plant only smaller in scale. In addition, certain ancillary analytical equipment is provided.

A typical complement of equipment is shown in Table 4-I. The equipment volume, weight and power requirements were derived from Reference 3.

4.2 PILOT PLANT

The pilot plant requirements are derived by considering a case example. The case example contains the unit process operations discussed in Section 3.0 and considers a 90-day interval between Shuttle resupply flights.

It should be emphasized again that while the pilot plant requirements are being derived based on the urokinase example, the equipment complement is appropriate for other material processing where separation and culture growth are the main processes involved.

4.2.1 Process Time Line

Batch-type chemical processes comprise a sequence of unit operations. Each is usually associated with an item of processing equipment. Successive unit operations may overlap in time, and in any given train of equipment a new production cycle may be started at the beginning of the train

Table 4-I. R&D Laboratory Equipment

Equipment	Unit Weight (kg)	Envelope Vol. (m^3)	Peak Power DC (kW)	Peak Power AC (kVA)
Research Electrophoresis Unit	45.0	.045	1.0	--
Preparative Electrophoresis Unit	10.0	.010	.10	--
Fraction Collection Unit	2.0	.001		.002
UV Absorption Scanner	2.5	.008	--	0.10
UV Source	4.5	.004	--	0.10
pH Monitor	6.0	.012	--	.05
Glove Box	15.0	.030	--	--
Centrifuge	23.0	.120	--	.12
Mech Mixing Unit	2.0	.001	--	.10
Incubator	26.0	.118	--	.08
Lyophilization Unit	40.0	.041	.20	--
Dialysis Unit	4.5	.027	.10	.10
Liquid Syringe Pump	7.5	.001	.02	--
Metering Pump	2.5	.001	.10	--
Particle Counter	25.0	.150	.10	.10
Culture Tank	1.5	.01	.10	.10
Microscope	15.0	.03	.05	.05
Refrigerated Storage Unit	57.0	.277	--	1.0
Bio-freezing and Storage Unit	46.0	.122	--	.20
Buffer Supply Tank	2.0	.002	--	--
Electrolyte Supply Tank	2.0	.002	--	--
Waste Liquid Tank	1.0	.004	--	--
Gas Elimination Unit	2.0	.005	.05	--
Vacuum System	1.0	.001	.05	--
Fluid Cooling/Refrig. Unit	54.0	.147	3.0	--

before one or more prior batches have completed their passage through the train.

Let us assume, as in the Space Station, a limited time frame (the mission period) within which a batch production process is to be carried out, either as a single or multiple cycle, and in the latter case either as overlapping or consecutive cycles. The total mission time T can then be expressed as the sum of a series of terms:

$$T = n\ T_{max} + T_p + T_f + T_s \tag{1}$$

where T_{max} is the duration of the longest of the unit process time, n is the number of production cycles within the mission time, T_p is the sum of unit process times preceding T_{max} in any cycle, but excluding overlapping time segments. T_f is the sum of unit process times following T_{max} in any cycle, but excluding overlap, and T_s is the sum of all time gaps, including delay in startup of the first cycle, gaps between process steps in any single cycle, total of gaps between successive T_{max} periods where $n > 1$, and time between the end of the last unit operation in the mission and the end of the mission itself.

The preceding assumes a single train of equipment. Where two or more trains are operated in parallel independently, they are treated as separate systems. In any given train, however, any unit process device may, for example, be doubled up for parallel operation and the analysis accommodates this modification.

T_{max}, representing the production-culturing step, is taken in our analysis as a constant, equal to 30 days. This derives from the known useful life of 30 to 40 days for this culture, and the desirability of using the culture for maximum yield and with minimum handling associated with repeated filling, cell removal, etc.

T_s is taken in our case example as twelve days. This is a reasonable estimated value representing combined operator time for production startup, termination and delays between unit operations for handling, material transfer, etc.

The unit operation times that makeup T_p and T_f are, in most cases, dependent on the volume of material processed through the system, since each unit operation equipment (even when doubled up, etc.) is able to process material only at a given average rate.

Individual terms associated with each of the unit process operations are summarized in Table 4-II.

$\underline{K}$ values such as $\underline{K}_B$, $\underline{K}_C$, etc., are proportionality constants.

$\underline{G}$ is the gain, or biological multiplication of the number of urokinase-producing cells during the growth stage.

$\underline{V}_p$ is the volume, in liters per production cycle, of production culture liquor. Multiplied by the number of cycles, this provides a measure of total urokinase produced during the mission, since the urokinase will be a constant fraction of the liquor volume when the culture matures. For convenience, V_p is used in our calculations as a common reference base for production scale.

Considering in turn the derivation of the process times listed in Table 4-II, we note:

4.2.1.1 CES Sample Workup

The independent time contribution of this process is negligible, since it largely overlaps Step B, the CES operation. This occurs because the raw, frozen cell mixture must be thawed and resuspended in small increments, say every 30 minutes to one hour (if not on a continuous automated basis), for feeding to the CES. This avoids long exposure of the cell sample to the CES medium, which may be damaging.

4.2.1.2 CES

The expression for T_B states the required CES operation time per cycle is proportional to production culture volume. It is based on CES processing capacity and sample characteristics listed under 3.2.2. It assumes a 1.5% packed cell fraction in the mature production liquor. The gain G, the cell multiplication factor during growth culturing, reduces the processing rate requirement relative to the subsequent volume of production liquor. N_{CES} is the number of CES systems, in the event that two or more are used in parallel.

Table 4-II. Unit Operation Time Requirements

		Required 24-hr Days, T, per Production Cycle	Days Req'd. Case Ex.
A.	CES Sample Workup	T_A included in allocation for T_B	Nil
B.	CES	$T_B = K_B V_P = (\frac{0.378}{N_{CES} \times G}) V_P$	1.39
C.	Centrifuge/Wash	$T_C = K_C V_P = (\frac{1}{72GN_C}) V_P$	0.51
D.	Growth Culture	$T_D = (2 \log G)/\log 2 = 6.64 \log_{10} G$	8.00
E.	Production Culture	$T_E = 30$	30.0
F.	Centrifuge/Decant	$T_F = K_F V_p = (\frac{1}{72 N_C}) V_P$	0.82
G.	Protein Purification	$T_G = K_G V_P = 0.05 V_P$	3.00
H.	Ultrafiltration	$T_H = K_H V_P = 0.05 V_P$	3.00
J.	Lyophilization	$T_J = K_J V_P = 0.021 V_P$	1.20
K.	Distributed Time	T_K = Constant	12.00

4.2.1.3 Centrifuge/Wash

The expression shown assumes a processing capacity of 3 liters per 1-hour run. N_C is the number of centrifuges, if two or more are used in parallel.

4.2.1.4 Growth Culture

The time required is determined by the number of doublings (2-day each) occurring in cell count before changeover to the production medium. We show later that there is an optimum value of G, given a set of system values assumed in any case example.

4.2.1.5 Production Culture

For the present analysis, and for reasons mentioned earlier, this is fixed at 30 days, a value near the maximum useful life of the production culture.

4.2.1.6 Centrifuge/Decant

Assumptions here are the same as for 4.2.1.3 and the expression differs only in lacking G in the denominator.

4.2.1.7 Protein Purification

The expression assumes that 20 liters of production culture supernatant can be processed per day. Assumed also is manual assistance by one operator, the use of small scale batch production equipment and the doubling up or paralleling of operations as appropriate to this production rate.

4.2.1.8 Lyophilization

Assuming a processing rate of 3.5 liter/day, and concentration of the protein from 1.5% to 20%, the proportionality factor K_J is given by:

$$K_J = \frac{1}{(20\%/1.5\%) \times 3.5 \text{ liter/day}} = 0.021$$

4.2.2 Case Example

Referring to Equation (1) in Section 4.2.1, and Table 4-II, we note that

$$T_P = V_P\ (K_B + K_C) + T_D \qquad (2)$$

$$T_f = V_p\ (K_F + K_G + K_H + K_J) \tag{3}$$

and $T_{max} = T_E$ (4)

Combining equations (1) with (2) to (4),

$$T = n\ T_E + V\ (K_B + K_C + K_F + K_G + K_H + K_J) + T_D + T_S \tag{5}$$

Now solving (5) for V_p, the quantity of production liquor that can be generated and processed per production cycle during the mission, we have:

$$V_p = \frac{T - n\ T_E - T_D - T_S}{K_B + K_C + K_F + K_G + K_H + K_J} \tag{6}$$

and values for T_E, T_D, K_B, K_C, K_F, K_G, K_H and K_J as shown in Table 4-II. We have then from Equation (6),

$$V_p = \frac{(T - n30) - (6.64 \log_{10} G) - T_S}{\frac{0.378}{GN_{CES}} + \frac{0.0139}{GN_C} + \frac{0.0139}{N_C} + 0.05 + 0.05 + 0.021} \tag{7}$$

By examination of Equation (7), one can see that there will be an optimum value of V_p with respect to G sinceG appears in both the numerator and denominator. Furthermore, the optimum gain factor, G, will most likely vary with N_{CES} and N_C.

For the case example, the following conditions are assumed

T = 90 days (time between resupply flights)

n = 2 (number of production cycles per resupply flight)

T_s = 12 days (summation of unit process gap time)

After substituting into Equation (7), the following urokinase process algorithm is obtained

$$V_p = \frac{2.71 - \log_{10} G}{\frac{0.057}{N_{CES}G} + \frac{0.0021}{N_C G} + \frac{0.0021}{N_C} + 0.0182} \tag{8}$$

An evaluation of Equation (8) is shown in Table 4-III. The evaluation is based on the constraint that N_{CES} and N_C are equal. However, this need not be the case and, in fact, there probably exists an optimum N_{CES}/N_C ratio.

Table 4-III. Urokinase Liquor Production Optimization

<u>Analytical Expressions</u>

$$V_P = \frac{2.71 - \log_{10} G}{\frac{.057}{N_{CES}G} + \frac{.0021}{N_C G} + \frac{.0021}{N_C} + .0182} \qquad T_D = 6.64 \log_{10} G$$

T_D	G	$N_{CES}=1=N_C$ V_P	$N_{CES}=2=N_C$ V_P	$N_{CES}=3=N_C$ V_P	$N_{CES}=4=N_C$ V_P	$N_{CES}=5=N_C$ V_P
0	1	34.1	55.5	70.2	80.9	89.1
2	2	48.3	70.8	83.8	92.3	98.3
2.6	2.5				93.9	99.1*
3.2	3			87.7	94.5*	99.0
4	4	60.1	79.2	88.5*	94.1	
4.6	5		80.1*	88.0		
5.2	6		80.0			
6	8	65.3				
6.3	9	65.4*				
6.6	10	65.2				

N_{CES} ≡ Number of electrophoresis units; N_C ≡ Number of centrifuges.

G ≡ Ratio of cells at growth step initiation to cells at end of growth step.

T_D ≡ Duration of growth step, days.

V_P ≡ Volume of urokinase liquor produced, liters, per production cycle.

*Indicates optimum gain factor for liquor production.

Table 4-III shows that as the number of electrophoresis units increases, the optimum gain factor decreases. At the same time, the amount of liquor produced increases. One might expect, therefore, that there is an optimum number of electrophoresis units to product an optimum amount of urokinase liquor. For example, if one assumes that the cost of production is proportional to the weight of the production facility, one can select the number of electrophoresis units which will produce the maximum liquor volume per unit of facility weight.

Facility weight data contained in Table 4-IV was used along with the information in Table 4-III to determine the specific liquor volume (liquor volume per unit of facility weight) as a function of the number of electrophoresis units. The results are shown in Figure 4-1. The maximum liquor volume curve is the production volume which results from the optimum gain factors.

The cell production rate used for this exercise considers that the cell population doubles every two days. Thus, the duration of the cell growth step in the overall process is expressed in terms of the gain factor as

$$T_d = 2 \log_{10}G/\log_{10}2, \text{ days}$$

This expression combined with the expression for the urokinase liquor production results in an optimum cell growth duration as shown in Figure 4-2.

If one assumes a 1.5 percent urokinase content in the liquor, and a recovery of two-thirds in the purification process, the urokinase production per liter of liquor is 0.01 kg/liter. This translates into 1.8 kg of lyophilized urokinase per mission cycle.

Table 4-IV. Bioprocessing Equipment

	Weight (kg)	Volume (m^3)	Power W (peak)
CES: Cooling system, mechanical	100		
Cooling reservoir fluid	20		
Cell and hydraulics incl. pumps	100	0.22	1500
Buffer reservoir and flow control	25		
Power supplies	15		
Collection system, filled	12		
Buffer reconditioner	45	0.04	100
Centrifuge, refrigerated	275	0.65	2000
Ultrafiltration system	15	0.15	200
Lyophilizer, using space as vapor and heat sink	120	0.5	200
(Lyophilizer, using mechanical pumps and refrigeration)	(400)	(0.7)	(3500)
Low temperature refrigerator	70	0.12	350

Based on the foregoing analysis, it would appear that the space station bioprocessing facility should contain more than one electrophoresis unit (likewise more than one centrifuge). The optimization results presented herein represent optimization of the number of electrophoresis units against only one set of variables. Since the true optimization which considers all variables is beyond the scope of this study, it is recommended that the space station study consider the bioprocessing facility described to contain three electrophoresis units and centrifuges.

It should be noted that the resu'ts of this optimization are based on two production cycles during a 90-day mission. Examination of Equation (7) shows that the maximum urokinase liquor production will occur if only one production cycle is used provided, of course, that there are no physical constraints or limits on the cell growth step of the process. However, even for one production cycle, multiple electrophoresis units are required to optimize production.

4.2.3 Space Station Resource Requirements

Required equipment weight, volume, unit apparatus power and operating time for the bioprocessing pilot plant are shown in Table 4-V.

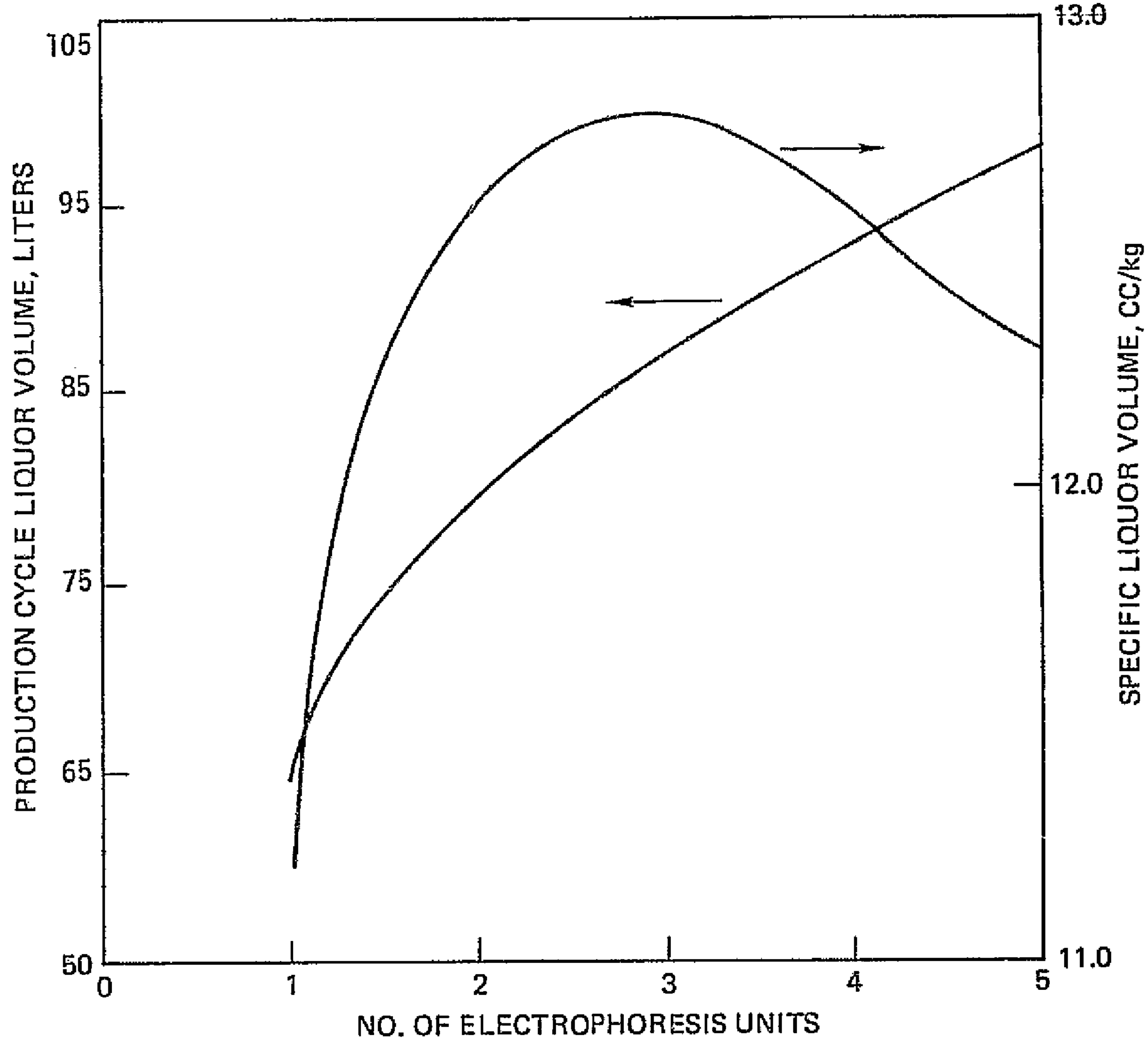

Figure 4-1. Electrophoresis Unit Optimization

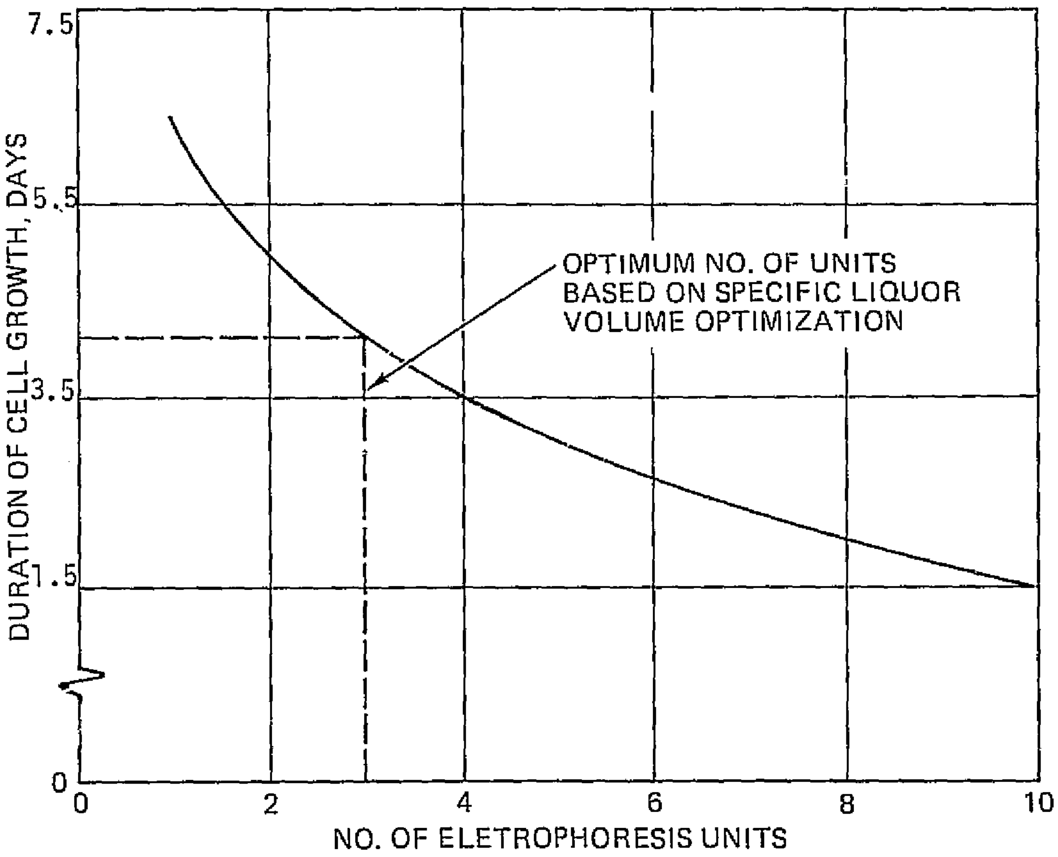

Figure 4-2. Cell Growth Process Optimization

Table 4-V. Bioprocessing Pilot Plant Requirements

Equipment	Weight (kg)	Volume (m^3)	Peak Power (kW)
Continuous Electrophoresis System (CES-3)	620	0.66	4.50
Buffer Reconditioner	45	0.04	0.10
Centrifuge, Refrigerated (3)	825	1.95	6.00
Growth/Production Culture Chamber	155	0.61	0.70
Protein Purification	205	0.95	0.27
Ultrafiltration System	20	0.20	0.27
Lyophilizer	160	0.68	0.27
Low Temperature Refrigerator	80	0.16	0.35
TOTALS	2110	5.25	

Requirements specified in the table originate with the case example discussed. Any other set of case variables will change the noted requirements.

The CES requirements are for three units as are the centrifuge requirements. The requirements for cell growth/production culture chambers consider the output from three CES units as does the other processing equipment whose weight and volume are directly related to production liquor volume.

Figure 4-3 shows the processing schedule. The required payload specialist time in the case example is estimated to be 25 man-days per mission cycle. The unit operations requiring specialist time are: (a) those involving sample workup and CES operation prior to the culturing operation, (b) the period where the two cycles overlap and (c) the final post-culturing operations. This is just the manpower required to run the process and does not include R&D product analysis or production process optimization.

A mission power timeline is shown in Figure 4-4. A sustained power of 13.7 kW is required for approximately 1.4 days during the second cycle CES operation that overlaps the first production cycle. Total energy requirements for the two production cycles are 9200 kWh. Average power for the processing equipment for the mission cycle is 4.2 kW.

5.0 ACKNOWLEDGMENTS

Major portions of this report were contributed by the Advanced Technology Operations of Beckman Instrument Instruments Inc., Anaheim, California.

6.0 REFERENCES

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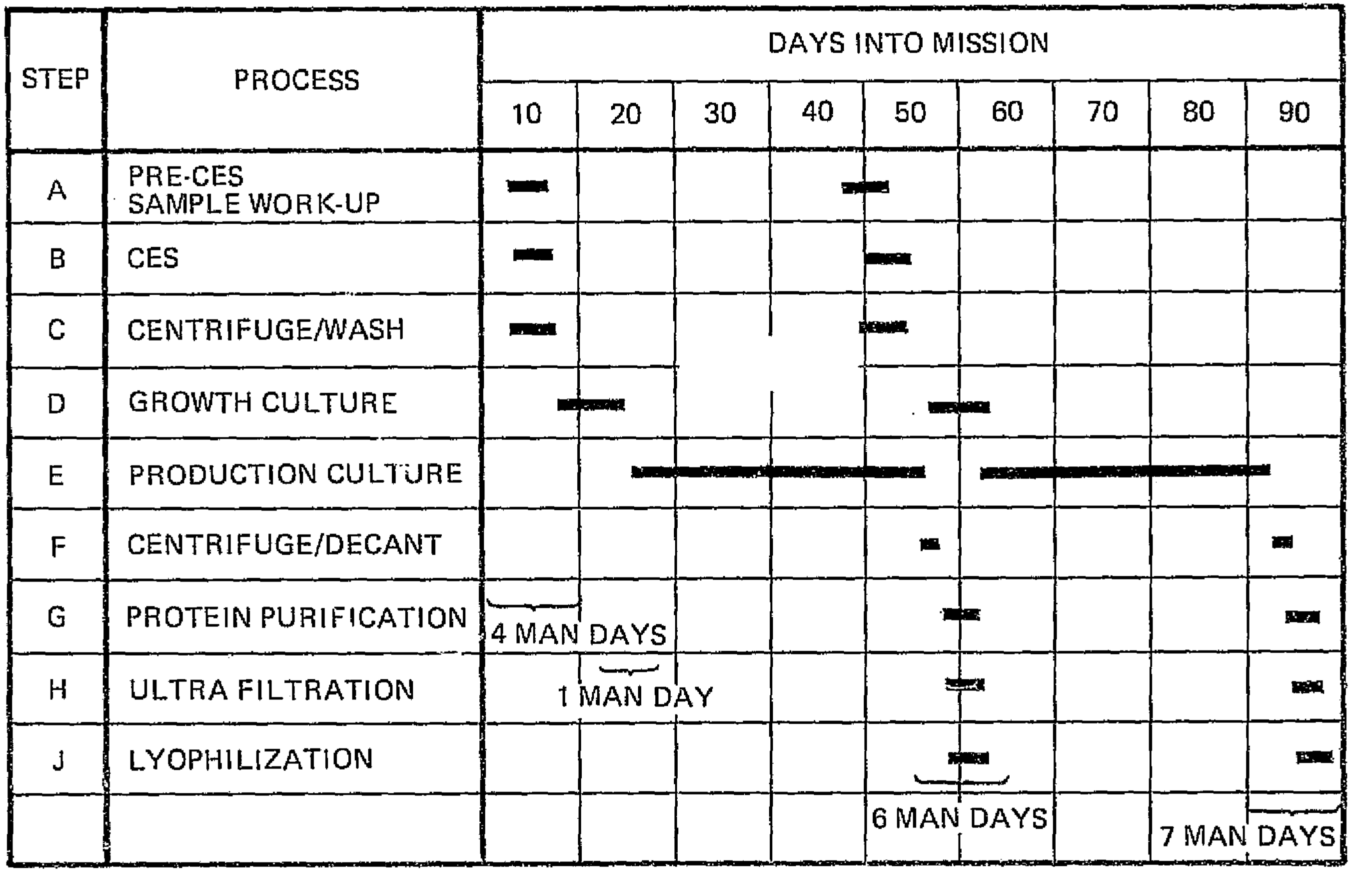

Figure 4-3. Processing Schedule Case Example
Estimated Mission Bioproduct Output

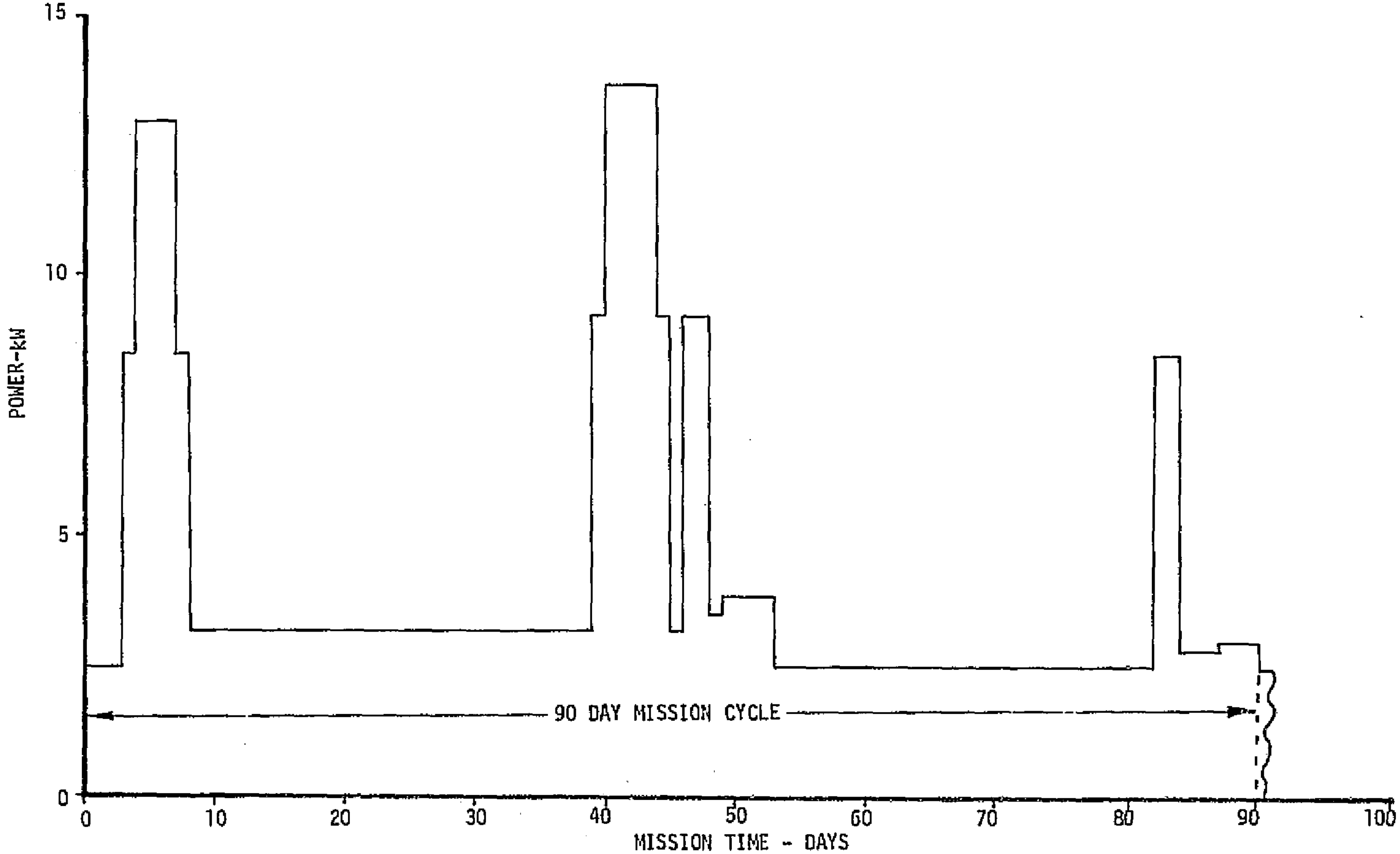

Figure 4-4. Bioprocessing Pilot Plant Power Timeline

CASE STUDY

INORGANIC PROCESSING

FIBER OPTICS GLASS PREFORM
FOR USE IN
COMMERCIAL COMMUNICATIONS TRANSMISSION
SYSTEMS

1. INTRODUCTION

A case study of glass preforms to be produced in space is presented in this report. As a case study, its sole purpose is to derive resource requirements for which Space Station subsystems can be developed. The resource requirements are developed from a simplified process flow for producing high purity fused silica glass by containerless melting in the low gravity environment of the Space Station.

The potential "product" will be a fiber optic glass preform coated on its surface with a glass cladding. This preform will be brought to Earth for cladding with another glass for structural purposes and then drawn into fibers for use in fiber optic transmission systems.

1.1 CASE STUDY BACKGROUND

The Space Station Study, Part 1, resulted in the identification of several areas of endeavor that could benefit from the resources potentially available via the Space Station. Among the several activity candidates is the field of space processing. Space processing encompasses the area of research into and development of materials which uniquely benefit from processing in space; the ultimate objective being products of significant economical value.

During the initial phase of the Space Station study, three areas of activity were identified for further examination as case studies. The three areas were production of biological products (e.g. the enzyme urokinase), production of electronic materials (e.g. high purity silicon ribbon) and production of special high purity glass (e.g. preforms for fiber optics). It is the latter of these three examples that is the subject of this glass case study document.

1.2 CASE STUDY OBJECTIVE

The important aspect of the glass case study is to develop requirements that are both reasonable and representative of glass product production. Thus, the selection of a fiber optic application for case study is not the important feature of the example. Rather, it is the fact that the fiber optic example represents a batch glass process and the process is

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representative of other potential glass products which would benefit from space processing (e.g., high CaO content lasing glasses). The use of a fiber optic case results from the on-going interest in fiber optics for communications systems.[1,2]

The results of the case study will aid in establishing Space Station requirements for the power generation and distribution subsystem, the thermal environmental control subsystem, the habitability subsystem and the control subsystem. The requirements derive from the power, energy, waste heat, volume and weight parameters established by the fiber optic glass production equipment and process.

1.3 CASE STUDY ELEMENTS

The approach taken in the example case study involves separation of the activity into several segments. The first segment consists of an examination of the new product development process in the glass industry to highlight where Space Station activity would logically enter the process.

In addition, other aspects of current technology as related to the case study were developed via a literature search. Methods currently in use for fiber production and related parameter limitations were investigated. This activity established the background for developing the approach to be taken in producing the fiber optic glass preforms in the Space Station supported facility.

The second segment of the study was concerned with the in-space glass preform process. Once an approach for preform production was established, the individual processing steps were defined in terms of equipment and equipment requirements. Wherever possible, the equipment requirements were parameterized in terms of a product variable (mass, size, etc.). Relationships were developed for the time required to accomplish each unit processing step as a prelude to total process analysis.

The final segment of the case study dealt with establishing the Space Station requirements. The case example developed uses a total process algorithm from which pilot plant requirements were derived. In addition, requirements for an R&D laboratory which would be a precursor to the pilot plant were specified. The background data for the R&D laboratory were mainly obtained from Reference 3.

2. FIBER OPTICS

The fiber optic field is becoming increasingly interesting for communications and data transmission because of the large bandwidths available at lightwave frequencies. Accordingly, space production of high purity fiber optic material was chosen for the case study. This section of the case study discusses new product development process within the glass industry, where Space Station fits into the development scenario and how fiber optics benefit from space production.

2.1 NEW PRODUCT DEVELOPMENT*

In any industry there is no standard, cookbook type of approach to new product development and the glass industry is not unique in this regard. Thus many of the concepts, philosophies, etc. discussed here will be applicable to about any industry. However, in elaborating on the general concepts, specific examples, perhaps unique to the glass industry, will be presented.

In discussing new product development, one will encounter varying thoughts as to what constitutes a new product. For the sake of this discussion, a new product will be defined as an item which has never been produced on a commercial basis in the past and requires a non-neglibile amount of scientific and/or engineering work to be fabricated.

In view of the above new product definition, new product development will be analyzed in concert with the initial type of scientific and/or engineering activity associated with the new product development. Four categories of technical endeavor are generally used to distinguish the different types of new product research and development.

1) Unoriented Basic Research (UBR).Research primarily aimed at understanding some phenomenon with no direct product or material development goal in mind. This type of activity generally constitutes pure research

2) Oriented Basic Research (OBR). Research directed primarily at ascertaining the mechanism of some process or material behavior, but with a material improvement or product development goal in mind.

3) Applied Research (AR). Research primarily aimed at materials and/or process improvement, with understanding playing a secondary role; a specific product goal is usually well in mind.

*Section 2.1 was provided by Owens-Illinois.

4) Development and Engineering (D&E). Research aimed primarily at product fabrication and component integration; at lease initial stages of materials development completed.

Clearly, research programs cannot be (in all cases) unambiguously classified and it is recognized that a good deal of overlap may exist among the above mentioned categories. Nevertheless, it will be useful to rely on the classifications in order to discuss various paths of new product development. As one will see in the following discussion, only the latter two research categories are likely candidates for Space Station activity. The first two categories will most likely be confined to terrestrial laboratories or short duration orbital flights such as characterized by the Shuttle-supported Spacelab.

Route 1, UBR. UBR rarely leads directly to a new product. However, many of the revolutionary product innovations in the past 30 to 40 years have originated from UBR. Figure 2-1 traces some possible scenerios of the the development of a new product initiating from a UBR study. Most often a successful UBR program will suggest a new OBR program, and/or an AR program, or perhaps property measurements depending on the outcome of the initial research program. The solid lines in Figure 2-1 outline the most probable transfer for a given outcome. The most noteworthy features of new product development via Route 1 may be summarized by the following points: a) no concept of product formulated at inception of research; b) highest risk involved since program may be short-circuited at many points; c) perhaps longest "lag time" from inception of research to finished product; d) potential "payoff" could be enormous; e) since "basics" understood, have information to fall on if difficulties arise in some later stages of RD&E; f) usually employed only by large, high technology-oriented companies; g) requires highly talented R&D personnel and technical managers; h) requires a very well coordinated but semi-autonomous RD&E effort.

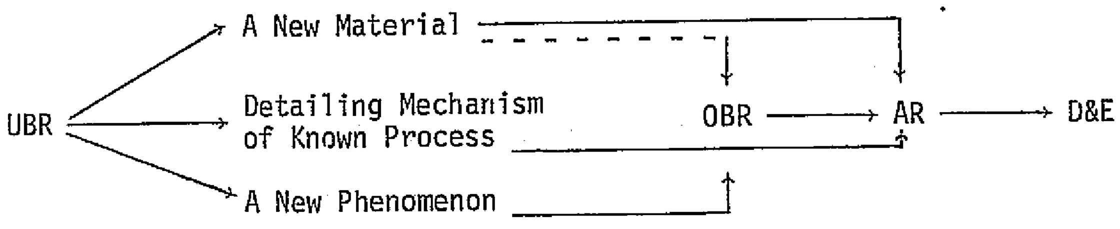

Figure 2-1. Unoriented Basic Research

An example of this route in the glass industry might be a study of the relationship between liquid-liquid phase separation in glasses and crystal nucleation. The key to a full understanding of glass-ceramic formation lies in the complete elucidation of the role which liquid-liquid phase separation plays in glass-ceramic formation. Such a study could lead to the ability of precisely controlling the microstructure and composition of glass-ceramics. This, in turn, would allow one to produce glass-ceramics with certain desired physical properties, since the latter depend crucially upon the composition, crystal density, and crystallite size. Product development could occur rapidly after such a research program if materials with unexpected, but very desirable, physical properties were developed, e.g., very hard material, or any strong, or exceptional corrosion resistance, etc. Mostly like the results of such work would be documented and any new materials, with its property measurements, would be filed for possible future retrieval. How such information could be used will be discussed later.

Route 2, OBR. Route 2 is quite close in spirit to Route 1 and hence will not be outlined in detail. We shall merely indicate some features which distinguishes the latter two routes. At the inception of the OBR, a product notion or material development or improvement may be envisioned. Thus, the risk of this procedure of product development is somewhat diminished. On other hand, as a direct consequence, the probability of finding a totally new phenomenon is also reduced. A successful OBR program will usually lead to an AR study although in rare instances, could lead back to an UBR study.

With regard to glass research, the difference from Route 1 would be in how one exercised the options with respect to the choice of glass forming systems studied, i.e., one would choose a system (or systems) with a probability of providing the combination of physical properties desired, e.g., one could explore the possibility of replacing very expensive ferroelectric single crystals with amorphous systems which could be inexpensively formed. In choosing the composition to study, one would certainly include the components of the single crystal plus components which are good glass formers and which did not violently react with the other components.

Route 3, AR. Perhaps most industrial new products result from an applied research program. Some of the possible outcomes of an applied research program are shown in Figure 2-2. In addition, some of the paths which one could follow in proceeding to a new product are shown. Products produced in this fashion are probably characterized by the following features:

- Results of research more predictable than outcome via Routes 1 and 2
- Product concept should be more well defined at outset, since research outcome will hold fewer "surprises"
- Smaller risk involved if product idea well conceived
- Coordination with D&E important to fulfill final goal
- Interaction with OBR essential if one has any hopes of producing "major breakthrough"
- Lag time shorter to product
- Competition from other firms probably greater.

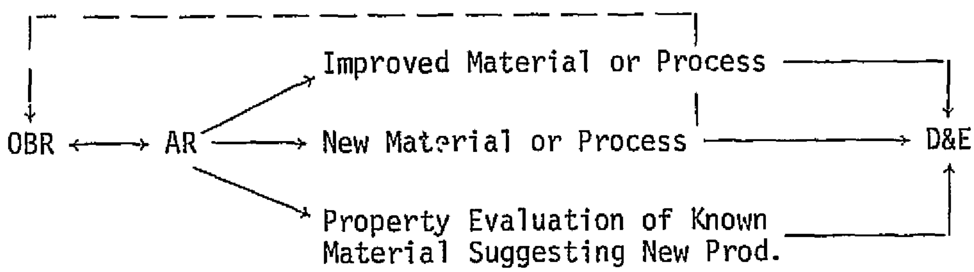

Figure 2-2. Oriented Basic Research

The major difference in this route is that the definition of the properties required become very specific, e.g., a specific application might require a lead-free, low melting dielectric glass. Lead free is specified because the application requires melting in a reducing atmosphere. Although lead containing glasses are excellent dielectric glasses, when melted in a reducing atmosphere the lead oxide is reduced to metallic lead, thereby ruining the dielectric properties. The design criteria in this case would be stated in terms of only a few properties, i.e., the glass should readily flow ($\log \eta = 2$)*at 100^{o}C and it should have a dielectric constant >5.0 and a loss coefficient of <.01 percent at 150^{o}C.

*η is material viscosity. Log η = 2 is usually referred to as "log 2" temperature.

One would first check the properties of glasses previously studied (refer back to comments made on filing properties for future retrieval). If such a search does not produce a glass with properties near those needed, the first mental step the glass thechnologist might take is to review the properties of the major oxide glass forming systems, silicates, borates, phosphate, germanates, and decide what major system to study. In this case, he might choose the borate system to get a low melting glass or he might decide to look at borate-silicate mixtures to get more durable low melting glasses. He will then look for modifiers whose concentrations can be adjusted, along with those of B_2O_3 and SiO_2, to design to the properties needed. He might examine phase diagrams to ascertain some clues with regard to obtaining low melting eutectic composition.

Finding sufficiently low melting eutectic compositions (in this case, below 1000^oD) he might melt those eutectic compositions and measure their log 2 temperatures, dielectric properties, and as a check, their liquidus temperatures. If he suspects the chemical durability of some of the higher borate glasses is poor, he might also measure chemical durability.

The study may then be concentrated on one or two eutectic compositions, and a second series of melts containing small amounts of single modifiers might be melted. This second series is intended to indicate how modifiers affect the properties of these glasses. Typical questions to be answered might be: does Al_2O_3 increase the log 2 temperature drastically? Does it improve chemical durability significantly? Does it change the electrical properties?

After the affects of modifiers have been determined in four component glasses, the glass technologist will combine modifiers in five or six component glasses to try to get the benefits of each modifier. One would start by picking the single modifier which most likely will increase chemical durability, the one that will most likely lower the log 2 temperature, the one that will most likely lower the loss coefficient, etc. Other combinations would also be tried with the hope of observing some synergistic effects.

Thus, the process of designing a glass with specific properties might involve a series of steps from simple ternary compositions to complex multicomponent compositions. If the desired properties are only obtainable through compositions that cannot be produced in the terrestrial environment because of density segregation, limited glass formation region or crucible contaminate sensitivity, a Space Station facility enters the picture.

As illustrated in the figures outlining the various paths of product development, the final step is D&E or (if you prefer) scale-up from laboratory to production. The decision to take the final step is based on economic considerations, customer reactions, availability of facilities to manufacture the new product, assessment of potential competitors' ability to compete and other nontechnical features.

With regard to the scale-up of a new glass composition, we must overcome the problems associated with going from melting glass in 1 to 10 pound crucibles (which are typical for the new composition research described above) to melting glass in quantities carying from 100 lbs/day to 200 or 300 tons/day. In order to bridge the gap between the small melts used in research and the large melts used in production, pilot scale melting and forming operations are of ten utilized. For example, if the glass is going to be melted in a tank which yields 100 tons/day of glass, a pilot scale tank holding 5 to 10 tons of glass will be utilized to test the corrosiveness of the glass on refractories and other properties. Forming equipment placed in front of the pilot scale tank will be used to measure the forming properties of the glass.

Often the initial forming trials will be done by hand. Glass will be gobbed out of the tank in the traditional manner and pressed, blown or drawn by hand to get some idea of how well the glass can be worked. In general, the forming characteristics of the glass will have been determined in the initial research program by measuring the various viscosity points of the glass. However, the tendency to devitrify or the degree to which the glass might wet other materials or any of dozens of other problems are often first observed in handworking.

Using handworked samples, molds or other process equipment can be developed. If the glass presents a difficult forming problem, its composition might once again be modified at this stage in order to yield a glass having the necessary end product properties and also the necessary forming properties. For example, if the glass becomes rigid too quickly upon cooling, different oxides will be added in order to lower the fiber softening point of the glass, giving it a longer glassworking range.

After the completion of satisfactory pilot-trials the glass would be introduced into a full-scale furnace and again adjusted to compensate for the different conditions in that larger furnace. For example, the residence time in the larger furnace might be longer than in the pilot scale furnace and the composition might have to be adjusted to allow for increased volatilization losses. Composition adjustments may also be needed in order to compensate for forming needs.

The series of steps just discussed are shown in Figure 2-3 along with the other activities associated with new product development. The initial activity might start with an identified glass from an OBR program or a definition of a new product requirement. The next set of activities are of a preliminary development nature. The glass property and melting/forming studies would take place in either the Space Station R&D laboratory or the Shuttle/Spacelab facility. After the initial decision point, all on-orbit activities denoted in Figure 2-3 would take place in the glass forming pilot plant module connected to the construction base (Space Station).

2.2 GROUND BASED TECHNOLOGY

The progress made in fiber optic technology over the last 10 years has been significant. From light attenuations on the order of several thousands of db/km, improvements in fiber technology have reduced losses to the order of 2 db/km in fused silica. A laboratory process wherein a chemical vapor deposition technique is used to produce cladded fused silica fibers results in the loss characteristic shown in Figure 2-4[4]. The lower limit to light attenuation is set by Rayleigh scattering. A Rayleigh scattering characteristic[5] for pure fused silica is shown in Figure 2-4 for comparison with the CVD process. The absorption peak exhibited by the CVD processed fiber at approximately 930 nm is a result of water contamination (OH^-) in the ppm range and the increased loss above Rayleigh scattering is due primarily to contaminants[4]. By comparison then, one can see that the loss improvement to be gained by obtaining ultrapure fiber optic material varies from approximately 3 db/km near 600 nm to 1.5 db/km near 1050 nm.

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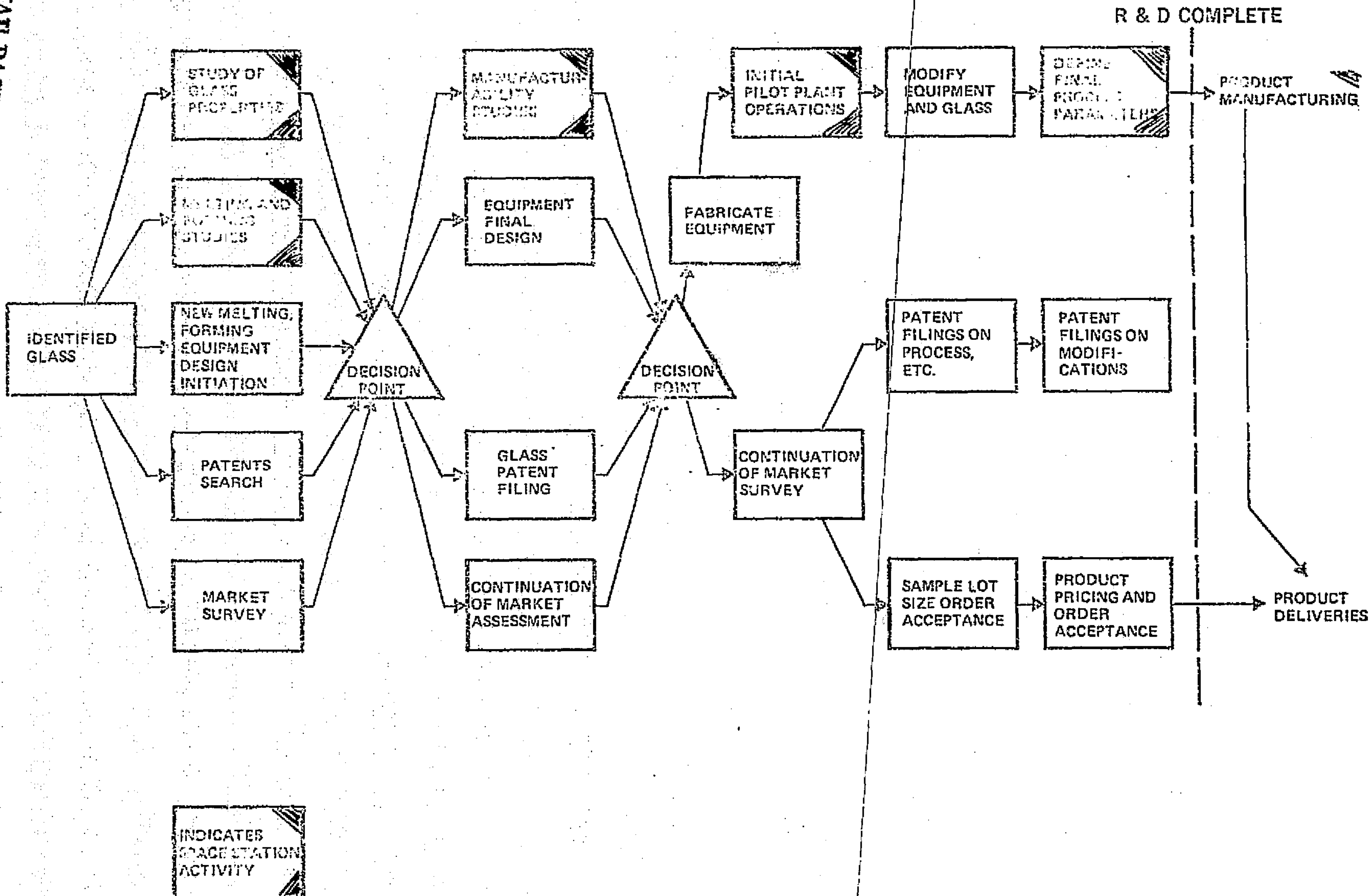

Figure 2-3. Product R&D in Glass Industry

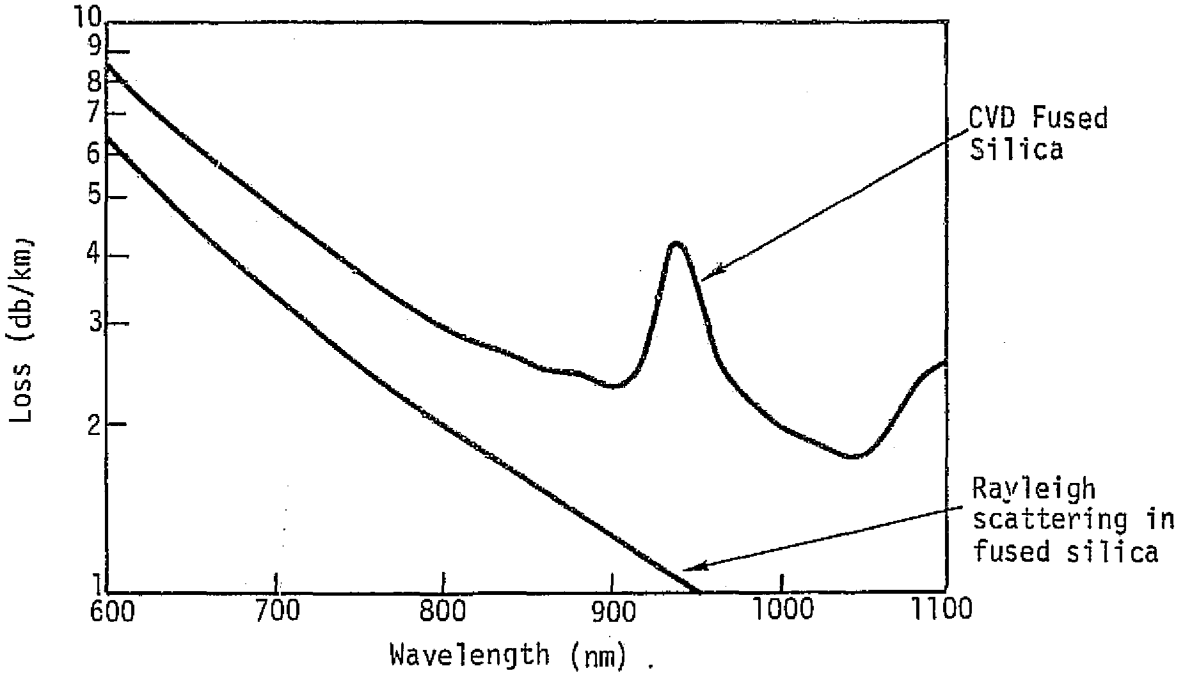

Figure 2-4. Fiber Loss Characteristic

Two attractive candidates for light sources in communications systems are the neodymium doped yttrium aluminum garnet (YAG) laser and the gallium arsenide injection laser. The YAG laser produces coherent light at 1060 mm and, by doping the gallium arsenide with aluminum, coherent light in the 800-900 nm region can be obtained.

Thus, one can see that ground-based technology as represented by experimental/laboratory results is rapidly approaching fundamental limits. However, if one postulates that the fiber characteristics shown in Figure 2-4 represent the limits of commercial ground-based technology (a postulation which is probably false or will be within a few years), then perhaps some technical benefits could be derived from using containerless melting in space to produce ultrahigh purity material. The issue as to whether any cost benefits are to be derived from space processing of fiber optic material is not pertinent to the purpose of the case study.

3. FIBER OPTIC PREFORM PRODUCTION

A simplified process flow analysis for producing fiber optic glass preforms in the low gravity environment of the Space Station is discussed in this section of the case study report. The purpose of the process flow analysis is to develop processing equipment requirements from which Space Station support requirements can be developed.

3.1 PROCESS DESCRIPTION

The glass preform production process could produce the preform shown in Figure 3-1. The preform produced in space consists of a pure fused silica core with a borosilicate glass cladding which has an index of refraction lower than the fused silica. This preform, with aspect ratio of four, establishes the sizes of the various furnaces used in the production process. Following return of the product to Earth, an outer cladding is added during the fiber drawing process. The outer cladding is of a glass with good tensile properties and provides the strength necessary for loads imposed on the fiber during optic cable manufacture and installation.

The process flow is shown in Figure 3-2. As shown, the portion of the process to be conducted in space can be generalized by five unit process steps as follows:

1) Glass formation
2) Preform shaping
3) Preform annealing
4) Preform cladding and annealing
5) Preform shipment preparation.

Each of these unit processes will be discussed in the subsequent section of the report.

It should be noted that the cylindrical shape of the preform is applicable to many glass products. For example, glass lasing rods and components of electro-optical devices could be candidates for space production and their form would basically be cylindrical. Thus, the unit process equipment to be discussed in subsequent paragrpahs is not unique to the fiber-optic preform.

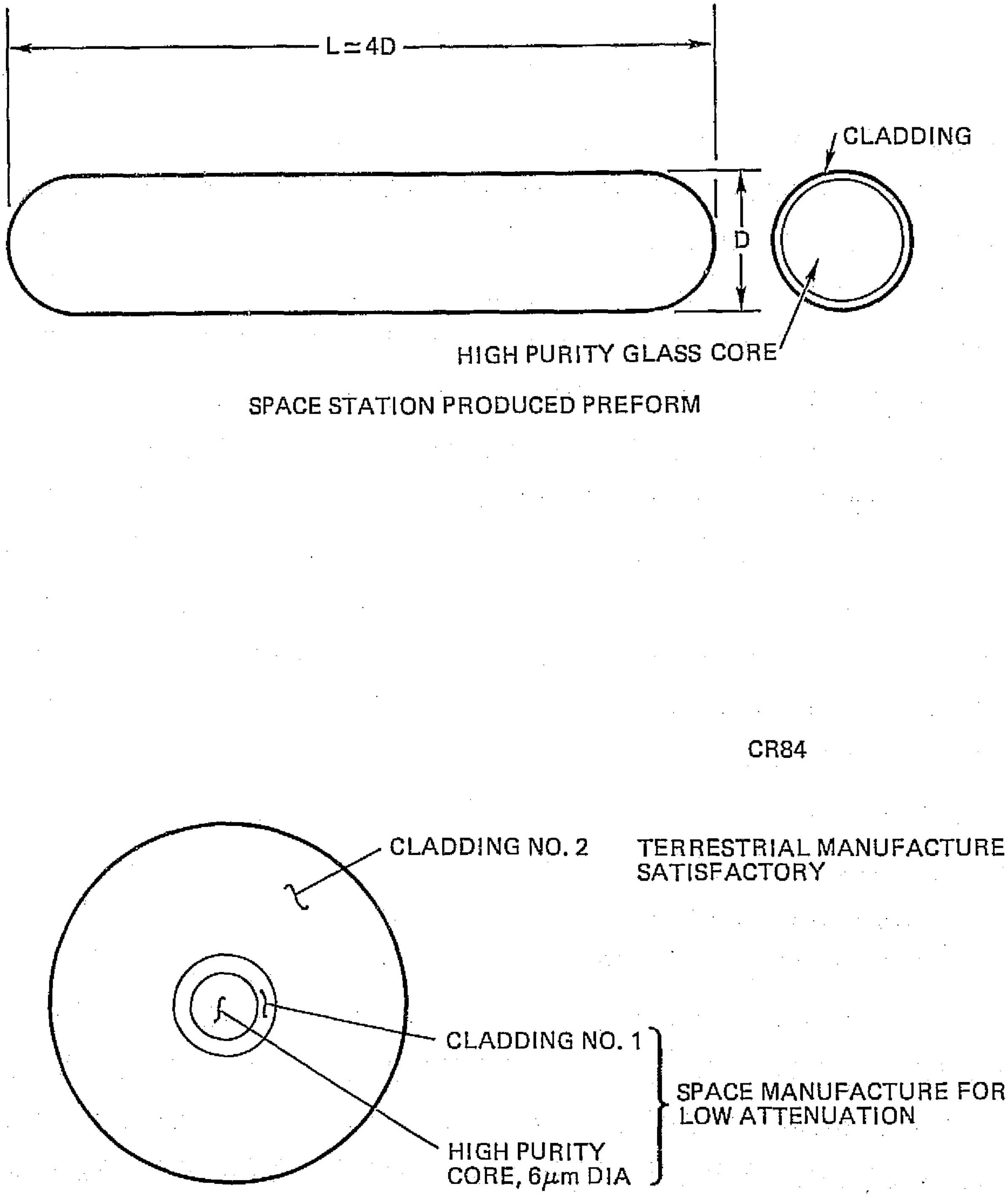

Figure 3-1. Preform Characteristics

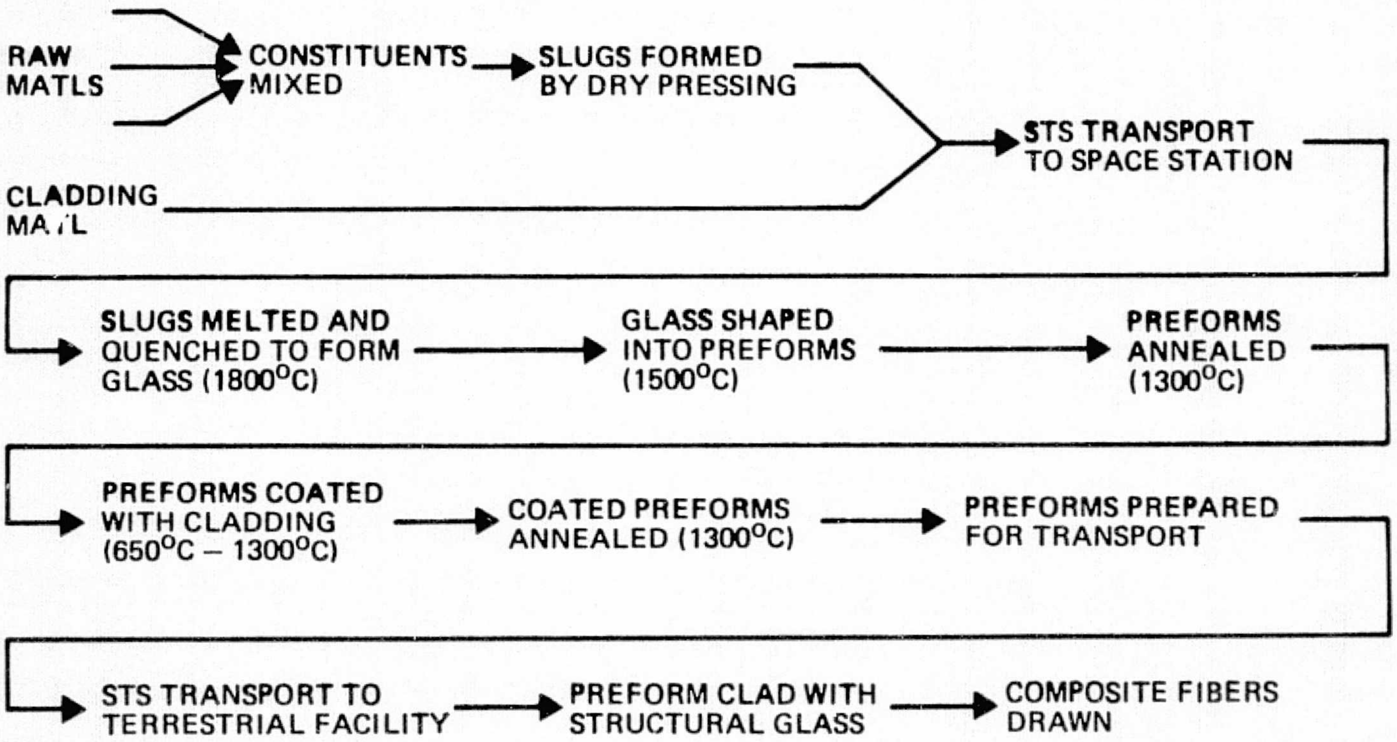

Figure 3-2. Fiber Optics Preform Production

3.2 UNIT PROCESS APPARATUS

Each of the high temperature unit processes shown in Figure 3-2 require processing equipment with differing requirements. In general, the requirements are established by the temperature at which the process takes place and the shape of the charge during the process.

3.2.1 Glass Formation

The glass formation process is conducted in a furnace which has the capability of maintaining the melt in a stable position without contact so as to not introduce contaminants into the melt. Various methods are available for providing such a containerless positioning furnace. However, since the glass melting will most likely require an oxidizing environment, some version of acoustic positioning would probably be adequate. In general, the following equipment would be required:

1) Containerless processing furnace (acoustic)
2) Melt quenching apparatus
3) Vacuum system (gas purging)
4) Atmosphere supply (gas tanks)
5) Glass handling apparatus
6) Coolant supply.

Specific requirements in terms of power and volume for the furnace are shown in Figure 3-3. The furnace power consumption and occupied volume are direct functions of the mass to be melted in the furnace (furnace charge). For example, a furnace melting a 1-kg mass would have an outside dimension of 70 cm, an equilibrium power consumption of 8 kW, and occupy a volume of approximately 0.02 m^3.

3.2.2 Preform Shaping

The shaping of the glass (basically spherical shape) into a cylindrical preform must also be accomplished in a non-contact manner to avoid contamination of the pristine surface prior to cladding with the lower refractive index glass coating. The shaping could be accomplished in the same furnace as the glass formation or a separate furnace with shaping

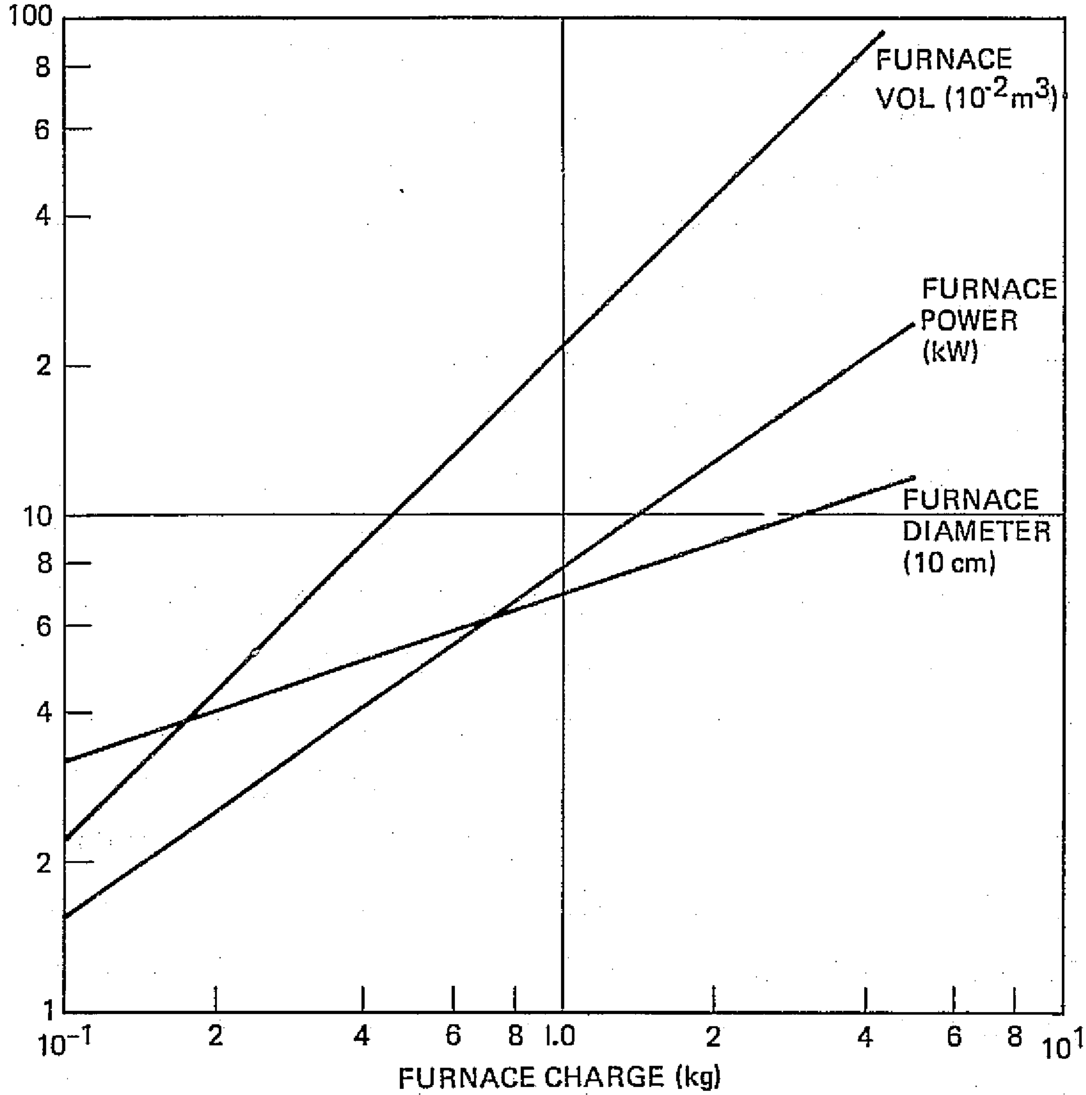

Figure 3-3. Containerless Processing Furnace Characteristics

capability could be used. The latter approach might be the more prudent one since the shaping process is likely to take significantly longer than the glass formation process. This approach would allow one glass formation furnace to feed multiple preform shaping furnaces.

In general the following equipment would be required:

1) Tube furnace
2) Shaping apparatus (contained within furnace)
3) Preform handling apparatus
4) Atmosphere supply
5) Coolant supply.

The furnace characteristics are shown in Figure 3-4. This furnace is somewhat smaller than the glass formation furnace. Preform aspect ratio (L/D) of four is anticipated.

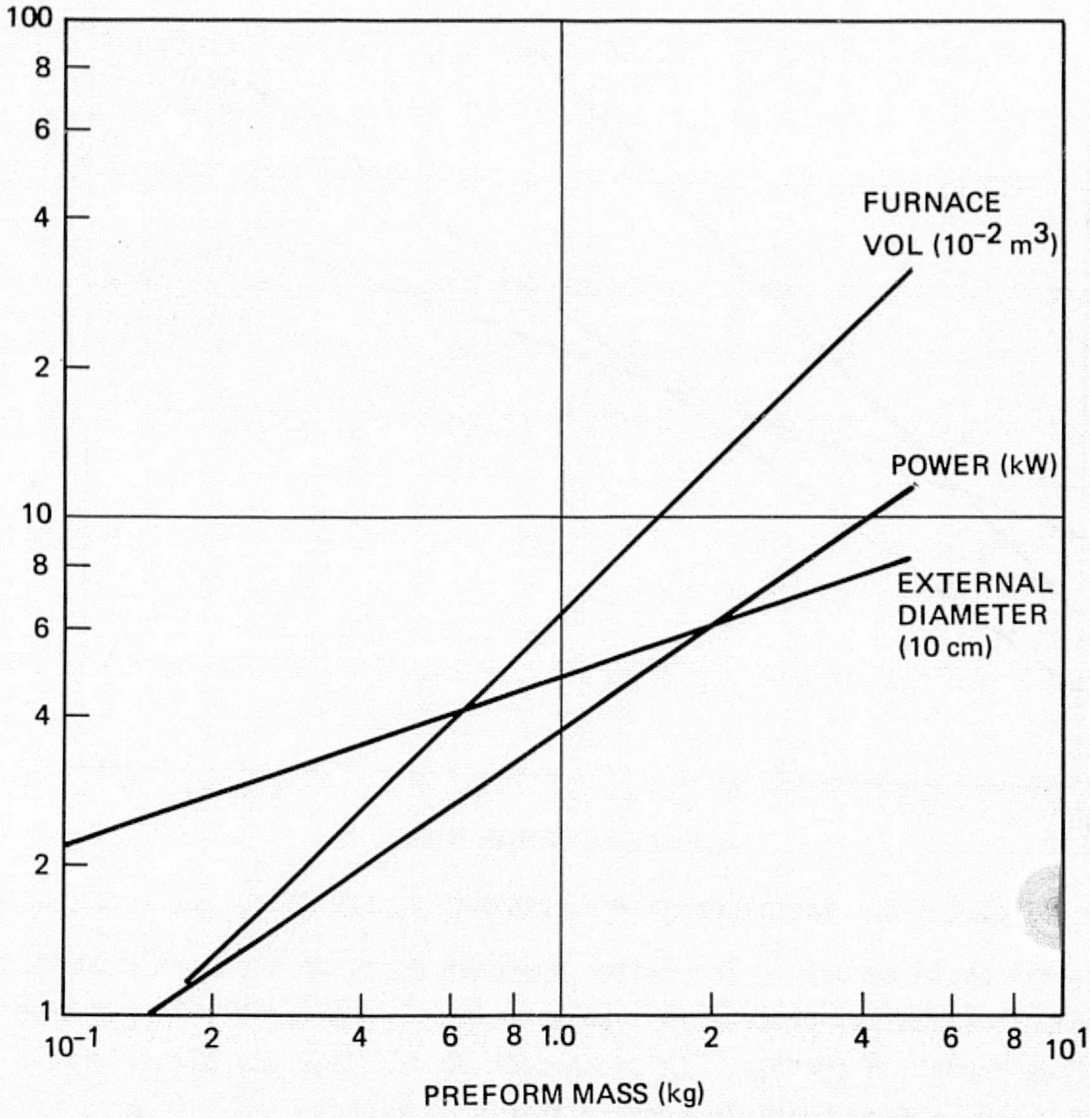

Figure 3-4. Shaping Furnace Characteristics.

3.2.3 Preform Annealing

The preform, once shaped, must be annealed to relieve the residual strains induced in the shaping process. The annealing operation can be accomplished in the shaping furnace following shaping, in a separate annealing furnace or in the cladding furnace. Since the annealing process occurs at a temperature 300 to 500^{o}C below the melt point, the preform can be supported during the process without introducing contamination into the glass.[7] Thus, the annealing process could best be accomplished in the cladding furnace prior to introduction of the cladding material into the furnace. Consequently, there are no requirements for annealing equipment other than that already in existence as part of the cladding operation.

3.2.4 Preform Cladding

The preform cladding process could be accomplished in several ways. One way would be to heat the cladding material above its flow point locally in an enclosure and bring it into contact with the glass preform as the melt is slowly translated past the glass preform. Maintaining the heated enclosure at a temperature above the cladding softening point but below the cladding melt temperature would allow a coating of cladding material to be "wiped" onto the glass preform.

The required apparatus would be a "tube type" furnace in size similar to that used in the shaping process. A furnace other than the shaping furnace should be used since it is anticipated that the cladding operation will be the time limiting operation. The furnace would be equipped with a preform rotation device and a cladding material holder/heater/translating drive screw device.

The cladded preform would be annealed in this same furnace to relieve any residual strains resulting from the cladding operation.

3.2.5 Preform Shipment Preparation

The preforms would be prepared for shipment by individual placement in protective containers. The individual containers would be aggregated into shipment containers for transport via the STS. Consequently, storage space for containers and a packaging work station must be provided.

3.3 PROCESS ANALYSIS

In order that the unit process equipment requirements can be uniquely specified, a means of establishing the size of the preform mass is required. Since the total processing time is fixed by the Space Station resupply interval, a process relationship establishing the number of process cycles which can be accomplished when coupled with the total mass to be produced will establish the mass per cycle (furnace charge). Consequently, a processing time relationship has been established for each unit process.

3.3.1 Glass Formation Time

The glass melting time consists of the furnace and charge heatup times, time at temperature and quench time. For the purpose of this case study, the quench time is considered small with respect to the time at temperature

$$T_1 = k_1 m + C_f M_f \Delta T_1 / \dot{q}_f + C$$

where

$k_1 = C_m \Delta T_m / \dot{q}_{heating}$ and

C is time at temperature.

3.3.2 Preform Shaping Time

The preform shaping time is established by the diameter of the preform, the mass to be shaped, and the rate of shaping and furnace heatup time

$$T_2 = k_2 m^{1/3} + C M_{f2} \Delta T_2 / \dot{q}_{f2}$$

where

$$k_2 = \frac{[-(\pi R_p \rho m)^{1/3} + (6/\pi \rho m)^{1/3}]}{2\dot{R}}$$

$\dot{R} \equiv$ shaping rate

$R_p \equiv$ preform aspect ratio.

3.3.3 Preform Annealing Time

The preform annealing time is a function of the allowable residual strains. The internal stresses are generally reduced to low values when the glass is held at the strain point temperature for 4 hours.[7]

$$T_3 = k_3$$

where k_3 is the strain point hold time.

3.3.4 Preform Cladding Time

The preform cladding time is a function of several cladding material properties such as viscosity, wetting angle, cladding melt temperature, etc. For purposes of this case study, cladding time is assumed to be a constant within a reasonable range of preform diameters.

$$T_4 = k_4$$

where k_4 is a constant.

3.3.5 Preform Shipment Preparation Time

The shipment preparation time is a function solely of the number of preforms to be shipped.

$$T_5 = k_5$$

where k_5 is the total preparation time for all preforms produced.

3.4 PROCESS CYCLE

The process cycle time can be expressed as a series summation of the individual steps if they are not overlapping. If, however, one step is significantly longer than the sum of those preceding it, the production time for multiple cycles can be written as the summation of those steps prior to the maximum step time, a multiple of the maximum step time, those steps following the maximum step time and the time gap between repeats of the maximum step time.

$$T = \sum_{i=1}^{m} T_i + nT_{m+1} + \sum_{i=m+2}^{k} Ti + T_{gap}$$

For the case example under question, the preform cladding (step 4) is assumed to be the longest process step. Thus, the total process time becomes

$$T = k_1 m + C + K_2 m^{1/3} + K_4 N + k_5 + T_{gap} + k_3$$

$$+ M_{f1} C_{f1} \, \Delta T_1/\dot{q}_{f1} + M_{f2} C_{f2} \Delta T_2/\dot{q}_{f2} \qquad (1)$$

where T_{gap} represents the time spacing between repeating cycles of step 4 and n = N since step 4 governs the number of preforms produced.

The mass of each preform, m, can be related to N and the total mass to be produced, M, as follows:

$$m = \frac{M}{N} .$$

Rewriting equation (1) in terms of M/N one gets

$$T = \frac{k_1 M}{N} + C + k_2 \left(\frac{M}{N}\right)^{1/3} + k_4 N + k_5 + T_{gap} + k_3 + A + B$$

where

$$A = M_{f1} C_f \ \Delta T_1 / \dot{q}_{f1}$$

$$B = M_{f2} C_f \ \Delta T_2 / \dot{q}_{f2}.$$

Solving for N one obtains

$$N^4 + \left(\frac{Z - T}{K_4}\right) N^3 + \left(\frac{K_1 M}{K_4}\right) N^2 + \frac{K_2 M^{1/3}}{K_4} = 0$$

where $Z = C + K_3 + K_5 + A + B + T_{gap}$, and after solving for roots results in

$$N_1 = \left(\frac{T - Z}{2K_4}\right) + \frac{1}{2} \left[\left(\frac{Z - T}{K_4}\right)^2 - \frac{4K_1 M}{k_4}\right]^{1/2}$$

$$N_2 = \left(\frac{T - Z}{2K_4}\right) - \frac{1}{2} \left[\left(\frac{Z - T}{k_4}\right)^2 - \frac{4k_1 M}{k_4}\right]^{1/2}$$

one can readily see that if

$$\left(\frac{Z - T}{k_4}\right)^2 \gg \frac{4k_1 M}{k_4}$$

there is only one root of interest, namely N_1. The number of preforms that can be produced is governed by the total mass M, the total time available T and the constants of proportionality of the unit process steps, K_i. The sizing of the processing furnaces, the power consumption, and the occupied volume are all derivatives of the number of preforms to be produced.

4. SPACE STATION REQUIREMENTS

The Space Station subsystems requirements relative to space processing derive from two types of space processing facilities: R&D laboratory and pilot plant. The R&D laboratory provides the capability to conduct experimentation on experimental size glass compositions. In contrast, the pilot plant is a facility established to produce a single glass composition of specific form. The pilot plant operation establishes the optimum production parameters for operation of a full scale production facility.

This section of the case study report sets forth Space Station subsystems requirements to support the two types of facilities. The R&D laboratory requirements are derived mainly from the data of Reference 3 while the pilot plant requirements come from both Reference 3 and a pilot plant case example developed from the characteristic equations developed in Section 3.

4.1 R&D LABORATORY

The R&D laboratory consists of the preparative, processing, and analytical equipment necessary to conduct experimentation on glass compositions unique to space development. Glass compositions that are beyond the terrestrial glass formation region could be melted, have their optical properties measured, and their forming characteristics identified in this facility.

Table 4-1 presents a list of facility equipment and pertinent equipment characteristics. The R&D laboratory equipment will occupy approximately 1.9 cu m and have a cumulative weight of 700 kg. The laboratory power requirements cannot readily be expressed in terms of a power timeline because of the wide variations possible in the experimental activity. However, a review of the equipment list shows that sustained power levels of 5 to 6 kW are possible and peak power could be 2 to 3 kW higher.

The contactless melting furnace is likely to operate with a core temperature approaching 2200°C and would be the maximum temperature source. Such a furnace is described in Reference 3. The thermal oven is likely to operate with a maximum temperature of 500°C and is primarily

Table 4-1. Glass R&D Facility Equipment Requirements

Equipment	Volume (m^3)	Weight (kg)	Power (kW) Peak/Sustaining
<u>Processing Enclosures</u>			
Contactless melting furnace[1]	0.190	65	3.0/2.0
Thermal oven	0.027	25	1.0/0.50
Glove box	0.027	25	N/A
<u>Preparative Apparatus</u>			
Slip casting unit	0.001	2	N/A
Rheological unit	0.005	5	0.05/0.05
Ph monitor	0.012	15	0.05/0.05
Grinding/polishing unit	0.082	110	0.35/0.20
Mechanical mixing unit	0.001	3	0.05/0.10
Mass measurement unit	0.001	5	0.10/0.20
<u>Process Control</u>			
Pyrometers	0.027	9	0.05/0.05
Pressure controllers	0.061	7	0.05/0.05
Thermocouples	N/A	Negligible	N/A
Microprocessor system	0.041	31	0.30/0.30
Residual gas analyzer	0.061	34	0.25/0.25
Gas supply system	0.5	30	0.1/0.1
SCR power controllers	0.060	15	0.2/0.2
Particulate filter system	0.005	5	N/A
Vacuum system	0.020	45	0.5/0.3
<u>Analytical Instrumentation</u>			
IR spectrophotometer	0.041	45	0.2/0.2
X-ray fluorometer unit	0.045	42	0.2/0.2
Refractometer-spectrometer	0.035	41	0.1/0.1
UV visual spectrophotometer	0.038	45	0.2/0.2
Mass spectrometer	0.035	35	0.3/0.3
Binocular microscope (100x)	0.026	23	0.1/0.1
Differential thermal analyzer	0.030	25	0.25/0.25
R&D Laboratory Totals	1.95[2]	750[3]	

[1] Includes contactless quenching and shaping apparatus

[2] Includes 45 percent packing density factor

[3] Includes 10 percent miscellaneous allowance but does not include structural supporting hardware.

used for drying slips for slip casting experiments and elevated temperature property measurement activity. All other equipment operation is near laboratory ambient temperature.

4.2 PILOT PLANT

The pilot plant processing chamber requirements in terms of the production constraints were developed in algorithm form in Section 3. In order to convert these parametric requirements into specific design parameters one must postulate a case example.

If one considers the results of part 1 of the Space Station study, a repeater cost savings of 59 million dollars can be achieved for every 1000 kg of fiber optic material produced with a 1 db/km lower signal attenuation.

Pilot plant operation is usually scaled an order of magnitude lower than the full scale production level. Consequently, a total mass M of 100 kg produced in a single resupply period is a reasonable estimate of production for the pilot plant in a 90-day mission. By exercising the production process algorithms one can define the pilot plant requirements.

4.2.1 Constants Evaluation

$$k_1 = C\ \Delta T/\dot{q}_{heating} = \frac{(0.3)\ (2830)}{\dot{q}_{heating}\ (3415)\ (24)\ (0.456)}$$

$$k_1 = 0.024/\dot{q}_{heating}\ \frac{\text{kW-days}}{\text{kg}}$$

$$k_2 = [(6/\pi\ \rho_m)^{1/3} - (\pi R_p \rho_m)^{-1/3}]/2\ \dot{R} = \frac{[(6/3x10^{-3}\pi)^{1/3} - ((4)(3x10^{-3})\pi)^{-1/3}]}{2\ (1)\ (24)}$$

$$k_2 = 0.12\ \text{day} - \text{kg}^{-1/3}$$

$K_3 = 0.17$ days

$k_4 = 0.42$ days/preform

$k_5 = 2$ days

$$A = M_{f1} C_{f1}\ \Delta T_1/P_{f1} = (725)\ (0.2)\ (2830)/(12)\ (3415)\ (24)$$

$$A = 0.42\ \text{day}$$

$$B = M_{f2}C_{f2}\ \Delta T_2/P_{f2} = (295)\ (0.2)\ (1580)/(6)\ (3415)\ (24)$$

$$B = 0.19 \text{ day}$$

$$C = 0.17 \text{ day}$$

$$T_{gap} = 0.05 \text{ day.}$$

4.2.2 Preform Production Rate

As noted in 4.2.1, the process constant for the glass formation is a function of the net heating rate, $\dot{q}_{heating}$, which is related to the surface area of the materials charge and the furnace heating rate. If one assumes, however, that the constituent materials are inserted into the furnace after the furnace has reached temperature, the average heating rate over the time of material warmup is approximately 1/3 of the maximum heating rate. Thus,

$$\dot{q}_{heating} = \frac{1}{3}\sigma\ \varepsilon_m A_m\ (T_f^{\ 4} - \dot{T}_{amb}^{\ 4}) \text{ and}$$

$$A_m = \pi\ \left(\frac{4m}{\pi\rho_m}\right)^{2/3}$$

where

m = preform mass if one assumes the initial charge is a pressed cylindrical slug with aspect ratio of 1.

For $T_f = 1600^0C$, m = 1 kg as an initial guess, $\varepsilon_m = 0.85$

$$\dot{q}_{heating} = 3.8 \text{ kW}$$

and therefore, $k_1 = 0.006$ day/kg.

Evaluating N for M = 100 kg, Z = 3 days, and T = 90 days

$$N = \left(\frac{90 - 3}{(2)(0.33)}\right) + \frac{1}{2}\left[\left(\frac{90 - 3}{0.33}\right)^2 - \frac{(4)\ (0.006)\ (100)}{0.33}\right]^{1/2}$$

N = 264 preforms.

The mass of an individual preform is

$$m = \frac{M}{N} = \frac{100 \text{ kg}}{264} = 0.378 \text{ kg.}$$

Since m is not equal to the 1.0 kg originally assumed to calculate k_1, k_1 will increase by a factor of 1/m. However, the number of preforms N is only weakly affected by k_1 and thus the iteration on k_1 will not be considered in this exercise.

4.2.3 Pilot Plant Requirements

The fiber optics preform production facility would produce approximately 260 preforms weighing 380 grams each. The requirements of the facility furnaces in terms of power and volume are shown in Figures 3-3 and 3-4.

Equipment requirements are summarized in Table 4-2. The summary pertains only to the example case and the specific values chosen for parameters. The weight and volume associated with supporting subsystems and/or structure are not included in the values of Table 4-2. The weights of the furnaces and enclosure are based on typical furnace weights of 1280 kg/m^3.

The pilot plant resource requirements are shown in Table 4-3. The power types are assumed to be provided from power conditioning equipment associated with the Space Station power distribution subsystem. Should this not be the case, the appropriate power conditioning apparatus will have to be added to the pilot plant equipment (Table 4-2).

A pilot plant operational timeline is shown in Figure 4-1 along with the plant power requirements timeline. The figure shows three complete cycles of the unit operations and additional cycles would result in a continuation of the repetitive portion of the power timeline.

The power timeline shown represents a sequencing of the individual apparatus power in a mode which results in the maximum instantaneous power requirement. The repeating cycle peak power requirements can be reduced 8-10 kW by maintaining the glass formation and annealing furnaces at temperature continuously after the initial warmup.

The power timeline includes power allocation for process control, data formatting, ancillary unit processes equipment, and inspection/analytical equipment. A total of 2.5 kW is allocated for these items and is considered to be a constant load.

Table 4-2. Glass Pilot Plant Equipment Requirements

Equipment	Volume (m)^3	Weight (kg)	Power (kW) Peak/Sustaining
<u>Processing Enclosures</u>			
Contactless melting furnace[1]	0.26	345	12.0/5.2
Contactless shaping furnace	0.10	160	6.0/3.0
Annealing furnace	0.10	145	6.0/1.5
Cladding furnace	0.20	295	8.5/5.1
<u>Process Control</u>			
Pyrometers (2)	0.054	18	0.10/0.10
Thermocouples	N/A	Negligible	N/A
Pressure controllers	0.061	7	0.05/0.05
Microprocessor system	0.041	37	0.30/0.30
<u>Atmosphere Control</u>			
Gas supply and manifold	1.50	90	0.30/0.30
Residual gas analyzer	0.061	34	0.25/0.25
Vacuum system	0.020	45	0.50/0.30
Particulate filter system	0.008	5	N/A
<u>Inspection</u>			
Laser optical scattering system	0.145	102	1.2/0.25
Shape comparator	0.027	35	0.20/0.20
Binocular microscope	0.054	23	0.15/0.15
Thickness measurement system	0.027	35	0.25/0.25
<u>Manipulators</u>			
Glass handling	0.027	15	0.1/0.1
Rotation drive assembly	0.027	25	0.15/0.15
Cladding heater translator	0.013	7	0.1/0.1
<u>Material Storage</u>			
Raw material	0.010	100	N/A
Product	0.010	100	N/A
Packaging/containers	0.025	15	N/A
Totals	4.0[2]	1725[3]	

[1]Includes contactless quenching apparatus

[2]Includes 45 percent packing density factor

[3]Includes 10 percent miscellaneous allowance but does not include structural supporting hardware

Table 4-3. Pilot Plant Resource Requirements

Resource	Level	Remarks
<u>Volume</u>		
Equipment	4.0 m^3	
Work area	38.0 m^3	Includes equipment
<u>Power</u>		
Peak	26.0 kW	Power types required are 230 VAC, 60-400 Hz, 3σ 115 VAC, 60 Hz, 1σ 28 VDC, regulated
Sustained	21.5 kW	
Average	17.0 kW	
<u>Weight</u>		
Equipment	1725 kg	Does not include weight of structural supporting hardware
<u>Crew</u>	2 crew persons/shift	

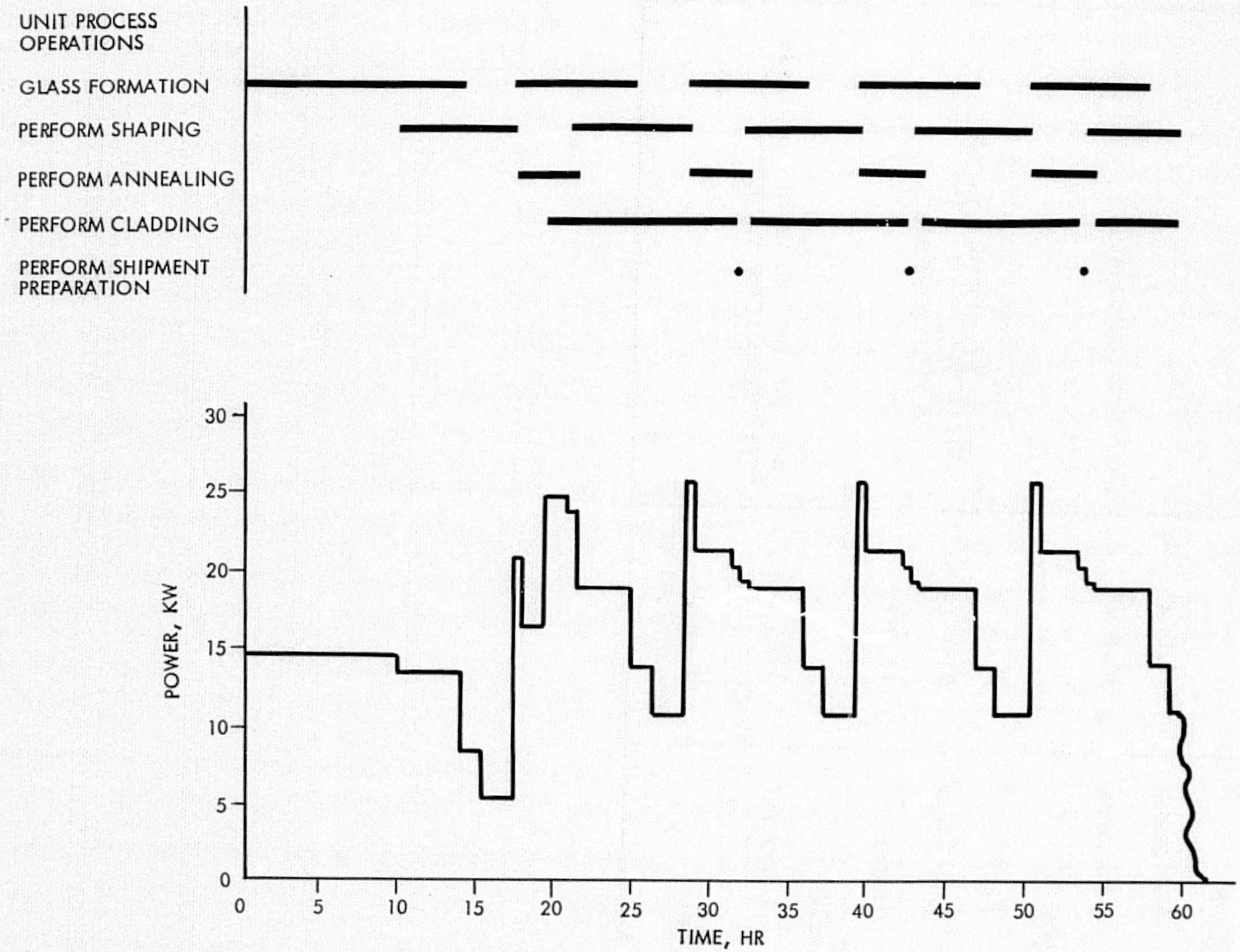

Figure 4-1. Pilot Plant Unit Operation and Power Timelines

The crew requirements can be determined from the operational timeline (Figure 4-1). Several operations in the overlapping unit processing operations occur simultaneously. To operate the pilot plant with a single operator would involve a less than optimum production rate. On the other hand, there is insufficient activity for two operators in any given 8 to 12-hour shift.

A conservative estimate for crew requirements would be 1.25 crewmen per shift. This requirement could be met by utilizing one of the Space Station crew persons to augment a full time glass technologist. The Space Station crew member would be given sufficient training to accomplish a limited set of tasks in support of pilot plant operation.

5. REFERENCES

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Part 13

CREW AND HABITABILITY SUBSYSTEM
(OPTION L)

Section 1
GENERAL CONCEPT

The Crew and Habitability subsystem is that portion of the space construction base system that provides the crew support equipment, furnishings, supplies and services, and procedures necessary to assure efficient, comfortable, and safe living and working conditions for the space construction base crew. Figure 1 shows the eight general categories of the Crew and Habitability subsystem, along with the elements included under each category. Seven of the categories identified in Figure 1 are associated with requirements and design of hardware, supplies, and the architectural arrangement within which they are provided. The eighth category, CREW, is concerned with the selection and training of crew members, the scheduling of their activities, and the provision of support aids such as checklists, performance aids, and manuals.

SCB PHASE-B-DERIVED SUBSYSTEM

CREW AND HABITABILITY SUBSYSTEM

- **PERSONAL EQUIPMENT**
  - CLOTHING
  - TOWELS, WASH CLOTHS
  - GROOMING AIDS
  - PERSONAL DOSIMETERS
  - INDIVIDUAL URINE RECEIVERS
- **GENERAL AND EMERGENCY EQUIPMENT**
  - TOOLS
  - PORTABLE LIGHTS
  - RADIATION DETECTORS
  - O_2 MASKS
  - FIRE EXTINGUISHERS
  - MOBILITY/RESTRAINT
  - FIRST AID KITS
  - EMU/PLSS/MMU
  - EMU DONNING STATION
  - EMU RECHARGE STATION
  - AIRLOCKS
- **RECREATION, EXERCISE, AND CREW CARE**
  - EXERCISE EQUIPMENT
  - RECREATION AREA/FURNISHINGS
  - RECREATION EQUIPMENT/SUPPLIES
  - MEDICAL-DENTAL AREA/FURNISHINGS
  - MEDICAL EQUIPMENT
  - MEDICAL SUPPLIES
- **FOOD MANAGEMENT**
  - FOOD SUPPLY
  - FOOD STORAGE
  - OVENS, HEATERS
  - FREEZER, REFRIGERATOR
  - FOOD TRAYS
  - PREPARATION UTENSILS
  - EATING UTENSILS
  - GALLEY WIPES
  - DINING FURNISHINGS
  - GALLEY FURNISHINGS
  - POTABLE WATER CONDITIONING
  - POTABLE WATER DISPENSING
  - POTABLE WATER STORAGE
- **PERSONAL HYGIENE/WASTE MANAGEMENT**
  - PH COMPARTMENT
  - WM COMPARTMENT
  - HAND WASHER
  - SHOWERS
  - MIRRORS
  - COMMODE/URINAL
  - WIPES
  - WASTE COLLECTION
  - WASTE CONDITIONING
  - WASTE DISPOSAL
- **STOWAGE/HOUSEKEEPING**
  - INVENTORY MANAGEMENT
  - TRASH MANAGEMENT
  - STOWAGE CONTAINERS
  - VACUUM CLEANER
  - WASHING MACHINE
  - TRASH BAGS
- **WORK STATIONS**
  - CONTROL/MONITOR STATIONS
  - EVA WORK STATIONS
  - WORKBENCH
  - CONFERENCE/MEETING AREA
- **SLEEP FACILITIES**
  - SLEEP COMPARTMENT
  - BUNKS
  - SLEEP RESTRAINTS
- **CREW**
  - CREW SIZE AND COMPOSITION
  - CREW SKILLS
  - CREW TASKS/TIMELINES
  - JOB PERFORMANCE AIDS
  - ANTHROPOMETRY
  - WORK/REST CYCLES
  - DUTY TENURE AND CAREERS
  - TRAINING

Figure 1. Crew Habitability Subsystem (U)

Section 2
SUBSYSTEM REQUIREMENTS

This section defines the requirements baseline for the space construction base Crew and Habitability subsystem. Section 2.1 summarizes the subsystem baseline requirements as defined in JSC Phase-B documentation. Section 2.2 identifies necessary modifications to the JSC baseline to accommodate the MDAC Option L 7-man initial space construction base. Section 2.3 describes the impact on the initial Option L subsystem occasioned by growth to a 14-man and to a 21-man station. Section 2.4 identifies subsystem areas that require further investigation before firm requirements decisions can be made.

2.1 JSC PHASE "B" BASELINE REQUIREMENTS

The subsystem baseline requirements discussed in this section represent the requirements as defined in the Crew and Habitability subsystem section (Section 8) of the JSC Phase-B Document No. SD 71-217-4.

2.1.1 General Requirements

The requirements listed in this section represent requirements that are general in nature and may influence all or some of the more specific requirements as summarized in Section 2.1.5 and subsequent sections.

2.1.1.1 Ceiling Height

The ceiling height in all general mobility areas above deck will be a minimum of 2.08m (82 in.). Below deck, the minimum height for general mobility areas will be 1.57m (62 in.) with no protrusions.

2.1.1.2 Equipment Installations

All equipment installations, including interior partitions, will be capable of use for pushoffs, and will be capable of reacting to crew impact loads of 136.03 kg (300 lb) applied in any direction.

All equipment within the Space Station will be installed so that access to the pressure hull can be achieved for inspection or repair. The access provisions will be such that a suited/pressurized crewman can gain access to the pressure hull.

2.1.1.3 Anthropometry

Crew member pertinent dimensions for a 5th and 95th percentile crew member, male or female, will be used for developing Space Station interior design and arrangements per Figures 2 and 3. These standard anthropometric dimensions are for a crew member wearing lightweight clothing. Standing height, eye height (standing), and shoulder heights (standing) will be increased by 2.54 cm (1 in.) by the addition of shoes.

2.1.1.4 Acoustics

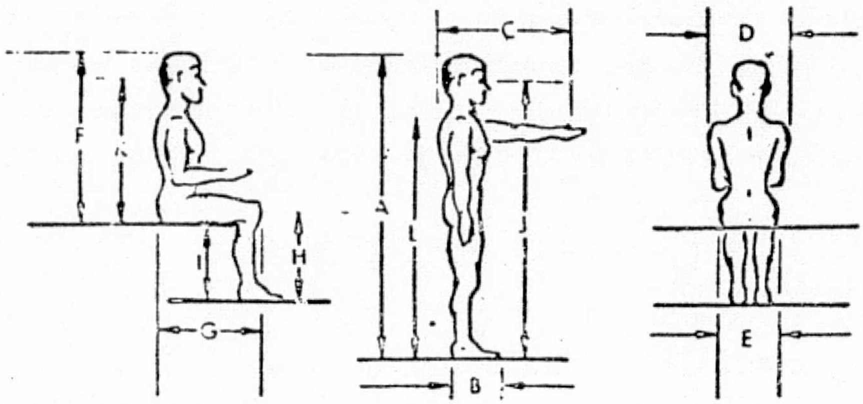
Noise levels will not cause discomfort to crewmen, or interfere with communication between crewmen at normal voice levels up to distance of 5.5m (18 ft). Continuous noise levels will not exceed 50 dB in the speech interference level (SIL) range (600 to 4,800 Hz), 70 dB at frequencies below SIL, or 60 dB at frequencies above SIL.

2.1.1.5 Lighting

Overhead diffused general lighting will be provided in all living and work areas in the range of 30 to 50 Foot-candles. Supplementary lighting will be provided in special areas such as specific work/maintenance areas and galley work surfaces. In the primary and backup medical areas over the examination/treatment bench, supplementary lighting will be provided in the form of diffused 500 to 1,000 Foot-candles.

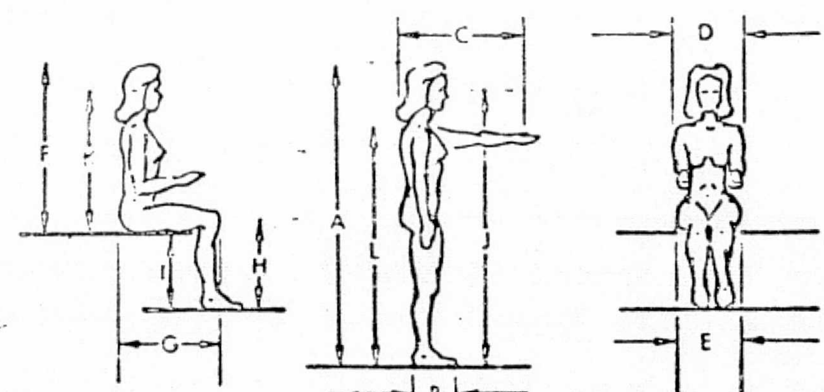
Exterior illumination for EVA operations will be a minimum of 2 Foot-candles along EVA surface paths and 7 Foot-candles at work surfaces.

General requirements for running lights will be provided to determine station orientation within 609.6m (2,000 ft). Acquisition lights will be provided to obtain rendezvous position at distances greater than 609.6m (2,000 ft). These exterior lights will be activated approximately 90 min before Orbiter renddezous and berthing, and for 90 min after Orbiter departure from the MSS. The exterior lights can be extinguished after the Orbiter deorbit maneuver.



DIMENSION	PERCENTILE 5 CM	5 IN	95 CM	95 IN	DIMENSION	PERCENTILE 5 CM	5 IN	95 CM	95 IN
A – STANDING HEIGHT	167.4	65.9	187.7	73.9	H – KNEE HEIGHT	51.8	20.4	59.9	23.6
B – FOOT LENGTH	25.2	9.9	29.0	11.4	I – POPLITEAL HEIGHT	40.6	16.0	47.0	18.5
C – FUNCTIONAL REACH (THUMBTIP)	82.3	32.4	97.3	38.3	J – EYE HEIGHT (STANDING)	155.2	61.1	175.3	69.0
					K – EYE HEIGHT (SITTING)	76.2	30.0	86.1	33.9
D – SHOULDER BREADTH	44.2	17.4	52.6	20.7	L – SHOULDER HEIGHT (STANDING)	135.6	53.4	154.9	61.0
E – HIP BREADTH (SITTING)	34.3	13.5	41.7	16.4					
F – SITTING HEIGHT	88.1	34.7	98.5	38.8	M – WEIGHT, KG (LB)	64.55	(142.2)	95.62	(210.8)
G – BUTTOCK – KNEE LENGTH	56.1	22.1	65.8	25.9					

Figure 2. Male Crewman Pertinent Dimensions



DIMENSION	PERCENTILE 5 CM	5 IN	95 CM	95 IN	DIMENSION	PERCENTILE 5 CM	5 IN	95 CM	95 IN
A – STANDING HEIGHT	156.7	61.7	175.0	68.9	H – KNEE HEIGHT	47.5	18.7	54.6	21.5
B – FOOT LENGTH	22.1	8.7	25.9	10.2	I – POPLITEAL HEIGHT	37.3	14.7	44.4	17.5
C – FUNCTIONAL REACH (THUMBTIP)	75.4	29.7	86.6	34.1	J – EYE HEIGHT (STANDING)	144.8	57.0	162.1	63.8
					K – EYE HEIGHT (SITTING)	70.4	27.7	78.7	31.0
D – SHOULDER BREADTH	37.9	14.9	44.7	17.6	L – SHOULDER HEIGHT (STANDING)	122.4	48.2	141.7	55.8
E – HIP BREADTH (SITTING)	34.3	13.5	42.9	16.9					
F – SITTING HEIGHT	82.3	32.4	91.4	36.0	M – WEIGHT, KG. (LB)	46.27	(102.0)	67.2	(148.2)
G – BUTTOCK – KNEE LENGTH	53.6	21.1	61.5	24.2					

Figure 3. Female Crew Member Pertinent Dimensions

2.1.1.6 Metabolic Criteria

A nominal metabolic load of 3,000 cal per man-day, equivalent to 11,900 Btu per man-day; a nominal oxygen consumption of 0.83 kg (1.84 lb) per man-day; nominal carbon dioxide production of 1.02 kg (2.25 lb) per man-day; and a nominal water balance of 3.18 kg (7.0 lb) per man-day will be used for design purposes.

2.1.2 Personal Equipment

2.1.2.1 Clothing and Linens

Crew apparel will include those garments customarily worn by the crew in a shirtsleeve mode of operation. Consideration of a mixed crew will be made in the selection of the garments. All apparel and linens will be expendable types.

2.1.2.2 Grooming Aids

Maximum allowable weight per crewmember for grooming aids of personal choice is 11.79 kg (26 lb).

2.1.2.3 Dosimeters

Individual radiation dosimeters will be provided and worn by each crewman.

2.1.3 General and Emergency Equipment

2.1.3.1 Tools

As specified in the baseline requirements, tools will be provided for generalized repair of station hardware and subsystem. The weight allotment for a set of tools is 68 kg (150 lb); the set will be located in the core module.

2.1.3.2 Portable Lights

Rechargeable portable lights are provided in each module. The portable light will provide 100 Foot-candles at 10-ft distances. After three hours operation the light will provide at least 50 Foot-candles at 10-ft distance.

2.1.3.3 Radiation Detectors

Three small Victorian-type detectors will be provided and installed so that they are equally scattered about the MSS cluster. These should have the capability of being read and reset once per week. A master radiation detector will be centrally located in the MSS cluster and integrated with the central computer.

2.1.3.4 Oxygen Masks

One emergency mask per crewman is provided in each module (three in each module).

2.1.3.5 Fire Extinguishers

Number and location of fire extinguishers TBD.

2.1.3.6 Mobility/Restraint

Mobility aids and restraint will be provided to support normal crew station operations including restraint for large equipment items.

2.1.3.7 First-Aid Kits

First-aid kits will be provided in each module.

2.1.3.8 EMU/PLSS/MMU

A total of four constant volume type pressure garment assemblies (PGA) and their support equipment will be provided. Each PGS will contain a 100% oxygen environment at an operating pressure of 5.52 ± 0.35 N/cm, gage (8.0 ± 0.5 psig). A total of four PLSS/OPS and four umbilicals will be provided in support of the PGA's. EVA/IVA support will be provided by the ECLSS, compatible with the constant volume 5.52 N/cm, gage (8.0 psig) PGA and PLSS/OPS.

2.1.3.9 Housekeeping Equipment

A vacuum cleaner and compactor will be provided for periodic as well as nonroutine housekeeping activities. Facility may be divided among several modules.

2.1.4 <u>Stowage</u>

2.1.4.1 Inventory Management

Inventory control terminals for cargo management and food management will be conveniently located to cargo module docking ports and the galley respectively.

2.1.4.2 Trash Management

Compactors for waste (trash) collection and disposal should be conveniently located to galley, work shops, labs, personal hygiene, and crew medical and health care facilities.

2.1.4.3 Stowage Compartments

Compartments for stowing personal equipment, general engineering equipment, recreational and emergency equipment, and consumables will be provided. They shall be located at or within easy access to the location of maximum use.

2.1.5 Furnishings

2.1.5.1 Control/Man Workstations

In the control center of SM1 and SM4, provisions will be made for two seating restraints/chairs. In addition two book shelves/cases, a two-way radio, a TV camera and monitor will be provided.

2.1.5.2 Sleeping Facilities

Sleeping restraints/bunks in each crew and commander's stateroom will be provided. There will be two sleeping restraints/bunks in the crew staterooms and two in the commander's stateroom. Such a facilities allocation is necessitated to accommodate an alternate crew of six for relatively short periods during crew changeover. In addition to sleeping restraints, there will be one seating restraint in each crew stateroom and four seating restraints in each commander's stateroom. The latter requirement permits conference/work session capability at this location.

In addition to above items, crew staterooms will include television, two-way radio, book shelf, and storage units.

2.1.5.3 Dining and Work Furniture

Ten seating restraints/chairs will be provided in the dining/recreation area. Also two dining surface/tables will be provided in the dining/recreating area. A special surface/table will be provided in the recreation area.

There will be one small work surface/desk (76.2 x 45.7 x 91.4 cm or 30 x 18 x 36 in.) in each crew and commander's stateroom. In addition, one larger work surface/desk (76.2 x 91.4 x 101.6 cm or 30 x 36 x 40 in.) will be provided in each commander's stateroom.

2.1.6 Recreation, Exercise, and Crew Care

2.1.6.1 Exercise Equipment

To support isometric and isotonic exercise programs, a bicycle ergometer, bungee-type devices, and support bars will be provided in modules SM1 and SM4.

2.1.6.2 Recreation Area Furnishings

The passive recreation area, collocated in the dining area, is primarily used for movie/television viewing, reading, and listening to music. It will use the dining area furnishings as described in Section 2.1.5.6.

2.1.6.3 Recreation Equipment and Supplies

Active and passive type recreation equipment and supplies will be provided for the crewmen. The complement will include the following: color television sets, motion picture projector and screen, film library, reading material, tape deck and library, craft material, table games, and puzzles.

2.1.6.4 Medical Care

A primary and a backup medical care facility will be provided, one located in each of the two pressure volumes. The primary medical care facility will include diagnostic and treatment equipment necessary to maintain life of a seriously ill or injured crewman for 96 hours awaiting orbiter arrival for evacuation of the crewman to earth. In addition, this equipment will assist the medical doctor/technician in the decision to call for an interim Orbiter flight for the evacuation of the crewman to earth. An additional requirement of these facilities is to provide for the qualification of the crewman's stay-time in the weightless environment.

The backup facility will provide for crew care in the event the module or pressure volume containing the primary facility is rendered unavailable for short periods of time.

2.1.6.5 Medical-Dental Area Furnishings

Medical-dental furnishings will include: sink and disposal cabinetry, analytical equipment storage cabinet with counter, and pharmaceuticals and equipment storage cabinets.

2.1.6.6 Medical Equipment and Supplies

Medical and dental equipment and supplies are provided for routine crew monitoring as well as for diagnosis and treatment of injury and illness. The medical and dental equipment includes X-ray, drugs, dressing, bandages, wraps, splints, cold packs and heat pads, body and specimen mass measurement devices, rotating litter chair, lower body negative pressure unit, biomonitoring and display equipment, behavioral evaluation equipment, laboratory analysis equipment, refrigerator and freezer, oven, and sterilizer.

2.1.7 Food Management

Daily caloric requirements will be as follows: normal diet, 3,000 cal per man-day, and 2,600 cal per man-day contingency diet. These diets will be satisfied by provisions for stowage, preservation, and preparation of foods in the following proportions: freeze-dried foods, 45%; frozen food, 30%; thermo-stabilized food, 20%; and fresh foods, 5%. As a backup to these requirements, and located in a module in the other pressure volume, preparation and stowage of thermo-stabilized and freeze-dried foods will be provided. Approximately 25% of the total of freeze-dried and thermo-stabilized foods will be provided in the backup galley area.

The station incorporates a primary galley in SM3 and a backup galley in SM2. The primary galley can prepare all types of food. The backup galley has been limited to reconstitution of dried foods and the warming of thermally stabilized foods by a hot plate with a capacity of 14 days.

2.1.7.1 Ovens and Heaters

An electrical resistance oven and a microwave oven are provided.

The performance requirement for the resistance oven is to be capable of heating 2.27 kg (5 lb) of frozen food from -17.75°C (0°F) to 71.11°C (160°F) in 30 min.

2.1.7.2 Freezers and Refrigerators

A freezer with following performance parameters will be provided:

Storage temperature: -23.3°C (-10°F) to -15.0°C (5°F)

Storage capacity: 353.8 kg (780 lb) - 1.06 m^3 (37.5 ft^3) total volume

Storage of experiments: 0.028 m^3 (1 ft^3) at -17.75°C (0°F)

A refrigerator with following performance characteristics will be provided:

Storage temperature: 4.44 ± 2.77°C (40 ± 5°F)

Storage capacity: 0.42 m^3 (15 ft^3) total

Storage of experiments: 0.0085 m^3 (3 ft^3) at 4.44°C (40°F)

2.1.7.3 Preparation Utensils

The rehydration of dehydrated food items will be provided for by several one-hand-operated, metered dispensing devices with volume control. Hot water will be used for items such as soup mixes, hot beverages, fruits, desserts, starches, cereals, etc. Temperature drop in the transport line between the accumulator tank and the dispenser will be minimized. The fit between the outlet orifice of the dispenser and the inlet valve of the food packages will be designed to prevent water or food leakage into the space cabin.

2.1.7.4 Eating Utensils/Cleanup Equipment

The food serving and cleanup subassembly consists primarily of serving trays and eating utensils. This subassembly interfaces with the waste processing subassembly for food packaging, waste food, and other waste disposal. In addition, a chamber sink is provided in the galley to assist in preparation and cleanup operations.

2.1.7.5 Potable Water

The JSC Phase-B baseline requirement indicates that sufficient potable water will be provided to maintain water balance. Based on the 3,000 cal (11,900 Btu) per man-day metabolic load, the human water balance shown in Table 1 will be used for design purposes.

Potable water purity requirements will be in accordance with Table 2. Capability to provide hot water at 68.33 ± 2.77°C (155°F ± 5°F) and cold water at 10 ± 2.77°C (50°F ± 5°F) for crew usage in both personal hygiene areas and food preparation areas will be a design requirement.

The hot and cold water transfer devices are wall-mounted metering units which interface with the pressurized water supply subsystems. The units consist of hot and cold water accumulators, ON/OFF controls, volume cylinders, pistons, and valves to facilitate discharging of preselected quantities of potable water from 0.03 to 2.5 kg (1 to 80 oz). Water is

Table 1

HUMAN WATER BALANCE

Human Water Balance	Cabin Pressure 10.14 N/cm² 14.7 psia	Cabin Pressure 6.9 N/cm² 10.0 psia
Water gain - kg (lb)		
Water of oxidation (from food)	0.35 (0.78)	0.35 (0.78)
Beverages plus water in food	2.87 (6.32)	2.94 (6.47)
Totals	3.22 (7.10)	3.29 (7.25)
Water loss - kg (lb)		
Insensible (lungs + latent)	1.11 (2.44)	1.22 (2.69)
Sensible (perspiration)	0.48 (1.06)	0.44 (0.96)
Urine	1.57 (3.45)	1.57 (3.45)
Water in feces	0.07 (0.15)	0.07 (0.15)
Totals	3.22 (7.10)	3.29 (7.25)

Table 2

AEROSPACE POTABLE WATER SPECIFICATION

Chemical Requirements	Milligrams/Liter or Parts Per Million	Source
Total solids	1000.0	SSA
Cadmium	0.05	SSA
Chromium, hexavalent	0.05	SSA
Copper	3.0	SSA
Lead	0.2	SSA
Silver	0.5	SSA
Iron	1.0	AF
Manganese	0.1	AF
Zinc	15.0	AF
Mercury	0.005	NASA
Nickel	1.0	NR
Chemical oxygen demand	0.5	NR
Selenium	0.05	USPH
	Units	
Color	15.0	AF
Turridity	25.0	AF
Taste and odor	Odor No. 3.0	AF
Ph	6.0-8.0	NASA
Microorganisms	Essentially no coliforms	USPH
Particulates	Level 3	NR MA0610-017

transferred into the food or beverage bag by pushing the bag inlet valve against the discharge port of the spring-loaded transfer device.

2.1.8 Personal Hygiene

The personal hygiene facilities will be divided equally between the two pressure volumes and will be located conveniently with respect to staterooms. There will be a maximum of one personal hygiene facility in each pressure volume, which will include as a minimum the following equipment:

1. One grooming station with sink, hot and cold water mixing capability, teeth brushing facility, soap dispenser, face and hands washing, body sponging, etc.
2. One standup urinal
3. One toilet with urinal (female adaptation)
4. One shower (may be included in only one of the two facilities for the initial station (6 men)
5. Equipment and facility will be arranged for maximum privacy in consideration of a mixed crew (male/female).

2.1.9 Crew

As the baseline Space Station mission is primarily concerned with space research and experimentation, the baseline crew makeup will consist of three operations personnel, two support personnel, and one scientist. As a general rule, each crewmember is assumed to have a basic skill background (8 to 10 years of training and experience) and a capability of achieving a level of proficiency in two similar fields.

A nominal 6-man crew makeup and work allocations for the initial station is presented in Table 3.

Crew duty cycles will be based on a 24-hr period, distributed in a manner to which man has already adapted. Table 4 summarizes the nominal crew duty cycle.

Twenty-seven skills have been identified that are necessary for conduct of experiment operations. Three additional skills have been identified for spacecraft operations.

Table 3

NOMINAL SIX-MAN CREW MAKEUP AND WORK ALLOCATIONS (INITIAL STATION)

Position	Basic Skill and Background	Compatible Experiment Skill Area	Average Man-Hours/Day	
			Station	Experiments
1. Commander	Engineering test and operations command control	Advanced technology, material processing	4.3	5.7
2. Flight controller	Engineering (electronics) navigation and orbital operations	Commander navigation, advanced technology (FF RAM monitoring and control)	5.9	4.1
3. Systems engineer	Engineering (aeronautical/mechanical)	Life support, man-machine interfaces, etc.	5.9	4.1
4. Electromechanical technician	Engineering (mechanical) test and maintenance	Various equipment operator skills	3.8	6.2
5. Electronic engineer	Engineering (electronic) test and maintenance	Various equipment operator skills	3.8	6.2
6. Experiment coordinator or scientist	Medicine or astronomy or physics, etc. program phased	Generally by discipline	1.1	8.9
		Total	24.8	35.2

Table 4

NOMINAL CREW DUTY CYCLE

Activity	Man-Hours
REQUIREMENTS	
Work	10.0
Eating	2.5
Sleep	8.0
Personal and hygiene	1.0
Recreation, medical, and exercise	2.5

SCHEDULE CRITERIA

Medical and Exercise. Consecutive so as to reduce equipment duplication and ensure availability of medical skills.

Work and Sleep. Concurrent with slight differences in start and stop times to reduce loading of eating and personal hygiene facilities. Second- or third-shift assignments are exceptions to this rule, and are only for demonstrable requirements.

Eating. Crew size: 5 or less, concurrent
6 to 12, 50 to 100%

Personal Hygiene. Random periods, 45, 45, and 60 min, with peak a.m. and p.m. periods and a staggered schedule of about 33% of the crew to reduce facilities use. Periods - 15 to 20 min a.m. and p.m., 5 to 10 min in between.

Recreation. Concurrent, generally, to permit as much social interaction as crew desires. Percent using facilities varies with crew size, as follows: 7 to 12 - 70 to 100%, concurrent.

It is estimated that the average crew of six for the initial Space Station will normally require proficiency in no more than 7 to 9 specialty areas, although some additional skills may be required for backup. In general, all critical skill specialties will be backed up by one or more overlap crewmen. Table 5 presents candidate crew skills versus discipline.

2.2 MODIFICATIONS TO BASELINE REQUIREMENTS NECESSARY FOR OPTION-L APPLICATION

In this section, the changes to the baseline requirements (discussed in Section 2.1) that are necessitated by the SCB 7-man Option L application are discussed. Only requirements deviating from the baseline requirements will be considered — in all other cases a no-change condition in requirements is implied.

2.2.1 Ceiling Height

Based on a 14-ft inside diameter of the modules, the ceiling height in general mobility areas will be 84 in. No multideck configurations for general mobility areas are being considered at this time. (JSC Phase-B baseline described in Section 2.1.1.1.)

2.2.2 Acoustics

Continuous noise levels will not exceed NC-50 as interpolated from Figure 4 (JSC Phase-B baseline described in Section 2.1.1.4).

2.2.3 EMU/MMU

The Space Shuttle EMU (Extravehicular Mobility Unit), which includes both a pressure garment assembly and the primary life support system, will be the baseline EVA garment for Option L 7-man Space Construction Base. It will have the characteristics identified in Table 6 and will be used in accordance with the EVA groundrules shown in Table 7.

The Space Shuttle MMU (Manned Maneuvering Unit) will be available for EVA translation to space construction base elements which are detached from the station cluster or are at such distances from the airlock that crane operations or manual translation is not feasible. Where an MMU is used a second MMU must be available and manned for emergency retrieval of the astronaut using the first MMU.

Table 5

CANDIDATE CREW SKILLS VERSUS DISCIPLINE — EXPERIMENT OPERATIONS

Discipline	Biological Technician	Microbiological Tech	Biochemist	Physiologist	Astronomer/Astrophysicist	Physicist	Nuclear Physicist	Photo Tech Cartographer	Thermodynamicist	Electronic Engineer	Mechanical Engineer	Electromechanical Tech	Medical Doctor	Optical Technician	Optical Scientist	Meteorologist	Microwave Specialist	Oceanographer	Physical Geologist	Photo Geologist	Behavioral Scientist	Chemical Technician	Metallurgist	Material Scientist	Physical Chemist	Acronomist	Geographer
Life Sciences	X	X	X	O								X	X	X							O	X					
Astronomy					X	X				X		X		X	X												
Communication Navigation										O		O		O			O										
Earth Observations								O				X		X		O		O	O	O						O	O
Technology						X			X	X	X	X															
Space Physics					X	X	X		X			X													X		
Materials Sciences	X											X										X	O	X			

O Initiated Subsequent to Growth Buildup

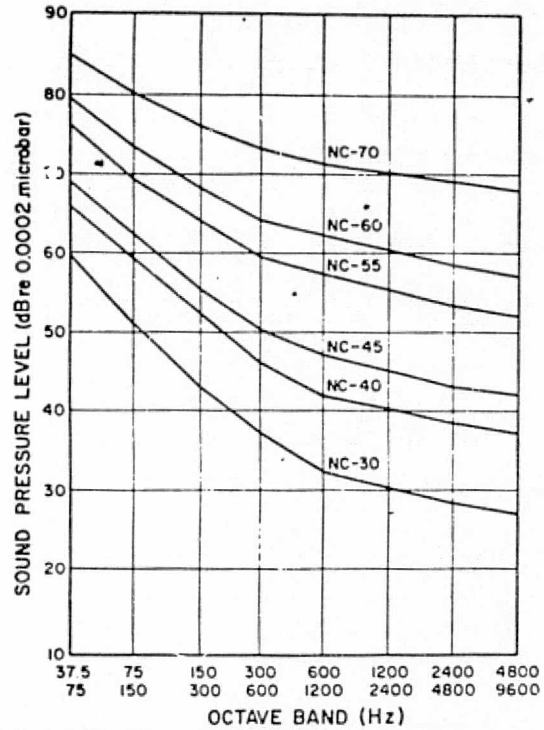


Figure 4. Noise Criteria (NC) Curves for Speech Communication

Table 6

EMU CHARACTERISTICS

Major Components

- Liquid-cooled ventilation garment (LCG)
- Hard upper torso (with PLSS/SOP and DCM attached)
- Lower torso (includes boots)
- Gloves
- Helmet (with EVA visor assembly and communications carrier assembly)

Noncustomized

PLSS/SOP

- Primary oxygen
- Gas ventilation circuit
- Water transport loop
- Feed-water loop
- Electrical systems (includes 16.8V battery)
- Secondary oxygen pack (SOP)

Display and Control Module (DCM)

- Status displays and controls

Weight - 76.75 kg (169.2 lb)

Maximum Depth (front of DCM to back of PLSS/SOP) - 0.502m (19.75 in)

Maximum Breadth (at elbows) - 0.711m (28 in)

Table 7

EVA GROUND RULES

EVA work period:	Two 3-hr periods with 2-hr interim
Pre/post preparation:	Pre-AVA - 40 min
	Post-EVA - 30 min
10-min rest period every 2 hr (in-suit refreshment)	
No prebreathing	
6-day week	
One/two EVA crewmen cherry picker	
Energy expenditure:	900 avg Btu/hr
Suit can be designed for adequate protection in all orbits, exclusive of solar flares	
Minimum 2-man EVA crew	
No backup man in airlock	
Suit life extended above present technology	
Suit can be washed/dried between shifts	
Independent life support suit (no umbilical)	

2.2.5 Furnishings

2.2.5.1 Control/Man Workstations

One primary and one emergency control center will be located in the habitation module and crew support module respectively. Two seating restraints will be provided in each of the control centers. The modules will also include two book cases, an intercom, and a TV camera and monitor. (JSC Phase-B baseline described in Section 2.1.5.1.)

2.2.5.2 EVA Work Stations

For routine EVA construction activities, work stations will be provided at the site of activities to restrain two space suited astronauts and permit two-handed assembly operations. These work stations will also provide restraints for tools and equipment, will provide lighting for local illumination, and will provide supplemental controls for remote crane operation.

For anticipated frequent EVA maintenance, astronaut restraint devices will be provided at the site of the maintenance and will be designed in such a way as to permit two-handed maintenance operations. Hand rails and hand holds will be provided on the module exterior to facilitate safe and efficient translation to the maintenance work site.

To accommodate unanticipated and infrequent EVA maintenance, portable restraints will be provided which can be emplaced by the astronaut at any potential EVA maintenance site. Liberal provisions of EVA handrails and handholds on the SCB exterior surfaces will be utilized as a design goal to permit access to exterior locations where EVA maintenance may be required.

2.2.5.3 EMU Donning Stations

Donning stations, which also serve for EMU storage, will be provided at appropriate locations within the SCB to enable efficient and rapid EMU donning and doffing. Donning stations will be located adjacent to, but not inside, the airlocks through which EVA will ordinarily be conducted. Sufficient volume will be provided at the stations to permit the full range of astronaut movements required in donning and doffing the EMU.

2.2.5.4 EMU Recharge Stations

The process of EMU recharging includes replenishment of consumables, draining of condensate, battery recharging, and space suit drying. A sufficient number of EMU recharge stations will be provided to accommodate the maximum number of EMU's which will be undergoing recharging at any one time.

Recharge stations will be located adjacent to, but not inside, the airlocks from which EVA crewman egress, and in close proximity to the suit donning stations.

2.2.5 Sleeping Facilities

Sleeping restraint/bunks in each crew stateroom will be provided. There will be two sleeping restraints/bunks in the crew staterooms. Such facilities allocation may be necessitated to accommodate an alternate crew of seven for relatively short periods during crew changeover. In addition to sleeping restraints, there will be one seating restraint in each crew stateroom. In addition to the above items, the staterooms will include an intercom book shelf, and storage unit. (Phase-B baseline described in Section 2.1.5.5.)

2.2.6 Dining and Work Furniture

Seven seating restraints/chairs and one dining surface will be provided. In addition, a recreation and lounge table will be provided. There will be one small work surface/desk in each crew stateroom. (Phase-B baseline described in Section 2.1.5.6.)

2.2.7 Recreation, Exercise, and Crew Care

2.2.7.1 Exercise Equipment

To support isometric and isotonic exercise programs, a bicycle ergometer, bungee-type devices, and support bars will be provided in the crew support module. (Phase-B requirement described in Section 2.1.6.1.)

2.2.8 Medical Care

One medical care facility, located in the crew support module, will be provided. This facility will include diagnostic and treatment equipment necessary to maintain life of a seriously ill or injured crewman for 96 hr, while awaiting Orbiter arrival for evacuation. An additional requirement of this facility is the qualification of the crewman's stay-time in weightless environment. (Phase-B baseline described in Section 2.1.6.4.)

2.2.9 Food Management

One galley, located in the Crew Support Module, will be provided. There will be no backup galley. (Phase-B baseline described in Section 2.1.7.)

2.2.10 Freezers and Refrigerators

A freezer with following performance parameters will be provided:

Freezer

Storage temperature: -23.3°C to -15.0°C (-10°F to +5°F)

Storage capacity: 353.8 kg (780 lb) - 1.06 m^3 (37.5 ft^3) total volume

Refrigerator

Storage temperature: 4.44 ± 2.7°C (40 ± 5°F)

Storage capacity: 0.42 m^3 (15 ft^3)

Requirement for storing experiments in the freezer and refrigerator has been eliminated as this requirement, if needed in the SCB, would be integrated with the modular laboratory facilities (Phase-B baseline described in Section 2.1.7.3.)

2.2.11 Personal Hygiene

The personal hygiene facilities will be divided between the habitation module and crew support module as follows: one combination toilet/urinal (female adaption) in the habitation and crew support module; one grooming station in habitation and crew support module; and one shower facility in the habitation module. All facilities will be arranged for maximum privacy in consideration of mixed crew. (Phase-B baseline described in Section 2.1.8.)

2.2.12 Crew

The SCB will have a crew of seven. Since the primary mission of the base is construction in space, the crew makeup will include the following skills: command/control (commander), EVA specialist, crane operator, fabrication/assembler, medical technician, and electrical/mechanical technician. In general, all critical skill specialties will be backed up by one or more overlap crewmen.

The construction crew duty cycles will be based on two overlapping 10-hr shifts per 24-hr. The schedule is depicted in Figure 5.

2.3 IMPACT OF GROWTH ON SCB

In this section the impact of growth in crew size on the SCB configuration and requirements is considered. Specifically, areas sensitive to growth changes to 14-man and 21-man crew configurations are discussed.

2.3.1 Areas Sensitive to Growth to 14-Man Configuration

One additional habitation module is required to provide sufficient crew accommodations and personal hygiene facilities.

It is anticipated that an additional crew support module will not be required, as expected flexibility in crew scheduling will permit the crew support module to be effectively used by the increased crew size. The control center volume of the second habitation module could be utilized for some functions normally supported by the crew support module (e.g. provide space for limited meal service).

As a result of the increased crew size, a backup medical facility may have to be provided. The backup facilities would be used primarily for first aid, short term emergency treatment, and storage of backup pharmaceuticals.

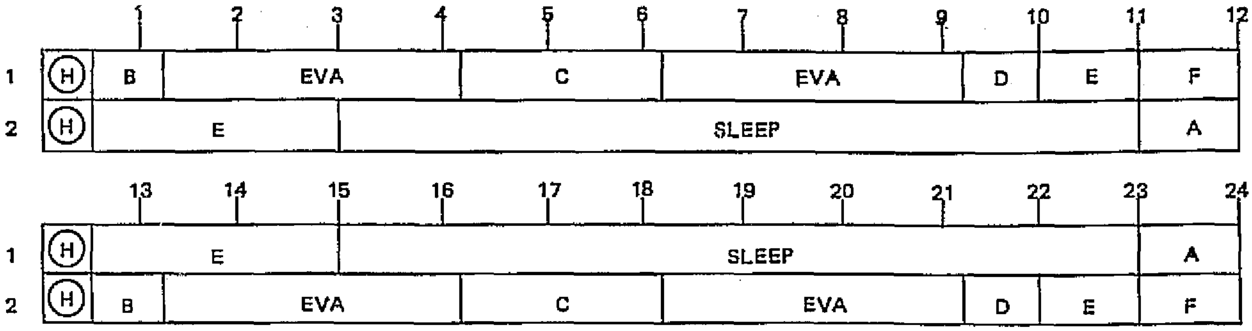
As the demand on housekeeping facilities increases, an additional vacuum cleaner and trash compactor will be required to provide flexibility for routine as well as nonroutine housekeeping activities.

With the increase in personal hygiene facilities by one combination toilet/urinal, one grooming station, and one shower, the need for additional facilities is not expected in crew support module.

2.3.2 Areas Sensitive to Growth to 21-Man Configuration

With the increase in crew size to 21, two additional habitation modules (a total of three) will be required to provide for sufficient crew station volume and personal hygiene facilities.

One additional crew support module (a total of two) will be required to provide adequate galley, dining/recreation, and medical/exercise facilities to support the crew size of 21.



LEGEND

A – PERSONAL HYGIENE AND BREAKFAST (1 HOUR)

B – TRANSFER AND PRE-EVA (45 MIN)

C – MIDSHIFT BREAK (LUNCH, PERSONAL HYGIENE, DOFF/DON SUIT, REST) (2 HOURS)

D – POST-EVA (45 MIN)

E – EXERCISE, RECREATION, PERSONAL HYGIENE

F – DINNER (1 HOUR)

H – PRE/POST SHIFT BRIEFING (COMMANDER AND BOTH CONSTRUCTION CREWS) (30 MIN)

Figure 5. 24-Hour Schedule – Two 10-Hour Overlapping Shifts

2.4 AREAS REQUIRING FURTHER INVESTIGATION

One of the areas requiring further investigation is the tradeoff of expendable vs. washable clothing and linens to determine which concept will provide the greatest savings in weight and storage volume, as well as crew acceptability.

The possible reduction in weight and storage volume requirements will be evaluated with respect to weight and volume penalties imposed by the addition of a washing machine and a dryer, as well as the resultant increase in power and waste management requirements. Also, increased water requirement must be considered.

Another trade-off that merits consideration is that of private crew staterooms vs. dormitories for larger crews. Dormitory-type quarters would permit more economical (in terms of volume) arrangement of sleep facilities, permitting more flexibility in the configuration and arrangement for dining/recreation, galley, and medical/exercise facilities. Potential crew acceptability and the impact of free volume gains on facility layout will be investigated.

As some potential weight and stowage volume savings are indicated, the possible replacement of disposable eating utensils with reusable type utensils will be investigated. The impact on stowage volume, weight, water, and power requirements as well as crew hygiene will be considered.

Section 3
SUBSYSTEM DEFINITION (SCB 7-Man Option L)

In this section, the 7-Man Space Construction Base Option L (Figure 6) crew and habitability subsystems are defined in more detail to reinforce and expand on the requirements that were discussed in Section 2 of this document.

3.1 PERSONAL EQUIPMENT

The primary function of the personal equipment subassembly is to provide the crew with clothing, linens, grooming aids, and a personal radiation dosimeter.

The crew apparel and linens are made of absorbent material which provide warmth and comfort in a shirtsleeve atmosphere provided by the environmental control subsystem. The crew personal effects include toilet articles, grooming aids, cleaners, and other items of the individual crewman's choice. All items of personal effects are selected by the individual crewman; however, these items must be consistent with the design and performance capability of other onboard subsystems.

Crew apparel includes those garments customarily worn by the crew in a shirtsleeve mode of operation, with consideration being given to a mixed crew utilization. Based on a utilization rate of one change of socks and undergarments every other day, one change of overclothes per week, and one change of linens per week, the estimated weight and volume of the crew apparel and linens is approximately 42.18 kg (93 lb) and 0.89 m^3 (4.2 ft^3) per crewman.

Individual radiation dosimeters will be worn by each crewmember at all times. The approximately weight per dosimeter is 0.045 kg (0.1 lb).

3.2 GENERAL AND EMERGENCY EQUIPMENT

Tools have been provided for generalized repair of station hardware and subsystems. The basic complement of tools weighs approximately 68 kg (150 lb) and is centrally located in the core module. Rechargeable portable lights weighing approximately 2.7 kg (5 lb) each are provided in all occupied modules. The general and emergency equipment subassembly also contains the pressure suit assemblies, including water cooled garmets, as well as portable life support systems to support extravehicular activities. Mobility aids and restraints are provided in each module to support normal crew station operations, including restraint for large equipment items at an estimated cost in weight of 54.43 kg (120 lb). In addition to the personal dosimeters, resettable radiation detectors are distributed at critical locations in the station to provide a status of the radiation environment to which the crew is subjected. The number and location of the detectors remains to be determined at this time, however the estimated weight per detector is 4.54 kg (10 lb).

Emergency oxygen masks with integral oxygen bottle will provide 10 min oxygen supply. Seven oxygen masks (one for each crewman) will be located in the core module to provide a central location that is within rapid and easy access from either the habitation or crew support module. Since the fabrication and assembly module is further removed from the habitation/crew support/core module complex, precluding rapid access, four additional oxygen masks will be provided in that location (it is anticipated that no more than four crewmen will occupy the fabrication and assembly module at any given time). The estimated unit weight per emergency oxygen mask/bottle is 1.13 kg (2.5 lb), thus the total weight for 11 units is 12.47 kg (27.5 lb).

The housekeeping subsystem provides biocide wipes to enable the crew to maintain the interior cleanliness of station surfaces. Receptacles for the collection and stowage of wet and dry debris are provided. Particles in air will be filtered and as a result, periodic replacement of filters will be necessary. A vacuum cleaner is provided for periodic as well as nonroutine housekeeping activities. The vacuum cleaner facility is divided among the several modules.

3.3 STOWAGE

Stowage encompasses all phases of loose equipment management. Loose equipment is defined as items that are not permanently attached to the spacecraft. Loose equipment management involves location, restraint, launch protection, and on-orbit utilization and inventory control of all items of equipment handled and moved by the crew.

Daily usage items, such as collection and hygiene equipment associated with waste management, trash collection equipment, food preparation and consumption equipment, and personal equipment are located at or within easy access to the location of maximum use. To illustrate the magnitude of stowage facilities requirements, based on a crew size of 7 and a 30-day on-station stay, the following data are provided:

- Dry food: 287.12 kg (633 lb) (with no contingency plans included)
- LiOH cartridges: Number of cartriges required = 105
  Total weight of cartridges - 304.81 kg (672 lb)
- Liquid Waste Tankage: Weight of liquid waste to be stored 857.3 kg (1,890 lb)
  Number of storage tanks required = 13
  Weight of storage tanks empty - 203.4 (448.5 lb)

Stowage facilities for general resupply items are provided with provisions for grouping together of like items with access independent of other groups.

For the purposes of stowage management and on-orbit inventory, a stowage location identifier system is provided that is capable of rapid change and is compatible with logical cataloging.

To facilitate usage, the stowage facilities are capable of being packed outside of the vehicle and then installed without disturbing the stowed contents.

A food stores inventory capability is provided by means of an on-time terminal, located in the galley facility, which is connected to the Space Station central computer. The inventory hardware will be an ISS remote terminal unit. The approximate weight and volume of such a unit is 18.14 kg (40 lb) and 0.04 m^3 (1.5 ft^3) respectively.

3.4 FURNISHINGS

3.4.1 Control Center Facility

All crew compartments shall be designed for maximum habitability. The recommended volume allocated for the control center facility, located in the habitation module is 6.1 m^3 (215 ft^3). Of that volume, 2.97 m^3 (105 ft^3) should be available for operations space with the remaining 3.11 m^3 (110 ft^3) taken up by equipment. Two seating restraints/chairs are provided at the control console at an estimated cost in weight of 4.54 kg (10 lb) each.

3.4.2 EVA Work Stations

Work stations to be used for routine construction EVA will be partially enclosed work platforms capable of being mounted on the end of the crane arm, in the cherry-picker mode. Each such work station will be a minimum of 1.22m (4 ft) deep and 1.83m (6 ft) wide and will provide the following capabilities:

- Support for two EVA crewmen
- Storage and restraints/tethers for small parts and tools
- Crew restraints and mobility aids
- Voice Communications and data entry
- Surveillance TV
- Services such as power, pneumatics, and fluids
- Remote control of crane operation

Crew restraints provided at this work station will include foot restraints and waist restraints. Design of the restraints shall permit the EVA crewman to extricate himself from the restraints and translate to work locations outside the envelope of the work station platform. Crew tethers with a minimum length of 6.1m (20 ft) shall be provided, with one end firmly attached to the work station structure, and the free end capable of being attached to the EVA crewman to permit him to safely leave the work platform and translate to another work location.

EVA work stations for anticipated frequent maintenance shall be permanently premounted at the planned work location. Each work station shall consist of a Skylab type EVA foot restraint (as shown in Figure 6) and a handhold (see Figure 7) above the foot restraint between waist and shoulder level, for crew ingress/egress. The foot restraints shall be capable of withstanding a 445 N (100 lb) working load in any direction. Illumination at the maintenance location may be provided by permanently installed lights or by portable lamps and shall provide a minimum of 55 lumens per square meter (5 Foot-candles) at the working surface.

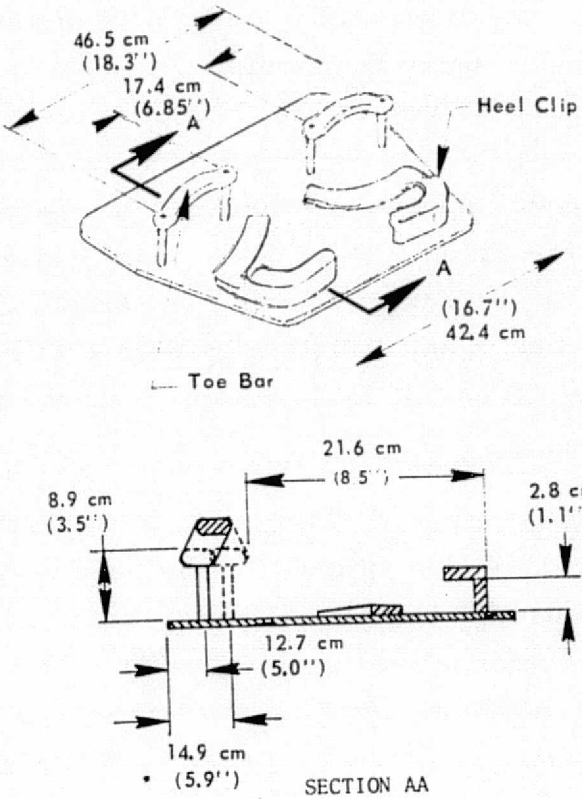


Figure 6. EVA Foot Restraint

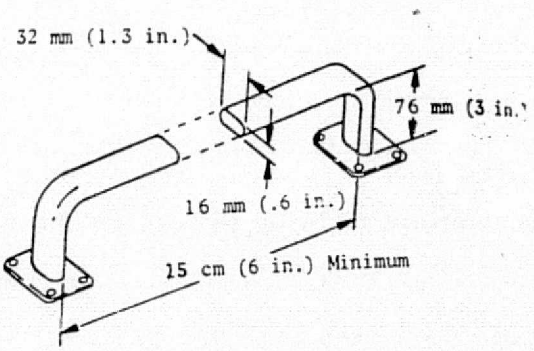


Figure 7. EVA Handhold

EVA work stations for infrequent or unanticipated EVA maintenance shall be provided by portable, crew mounted restraints. Portable foot restraints, portable handholds, and chest or waist tethers shall be provided, with the crew having the option to use the most appropriate restraint (or combination of restraints) for the specific maintenance application. Tethers alone may be used when minimum crew activity at the EVA maintenance site is required, and when used they shall be one-hand operable and capable of withstanding a 2, 046 N (460 lb) working load in any direction. Tethers will be adjustable to position the crewman from 75 cm (30 in) to within 30 cm (12 in) of the attach point.

3.4.3 EMU Donning Stations

Suit donning and doffing will be done at specific locations designated as Donning Stations. Four donning stations will be located in the fabrication and assembly module to accommodate the shift overlap of the two 2-man EVA construction crews. Additional donning stations will be located adjacent to (or inside) other airlocks from which EVA may be initiated.

Each donning station will be equipped with foot restraints to secure the suit at the boots and with handholds to stabilize and control the crewman's motions during donning and doffing. Each donning station will provide restraint devices for holding loose EMU items (e.g., helmet, gloves) during the don/doff process. Donning stations shall be located a maximum of 1.83m (6 ft) from the recharge station so that the 2.13m (7 ft) service and cooling umbilical (SCU) can be used during donning and doffing for suit cooling and oxygen purge.

3.4.4 EMU Recharge Stations

EMU Recharge Stations support EVA by providing the following facilities and capabilities to the EMU:

- Oxygen
- Suit cooling during pre- and post-EVA
- Power
- Audio communications
- Feedwater resupply
- Condensate water drain

- Battery recharging
- Suit drying

One recharge station (with appropriate controls, displays, and inlet/outlet connections for water, oxygen, communications, and power) will be located adjacent to the airlock used for routine EVA and within 1.83m (6 ft) of the EMU donning stations. A sufficient number of SCU's will be provided to accommodate the maximum number of EMU's undergoing recharge (including drying) at any one time.

The recharge station will supply 620 N/cm^2 (900 psig) oxygen for purging and recharging the EMU, for recharging portable oxygen masks, and for crewman prebreathing (if required). For each EVA, pre-EVA purging will require 0.38 kg (0.83 lb) of oxygen and post-EVA recharging will require 0.73 kg (1.6 lb) of oxygen for each EMU.

The recharge station will contain LCG fittings for supplying cooling to the crewman during pre- and post-EVA periods when the PLSS cooling system is not functioning. An LCG heat exchanger will be provided through which the EMU pump will circulate LCG water, rejecting up to 140 gm-gal/sec (2,000 Btu/hr) per crewman.

The recharge station will supply 17.0 ± 0.5 Vdc power for operation of EMU components such as pumps and fans (bypassing the EMU battery), and for recharging the EMU batteries. Capability will be provided for recharging a minimum of four EMU batteries simultaneously, either installed in the EMU or when removed from the EMU.

The recharge station will provide hardline communications to and from the partially suited crewman during pre- and post-EVA operations. RF voice communication is provided separately for the fully-suited crewman.

The recharge station will supply potable water for post-EVA recharging of EMU feedwater reservoirs and will drain condensate water collected during EVA from the EMU. Each EMU will require 4.1 kg (9.0 lb) of potable water and will be drained of approximately 0.9 kg (2 lb) of condensate water following each EVA.

The recharge station will supply heated air (or oxygen) up to a maximum temperature of 48.9°C (120°F) to reduce the water remaining in the suit to ≤50g (0.11 lb) and the relative humidity in the suit to ≤55%, the maximum permitted for suit storage.

3.4.5 Crew Staterooms

The individual crew staterooms, located in the habitation module, provide personal quarters for sleeping, relaxing, storage, and communication. The recommended volume for individual crew staterooms is 5.55 m^3 (196 ft^3), with approximate dimensions of 2.13 x 2.13 x 1.22m (7 x 7 x 4 ft), and should have a capability for dual occupancy for short periods during crew change-overs. Approximately 0.93 m^3 33 ft^3 of that volume will be occupied by furnishings and personal equipment. The furnishings include two sleeping restraints, one work surface, storage drawers, and a closet.

3.4.6 Dining/Recreation Area

The dining/recreation (passive) area should be located adjacent to the galley and should be conveniently accessible from crew quarters. In addition to providing a dining facility for a crew of seven, it should also provide space and equipment for passive recreation such as watching movies or television, listening to music, or reading. Consequently, facilities for storage and use of a projector and audio video unit should be provided. Assuming that the dining surface/table can be stowed (e.g., raised into the ceiling) for converting the dining room into a recreation room configuration, the recommended volume of the dining/recreation should be 15.23 m^3 (539 ft^3) with approximate dimensions of 2.13 x 2.13 x 3.66m (7 x 7 x 12 ft). The following furnishings for the dining/recreation facility should be included: nine seating restraints/chairs (two of which are at lounge table), one dining surface/table 2.1 x 0.76m (84 x 30 in.), and one recreation/lounge table (0.76m or 30 in. dia).

3.5 RECREATION, EXERCISE, AND CREW CARE

A separate area for exercise/active recreation should be provided. This area will provide space for conduting exercises and competitive activities. The recommended volume for such a facility is 20.67 m^3 (730 ft^3). Active and passive-type recreation equipment and supplies are provided in the dining/recreation facility of the crew support module. The complement includes the following: color television sets, motion picture projector and screen, film library, reading material, tape deck and library, craft materials, table games, and puzzles. It is estimated that the total weight of passive recreation devices is 90.72 kg (200 lb). The isometric and isotonic exercise equipment includes a bicycle ergometer, bungee-type devices, and support bars with an approximate total weight of 22.68 kg (50 lb) with volumes to be determined.

A separate medical facility located in the crew support module is provided, as it needs to be located apart from noise and contaminant-producing activity areas. Because of the large number of equipment items required to support this facility, its overall volume allocation is estimated at 12.69 m^3 (448 ft^3) with approximate dimensions of 2.44 x 2.44 x 2.13m (8 x 8 x 7 ft). Of that volume, approximately 8.5 m^3 (300 ft^3) will be taken up by equipment.

All medical and dental equipment and supplies are located in the medical/exercise area in the crew support module, and provide for routine crew monitoring as well as diagnosis and treatment of injury and illness. The weight and size of the medical equipment is summarized in Table 6.

3.6 FOOD MANAGEMENT

The food management facility provides for food storage, preparation, cleanup and inventory control. It accommodates a large range of food types, cooking operations, and crew use modes.

The station incorporates a galley in the crew support module adjacent to the dining/recreation facility. The overall volume allocation for the galley facility is approximately 12.49 m^3 (441 ft^3), based on an area requirement of 5.85 m^2 (63 ft^2) and a ceiling height of 2.13m (7 ft).

The galley provides a freezer adequate for 120 days of frozen food storage, a refrigerator, and room-temperature food storage adequate for 120 days. The significant preparation equipment includes an electrical resistance oven, a microwave oven, and a dried-food reconstitution unit. Reusable trays with disposal utensils are provided.

The food storage subassembly includes provisions for room-temperature food storage, refrigerated-food storage, and frozen-food storage adequate for 120 days. The room-temperature food storage includes dried food and thermostabilized food, with 100 ft^3 of storage provided in the galley facility.

The freezer consists of insulated compartments capable of storing 2.83 m^3 (100 ft^3) of packaged food at -23° to -15°C (-10° to +5°F). Several access doors are provided to minimize heat gain during food removal.

Since 5% of the total food supply is provided in the form of fresh foods, a refrigerator capable of storing 0.43 m^3 (15 ft^3) at 4.44° ± 2.77°C (40 ± 5°F) is provided. At that temperature, fresh food can safely be stored for a period of 2 weeks. This provides the crew with 2 weeks of fresh food supply following each resupply mission.

The food preparation subassembly consists of an electrical resistance oven, a microwave oven, a hot and cold water unit for dried food reconstitution, and miscellaneous preparation utensils.

The resistance oven has the capability of heating seven man-meals from a frozen condition to 71.11°C (160°F) in 0.5 hours. The microwave oven provides flexibility and operations by providing capability to thaw frozen food, heating snack items quickly and providing single man-meal preparation. It consists of an insulated envelope capable of heating one to six meals.

The rehydration of dehydrated food items will be provided for by several one-hand-operated, metered dispensing devices with volume control. Hot water will be used for items such as soup mixes, hot beverages, fruits, deserts, starches, cereals, etc. Temperature drop in the transport line between the accumulator tank and the dispenser will be minimized. The fit between the outlet orifice of the dispenser and the inlet valve of the food packages will be designed to prevent water or food leakage into the space cabin. The hot and cold water transfer devices are wall-mounted metering units which interface with the pressurized water supply subsystems. The units consist of hot- and cold-water accumulators, ON/OFF controls, volume cylinders, pistons, and valves to facilitate discharging of preselected quantities of potable water from 0.028-2.47 kg (1 to 80 oz). Water is transferred into the food or beverage bag by pushing the bag inlet valve against the discharge port of the spring-loaded transfer device.

The food serving and cleanup subassembly consists primarily of serving trays and eating utensils. This subassembly interfaces with the waste processing subassembly for food packaging, waste food, and other waste disposal. In addition, a chamber sink is provided in the galley to assist in preparation and cleanup operations.

Food recording is required for food stores inventory and crew intake and medical measuring. An on-time terminal located in the galley facility connected to the space station ISS central computer is provided. The inventory control subassembly will graphically display (on command) an individual crewman's chart for food stores data, or other selected data. Software will be required to implement the inventory control and facilitate (MBLMS) crew monitoring. The inventory control hardware will be an ISS remote terminal unit.

The physical characteristics of the food management equipment are as follows:

Function Subassembly	Weight kg (lb)	Volume m^3 (ft^3)
Freezer	136.1 (300)	2.46 m^3 (87)
Refrigerator	54.43 (120)	0.34 m^3 (12)
Resistance oven	36.29 (80)	0.11 m^3 (4)
Microwave oven	34.02 (75)	0.14 m^3 (5)
Reconstitution unit	12.25 (27)	0.02 m^3 (0.7)
Inventory control	18.14 (40)	0.04 m^3 (1.5)
Utensils	102.51 (226)	0.09 m^3 (3)
Totals	393.72 (868)	3.21 m^3 (113.2)

3.7 PERSONAL HYGIENE

Personal hygiene accommodations, consisting of waste management, grooming, and shower facilities, are divided between the habitation and crew support modules to permit convenient access with respect to crew staterooms, dining/recreation, and medical facilities.

The waste management facility provides a safe, reliable system that provides for collection and disposal of biological wastes without contaminating the cabin environment with waste material. The facility includes hardware for waste collection, processing, and stowage/disposal as well as odor and particulate control. One each combination toilet/urinal with a facility volume of 1.84 m^3 (65 ft^3) is provided in the habitation and crew support module. They are completely enclosed to afford maximum privacy and accommodate both male and femal crewmembers.

Also one each grooming station with a facility volume of 0.85 m^3 (30 ft^3) is provided. The grooming facility is equipped with a sink, hot- and cold-water mixing capability, teeth brushing facility, and soap dispenser to permit face and hands washing, body sponging, and other miscellaneous body grooming activities. It should be conveniently located with respect to crew staterooms and waste management facilities, and should offer complete privacy.

To accommodate the 7-man crew, one shower facility is provided in the habitation module as the anticipated shower utilization frequency is 1 shower/ man/week. The shower facility volume is 1.70 m^3 (60 ft^3), and is provided with a handheld spray head to ensure coverage of all body areas. It also includes foot restraints to permit freedom of both hands for washing. The minimum water consumption per shower is 2.72 kg (6.0 lb) at a temperature of 37.78° - 43.33°C (100° - 110°F).

3.8 CREW

The Space Station shall be under the control of one man who is responsible for the safety and operation of the vehicle. This station commander should delegate command responsibilities to other individuals to assure operation and safety of all on-board systems and the accomplishment ofmaintenance and housekeeping tasks, and to perform all basic flight operations and mission tasks. This delegation provides backup to the command function and makes maximum use of onboard specialties.

The SCB crew is divided into two functional categories: flight operations crewmen and support technicians. These terms are oversimplifications used to designate basic skills and background requirements, and should not be interpreted as specific areas in which a crewman will be exclusively utilized. Since the primary purpose of the SCB is construction in space, it is expected that the entire crew will be involved to some degree in the space construction activities.

In the case of the support personnel, the term "technician" generally implies an individual well grounded in the technical skills, including manual and cerebral activities. It is not intended to rule out academic training as technicians with engineering degrees would probably participate in the initial SCB missions.

The SCB will have a crew of seven and the following skills will be included in the crew makeup: command/control, EVA specialist, crane operator, fabrication/assembler, medical technician, and electrical/mechanical technician. All crewmembers will be EVA-qualified.

The crew duty cycles will be based on two overlapping 10-hour shifts per 24-hours to permit most efficient crew utilization. The nominal crew schedule has been presented elsewhere in this document (Section 2.2.12, Figure 5).

Part 14

HABITABILITY SUBSYSTEM CONSIDERATIONS FOR SHUTTLE-TENDED OPTION L'

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HABITABILITY SUBSYSTEM CONSIDERATIONS FOR SHUTTLE-TENDED OPTION L'

In the Shuttle-tended space construction base, the Shuttle's habitability subsystem is the only means of providing crew support equipment, furnishings, supplies and services, and procedures necessary to assure safe living and working conditions for the space construction base crew. The following paragraphs are concerned with the habitability requirements that are specific to Option L', how they compare with the Shuttle's habitability capabilities, and the impact Option L' requirements have on Shuttle's baseline habitability subsystem.

Section 1
HABITABILITY SUPPORT REQUIREMENTS

In this section, the crew habitability requirements as they apply to the Shuttle-tended space construction base are presented. The habitability requirements specifically discussed include free volume, food management, personal hygiene, sleeping accommodations, recreation, stowage, and extravehicular activity considerations.

In considering the habitability requirements in support of a space construction base for extended periods of up to 180 days in the Shuttle-tended mode, the following assumptions are made: (a) two overlapping 10-hour shifts per 24-hours will be utilized for most efficient space construction base operation; (b) to support a two shift operation, a 10-man crew will be required, consisting of three flight crew and seven support (construction) personnel; and (c) participation of the flight crew in construction work is not expected.

1.1 FREE VOLUME

To minimize crew impairment during Shuttle-tended space construction missions of 30-180 days duration, the minimum free volume requirement per crewman is 4.96 - 5.66 m^3 (175 - 200 ft^3). Consequently, for a crew of 10, the total free volume requirement is 49.6 - 56.6 m^3 (1,750 - 2,000 ft^3). Free volume is defined as the space available in a specific location for body movement and transfer within the location, ingress to and egress from the location, and performance of tasks at the location.

1.2 FOOD MANAGEMENT

The following metabolic criteria will be used for food management design purposes: A nominal metabolic load of 3223 kcal (12,800 Btu) per man-day; a nominal oxygen consumption of 0.94 kg (2.08 lb) per man-day; nominal carbon dioxide production of 1.17 kg (2.58 lb) per man-day; and a nominal water intake of 5.55 kg (12.23 lb) per man-day.

The food management requirements will be satisfied by provisions for stowage, preservation, and preparation of foods in the following proportions: freeze-dried foods, 45%; frozen food, 30%; thermo-stabilized food 20%; and fresh foods, 5%.

1.3 PERSONAL HYGIENE

The personal hygiene facility shall be provided in the middeck area, which will include, as a minimum, the following equipment: (a) one grooming station; (b) one commode/urinal; and (c) one shower. The equipment and facilities will be arranged for maximum privacy in consideration of a mixed crew.

1.4 SLEEPING ACCOMMODATIONS

Five sleeping accommodations will be provided, assuming that "hot bunking" between shifts will be utilized. They should be located apart from noise and contaminant-producing work areas and should permit convenient access to the personal hygiene facility. As an alternate, curtains, earmuffs, and eye shades may be utilized to provide privacy. A horizontal orientation of the restraints/bunks with reference to the floor should be maintained wherever possible to provide a familiar and therefore a more reassuring visual orientation to the surroundings.

1.5 RECREATION, EXERCISE, AND CREW CARE

Active and passive type recreation equipment and supplies will be provided for the crewmen. The complement should include the following: color television set, reading material, tape deck and library, table games, and puzzles. Also, exercise equipment should be provided and should include a bicycle ergometer, bungee-type devices and support bars.

Sufficient medical care capability will be provided to cope with minor injury or illness. (It is assumed that in case of major injury or illness, the Orbiter would return to earth.)

1.6 STOWAGE

Adequate stowage facilities will be provided for the purpose of loose equipment management. Stowage compartments shall provide restraint, launch protection, and on-orbit utilization and inventory control. The stowage compartments will be located in the immediate vicinity to location of maximum functional use as much as practicable.

1.7 EVA

The requirements which the Orbiter-tended SCB imposes on the Shuttle EVA system are identified in this section and categorized under the appropriate elements of the Shuttle EVA system. One general requirement, not specific to the EVA system, is that continuous visual surveillance of EVA crewmen must be provided.

1.7.1 Extravehicular Mobility Unit (EMU)

- Support a minimum of two 2-man EVA's per day, each of 6 hr duration.
- Provide one EMU for each SCB crewman who will do routine EVA.
- Support average metabolic rates while EVA of 900 Btu/hr.
- Provide radiation protection for crewmen performing EVA under planned conditions of use.
- Recharging/drying period of not more than 14 hours between suit uses.
- Provide capability for in-suit liquid nourishment.
- Provide independent (not umbilical supported) life support system.
- Provide for urine and fecal collection during EVA.
- Provide 30-min emergency oxygen supply.
- Permit duplex voice communication between EVA crewmen and between EVA crewmen and the Orbiter and ground.

1.7.2 Airlock/Docking Module/Tunnel Adapter

- Provide volume in which two men can simultaneously don/doff EMU's.
- Permit reentry of EVA crewmen for 2-hr break between EVA sojourns without the necessity for prebreathing prior to resuming EVA.
- Be repressurizable from both inside the airlock and from the exterior of the airlock.
- Permit a maximum repressurization time (emergency) of approximately 1 min.

1.7.3 Manned Maneuvering Unit (MMU)

No SCB requirements identified.

1.7.4 EVA Restraints/Mobility Aids

- Restraints for loose equipment while donning/doffing.
- Restraints for crewmen while donning/doffing.
- Mobility aids for EVA translation.

1.7.5 EVA Lights

- Provide illumination for EVA translation routes.
- Permit EVA (including airlock ingress/egress) during both light and dark periods.

1.7.6 Airlock Support Subsystem

- Support EVA periods by individual crewmen separated by a maximum of 14 hr.
- Provide storage for a minimum of four complete EMU's (including pressure garment and PLSS/SOP).
- Provide recharge capability (including battery recharge) for four EMU's simultaneously.
- Accommodate prebreathing.
- Provide communication between EVA crewmen in airlock and monitoring personnel.
- Provide interface between Orbiter ECLS and EMU for pre- and post-EVA suit cooling, post-EVA recharging of suit water and oxygen systems, and post-EVA draining of condensate.

Section 2

SHUTTLE HABITABILITY SUBSYSTEM DEFINITION

In this section, the Shuttle habitability subsystems are defined with respect to the subsystem requirements discussed in Section 1. This section forms the basis for evaluation of the impact that the requirements posed by Option L' have on the Orbiter's baseline capabilities.

2.1 FREE VOLUME

The combined free volume of the Orbiter's flight and middeck areas is estimated at 28.32 m^3 (1,000 ft^3). With a maximum baseline crew of 7, this would be equivalent to approximately 4.1 m^3 (143 ft^3) of free volume per crewman. Based on experimental free volume – duration tolerance data, this represents an acceptable value when mission durations of up to 30 days are considered.

2.2 FOOD MANAGEMENT

The baseline Orbiter food management design is based on the nominal metabolic criteria presented in Table 1. The food management requirements are satisfied by provisions for stowage, preservation, and preparation of foods.

The food management subsystem consists of a galley area which provides a food preparation center, including food and equipment storage, hot and cold water dispensers, food trays, holding oven, water heater, and waste storage.

The nominal storage volume for ambient food storage is 0.22 m^3 (7.6 ft^3). An additional 0.08 m^3 (2.9 ft^3) storage volume is provided for a 4-day contingency capacity. Thus, the total stowage for foods is 0.3 m^3 (10.5 ft^3).

The hot and cold water dispensers are capable of delivering water at a rate of 27.2 kg/hr (60 lb/hr) at temperatures of 65° ± 3°C (149° ± 6°F), and 9° ± 3°C (48° ± 5°F), respectively.

Table 1.

CREWMAN METABOLIC BALANCE, NOMINAL kg/man-day (lb/man-day)[1]

Input			Output		
Solids		0.59 (1.30)	Solids		0.10 (0.22)
Food	0.59 (1.30)		Urine	0.06 (0.13)	
			Feces	0.03 (0.07)	
			Sweat solids[2]	0.009 (0.02)	
Liquids (water		2.84 (6.27)	Liquids (water)		3.18 (7.00)
Drink[3]	1.70 (3.74)		Urine	1.50 (3.31)	
Food preparation[3]	0.89 (1.96)		Latent	1.58 (3.49)	
Hot	0.44 (0.98)		Sweat	0.70 (1.54)	
Cold	0.44 (0.98)		Insensible[4]	0.88 (1.95)	
Wet food	0.26 (0.57)		Feces	0.09 (0.20)	
Gases		0.80 (1.76	Gases		0.96 (2.11)
Oxygen	0.80 (1.76)		Carbon dioxide	0.96 (2.11)	
Total		4.23 (9.33)	Total		4.23 (9.33)

[1]Metabolic rate = 10,733 Btu/man-day (2705 kcal/man-day); respiration quotient = 0.87; cabin temperature = 70° F.

[2]One percent of sweat and skin diffusion.

[3]From potable water supply.

[4]Composed of lung latent loss (10 percent of total metabolic rate) plus skin diffusion (40 Btu/hr).

Fuel cell byproduct water is the only source of potable water except for a small quantity loaded onboard prior to launch. If the water usage requirements exceed the fuel cell production rates, the shortage will be accommodated by onboard storage. Should the fuel cell production rates exceed the water usage requirements, the excess can be accommodated by onboard storage or by overboard dumping through flash evaporators on the water dump nozzle. There are three potable water tanks with a volume capacity (maximum loaded capability) of 73.3 kg (168.3 lb) each.

A holding oven with a capacity of holding seven food trays at a temperature of 65° ± 3°C (149° ± 6°F) is provided. There are no provisions for a refrigerator or freezer.

Food trash containment is based on the following nominal quantities: (a) food waste, 0.18 kg (0.40 lb)/man-day, (b) food packaging, 0.53 (1.18 lb)/man-day, and (c) utensils and other, 0.0005 kg (0.01 lb)/man-day.

2.3 PERSONAL HYGIENE

The personal hygiene facility, located in the middeck area, includes one waste management compartment and one personal hygiene station. The waste management facility consists of one metabolic waste collector (commode/urinal) that weighs approximately 45.36 kg (100 lb), has a storage volume of 0.068 m^3 (2.4 ft^3) and a storage capacity designed for 210 man-days.

The personal hygiene facility is integral with the galley module and consists of a mirror, hand washer, and drain, and has a hot and cold water dispenser. Each crewman has a personal hygiene kit weighing 1.59 kg (3.5 lb). Hygiene storage containers are provided within the personal hygiene facility. No shower facility is provided.

Waste water from the urinal, personal hygiene station, and the airlock is normally stored in the waste water storage tanks. In an emergency, the waste water dump nozzle can be used to dump waste water directly overboard. There are a total of two waste water storage tanks with a volume capacity per tank of 76.44 kg (168.3 lb). Metabolic solid wastes are stored in the waste collector.

2.4 SLEEPING ACCOMMODATIONS

One vertical and three horizontal bunks are provided in the middeck section of the Orbiter. Sleep restraints and curtains are provided weighing approximately 1.63 kg (3.6 lb) per bunk. Sleeping bags for bunks weigh 0.85 kg (1.9 lb) per bag. To provide sound and light isolation during periods of sleep, ear muffs and light masks are provided with a total weight of 0.091 kg (0.2 lb) per-man.

2.5 RECREATION, EXERCISE, AND CREW CARE

Because of the relatively short baseline mission duration, the principal passion recreation will be provided by black and white/color television. Minimal biomedical/exercise equipment will be provided to permit the crew to engage in limited active recreation/exercise program. It is estimated that the television equipment weight 9.03 kg (19.9 lb) and the exercise equipment will weigh approximately 21.27 kg (49.9 lb). A 1.36 kg (3 lb) medical kit will be available for use in case of minor injuries and illnesses.

2.6 STOWAGE

Modular stowage containers with dimensions of 27.31 x 40.64 x 50.80 cm (10-3/4 x 16 x 20 in) will be provided.

2.7 SHUTTLE EVA SYSTEM

The Shuttle EVA system consists of the following elements:

- EMU (Extravehicular Mobility Unit)
- Airlock/Docking Module/Tunnel Adapter
- MMU (Manned Maneuvering Unit)
- EVA Restraints/Mobility Aids
- EVA Lights
- Airlock Support Subsystem

The Shuttle EVA system provides the following capabilities:

- Six hours continuous EVA at 1,000 Btu/hr for any one hour and 2,000 Btu/hr for periods not exceeding 15 minutes) plus 30 minutes for egress/ingress and 30 minutes contingency reserve.
- Two EMU's are provided.
- The airlock can accommodate two men donning or doffing simultaneously and unassisted.

- Airlock provides space for stowage of two EMU's.
- The EMU provides in-suit urine collection and in-suit beverage dispensing.
- Lighting in the airlock and payload bay.
- Restraints and mobility aids in airlock and payload bay.

The Shuttle EVA system imposes the following constraints on EVA operations:

- Requires 3.5 hours crew time pre-EVA (includes prebreathing).
- Requires 1.5 hours crew time post-EVA.
- Requires 12-16 hours for battery recharging.
- Requires 48 hours nominal (60 hours maximum) for suit drying.

The Shuttle Extravehicular Mobility Unit (EMU) is a non-customized pressure garment with attached Primary Life Support System and Secondary Oxygen Pack. It consists of a liquid cooled ventilation garment (LCG), hard upper torso, lower torso (with boots), gloves, and helmet (with extravehicular visor assembly). It has the equivalent of 0.3 grams/cm^2 of radiation protection. During pre and post-EVA the suit is connected to a recharging station by a 7 foot long Service and Cooling Umbilical (SCU) which provides electrical power, makeup oxygen, heat rejection, and hardline voice communication.

The Shuttle airlock is a modular structure 160 cm (60 in.) in diameter and 210.8 cm (83 in.) long, having a total volume of 150 cu. ft. (effective volume 130 ft^3). It provides two donning/doffing stations and space for stowage of two EMU's. Normal repressurization of the airlock, with oxygen from the Orbiter ECLS, is at 0.1 psi/sec rate and takes approximately 190 seconds. In an emergency the airlock can be repressurized at 1.0 psi/sec. in approximately 17.8 seconds. Depressurization of the airlock is performed by dumping the approximately 11 lbs of air to vacuum and takes approximately 5 minutes at a rate of 0.1 psi/sec.

The Shuttle Manned Maneuvering Unit (MMU) is a modular backpack device, readily attached to the EMU, for translation beyond the envelope of the payload bay (normal range 100 meters). It is stored and serviced at the Flight Support Station mounted in the payload bay. Total weight, including the Flight Support Station, is 127 kg (280 lb).

The airlock support system provides an Oxygen supply, power, instrumentation, suit cooling potable water supply, audio communciations, EMU and POS recharge station, and valves, gages, and switches for control functions. The airlock support system interfaces with the EMU via the Service and Cooling Umbilical.

Section 3

IMPACT OF SCB OPTION L' HABITABILITY REQUIREMENTS ON SHUTTLE SUBSYSTEM

In this section, the impact of SCB Option L' habitability requirements on Shuttle subsystem is discussed. Specifically, the increased requirements posed by larger crew size and longer mission duration are compared with current Orbiter habitability subsystem capabilities.

3.1 FREE VOLUME

To provide an acceptable crew confinement tolerance level for mission durations of up to 180 days, the minimum free volume per crewman should be 4.96 - 5.66 m^3 (175 - 200 ft^3). Therefore, for a crew of 10, the total free volume required is 49.6 - 56.6 m^3 (1750 - 2000 ft^3). As the Orbiter flight and middeck sections are capable of providing only approximately 28.32 m^3 (1000 ft^3) of the requirement, increased free volume must be provided by approximately doubling the current available Orbiter free volume. Consequently, an additional habitation module in the Orbiter payload bag is indicated to provide flexibility in locating galley, sleeping, hygiene, and dining/recreation facilities in a manner that the required minimum free volume can be accommodated.

3.2 FOOD MANAGEMENT

Because EVA has a major role in the early space construction base missions, significantly greater metabolic requirements are anticipated as it is estimated that to maintain crew metabolic balance, a nominal metabolic rate of 3,326 kcal/man-day (12,800 Btu/man-day) must be satisfied. This represents a 521 kcal/man-day (2,067 Btu/man-day) increase over the nominal Orbiter baseline metabolic rate allowance of 2,705 kcal/man-day (10,733 Btu/man-day). The significance of the increased metabolic requirement is illustrated in Table 2, where the nominal baseline metabolic balance data is compared with SCB Option L' requirements.

The principal impact for food management will be the need for increased food storage capacity and an increased demand for potable water as food intake (Table 2) is significantly increased. For example, the food intake is approximately doubled as it increases from 4.13 kg (9.11 lb) per day for the baseline crew of seven to 8.1 kg (17.86 lb) per day for the Option L' 10-man crew. Similarly, potable water intake is increased from 19.88 kg (43.84 lb) per crew-day to 51.5 kg (113.58 lb) per crew-day.

Also, provisions for a freezer, refrigerator, convection oven, and micro-wave oven should be made to permit utilization of freeze-dried, frozen, thermo-stabilized, and fresh foods.

The resistance oven should be capable of heating 2.27 kg (5 lb) of frozen food from -17.75°C (0°F) to 71.11°C (160°F) in 30 min. The freezer and refrigerator should have the following performance capabilities:

<u>Freezer</u>

Storage temperature: -23.3°C (-10°F) to -15.0°C (5°F)
Storage capacity: 353.8 kg (780 lb) - 1.06 m^3 (37.5 ft^3) total volume

<u>Refrigerator</u>

Storage temperature: 4.44 ± 2.77°C (40 ± 5°F)
Storage capacity: 0.42 m^3 (15 ft^3)

If the requirement for frozen and fresh foods were waived, refrigerator and freezer requirements could be eliminated and the total supplied food would be of the freeze-dried and thermo-stabilized type.

3.3 PERSONAL HYGIENE

To support a 10-man crew for up to 180 days, the baseline personal hygiene facility must be supplemented by one shower facility to provide a nominal capability of one shower per crewman per week. Approximately 1.70 m^3 (60 ft^3) of space should be allotted for such a facility.

The current Orbiter metabolic waste collector (commode/urinal) has a storage capacity designed for 210 man-days. For a Shuttle-tended space construction base, this would be marginally sufficient for 21 days. Increased storage capacity will be required to support a crew of 10 for 30-180 days (300-1,800 man-days) taking into account the waste generation rates as described in Table 2.

The Orbiter baseline waste water holding capacity is 152.88 kg (336.6 lb). Since waste water from the urinal, personal hygiene station, and the airlock is normally stored in the waste water storage tanks, the capacity of the latter must be greatly increased. To illustrate the impact on waste water collection system, the expected urine output alone is approximately 24 kg (52.9 lb) per crew-day.

3.4 SLEEPING ACCOMMODATIONS

To accommodate the Shuttle-tended SCB crew, a total of five sleep restraints/ bunks will be required. Sleeping in two shifts ("hot bunking") will accommodate 10 men. Each sleeping accommodation should have the following dimensions: depth, 0.51m (20 in); width, 0.91m (36 in); and length, 1.98m (78 in). Consequently, five sleeping accommodations will occupy a total volume of approximately 4.55 m^3 (162.9 ft^3). Since the Orbiter baseline configuration has only accommodations for four bunks, one additional bunk must be provided. As an alternate to locating sleeping accommodations away from noise-producing areas, curtains, earmuffs, and light masks will be provided. Thus, each sleeping accommodation will be equipped with the following items: (a) sleep restraints and curtain, 1.63 kg (3.6 lb) per bunk; (b) earmuffs and light masks, 0.09 kg (0.2 lb) per man; and (c) sleeping bag, 0.86 kg (1.9 lb) per bag.

3.5 RECREATION, EXERCISE, AND CREW CARE

To support a 10-man crew for periods up to 180 days, passive and active recreation facilities must be provided. As a minimum, the passive and active recreation equipment and supplies should include a color television set, reading material, tape deck and library, table games, and puzzles.

In addition, to maintain physical and physiological conditioning, action recreation (exercise) equipment must be provided and should include a bicycle ergometer and bungee-type devices with support bars. These are expected to weigh 22.68 kg (50 lb).

Table 2

COMPARATIVE CREWMAN METABOLIC BALANCE DATA

Baseline Metabolic Balance, Nominal kg/man-day (lb/man-day)			Option L' Nominal Metabolic Balance, kg/man-day (lb/man-day)		
Input			Input		
Solids		0.59 (1.30)	Solids		0.81 (1.79)
Food	0.59 (1.30)		Food	0.81 (1.79)	
Liquids (water)		2.84 (6.27)	Liquids (water)		5.15 (11.35)
Drink	1.70 (3.74)		Drink	3.57 (7.86)	
Food preparation	0.89 (1.96)		Food preparation	1.22 (2.70)	
Wet food	0.26 (0.57)		Wet food	0.36 (0.79)	
Gases		0.80 (1.76)	Gases		0.94 (2.08)
Oxygen	0.80 (1.76)		Oxygen	0.94 (2.08)	
Total		4.23 (9.33)	Total		6.90 (15.22)
Output			Output		
Solids		0.10 (0.22)	Solids		0.19 (0.41)
Urine	0.06 (0.13)		Urine	0.10 (0.22)	
Feces	0.03 (0.07)		Feces	0.07 (0.15)	
Sweat solids	0.009 (0.02)		Sweat solids	0.02 (0.04)	
Liquids (water)		3.18 (7.00)	Liquids (water)		5.55 (12.23)
Urine	1.50 (3.31)		Urine	2.40 (5.29)	
Latent	1.58 (3.49)		Latent	2.95 (6.50)	
Feces	0.09 (0.20)		Feces	0.20 (0.44)	
Gases		0.96 (2.11)	Gases		1.17 (2.58)
Carbon Dioxide	0.96 (2.11)		Carbon dioxide	1.17 (2.58)	
Total		4.23 (9.33)	Total		6.90 (15.22)

An area should be able to be converted for use as a passive recreation facility with seating and tables being provided to accommodate maximum number of crew.

There are essentially no provisions on the Shuttle for a recreation area and the passive and active recreation equipment, with the exception of a television set, is lacking. Consequently, the weight and volume requirements posed by this need will impact on the free volume, and stowage facility requirements, as provisions for equipment stowage and recreation area and furnishings will have to be taken into account.

3.6 STOWAGE

Principal impact regarding stowage requirements as imposed by Option L' is in the following areas:

Food storage — Increased crew size and metabolic rate over the Orbiter baseline provisions demand greater stowage provisions to be set aside for the purpose of food stowage.

Waste management — Generation of waste is increased requiring greater facilities for handling and storing metabolic wastes and trash.

Crew seats (flight seats) — Three additional seats have to be accommodated and stowed when in orbit.

Exercise equipment — Requirement for providing stowage of exercise equipment has been added as no allowances for such equipment has been made in the Orbiter baseline configuration.

To illustrate the impact of stowage facilities posed by the Option L' requirements over that of baseline, the following data are provided (based on 30-day on-station stay time).

Item	Baseline	Option L'
Dry food	123.8 kg (273 lb)	243.6 kg (537 lb)
LiOH cartridges		
No. of cartridges required	105	150
Total weight of cartridges	304.8 kg (672 lb)	435.5 kg (960 lb)
Liquid Waste Tankage		
Liquid waste to be stored	857.3 kg (1,890 lb)	1,224.7 kg (2,700 lb)
No. of storage tanks required	13	18
Weight of storage tanks, empty	203.4 kg (448.5 lb)	281.7 kg (621 lb)

3.7 EVA

The Shuttle EVA System described in Section 2.7 cannot meet the EVA requirements of Section 1.7 without considerable impact in terms of weight and volume on the Orbiter, primarily because more units are needed and the extensive EVA contemplated imposes severe penalties in terms of consumables. The individual hardware elements of the Shuttle EVA System are, however, usable for SCB applications with only minor design changes.

The major weight and volume impact is in consumables. The baseline EVA system provides consumables for only two payload dedicated 6-hour EVA's per mission. Table 3 summarizes the consumables requirements for SCB, assuming two-shift operations in which two 2-man 6-hour EVA's (divided into two 3-hr periods separated by a 2-hr rest period) are performed 6 days in every week.

Another impact is in weight and storage volume for EMU's. The baseline Orbiter system provides two EMU's and specifies these are to be used by the pilot and mission specialist. In two-shift SCB construction activities, four SCB crewmen will be routinely doing EVA, requiring that the Orbiter accommodate at least two additional EMU's, each weighing approximately 91 kg (200 lb) and occupying approximately 15 ft^3 of storage volume.

To accommodate the requirement for a 2-hr break between 3-hr EVA time segments (without additional prebreathing), it will be necessary to provide food and personal hygiene accommodations in the airlock prior to the initiation of each 6-hr EVA. It is assumed that these can be portable facilities and supplies requiring no permanent installation, connections, or design changes in the airlock.

The baseline EMU recharge station can accommodate two EMU's simultaneously. To meet this requirement, either the recharge station must be redesigned to accommodate four EMU's or a second recharge station must be supplied. The latter alternative would permit location of the second recharge station outside the airlock which would tend to alleviate the airlock volume problem.

The approximately 0.3 gm/cm^2 equivalent radiation protection afforded by the Shuttle EVA system EMU is probably not sufficient to protect crewman if EVA is performed routinely without regard to passage through the South Atlantic anomaly. It will, therefore, be necessary to schedule EVA construction activities to minimize EVA time in this area of higher radiation.

Table 3.

EVA CONSUMABLES/EXPENDABLES

	Amount/Man/EVA	Amount/EVA	Amount/Day#
Oxygen — kg(lb)			
Airlock depressurization (initial)	0.38 (0.83)	1.25 (2.75)	2.5 (5.5)
Airlock depressurization (after 2-hr break)		1.36 (3.0)	2.75 (6.0)
EMU purge	0.38 (0.83)	0.76 (1.68)	1.52 (3.36)
EMU recharge	0.54 (1.183)	1.08 (2.38)	2.16 (4.76)
Prebreathing	1.5 (3.3)*	3.0 (6.6)	6.0 (13.2)
Nitrogen — kg(lb)			
Airlock depressurization		3.74 (8.25)	7.48 (16.5)
Potable Water — kg(lb)			
EMU feedwater loop	4.76 (10.5)	9.52 (11.0)	19.04 (22.0)
Insuit drink bag	0.91 (2.0)	1.82 (4.0)	3.64 (8.0)
Electrical Power (amp-hr)			
Battery recharge	30.0	60.0	120.0
EMU support (pre/post EVA)	8.6	17.2	34.4
Suit drying	TBD	TBD	TBD
EMU Batteries — kg9lb)	TBD**	TBD**	TBD**
Contaminant Control Cartridge (CCC)			
kg (lb)	2.09 (4.6)	4.18 (9.2)	8.36 (18.4)
Dessicant Cartridges	TBD***	TBD***	TBD***

\# Assumes two 2-man EVA's per day

* Only a fraction expended

** Each battery weighs 9.8 lb. Battery life in terms of number of recharges is TBD (must be recharged after each EVA)

*** For suit drying — depends on weight of cartridges, number of cartridges used in ventilation loop, and whether cartridges are expendable or rechargeable in flight (if rechargeable additional electrical power will be required).

Section 4
IMPACT OF GROWTH TO 14- AND 21-MAN CREWS

While the total crew size of 10 is adequate for space construction base operations, other objectives may require an increase in crew size of up to 14 or 21 crewmen. In Shuttle-tended operations, increased demands are placed on the Shuttle's habitability subsystem capabilities, assuming that the Shuttle is the only means of habitability support.

The increase in crew size impacts virtually all aspects of the Shuttle habitability subsystem. The most severe of these would be the provision for adequate free volume, logistic support, and waste management. For example, the free volume requirements would increase from the 10-man requirement of 49.6 - 56.6 m^3 (1,750 - 2,000 ft^3) to 69.4 - 79.3 m^3 (2,450 - 2,800 ft^3) for a 14-man crew and 104.1 - 118.9 m^3 (3,675 - 4,200 ft^3) for a 21-man crew. In the latter case, the additional habitation module in the Orbiter payload bay that would be required would occupy a significant portion of the total volume available in the payload bag.

To illustrate the magnitude of the impact of the increased crew size on the Orbiter's food and waste management, some food intake and waste production quantities (based on a nominal metabolic balance of 2,705 kcal (10,733 Btu) per man per day) are presented below:

Function	14-man crew/day	21-man crew/day
Solid food intake	8.26 kg (18.20 lb)	12.38 kg (27.30 lb)
Water intake (drinking)	23.75 kg (52.36 lb)	35.62 kg (78.54 lb)
Urine output	21.02 kg (46.34 lb)	31.53 kg (69.51 lb)

To provide adequate personal hygiene and waste management facilities for a crew of 14, a total of two commode/urinal modules and two personal hygiene modules would be preferred. For a crew of 21, it would be a definite requirement. An additional shower may not be required if one shower/man/week (14-21 man-showers per week) is considered.

While a crew of 14 would be served by our galley module, two galley modules would be required to accommodate a crew of 21.

The number of bunks to be provided will increase — the number of required bunks will depend on the work-rest cycles used during a given mission, so that "hot bunking" could be utilized to the maximum.

Part 15

ORBITER UTILIZATION IN MANNED SORTIE MISSIONS

ORBITER UTILIZATION IN MANNED SORTIE MISSIONS

INTRODUCTION

This appendix examines the use of sortie mission 5 to achieve space construction base mission objectives.

The complete possible utilization mode spectrum of the STS/Orbiter in manned space is both broad and diverse. Figure 1 illustrates this point by defining primary characteristics of the three major manned mission modes: Sortie, Shuttle-Tended, and Permanently Manned Base. Manned sortie missions are defined as those in which the payload is either returned to earth (current Spacelab concept) or turned over to another Orbiter for continued operation (hand-off). In general, mission hardware is totally dependent on the Orbiter for basic services (habitability, power, data, communications, etc.) in this mode.

Shuttle-tended systems are capable of being stored in orbit for extended periods of time. Hence, operations may be intermittent with the mission hardware functioning as an unmanned spacecraft. Simple Shuttle-tended missions would depend heavily upon Orbiter services, and EVA for activity external to this vehicle. Hence they would be little different than a sortie mission but the hardware need not be returned to be stored for reuse. More sophisticated Shuttle-tended systems may utilize orbitally stored modules with a partial ECLS system to enlarge available "shirtsleeve" volume and be less dependent on Orbiter-supplied services. In fact, it is probably desirable for the Orbiter, which would still supply basic data, habitability, and communications services, to become dependent upon an orbitally stored electrical power supply in these cases.

The distinction between sophisticated orbitally stored systems and permanent manned bases, again, may not be sharp. In essence, a permanent base is

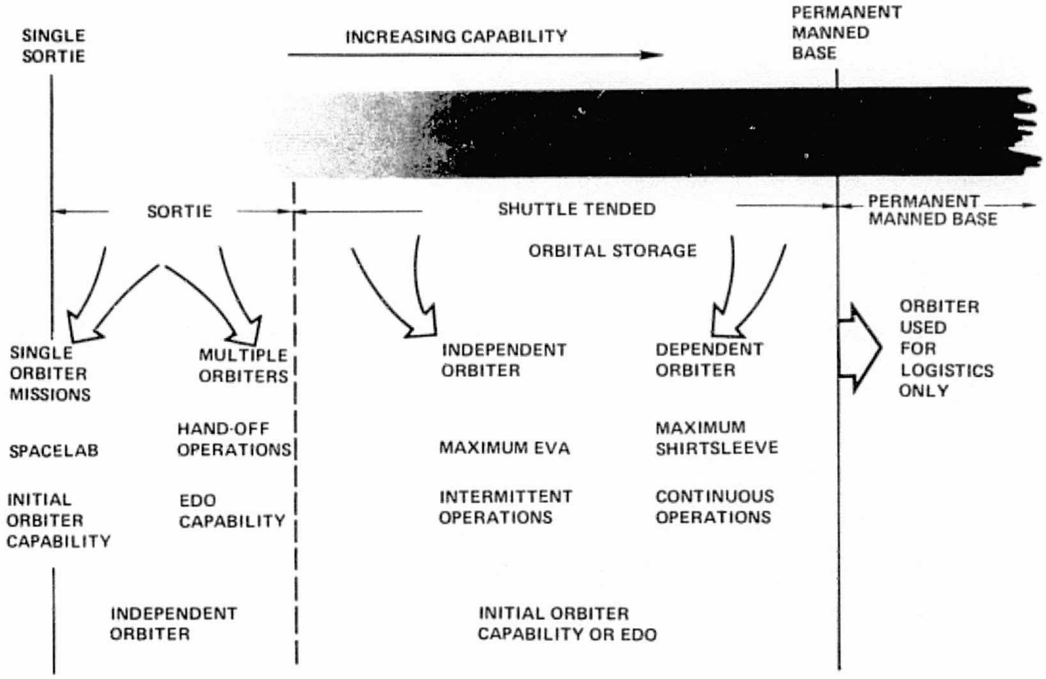

Figure 1. Operational Mode Spectrum

defined as a facility which depends upon the Orbiter to the minimum extent – i. e., logistics resupply only.

In general, this listing is in ascending order of mission capability. It is clear that programs planning a productive capability requiring hundreds of orbital man-months per year are best supported by some form of a permanent orbital base. On the other hand, it is equally obvious that if the total program requires only tens of orbital man days annually, a single sortie (or series of shorter sorties) would be the appropriate mode.

However, it is important to recognize that this association of mission mode with capability is primarily for reasons of economy and not generally demanded by technical mission requirements. An absurd example serves to illustrate the point: if a biological specimen must be exposed to a weightless environment for a period of years, the requirement could be met by a continuous series of 7-day Orbiter sorties with the experiment package handed off from vehicle to vehicle on orbit. Similarly, a requirement to continuously support dozens of construction workers in LEO could be met by docking multiple "Orbiter hotels" to a Shuttle-tended work platform. While these are reducto absurdum arguments, the point is clear: mission magnitude, as measured by duration or man-hour requirements, cannot be used as the sole basis for choice of mission mode.

To support this point, LEO missions detailed in the Space Stations Systems Analysis Study have been examined, and it has been determined that it would be both feasible and technically practical to undertake all in a sortie mission mode. As discussed later, this does, however, require elaborate operational procedures (multiple synchronized Orbiter missions). It was therefore concluded that technical missions requirements alone are not generally sufficient to determine best mission mode. Hence, the approach taken here is to examine the basic factors controlling program cost, concentrating on transportation requirements. Table 1 summarizes the order of discussion and outlines conclusions.

Table 1

DISCUSSION ROADMAP AND SUMMARY

Subject	Conclusion
Single Orbiter, initial capability	Mission limited by power system
Single Orbiter, EDO Spacelab	Capability limited by volume and load carrying capacity
Multiple Orbiter sortie EDO capability	Capable, but not cost effective compared to Shuttle-tended mode

SINGLE SORTIES WITH INITIAL ORBITER CAPABILITY

In Part I of the SSSAS, the productivity of Spacelab sorties in scientific missions was compared to that of a minimum 4-man Space Station. Since the productivity of manned space research missions is a function of both the allowable quantity of equipment (payload) and available man-hours, neither factor, by itself, is an adequate description of mission productivity. Hence, the definition of mission productivity potential is taken as the product of available man-hours and the available payload weight per Orbiter flight. While an optimum ratio of man-hours to payload weight will clearly vary from mission to mission, this chosen parameter is a meaningful statistic, since in the context of most manned space activity, the cases of man-hours without payload, or payload without man-hours, produce a parameter value of zero.

Figure 2 summarizes the results of this past effort. Orbiter flight duration was considered to be limited only by expendables, but subsystem capability remained that which will be initially flown. Hence, a major reason for the illustrated, relatively poor productivity of the sorties mission is fuel consumption of the electrical power system. This penalty is approximately 2 lb per kilowatt hour (1 lb of fuel plus 1 lb of container). Figure 3 indicates the effect: if landing payload is limited to the current 14,456 kg, the Orbiter-Spacelab has a zero payload on a 32-day mission if 8 kW is available for payload use. This is simply because some 6810 kg of empty fuel cell

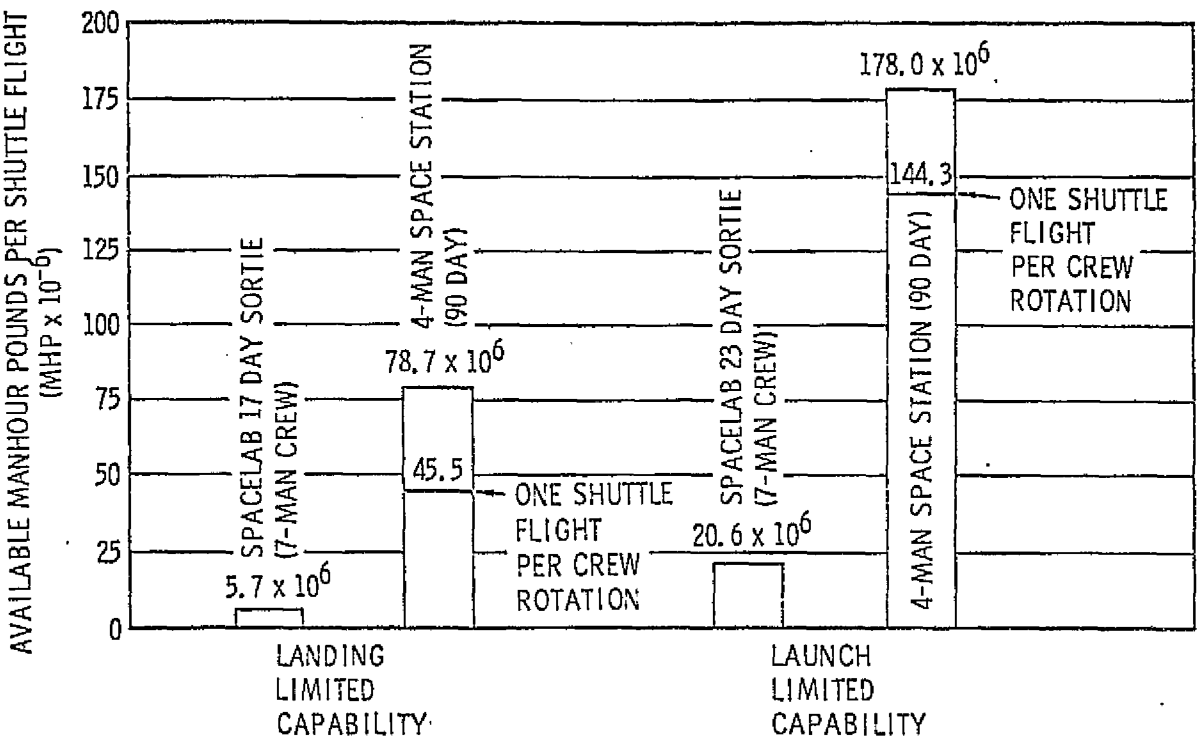

Figure 2. Spacelab Sortie/Space Station Comparative Capability Corrected for Learning (85%)

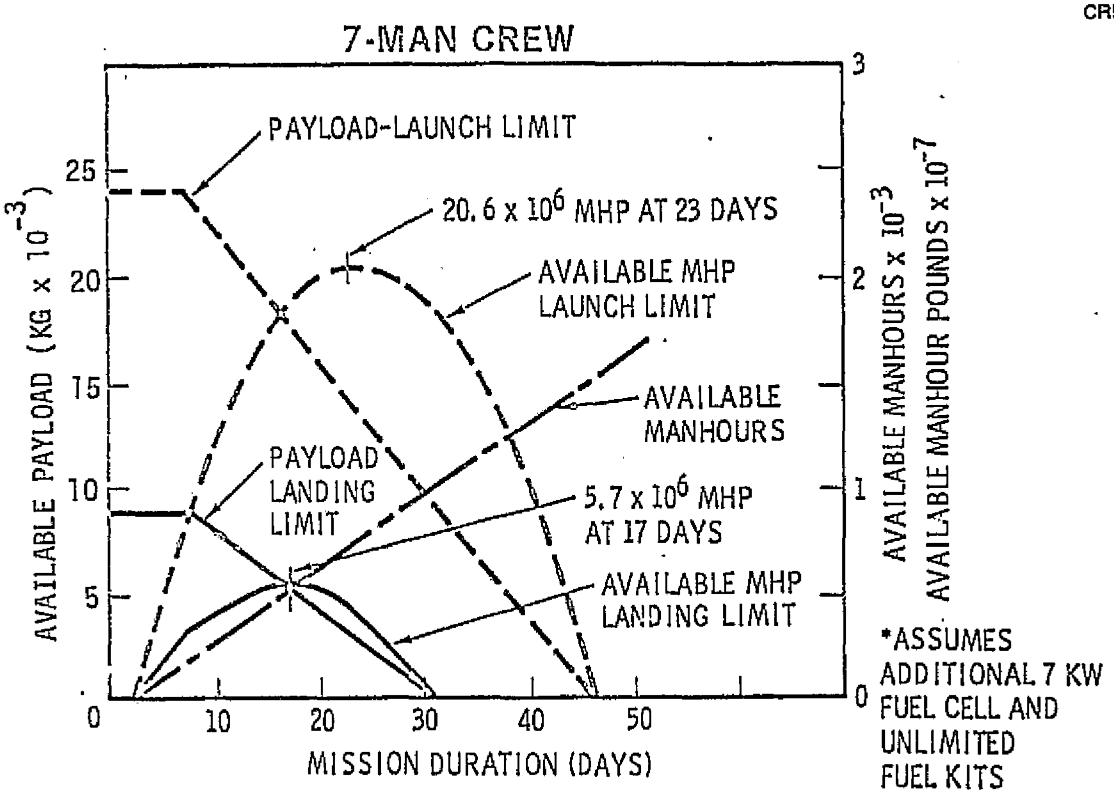

Figure 3. Orbiter + Spacelab Sortie Mission 19.2* kW (8 kW to Experiments)

reactant kits must also be returned with the Spacelab. Discarding these reusable kits on orbit would be wasteful since replacement costs would at least equal current estimated Orbiter operating costs.

EXTENDED DURATION ORBITER-SPACELAB

This limitation on the Orbiter-Spacelab combination to efficiently provide the large energy requirements of future manned missions has been recognized in the past. Various possible means of increasing its utility have been studied by NASA and the Rockwell Corp. The latter's recent EDO (Extended Duration Orbiter) concept is of most interest here. One of the configurations studies by Rockwell was simply the addition of a deployable solar cell auxiliary power system and modifications to the ECLS system to reduce consumables. Using the standard Spacelab configuration, this would seem to be a technically feasible way of extending the Orbiter's duration to some 90 days, while providing adequate power to the payload. However, volume limitations on living space (some 28 m^3 of free volume) would seem to limit crew size to about four on such long-duration missions, and available payload volume is further restricted by inclusion of the folded solar cell array (Table 2).

In reviewing the space construction base mission, it was found that: (1 none of the defined construction equipment and few of the required prefabricated

Table 2
EDO-SPACELAB CAPABILITY

4 Men 90-Day Duration	300 Man-Shifts
4540 kg Discretionary payload (MDAC)	
5450 kg Discretionary payload (RI)	
102 m^3 cargo volume (285 m^3 available at standard Orbiter)	

components will fit in the available volume; (2) assembly only of the 30m radiometer is estimated to require less than 300 man-shifts (2,400 man-hours). However, the multidisciplinary laboratory and sensor development objectives represent requirements that can be largely accomplished "a bit at a time." Of course this result is also generally applicable to the existing Orbiter-Spacelab concept (non-EDO).

Thus, it has been concluded that single Orbiter-Spacelab sorties, without use of multiple Orbiters or orbital storage of equipment (defined as Huttle-supported mode), can accomplish none of the SSSAS-defined construction or space processing missions. Further, the current concept of Orbiter-Spacelab is, in comparison to even a small 4-man station, quite inefficient in the less demanding laboratory missions it can undertake.

MULTIPLE-ORBITER SORTIE MISSION MODE

It is obvious from the previous discussion that an effective way of increasing the EDO-Spacelab mission would be use of multiple Orbiters. In this mode, one or more payload-carrying Orbiters would rendezvous with the EDO-Spacelab. If a mission allowed the technique, the payload carrier could be a standard (short-duration) vehicle that returns home immediately after docking its payload to the EDO.

In principle, this mode would not be restricted to small assemblies. If a large cluster is involved, use of distributed attitude control thrusters would be desirable, if not necessary. If large crews are required, it would of course be possible to utilize multiple EDO-Spacelabs. However, in this case it would probably be wise to outfit Spacelab modules for additional habitability volume and bring up any additionally required work space volume as separate payloads. This defines the extended-duration Orbiter hotel concept. The EDO hotel provides only "hotel" functions (crew housing) in addition to its normal function of providing utilities (electric power, communications, data services, etc.).

Specific mission hardware elements utilized by SSSAS objectives are listed in Table 3. In the sortie modes, each of these elements would be launched

Table 3

ASSUMED MISSION AND CONSTRUCTION SUPPORT HARDWARE

Objective	Hardware*
SPS	
TA-1	Plastic-tube fabrication facility TA-2 antenna fixture Fabrication and assembly module
TA-2	Plastic-tube facility Antenna fixture Solar collector fixture Fabrication assembly module
Biologicals development	Module (dedicated)
Fiber optics development	Module (dedicated)
Silicon crystal development	Module (dedicated)
30m radiometer	Fabrication and assembly module
Sensor development	Module (dedicated)
Living and working in space	N. A.
Multidiscipline Laboratory 6	Module (dedicated)

*As defined in MDAC's Program Option L

with a standard cargo Orbiter and rendezvous with an EDO hotel. As previously mentioned, this requires a complex operational procedure in some cases. Table 4 outlines the sequence of missions required by the solar power satellite Test Article-2. With this sequence, and currently estimated STS cycle times, the mission is possible with an inventory of two EDO hotels, two cargo Orbiters, and one launcher. The number of crewmen is, of course, inversely proportional to time on orbit for each EDO hotel. With a "rubber" vehicle, these parameters can be optimized for the mission.

EDO as a hotel and power source was applied in this analysis as support to eight objective elements (Table 3) in which the possibility of orbital storage was excluded (definition of sortie mission).

Table 4

MULTIPLE-ORBITER SORTIE MISSION SEQUENCE

SPS: TA-2

Time	Vehicle	Payload	Remarks
-2 weeks	Orbiter No. 1	Plastic tube fabrication	Loiters
-1 week	Orbiter No. 2	Fabrication and assembly module (FAM)	Orbiter No. 1 docks to FAM PTF berthed on FAM Orbiter No. 1 returns
0	EDO hotel	"X" crewmen	Docks to FAM Orbiter No. 2 returns
+1 week	Orbiter No. 1	Antenna fixture (AF)	Docks to FAM AF berthed on FAM Orbiter No. 1 returns
+2 weeks	Orbiter No. 2	Materials pallet (MP)	Docks to FAM MP berthed on FAM Orbiter No. 2 returns
+3 weeks	Orbiter No. 1	Solar collector fixture (SCF) parts	Docks to FAM SCF removed by crane Orbiter No. 1 returns SCF berthed on FAM
+4 weeks	Orbiter No. 2	Solar collector fixture parts	SCF handed off Orbiter No. 2 returns SCF parts installed
"N" weeks	EDO hotel No. 2	"y" crewmen	Facility handed off to EDO No. 2 EDO No. 1 returns

The requirements for each objective element in terms of EDO support and supply Orbiter support have been uniquely determined. An example of the requirements is given in Table 5 for one of the objective elements, TA-1. The required power and task size are associated with the EDO on-orbit requirements, whereas the weight and volume are tied into the number of supply Orbiters required to transport the equipment to orbit. For instance, the fabrication and assembly power level of 6 kW is an input into establishing the solar cell size of the EDO. The task size shown indicates the extent of the construction job in terms of the amount of man-effort required. Analyses to date on TA-1 have centered around an average shift size of 3 men, and implies that a maximum of 9 men could be used per three-shift, 24-hr period. This does not preclude any systems analysis based on more or less

Table 5

EXAMPLE OF REQUIREMENTS FOR TA-1

Power:

- Fabrication and assembly = 6 kW
- Test = 5 kW

Task Size:

- Fabrication and assembly = 267 man-shifts
- Test = 670 man-shifts
- Preferred shift size = 3 men

Associated Weight

- 10,532 kg + 14,456 kg (fabrication and assembly module)

Associated Volume

- Plastic-tube module (nearly one Orbiter)
- Antenna construction fixture (nearly one Orbiter)
- Fabrication + assembly control module (one Orbiter)

effort per shift; however, supporting analysis has not been performed to justify a work efficiency for other sizes equivalent to that of a 3-man shift, although it may be very plausible. An average shift length of 8 hr was assumed — based on a 50/50 split between EVA (6 hr/shift) and non-EVA (10-hr/shift). Three shifts per day of EVA can be accommodated if suit donning and doffing is done external to the airlock.

To allow a complete examination of all possibilities inherent in a multiple-Orbiter sortie mode, MDAC has derived a parametric definition of "EDO hotel" which assumes that its payload capacity, over and above habitability requirements, is dedicated to the solar cell auxiliary power supply. This then defines a family of EDO hotels with electric power, crew size, and mission duration parameters. Table 6 indicates the basic assumptions in this derivation.

Table 6

EDO HOTEL ASSUMPTIONS

Structural volume = 6 m^3/man (free volume) + 100% for subsystem support

Crew support = Crew above 4 men was assumed to have crew quarters in cargo bay, also included shower in pressure shell in bay.

Life support

- Spares redundancy baselined at 90 days and assumed linear for other durations
- Cryogenic gas storage
- Water recovery
- Regnerative CO_2 removal
- Water dump at entry

One OMS kit baseline for RCS, VCS

Cargo bay contains rubberized habitability module and tunnel, docking module, OMS kit, folded advanced solar cells, and (N_i/H_2) batteries

NO EVA, except as required through docking module

25% contingency on all EDO subsystems except docking module, tunnel, and OMS

The resources of the EDO hotel available for an objective element are a function of the size of the crew and the duration in orbit for the EDO. Figure 4 illustrates the resources expressed as solar cell power available for the objective elements. The right-hand scale gives the resources in terms of excess payload, assuming a trade factor of 0.114/w. The "zero power" line provides all the power, expendables, and living quarters to support a given crew size for the indicated duration. The used and excess expendables are assumed dumped overboard prior to EDO reentry in order to keep the landed weight at 14,456 kg of payload.

A cut across Figure 4 at a value of constant power increment available, will look like the upper curve on Figure 5 in which, for a fixed power level, the supportable crew size is given as a function of orbital duration. Also indicated is a curve representing the task size for a given objective element. The shaded area between the Abailable and Required curves represents the resource margin available over and above the objective element. It is limited in time by an orbital duration limit; e.g., the 180-day zero-g limit for men in orbit. The resource margin may be used in higher power margin, discretionary payload, or extra crew size.

Figure 6 represents a spread of the Required curves as a function of task size. It assumes that the crews rest 1 day out of 7. In order to reduce total job time, and therefore reduce the cost of ground operational support, the desired solutions will tend toward the larger crew complements until limited by available power (Figure 5).

Table 7 summarizes resource requirements for each of the objective elements in the study. It should be noted that three of them (1, 2, and 6) are construction projects with a defined task size. Three of them (3, 4, and 5) are extended-duration tasks at a given level of effort. The final three (7, 8, and 9) are level-of-effort tasks with open-ended duration.

The operating regions for the three construction projects are indicated in Figure 7 for the fabrication and assembly effort only (no test). For the larger projects (TA-1 and TA-2), the operation of more than one EDO in series was considered to shorten the time in orbit per EDO. For this type of operation, the expended EDO would mechanically (with Manipulators) hand

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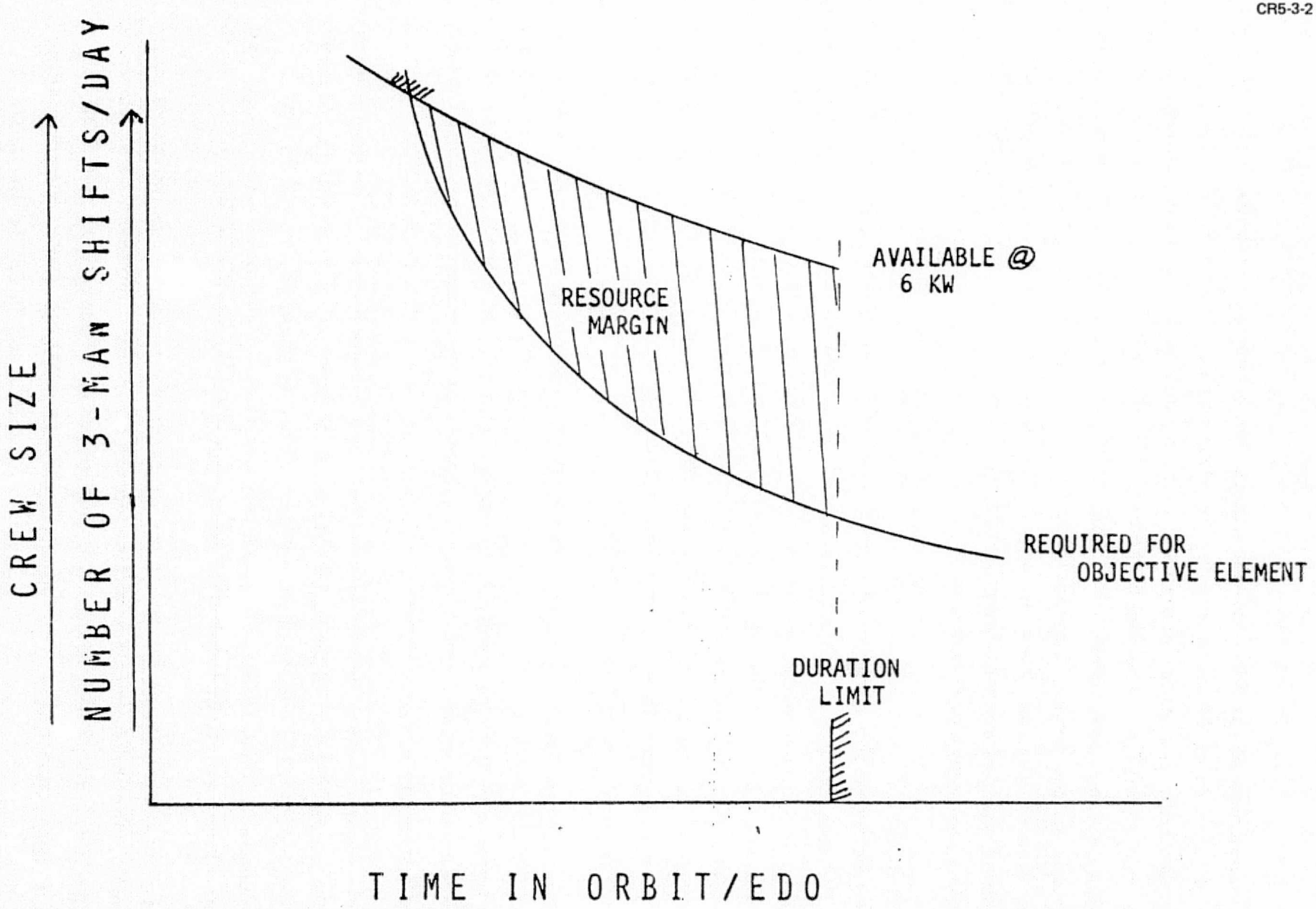

Figure 4. Example of Operating Region for Objective Element

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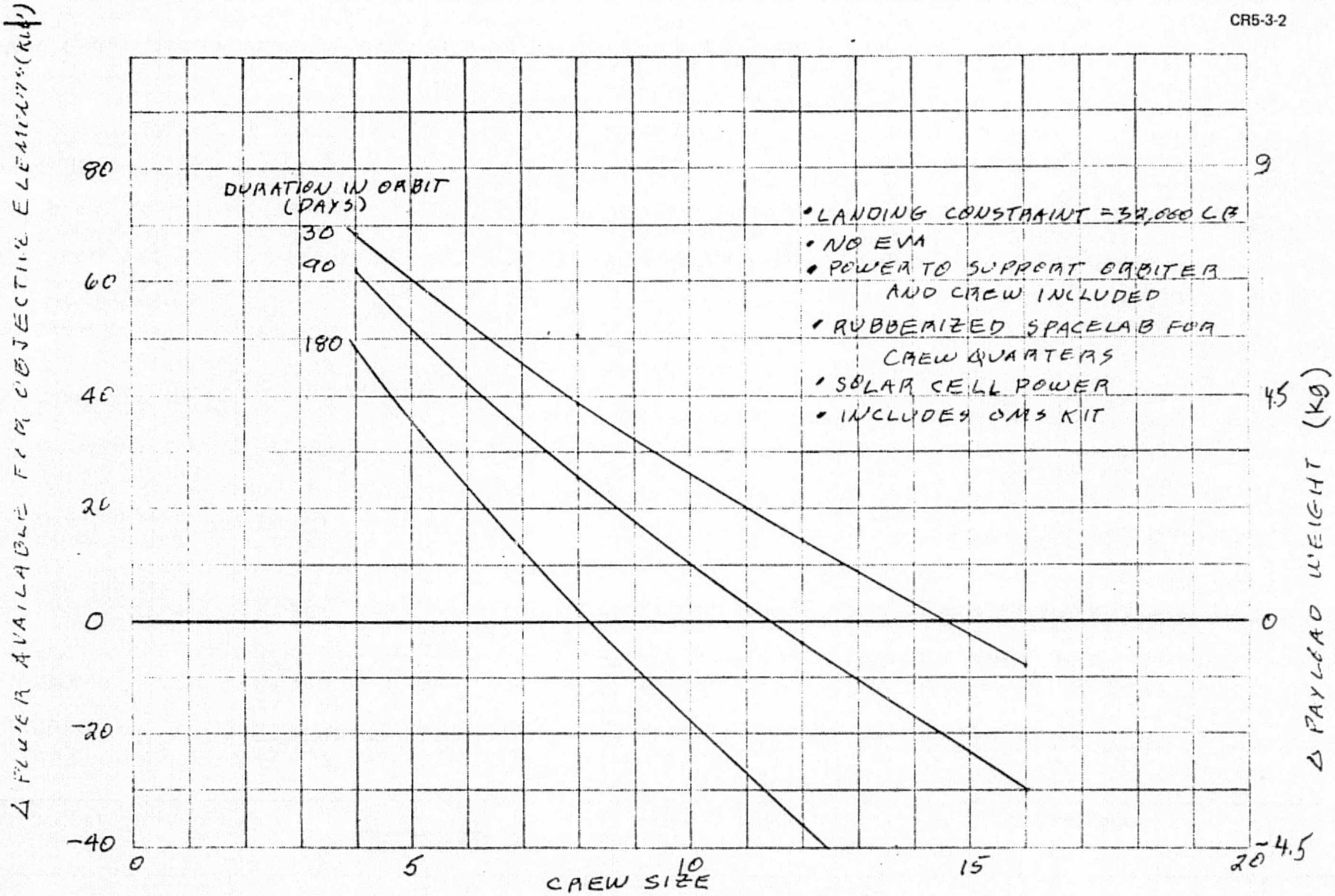

Figure 5. Available Resources of EDO

Table 7

EXTENDED-DURATION ORBITER (EDO) OBJECTIVE ELEMENT REQUIREMENTS

Objective Elements	Mass (kg) Weight (lb)	Power (kW)	Task Size
1. SPS TA-1	10,532 + 14,456* 23,170 + 32,000*	Fab/Assy = 6 Test = 5	267 man-shifts, fab/assy 670 man-shifts, test
2. SPS TA-2	23,694 + 14,456 52,127 + 32,000*	Fab/Assy = 9 Test = 2	504 man-shifts, fab/assy 1460 man-shifts, test
3. Biological Development	11,500 25,353	1.6	1.5 men, 2 shifts/day for 4 years
4. Fiber Optics Development	12,000 26,455	9.7	1.5 men, 2 shifts/day for 4 years
5. Silicon Crystal Development	14,500 31,967	17.0	2 men, 2 shifts/day for 4 years
6. 30m Radiometer	15,400 + 14,456* 33,880 + 32,000*	2.0	210 man-shifts, fab/assy 60 man-shifts, test
7. Sensor Development	13,400 29,542	10 kw	2 men, 1 shift/day continuous
8. Living and Working in Space	750 1,653	('84-'87) = 0.5ks ('87) = 1.0kw	1 man, 1 shift/day ('84-'87) 2 men, 1 shift/day ('87)
9. Multidiscipline Laboratory	35,200 77,603 (per year)	12 kw	3 men, 2 shift/day continuous

*Fabrication and Assembly module required for TA-1, TA-2 or 30m radiometer, requires one cargo Shuttle.

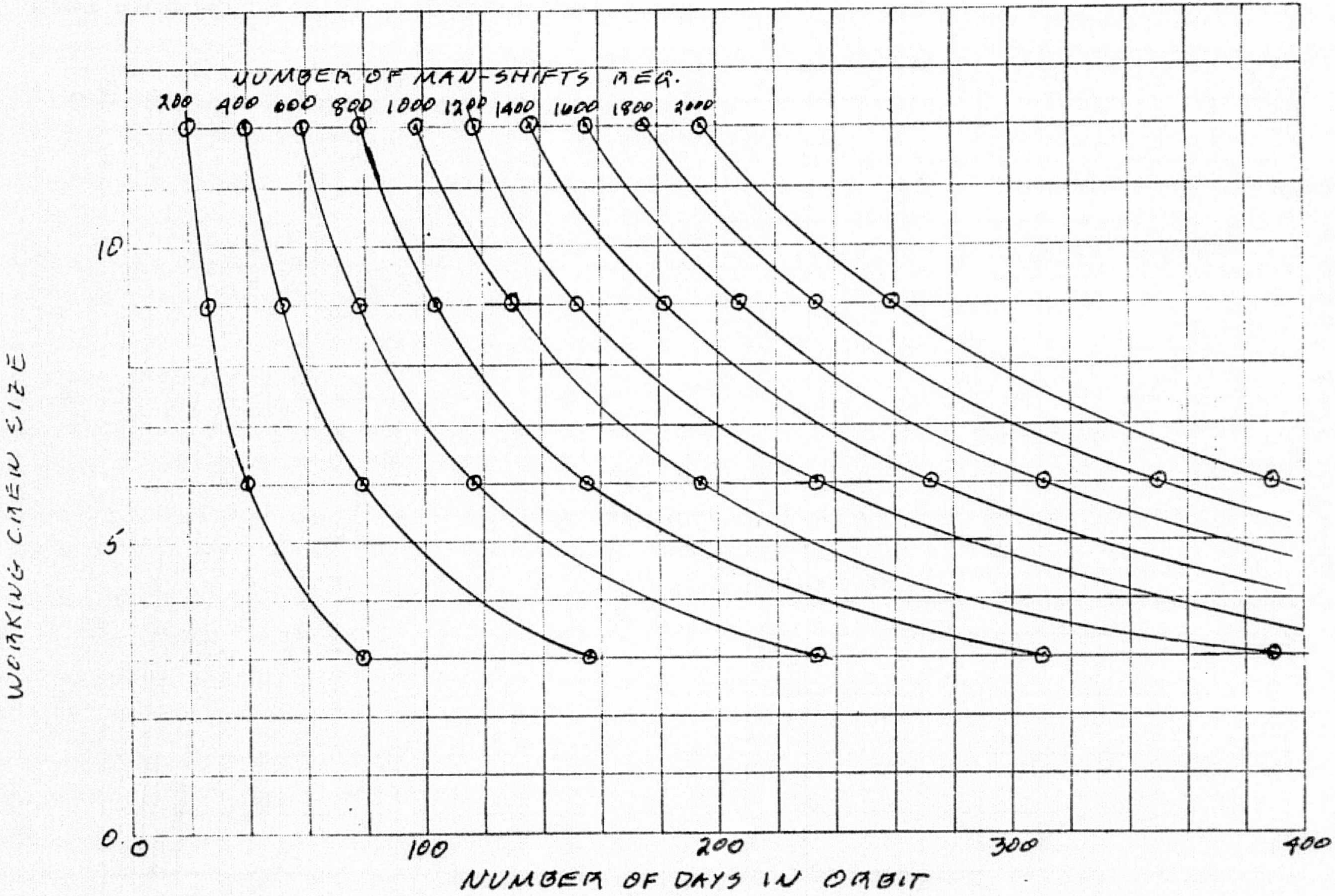

Figure 6. General Task Resource Requirements

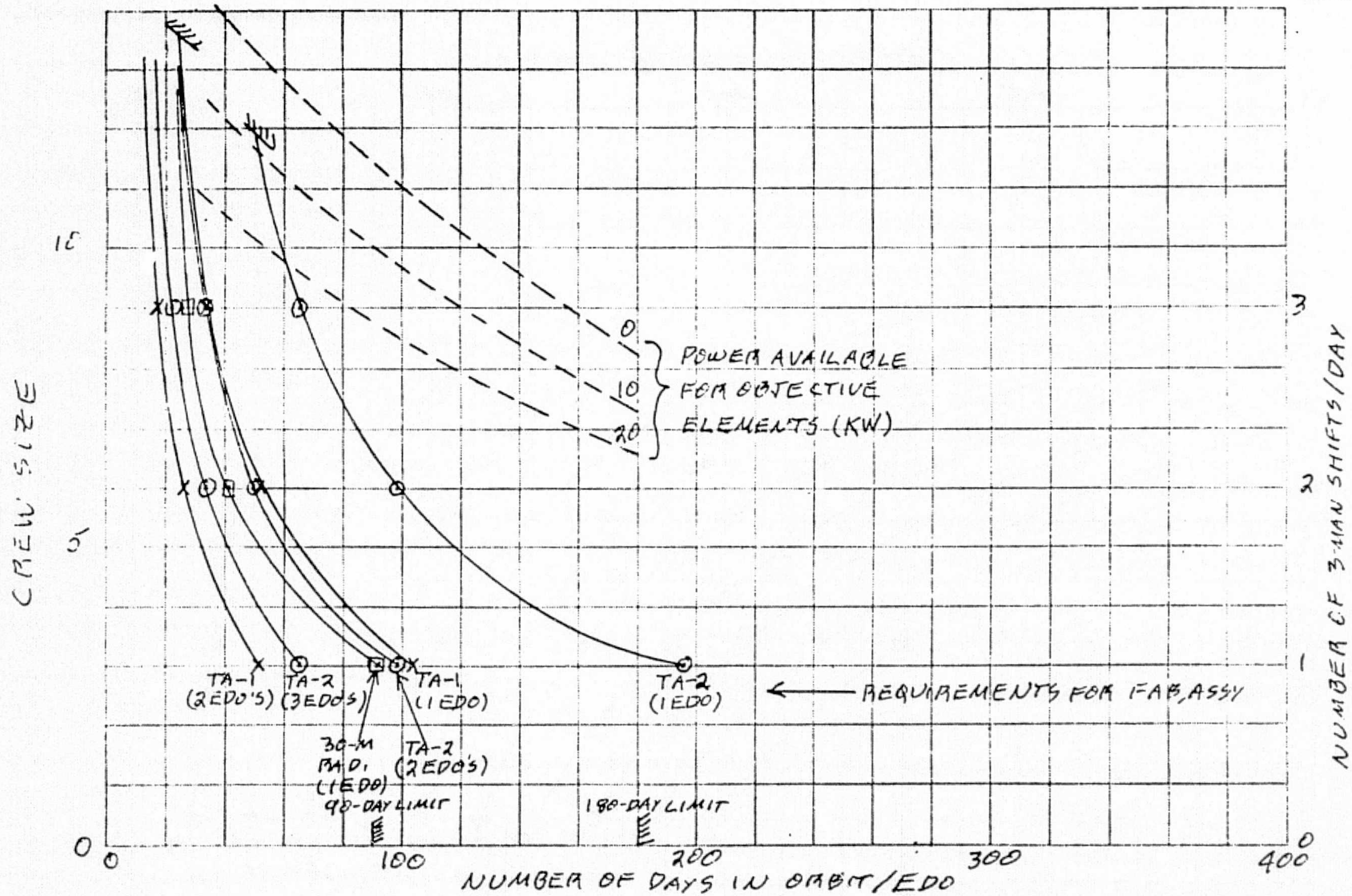

Figure 7. Operable Region for EDO Support of Objective Elements (Fab, Assy)

the objective element over to the next EDO, thus requiring two EDO's to be assigned to the program. The 30m radiometer task is small enough that only a single EDO need be considered.

The addition of the test time to the fabrication and assembly tasks considerably enlarges the requirements of the tasks, as shown in Figure 8. Here, it is apparent that TA-2 cannot be done with a single EDO since any chance that the Required and Available curves would cross is well past the 180-day limit on zero-g crew duration. The available resource margin area with two EDO's is small, and therefore risky.

A matrix of possible solutions for the three construction tasks (including test) is given in Table 8. Solutions to fit a 180-day limit are shown, and 90-day limit solutions are also provided in the event that Orbiter and EDO subsystem reliability issues may limit EDO duration. Some of the marginal (in duration or power) cases are shown to end some lsensitivity to the data. Of all the solutions, the most acceptable are generally those utilizing three shifts (to minimize ground support time) and more EDO's to minimize the single EDO MTBF requirements. The latter choice may, in fact, introduce other reliability factors requiring more series elements for mission success, and, given the appropriate Orbiter data, would be a good area for tradeoff.

At this point in the analysis, it should be pointed out that three areas of conservatism are involved:

- Redundancy (weight) allowances for Orbiter subsystem on-orbit life extension were assumed minimal, and will probably be a large factor in even 90-day life extensions.
- The assumed Orbiter basic power requirement of 2 kW is probably optimistic with some recent data indicating as much as 6 kW. This has the effect of lowering the Available curves so that 0 kW would be at an indicated value of 4 kW.
- The assumption on crew size is that the flight crew (pilot, co-pilot), etc.) are full working members of the construction crew. Additionally, the monitoring, operations, and maintenance of the Orbiter and its subsystems on-orbit were considered only as a negligible part-time effort performed by off-shift crewmen. If either or (both) of these assumptions do not hold, it is possible that

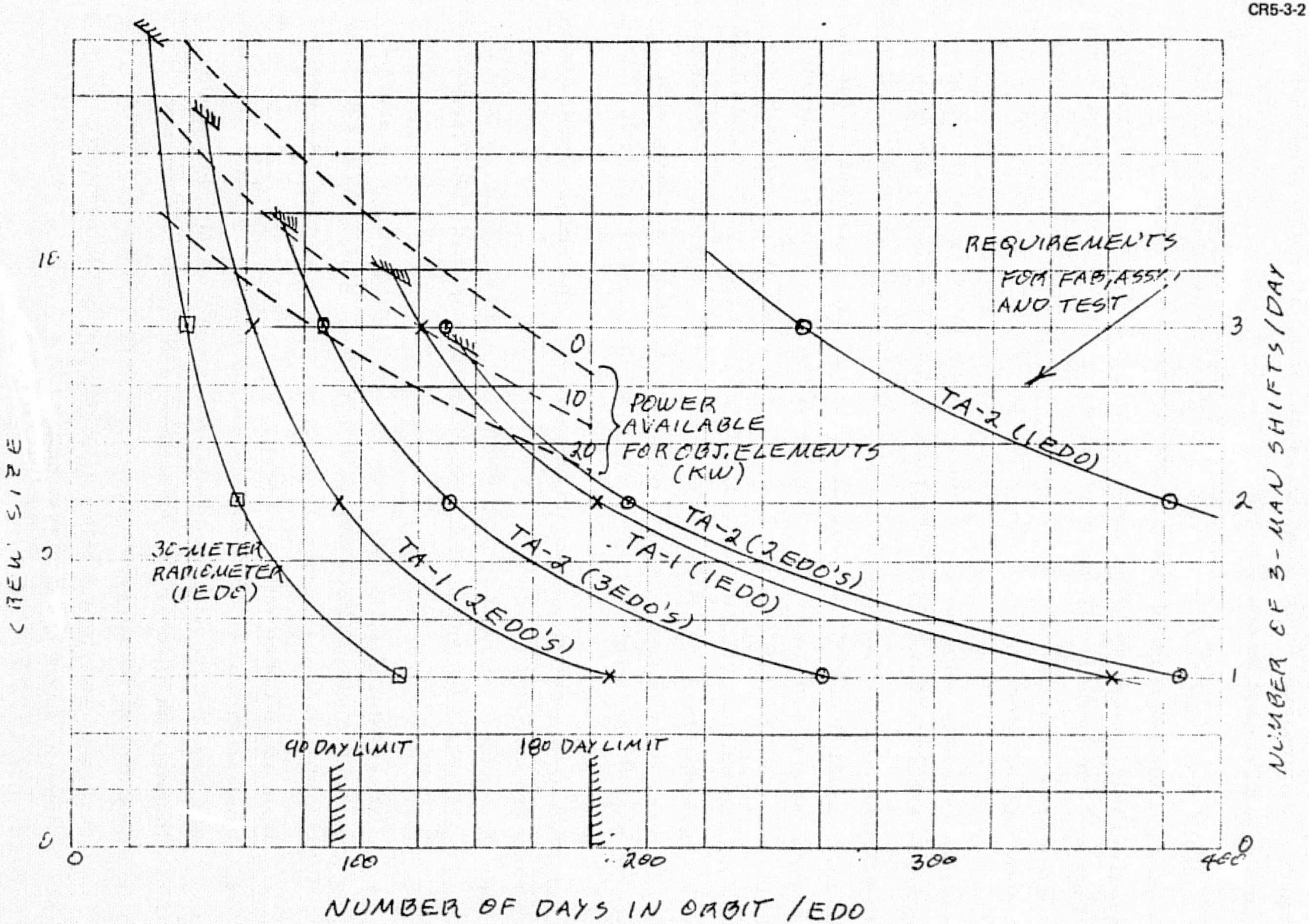

Figure 8. Operable Region for EDO Support of Objective Elements (Fab, Assy, Test)

Table 8

EDO CANDIDATE SOLUTIONS TO OBJECTIVE ELEMENTS
(ASSY, FAB, AND TEST)

Objective Element	Number of Series EDO's	90-Day Limit 1 Shift (3 Men)	90-Day Limit 2 Shifts (6 Men)	90-Day Limit 3 Shifts (9 Men)	180-Day Limit 1 Shift (3 Men)	180-Day Limit 2 Shifts (6 Men)	180-Day Limit 3 Shifts (9 Men)
TA-1	1	NS	NS	NS	NS	182* Days per EDO	121 Days per EDO
	2**	NS	91* Days per EDO	61 Days per EDO	182* Days per EDO	91 Days per EDO	61 Days per EDO
TA-2	1	NS	NS	NS	NS	NS	NS
	2**	NS	NS	NS	NS	NS	129* Days (Power Lim)
	3**	NS	NS	87 Days per EDO	NS	131 Days per EDO	87 Days per EDO
30m Radiometer	1	NS	58 Days per EDO	39 Days per EDO	116 Days per EDO	58 Days per EDO	39 Days per EDO

NS = No solution
* = Marginal case
** = Assumes mechanical handover

as much as three more men might be required to be accommodated in orbit over and above the construction crew. Since they produce no net work to the objective element, this would have the gross effect of raising the ordinate of the Required crews by the number of support men required. If this value is three men, it can be seen in Figure 8 that even the 3-EDO case for performing TA-2 is very marginal.

Thus, it is possible that the results presented so far can be considered somewhat optimistic.

The operational flights to initiate TA-1 and TA-2 are summarized in Table 9, along with an estimate of how many of the modules must be returned from orbit upon completion of the task. Modules needed for other orbital operations must be returned by cargo Orbiters because of the "no orbital storage" ground rule. Under the indicated assumptions, both of these objective elements can be achieved with a single launch facility and with two cargo Orbiters assigned. No consideration of backup Orbiters on standby was made in this analysis, but they would be required to increase the probability of success.

The summary of EDO solutions to the eight objective elements is given in Table 10 in terms of the nuber of EDO and cargo Shuttle flights, and the minimum operational fleet size (no spares). The construction-oriented objective elements are associated with the largest crews and the most elaborate cargo Shuttle support and return schedules. The manufacturing objective elements require a regular 180-day EDO rotation for the duration of the planned manufacturing period. Because of the high power requirement for these elements (especially the silicon lab), the combination of all three can only be done in 30-day increments. The remaining level-of-effort experimentally-oriented objective elements require nominal support.

CONCLUSIONS

A major conclusion of this substudy is that accomplishing SSSAS missions with sortie missions is technically practical. However, from a cost standpoint, the use of sortie missions without resorting to orbital storage (Shuttle-tended mode) would appear to be both wasteful and involve unwarranted risks, unless backup vehicles are always available.

Table 9

OPERATIONAL FLIGHT ORDER

Item	TA-1	TA-2
Plastic Tube Module	X	X
Fabrication and Assembly Control Module	X	X
Extended Duration Orbiter (EDO)	X	X
Antenna Construction Fixture	X	X
Construction Support and Materials	X	X*
Solar Collector Tooling and Fixtures		X
Solar Collector Beam Makers		X

*Not returned from orbit

Assumptions:

- Average launch interval 1 week
- Orbiter on-orbit duration = 1 week
- Orbiter turn-around = 2 weeks
- Launcher turn-around = 1 week

Conclusions:

- Minimum operational fleet = 2 Orbiters (with handover of No. 1 to No. 2 in orbit)
- One launcher facility

Table 10

EXTENDED DURATION ORBITER (EDO) APPLICATION TO OBJECTIVE ELEMENTS

Objective Elements	No. EDO Flts (180 days) / No. Min Operational Fleet	No. Shutter Flights / No. Min Op'l Fleet	No. Return Shuttle Flights
1. SPS TA-1	1 (9-man crew) / 1	2 + 1 * / 2	2 + 1 *
2. SPS TA-2	3 (9-man crew) / 2	5 + 1 * / 2	4 + 1
3. Biological Development	8** (3-man crew) / 2	1	1
4. Fiber Optics Development	8** (3-man crew) / 2	1	1
5. Silicon Crystal Development	8** (4-man crew) / 2	1	1***
6. 30m Radiometer	1 (9-man crew) / 1	2 + 1 *	1 + 1 *
7. Sensor Development	2/yr (2-man crew) pick-a-back / 2	1	1
8. Living and Working in Space	pick-a-back	0	0
9. Multidiscipline Laboratory	2/yr (6-man crew) / 2	2/yr / 1	2

*Fab. and Assy. module required for TA-1, TA-2 or 30-meter radiometer, requires one cargo Shuttle.
**3 space processing objectives can be combined with 48 flights (10-man crews) with 2 EDO's assigned.
***Return weight marginal

Most of the objective hardware in the SSSAS missions is reusable: construction equipment may be used over again and again; while the process development modules are devoted to specific vlentures, each has wide applicability to other projects using similar technology, and the multidiscplinary laboratory is versatile by design. Hence, each must be returned for its next mission unless they can be stored.

Secondly, without orbital storage capability, launch delay of any EDO hotel could abort the whole mission and cause loss of all mission hardware on orbit.

On the other hand, the inclusion of orbital storage capability for any assembly is seen as a relatively small complexity. Gravity gradient stabilization may be used without an active control system and thus most subsystems can hibernate and the only conscious system needed is a simple command receiver (to turn on the control system and telemetry).

It is therefore concluded that the shuttle-tended mode is preferred over the sortie mode when coordinated. Multiple-Orbiter flights are required.

Further, it is believed that the Shuttle-tended mode is a relatively efficient means of accomplishing many construction base or Space Station objectives and that a rational choice between Permanent Manned Base and Shuttle-tended modes cannot be made by the simple analyses discussed here. This more difficult comparison must be made on a cost-accomplishment basis. Part III of the SSSAS will examine, in some detail, the Shuttle-tended mode.

Part 16

JSC PHASE-B RCS CONSIDERATIONS IN DEFINITION OF SCB STANDARD MODULE CONFIGURATIONS (OPTION L)

Section 1
SUMMARY OF JSC PHASE "B" RCS DOCUMENT REVIEW AND ITS APPLICABILITY TO THE SCB

The Reaction Control Subsystem (RCS) sections of the JSC Phase-B documentation were reviewed. The following paragraphs summarize the report contents with respect to RCS requirements and performance characteristics, and present comments on necessary RCS modifications for the SCB, impact of SCB growth on the RCS, and areas of further investigation relative to RCS application to the SCB.

1.1 JSC PHASE-B RCS REQUIREMENTS DEFINTION

The RCS provides the forces necessary for control of the modular Space Station (MSS). It provides forces to stabilize the initial modules of the MSS during unmanned buildup for Shuttle docking and buildup operations. During manned operations, forces are necessary to overcome the momentum loss caused by atmospheric drag concurrent with control moment gyroscope (CMG) desaturation, to maneuver the station for experiments, to stabilize attitude for Shuttle docking, and to control docking and undocking torques introduced by the Shuttle.

The RCS also provides storage for gases common to it and other subsystems. Whenever possible, common gases are used by the RCS, environmental control life support subsystem (ECLSS), and electrical power subsystem (EPS). In this manner, the number of resupplied gases, types of tankage, types of equipment, and cost of development can be reduced.

The major functional assemblies of the RCS include the propellant accumulators, propellant feed control, and engine assembly.

The RCS was sized by evaluating the major requirements as established by buildup operations during the unmanned phase of MSS operations and the major requirements for normal manned operations.

The major criterion for the RCS during buildup is to maintain station control for docking to the Shuttle. In addition, the safety requirement states that control is required through two RCS failures; therefore, the initial module must include the full complement of RCS engine quads. However, the propellants needed to meet the buildup impulse requirements (Table 1) are supplied by the EPS high-pressure gas storage for the first two buildup phases and by the EPS electrolysis units for buildup Steps 3 through 7 (see Figure 1).

Table 1

BUILD-UP PHASE IMPULSE REQUIREMENTS

	Impulse Requirements (lb-sec/30 days)		
Buildup Step	Solar Panels Not Deployed	Solar Panels Deployed 25%	Solar Panels Deployed 100%
1	2,400		
2	10,200		
3		16,900	37,000
4		22,000	42,000
5		27,200	47,000
6		31,600	51,500
7		31,600	51,500

The impulse requirements during normal orbital operations are shown in Table 2.

Table 2

RCS IMPULSE REQUIREMENTS*

Area	Requirement (lb-sec/120 days)
Orbit makeup and CMG desaturation	166,000
Maneuvers	48,000
Shuttle on	28,000
Contingency (20%)	48,000
Total	290,000

*Normal Operations, 6-Man Level

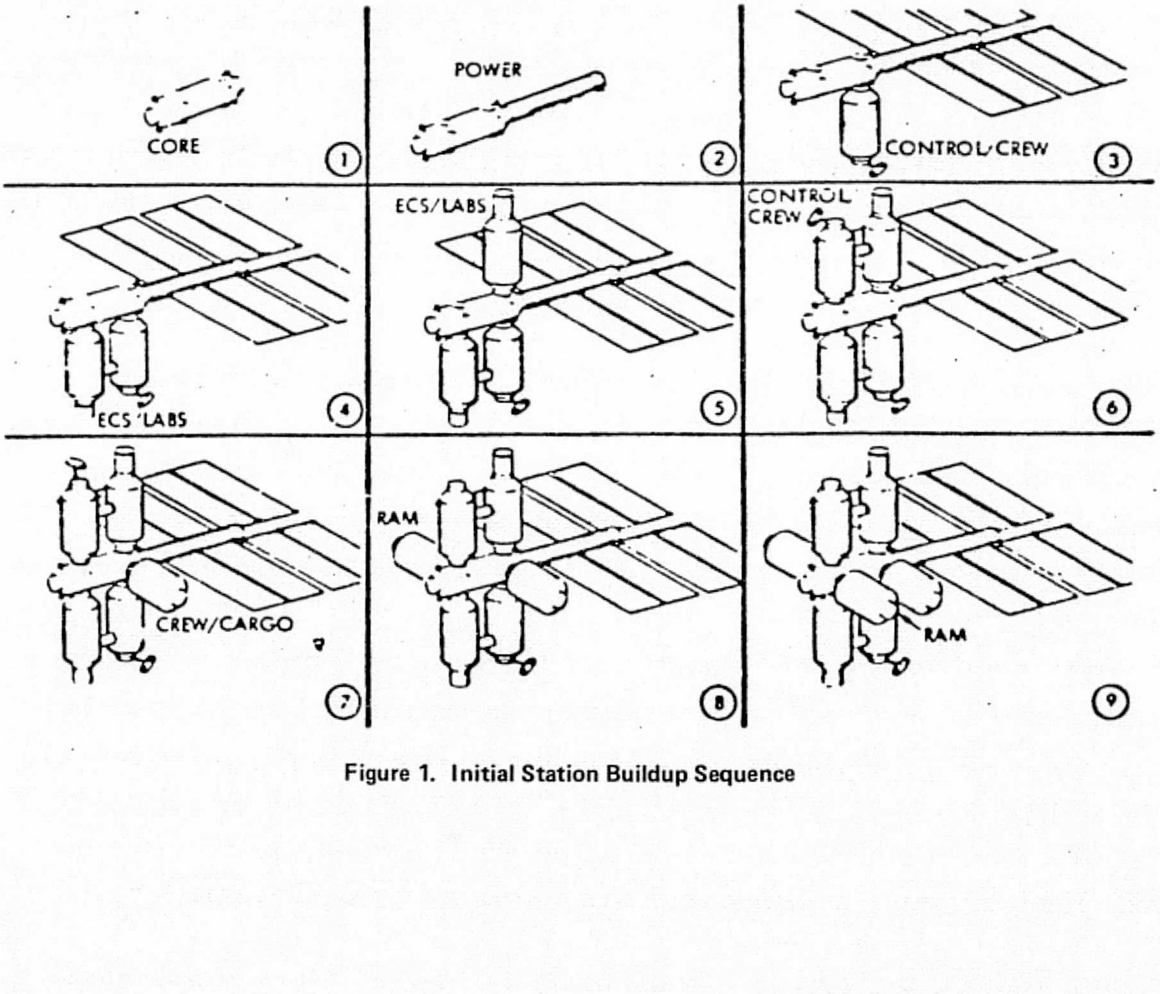

Figure 1. Initial Station Buildup Sequence

Additional requirements employed in sizing the RCS are as follows:

1. 55-degree orbit inclination, 240-nmi "design-to" altitude.
2. XPOP flight mode
3. No effluent dump for 12 hours during experiment operations.

These requirements, along with safety and total impulse requirements, determined the RCS accumulator sizes, engine locations, and firing durations.

When the EPS experiences a failure causing loss of power for electrolysis operation, no reactants can be produced for RCS operations. In the event of this emergency condition, 8,000-lb-sec of impulse are required to provide a single docking to the Shuttle. The RCS, in a cold-gas firing mode, can use the high-pressure oxygen gas stored in the power boom normally used for MSS repressurization. There are 195 lb of oxygen stored in the power boom. The RCS requires 123 lb of oxygen expelled through the thrusters, at an ISP of 65 sec to meet the emergency docking requirement.

1.2 JSC PHASE-B PERFORMANCE CHARACTERISTICS DEFINITION

The basic RCS (Figure 2) contains three major assemblies: propellant accumulator, propellant feed control, and engine. The propellant gases, hydrogen and oxygen, are produced by the ECLSS electrolysis unit and stored in accumulators located in SM-2 and SM-3. The accumulators store the reactants at 300 psia and provide storage to accomplish one-half the daily impulse. The accumulators also store ECLSS oxygen and hydrogen to maintain ECLSS functions during orbital dark operations. Storage is maintained for 12-hour intervals to satisfy the no-venting conditions during experiment operations. In the event of ECLSS electrclysis failure, the accumulators can be supplied from the EPS electrolysis unit; or in event of EPS electrolysis failure, the accumulators can supply hydrogen and oxygen to the EPS fuel cells.

The accumulator sizing is based on a two-sigma Jacchia mean atmosphere. Earlier studies based the sizing on the 1959 ARDC standard atamosphere. In order to utilize the atmospheric variations, a firm IOC of February 1982 is ncesssary to determine accumulator sizing because the atmospheric density varies with each year.

The RCS performance characteristics are defined in Table 3.

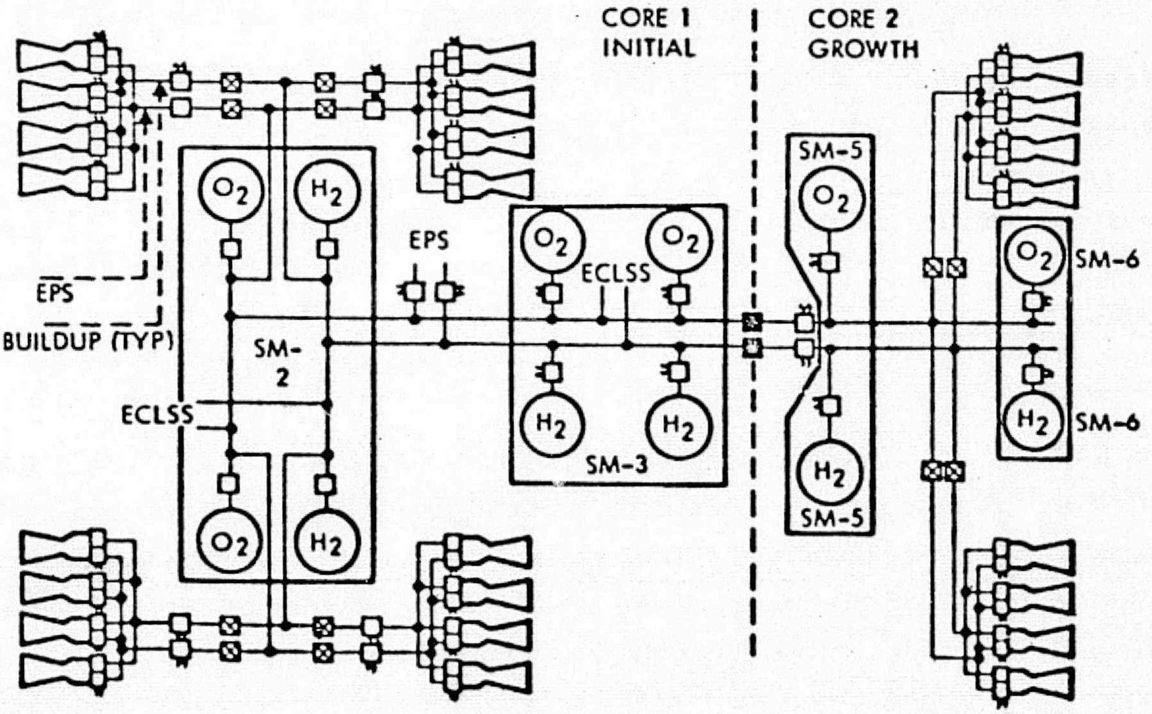

Figure 2. RCS Schematic

Table 3

RCS PERFORMANCE CHARACTERISTICS

Item	Characteristics
	ENGINES
Thrust	10 lb
Specific impulse	320 sec
Oxidizer/fuel ratio	8:1
Propellant temperature	70°F
Firing duration	60 sec/two thrusters every 12 hr
	ACCUMULATORS
Pressure (nominal)	300 psia
Pressure (minimum)	50 psia
Stored impulse	300 lb-sec/accum pair
Hydrogen storage	0.10 lb/accum RCS
	0.014 lb/accum ECLSS
Oxygen storage	0.8 lb/accum RCS
	0.109 lb/accum ECLSS

The ECLSS-RCS interface consists of the water electrolysis unit of the ECLSS with the RCS accumulator. Water is pumped from the integrated water storage tanks (cargo module tanks, potable water tanks on EPS energy storage water tanks) to an ECLSS electrolysis unit where it is electrolyzed into oxygen and hydrogen. The electrolysis is done at a high pressure (300 psia) so that compression of the gases is not required before they are passed to the accumulators. In addition, the repressurization oxygen stored in the power boom can be used by the RCS to provide vehicle stabilization in the event of a power loss that would disable the electrolysis units. The high-pressure oxygen will be used for cold gas thrusting through the engine quads.

The EPS-RCS interface includes the EPS electrolysis units supplying oxygen and hydrogen to the RCS if an ECLSS electrolysis failure occurs. The EPS supply is only a backup supply, and is not intended as a primary source of RCS oxygen and hydrogen during manned operations. However, the EPS is the primary supply of RCS hydrogen and oxygen during buildup before ECLSS equipment is available. The EPS supplies the RCS requirement from high-pressure gas (3,000 psia) that is aboard the first launch (core module).

The docking interface consists of hydrogen lines between SM-2 and SM-3 and the core module. These lines connect the accumulators to the engine quads in the core. Oxygen lines are shared with both the EPS and ECLSS, an integrated gas distribution concept which eliminates duplication of lines, reduces complexity, improves reliability through shared redundancy, reduces weight and provides for cost reductions. For the growth station, additional hydrogen connections are required between Core 1 and Core 2 and between SM-5 and SM-6 to Core 2. These lines connect the accumulators to the engine quads on Core 2.

The G&C subsystem provides the electronic driver units that actuate engine solenoid valves in firing the thrusters.

1.3 PHASE-B RCS MODIFICATIONS NECESSARY FOR STANDARD OPTION L SCB APPLICATION

The Standard Option L SCB is shown in Figure 3 and the differences between this and the Phase-B configuration can be seen by comparison with Figure 1. Specific modifications (if any) necessary for Standard Option-L application have not been assessed at this time. However, areas of potential impact have been identified. These areas are increased RCS thrust level and increased accumulator volume and pressure. The potential thrust level increase may be necessary because of increased SCB mass and moments of inertia. Accumulator volume increase may be required because of higher ECLSS oxygen/hydrogen output and increased impulse for SCB stabilization and maneuvering.

When control forces, minimum impulse bit, and oxygen/hydrogen RCS interface timeline requirements are established, a more detailed modification assessment can be accomplished.

1.4 IMPACT OF GROWTH IN SCB

When considering growth versions of the SCB, the areas of impact are essentially the same as those discussed in the previous paragraph regading modifications necessary for Option-L application. That is, thrust level increase

PROGRAM OPTION L PERMANENTLY MANNED

7-MAN FABRICATION/ASSEMBLY FACILITY

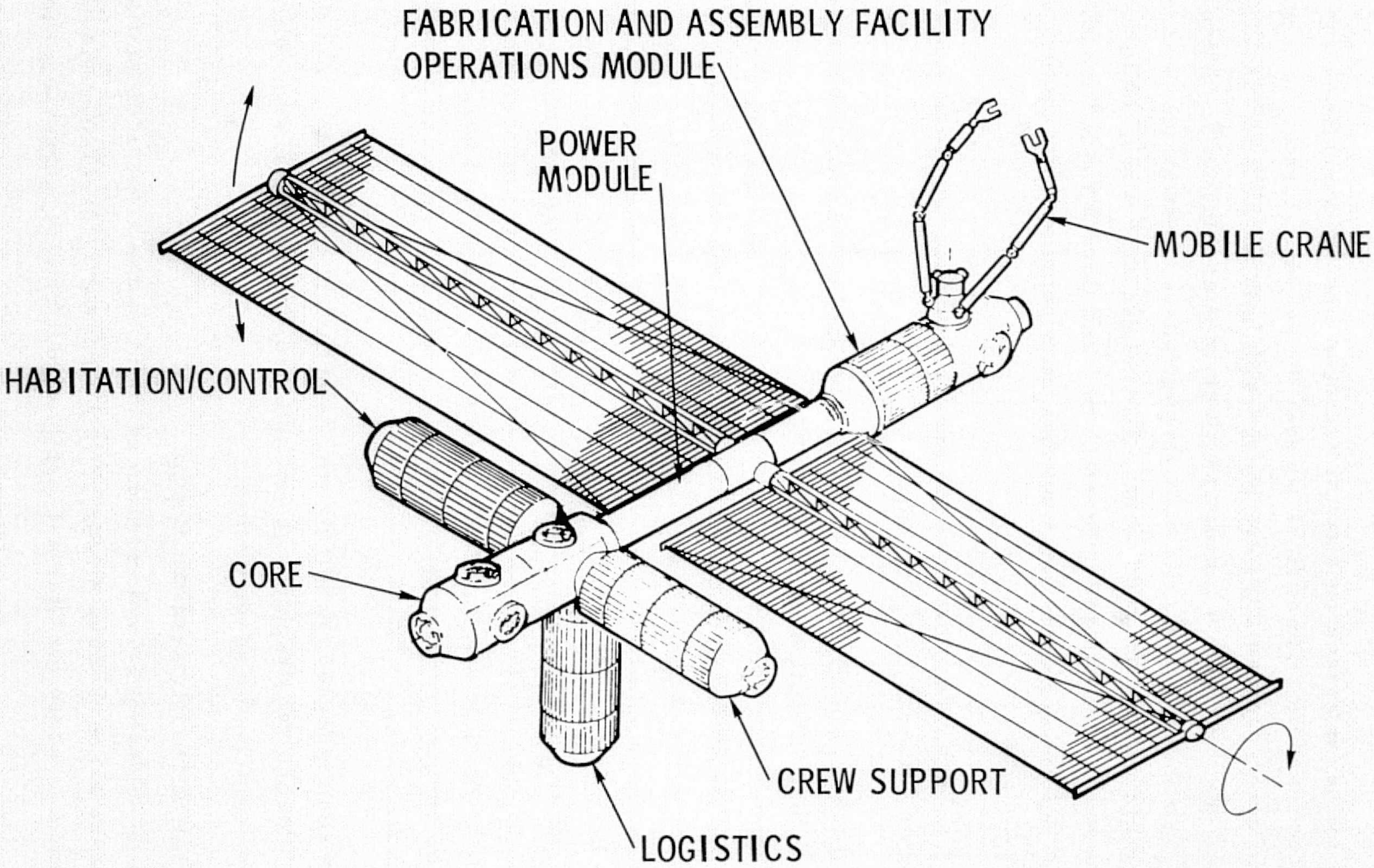

Figure 3. Standard Option L

may be required to compensate for a larger mass SCB with increased MOI's, and accumulator volume may have to be increased to store higher oxygen/hydrogen output from the electrolysis units. Specific values cannot be presented at this time since the requirements affecting these evaluations have not been established for the Option-L or growth configurations.

1.5 AREAS WHICH REQUIRE FURTHER INVESTIGATION

Areas identified herein which require further investigation directly reflect the discussions in the previous paragraphs. The RCS thrust level should be investigated to determine the thrust required to meet SCB maneuvering requirements, (rate and acceleration), stabilization (minimum impulse bit), drag makeup and CMG desaturation.

The oxygen/hydrogen accumulators are required to store the gases generated by the ECLSS electrolysis units to satisfy several functional requirements including: control and maneuvering impulse, ECLSS supply during dark operations, and EPS emergency supply. Therefore, the volume requirements should be investigated separately and in combination to determine the worst case volume requirement. Accumulator pressures can also be investigated if volume reduction would be of significant benefit.

Since the RCS, ECLSS and EPS are integrated, any investigations concerning oxygen/hydrogen mass, pressure, etc., must be closely coordinated between these three areas.

Section 2
SUMMARY OF RCS REQUIREMENTS AND PERFORMANCE FOR SCB 7-MAN OPTION L

The requirements and performance characteristics for the Option-L SCB RCS are summarized in Table 4. The requirements are partially qualitative since quantitative values have not been established for all of the requirements at this time. It has been assumed that the Phase-B-derived RCS will be adequate to meet the Option-L requirements, and those performance characteristics are shown in the table.

A block diagram of the RCS is shown in Figure 4.

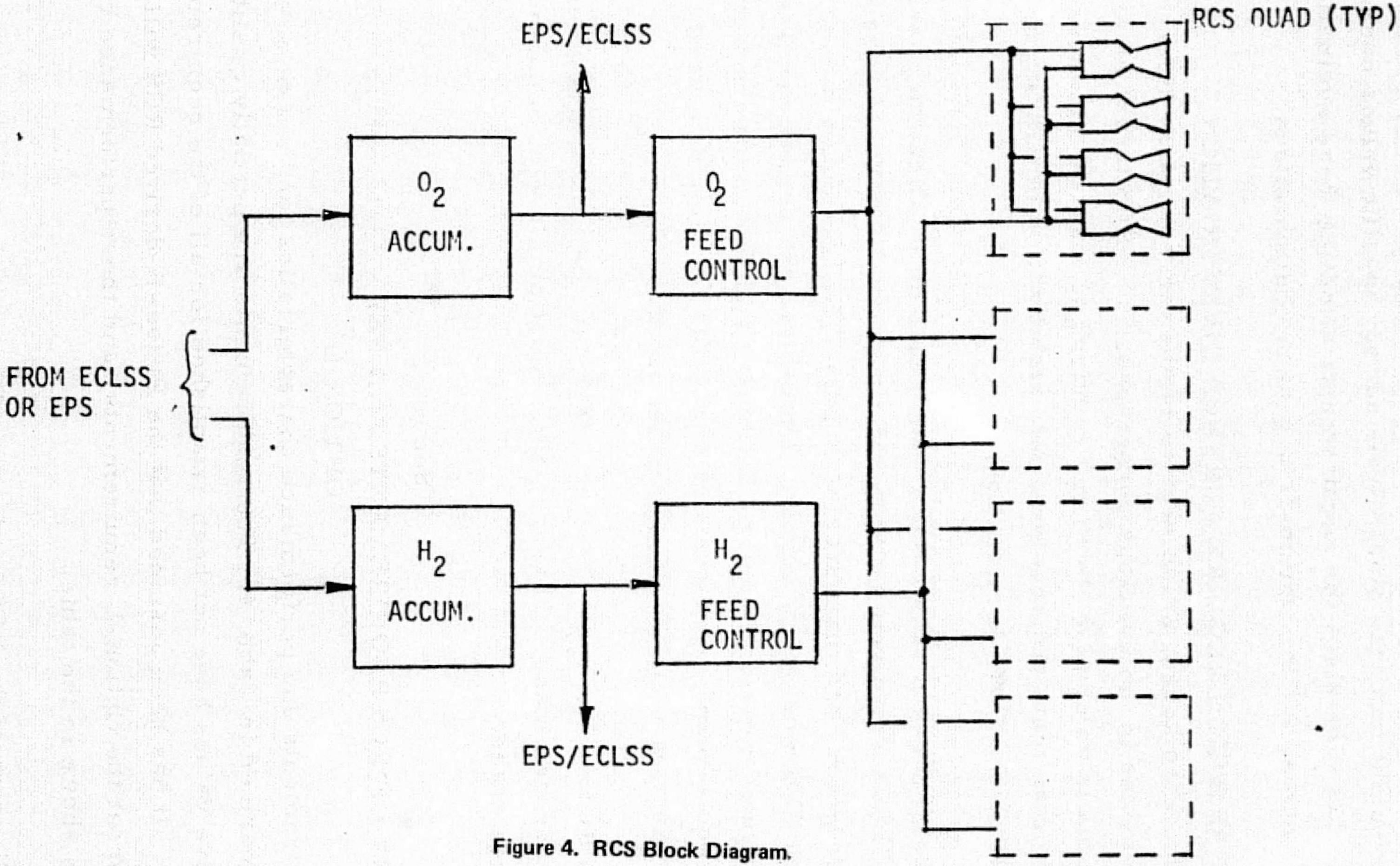

Figure 4. RCS Block Diagram.

Table 4

SCB PROPULSION INFORMATION - PHASE-B-DERIVED DESIGN

Assembly	Design Requirements	Performance
Accumulators	12 hr no vent (experiment)	300 to 50 psia operating range
	Store 1/2 daily impulse requirement	16 in ID H_2, 13 in ID O_2
	Store all ECLSS generated O_2/H_2	Weight: 22 lb H_2, 18 lb O_2
	Supply ECLSS during dark operations	Store 0.114 lb H_2, 0.909 lb O_2
	Emergency supply for EPS	
Propellant Feed Control	Regulate O_2/H_2 pressure	50 psia primary reg
	Control flow direction	55 psia backup reg
	Distribute O_2/H_2 to RCS quads	EPS to RCS check valves
		Aluminum alloy feedlines
Thruster Quads	Provide SCB stabilization, maneuvering, drag makeup, CMG desaturation, and control during Shuttle dock/undock	10 lb thrust, 320 sec ISP, 8:1 mixt. ratio, 140-hrs life
	Control required after two RCS failures	Redundant thrusters
		Thruster/quad isolation

Section 3

AREAS SENSITIVE TO OPTION-L GROWTH TO 14- AND 21_MEN CREW CONFIGURATIONS

For the Phase-B-derived RCS, the areas sensitive to growth are thrust level and accumulator volume. As discussed previously, these areas are affected due to SCB mass increases and greater ECLSS output. It is possible that growth versions could be accommodated by additional thrusters and accumulators.

Part 17

GUIDANCE, CONTROL, AND NAVIGATION SUBSYSTEMS

GUIDANCE, CONTROL, AND NAVIGATION SUBSYSTEMS

Section 1 — PROGRAM OPTIONS L AND L'

INTRODUCTION AND SUMMARY

This appendix contains a brief discussion of SCB Guidance, Control, and Navigation Subsystem (GC&NS) concepts relative to Program Options L and L'. The JSC Phase B GC&NS design is reviewed and discussed relative to SCB requirements. Preliminary SCB Option L and Option L' GC&NS designs are presented along with impacts associated with growth to larger SCB configurations.

Section 2
DISCUSSION OF GC&N SUBSYSTEMS

2.1 SCB PROGRAM OPTION L GC&NS SUBSYSTEM

The Guidance, Control and Navigation Subsystem (GC&NS) defined below was derived from the Johnson Space Center Modular Space Stations (MSS) Phase B design. The GC&NS Phase B design requirements were primarily derived from earth observation experiment considerations. The Space Construction Base (SCB) concept under study emphasizes space fabrication and assembly and preoperational testing of large satellites and providing low g-level environments for space processing of special materials. New design requirements must be defined for the SCB configurations considering the fabrication and assembly and space processing objective elements. The definition of these requirements with respect to the GC&N subsystem has not been completed and further definition progresses as the fabrication, assembly, and testing procedures are defined.

The Phase B GC&N subsystem is reviewed below and some discussion relative to modifications necessary to meet SCB requirements and growth impacts are presented.

2.1.1 JSC Phase B Design Review and Application to SCB Option L Configurations

2.1.1.1 Phase B Design Requirements and GC&N Subsystem Definition and Performance

The Phase B GC&N system design requirements were derived from earth observation experiment considerations. Attitude control requirements defined with respect to the local vertical, where,

Pointing

± 0.25 deg (long term)

± 0.1 deg (30 minutes)

Rate Stability

0.05 deg/sec (long term)

$<$0.01 deg/sec (30 minutes).

The required one-sigma navigation accuracies were:

Down range	±1.2 km
Cross range	±0.7 km
Altitude	±0.5 km

Figure 1 shows a functional block diagram of the Phase B GC&N subsystem. Sensors included:

1. Two gimballed star trackers.
2. A horizon sensor.
3. A strapdown IMU.
4. A manual sextant/telescope.

The alignment links shown were used to align the various sensors which were physically separated in the MSS. The control torques were generated with control moments gyros (CMG's) and RCS thrusters. Orbit-keeping maneuvers also used the RCS thrusters. The digital data-processing load was distributed among preprocessors and the main multiprocessor.

The two star trackers were double gimballed and based on the Kollsman Instrument Corporation Model KS 199. Their instrument accuracy was assumed to be 0.003 deg one sigma. Both units were used for IMU updates but one could be turned off and used as a standby redundant unit. They were located separately in the MSS core module hull about 90 deg apart along the hull circumference. An optical alignment link provides calibration capability.

The horizon sensor used four separate edge tracker heads and operated in the 14- to 16-micron carbon dioxide absorption band. The assumed instrument accuracy was 0.017 deg one-sigma and was based on a Quantic Industries Mod IV horizon sensor system. The horizon edge tracker heads were all mounted on a common rigid base which was in the same cross-sectional plane of the module as the star trackers. The IMU and sextant-telescope were mounted on the rigid base with the horizon sensor optics.

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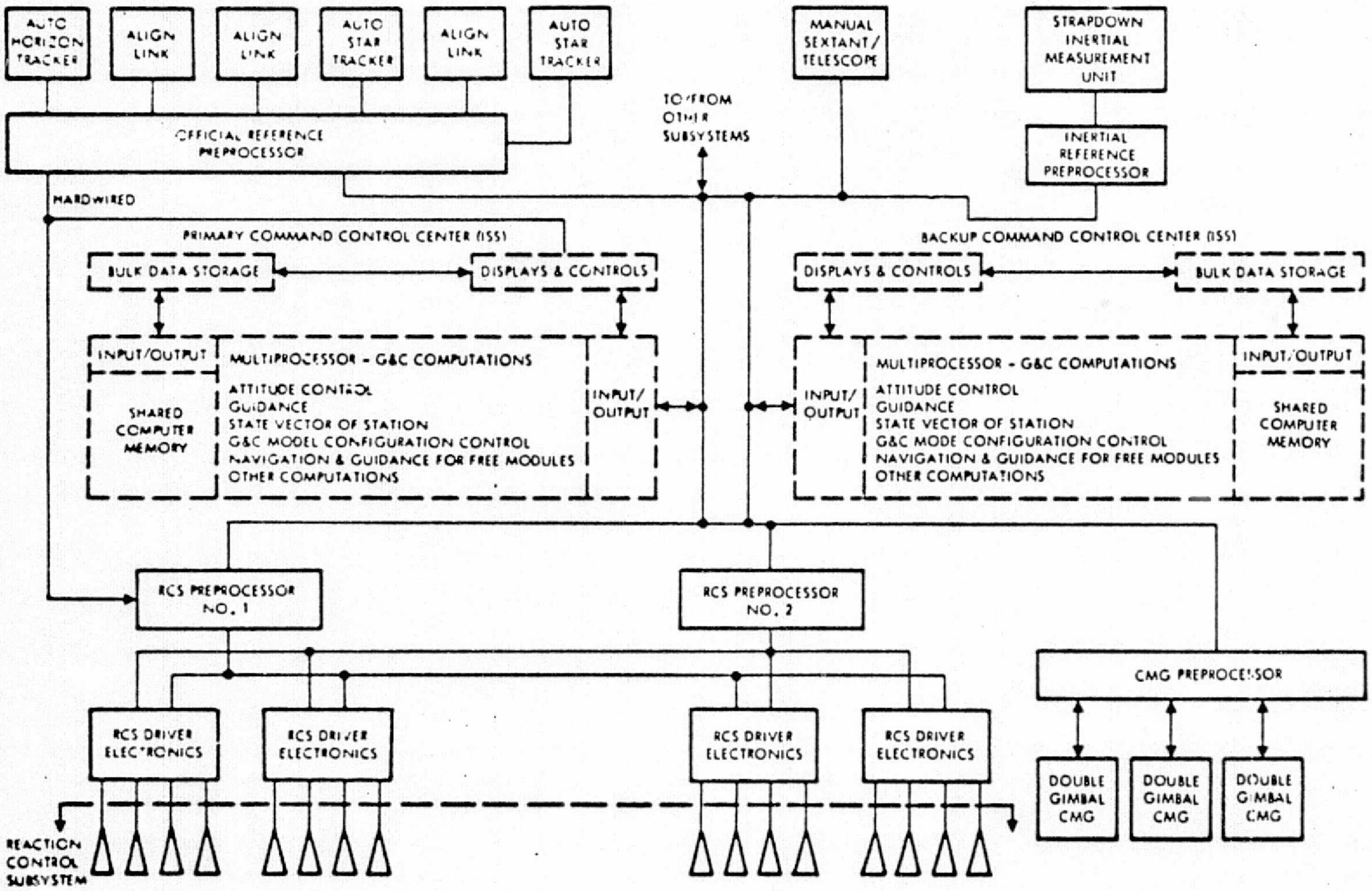


Figure 1. Phase B G&CNS Functional Block Diagram

The IMU used 6 single-degree-of-freedom strapdown rate integrating gyros and 6 accelerometers in a hexad arrangement. Both the accelerometers and gyros used pulse rebalance techniques. No performance data for the IMU was presented in the Phase B report.

The sextant/telescope was an Apollo optical unit assembly. It was mounted on the same reference base as the horizon sensors and IMU and its accuracy w was assumed 0.003 deg one-sigma.

The momentum exchange assembly consisted of 3 double-gimballed CMG's, each with an angular momentum of 1500 n-m-sec. The outer gimbals had unlimited rotation and the inner gimbals were limited to ±80 deg rotation. The CMG array had all the outer gimbal axes parallel (along the MSS longitudinal axis) and the initial inner axes oriented to evenly distribute the three angular momentum vectors in the orbit plane. Total weight of the momentum exchange assembly (including mountings and electronics) was 450 kg, and the average power requirement was 144w. The CMG sizing resulted in a desaturation frequency of once per 12 hours with all three CMG's operating and once per orbit with two operating.

The RCS thrusters were gaseous hydrogen/oxygen and the thrust levels were 45 newtons per thruster. The thrusters were clustered in groups of four and four clusters were included for a total of 16 thrusters. The clusters were mounted two on each end of the core module.

2.1.1.2 SCB Option L GC&N Subsystem Design Requirements and Subsystem Definition

The construction activity and objective element impacts of SCB Option L have not been determined, but a few general statements may be made.

A. The SCB must be able to maintain an attitude hold with sufficiently small rates to facilitate an Orbiter docking. This is a relatively short-term requirement and lateral velocities of the SCB docking interface of less than ±0.25 fps are required. An attitude hold within a few degrees is also adequate. The Orbiter will maneuver to make the dock.

B. The position and velocity of SCB must be known to a sufficient accuracy to allow a reasonable rendezvous procedure for the

Orbiter. Completely autonomous navigation by the SCB is not required, but onboard ephemeris generation is required to supplement the ground link updates.

C. During fabrication and assembly it may be desirable to orient the objective element structure in some specific way with respect to the sun (and earth) to minimize thermal deformations and other solar effects. These attitude holds may have a duration of days or weeks. The required accuracy of the attitude hold for this purpose will probably be on the order of a few degress with respect to the reference vector. The gravity-gradient moments associated with the SCB are large for some orientations. The impact to the attitude control system (and RCS) of holding a given sun relative orientation for long periods of time will have to be evaluated and analyzed with respect to the fabrication and assembly requirement.

D. The SCB must provide a stable base for objective element testing, space processing, and scientific experiments. Space processing requirements include a maximum lateral acceleration environment of 10^{-3}g's. The TA-1 and 30m radiometer testing procedures appear at this time to impose a pointing requirement on the order of 0.0005 deg. Pointing accuracies of this type must be accomplished with separate mission hardware.

E. The stability and control (S&C) subsystem must maintain dynamic stability under a wide variety of conditions. The mass properties will vary radically during the various phases of SCB construction and, in fact, will vary significantly (and relatively quickly) as a function of time as large masses are moved relative to one another. Stability must be maintained with and without the Orbiter docked. Many of the structures associated with the SCB will be flexible with low resonant frequencies, and the flexible structure characteristics will vary significantly during the SCB mission and, as with the mass properties, sometimes rapidly as a function of time. Much study is required in this area of control of the SCB.

F. The SCB must provide orbit-keeping capability. This capability is impacted by the various orientations that may be required during fabrication and assembly in that aerodynamic drag and optimum thruster location is a function of the orientation.

The GC&N subsystem described below (Figure 2) is based on the JSC Phase B design presented in Section 2.1.1.1. It is a preliminary design and will be modified as needed as the requirements are better defined. The horizon sensor has been gimballed to facilitate a variety of earth relative orientations. This is desirable since the principal axes of inertia can be misaligned tens of degrees from the vehicle geometrical axes and minimizing gravity gradient torques requires orienting with respect to the principal axes of inertia. Since the SCB mass properties vary significantly during th SCB mission, a fixed horizon sensor would be of limited value.

A manual sextant-telescope (also in the Phase B design) was included, and the Orbiter crew optical alignment sight (COAS) may be an equally acceptable instrument. There is no clear requirement for this instrument and it may be considered optional. This instrument could be used to do manual landmark tracking, star fixes, and free-flying satellite visual tracking. The translation and rotation hand controllers provide manual control of the SCB, and the controls and displays are the crew/GC&N subsystem interface with respect to mode control and GC&N status.

The two gimballed star trackers have been retained and star fixes will normally be possible without maneuvering the SCB to avoid the earth, moon, or sun or to find a desirable star pair. The star trackers will be used to update the IMU (inertial measurement unit) characteristics and to determine an accurate inertial attitude fix. The IMU is strapdown and internally redundant. A typical mode of operation would be the IMU and horizon sensor "gyro compassing" for attitude reference with occasional star fix updating of IMU drift. Ground tracking updates supplement the onboard position and velocity calculations.

Control moments and forces will be provided by RCS thrusters and control moment gyros (CMG's). The placement of the RCS thrusters was not determined for this study and will require extensive analysis because of the large center of mass variations. Firing thrusters in pairs forming pure couples (moment with zero net lateral force) may be desirable since the controlling moments are independent of center of mass location for a couple. Thruster location for efficient orbit keeping will also require study involving desired vehicle attitudes and center of mass variability.

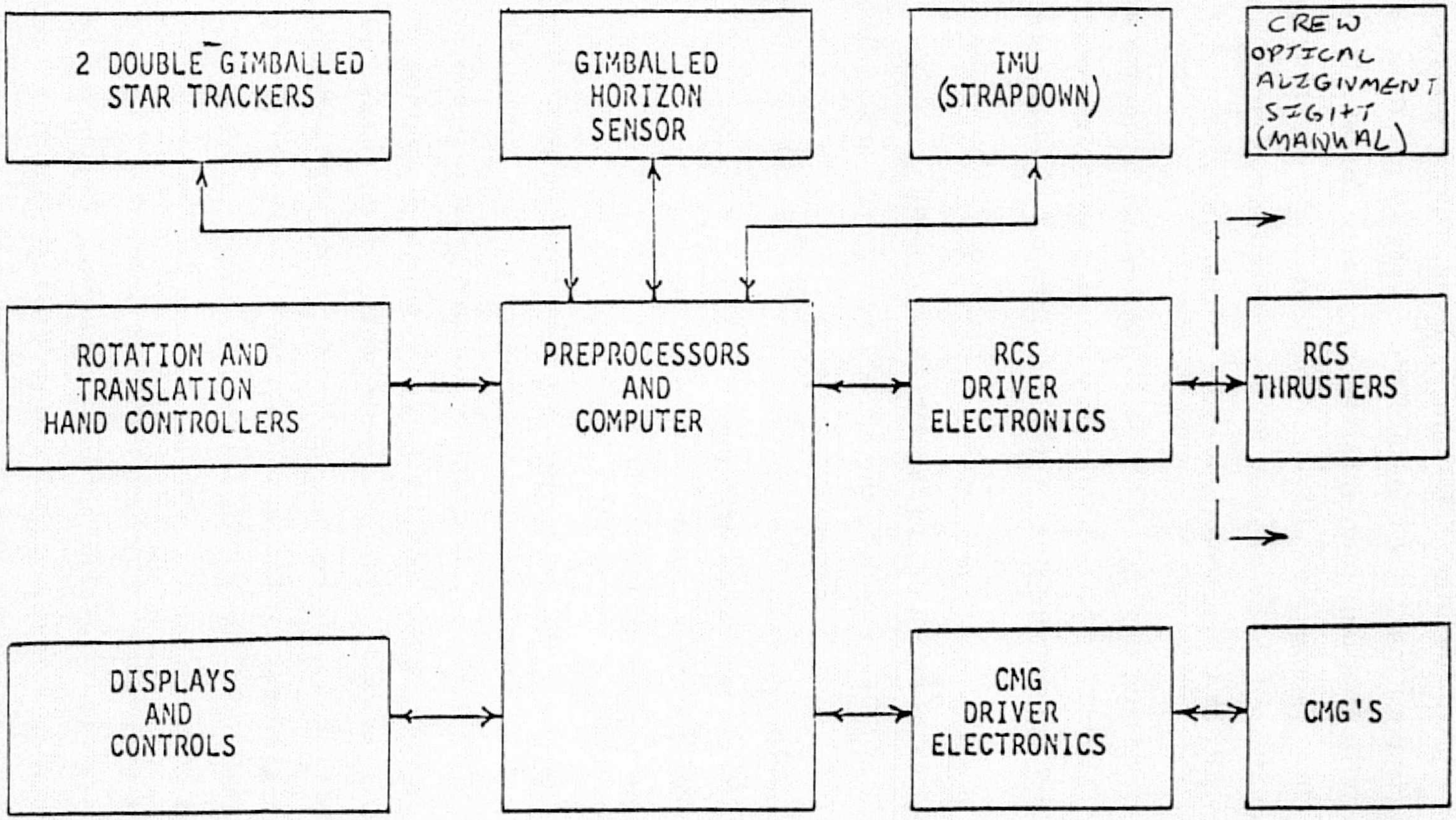


Figure 2. Phase B Derived SCB GC&N Subsystem

CMG sizing was not attempted because the requirements with respect to vehicle orientation were not defined. Volume 3, Book 2 documents an orientation study which indicated that, for non-minimum gravity gradient moment orientations, the gravity gradient moment gets very large for small attitude deviations from the zero moment condition. The feasibility of CMG control may have to be re-evaluated if the desired SCB orientations during fabrication and assembly result in large gravity-gradient torques.

The GC&N subsystem design described above is conceptual in nature and was defined based on the general requirements mentioned. As the requirements are refined, the GC&N subsystem design will be molded to fit those requirements.

2.1.2 Impact of Growth to Option L 14- and 21-Man Configurations

The primary impacts of SCB growth to the GC&N subsystem are in the areas of:

1. Mass properties.
2. Flexible structure characteristics.
3. Aerodynamic forces and moments.
4. Optical sensor field of view (FOV).

Gravity-gradient torque is proportional to the differences between the principal moments of inertia. Increasing the vehicle mass does not necessarily mean increasing the potential gravity gradient torques, but depends on where the mass is added. Adding mass to the configuration which extends its length tends to increase the gravity gradient moments while adding mass near the center of mass tends to makelthe configuration more compact and may even decrease the gravity-gradient torques. In light of the potential (depending on the orientation) for large gravity-gradient torques, the primary GC&N subsystem sensitivity is with respect to differential principal moments of inertia. "Long-thin" configurations may be restricted from some orientation relative to the earth where large gravity-gradient torques esist. Depending on other orientation requirements, a severe penalty in RCS propellant mass and/or CMG masses may result for the growth SCB's. Relocation of the RCS thrusters is a possibility but may not be a big impact.

The last three growth impacts mentioned above are not expected to be as severe as the potential gravity-gradient torque impact. Modifications to the GC&N subsystem software may be required to compensate for flexible structure effects and optical sensors may have to be added or moved. Compact configurations tend to result in more FOV obstruction because of the close proximity of the structure to the sensor. The aerodynamic forces and moments are most sensitive total solar panel area since the solar panels make up the majority of the wind exposed area of the SCB. The orientation study assumed an SCB configuration with $1160m^2$ of solar panel area and showed that the aerodynamic forces and moments were not excessive for the 1984-1985 time period. An increase in solar panel area of a factor of 2 or even 5 would not result in RCS propellant requirements that are prohibitive; on the order of 1000 kg/30 days based on the orientation study results documented in Volume 3, Book 2.

2.2 SCB PROGRAM OPTION L GC&N SUBSYSTEM

The Orbiter-tended L' options represent SCB configurations which are supported to varying degrees by Orbiter subsystems. Groundrules for the L' options include a maximum docked support duration of 30 days with a maximum SCB free flight (unmanned) duration of 90 days. The 10 L' configurations represent maximum Orbiter subsystem dependence (L'-1) through minimum Orbiter subsystems dependence L'-10 (with the L'-10 configuration having the capability of direct growth to the 7-man permanently manned SCB. A typical configuration (L'-5) is shown in Figure 3.

2.2.1 L' Options GC&N Subsystem Definition

The basic GC&N function for the SCB is similar to the Orbiter on-orbit GC&N function/capability. Therefore, it is conceptually possible to use the Orbiter GC&NS while in the docked mode with no SCB contribution. Possible problems which have to be evaluated incude Orbiter thrust impingement on the SCB, control torque loads at the docking interface, efficiency of using Orbiter RCS/VCS thrusters and dynamic stability of the docked configuration, including control axes cross-coupling effects. The cross-coupling effect can be significant since the docked configuration center of mass and principal moment-of-inertia axes may not coincide with the Orbiter control force and moment axes. Consider configuration L'-5 (Figure 3), for example, the 30m radiometer would be built on one of the two ends of the strongback. For this condition, the configuration center of mass is above and offset laterally

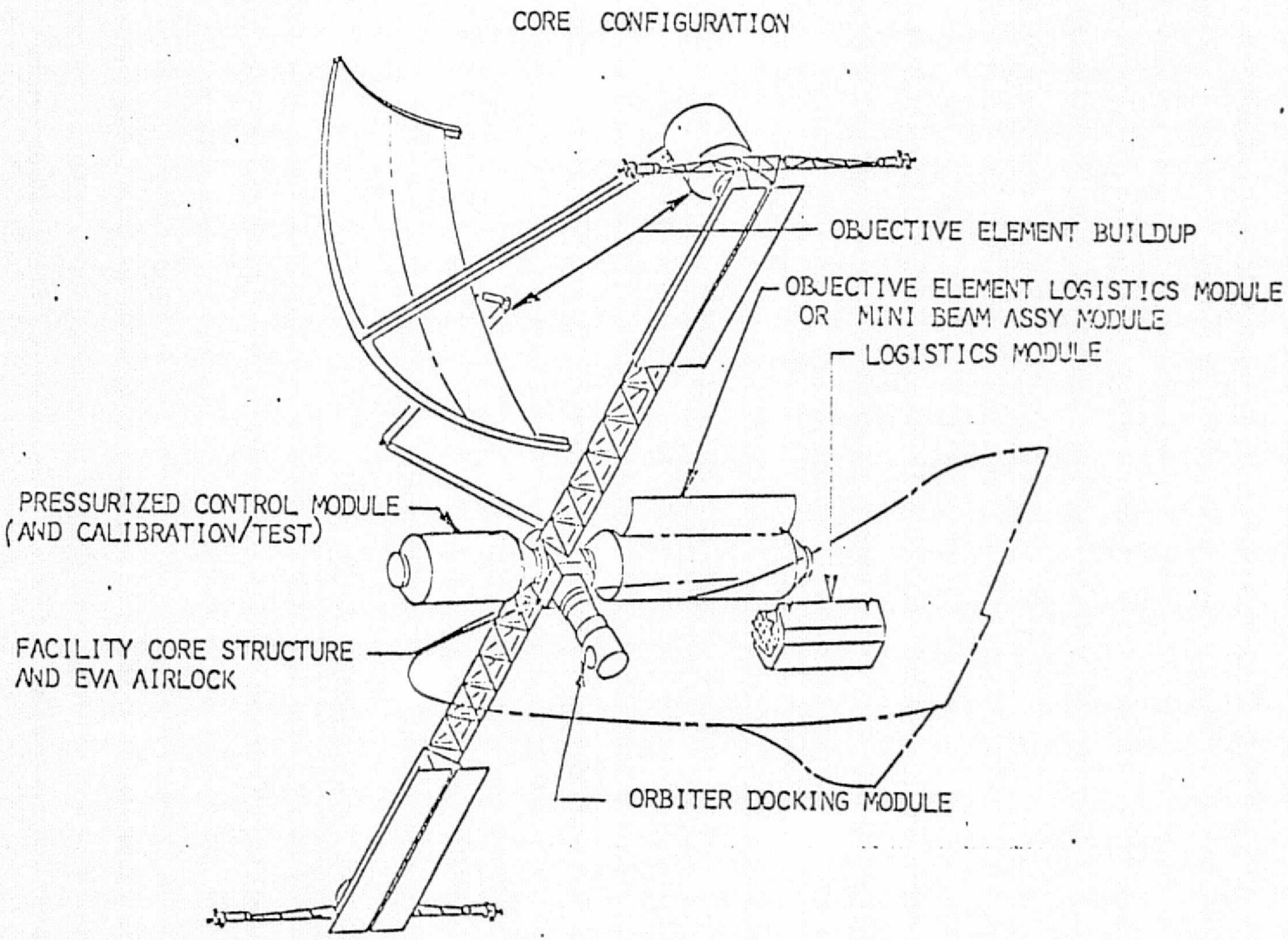


Figure 3. Shuttle-Tended SCB Configuration – Options L′-5 and L′-6

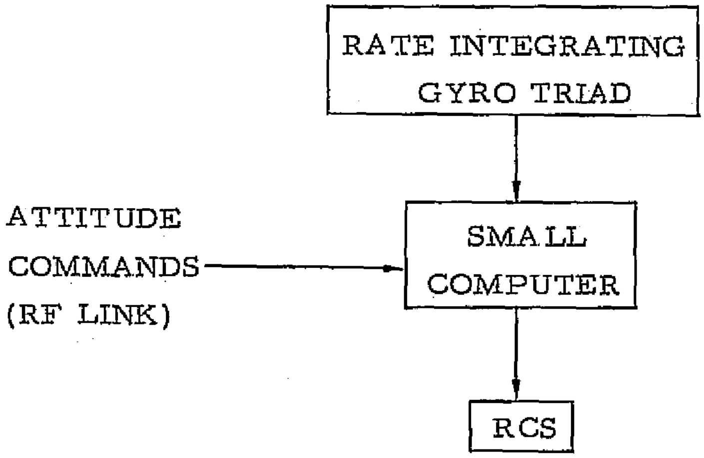
relative to the Orbiter and the configuration principal moment-of-inertia axes are nearly 45 deg offset from the Orbiter's principal axes. At a minimum, the Orbiter software would have to be modified to handle the example configuration.

The L' option SCB will be left unmanned and free-flying for up to 90 days. During this time, some minimal GC&NS would be required. The SCB must be able to orient and hold an attitude during docking and the position and velocity of the SCB must be known in order to rendezvous with it. The position and velocity can be determined from ground tracking data and updated when the Orbiter rendezvous radar locks onto the SCB. The SCB attitude hold capability is possible with a simple redundant gyro triad and RCS thrusters. The SCB reorientation to facilitate docking can be accomplished by including a horizon sensor on the SCB or by RF link from the Orbiter using visual cues. It may be possible to let the Option L' configurations drift (i.e., no control) during the free-flying 90 day-period for the undocking and docking sequences. A rate damping mode may be required, however, to limit angular rate which could result from outgassing, also dynamic and gravity-gradient disturbances. No orbit keeping will be required during the 90-day period for the 1985-1986 time frame when atmospheric density is predicted to be at a minimum.

The minimum SCB and maximum Orbiter dependence GC&NS Option L' concept is, therefore, total guidance, control, and navigation by the Orbiter GC&NS (and the Orbiter RCS/VCS) when in the docked configuration and with a free-flying SCB GC&NS as defined by the block diagram in Figure 4. The maximum SCB and minimum Orbiter dependence GC&NS concept is the SCB GC&NS described for the 7-man permanently manned configuration where the SCB GC&NS has control in the docked and undocked modes.

2.2.2 Growth Impact on the Option L' GC&N Subsystem

The effect of growth on the L' configurations will be to upgrade the L' SCB GC&N subsystem ultimately to the Option L System described in Section 2.1. Changeover from Orbiter control of the docked configuration to SCB control will be a major impact. The gyro triad wil have to be upgraded to a "navigation quality" instrument and the GC&NS computation load for the computer will have to increase. Also, optical sensors such as horizons and star



Orbiter GC&NS and RCS/UCS Used for Docked Configuration (Maximum Orbiter Dependence)

Figure 4. Option L' Free-Flying SCB GC&N Subsystem

trackers should be added. The inclusion of hand controllers for manual control of translation and rotation and a display and control package will be desirable for the growth configurations. Some or all of these upgrades could be used on the minimum SCB GC&NS shown in Figure 4 if cost tradeoffs with respect to predicted growth dictate.

Part 18

SYSTEM AND DESIGN TRADEOFFS

SYSTEM AND DESIGN TRADEOFFS

Throughout the Part 2 study phase, many key tradeoffs were performed. The results of those tradeoffs are discussed in the technical volume and the technical appendixes. A summary of the 15 key tradeoffs is given in Table 1. Each tradeoff is discussed in the following paragraphs.

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Table 1

SYSTEM AND DESIGN TRADEOFF SUMMARY

Section Location	Critical Issues	Trade Studies	Conclusions
1.1	Initial size and growth concept	Modular (small diameter) vs orbital assembly (large diameter)	Common design, Shuttle-compatible, modular concept.
1.2	Initial size and growth concept	Sensitivity of design to increasing crew size and shift operations	Size habitation and crew support modules for seven men and add more modules as needed.
1.3	Number of crew in habitation module	New concept vs modification	Sizing habitation module for 7 men provides more flexibility than 4-man module.
1.4	Low-cost module design	Alternate structural approaches	Isogrid design preferred over a viable monocoque configuration.
1.5	Handling of large structural elements	EVA (manual) vs automated	Preferred approach is combination of operator-commanded crane operations with assistance of EVA crewmen at both ends of transfer.
1.6	Machinery environmental, crew support, maintenance	• Pressurized, pressurizable vs unpressurized areas • Man-tended vs fully manned	Subsystems status displayed at control console. EVA acceptable method for external subsystem maintenance. All pressurizable modules remain pressurized except for emergencies.
1.7	Machinery selection	• Manual vs automated • Continuous flow vs assembled structure	Automatic whenever possible since crew time in orbit is expensive.
1.8	Orbiter docking location and module berthing	Three concepts for docking/berthing	Orbiter docking along SCB X-axis and module berthing to SCB ports using SCB crane or Orbiter RMS provides the most flexibility.
1.9	Orientation of space base	Configuration vs orientation vs solar (β) angle	Principal inertia axis orientation is preferred for long-term orientation. Low β-angles will drive subsystem sizing and resources.
1.10	Buildup of jigs and fixtures	Ground fabrication and assembly with orbital fabrication and assembly	Requires individual tradeoffs for comparing fabrication and transportation costs.
1.11	LEO vs GEO construction	Program cost comparison	LEO construction and transfer to GEO is preferred.
1.12	Low-g environment for space processing	Location vs isolation of processing	Attachment of the space-processing modules to the SCB is simplest.
1.13	Antenna construction concepts	Ground fabrication vs orbital fabrication and assembly	Ground fabrication followed by orgital assembly and test is most straightforward.
1.14	OTV performance optimization	Number of stages, propellant, staging approach	Best solution is two-stage, LH_2/LO_2 propellant, and common-stage design.
1.15	OTV propellant operations	Shuttle tanker vs depot support	Tanker mode is cheaper and simpler.

1.1 INITIAL SIZE AND GROWTH CONCEPT

The pressurized volumetric requirements were established by the analysis of the various objective elements and crew requirements associated with program Option L. A schedule for Option L was developed using study-derived criteria and constraints, and the time-sensitive requirements, crew size, and power were timelined. An initial pressurized volume requirement of $180m^3$ increased to a maximum of $1250m^3$ during the development and test phase and reducing to approximately $950m^3$ for the commercial production phase of operations. Also, an evaluation of crew requirements indicates a total pressurized volume of $720m^3$ for a 7-man crew to a total of $1900m^3$ for a 21-man crew. Each of these volumetric requirements are time-sensitive. Each of the volumetric requirements can be satisfied by two basic system options: a large-diameter space structure or a series of small diameter modules.

Evaluation of all objective elements did not reveal the requirement for a large volume facility, except for the OTV maintenance facility. Further investigations indicated that a maintenance approach to the OTV involving EVA operation will eliminate the large single volumetric requirement. If further study indicates a need for a hangar-type facility, the large structure concepts will be reinvestigated. The modular concept permits the SCB to take many forms and can be configured and reconfigured to house from 7 crewmen up to 21 and more. Also, it provides each objective element with flexibility in providing dedicated facilities with efficient capability of ground changeover as R&D programs proceed. Modules configured to perform various system functions can be introduced into the overall program within budget and schedule restrictions essentially on a noninterference basis.

Based on the time-phasing schedule requirements defined, the common design, Shuttle-compatible, modular concept was selected as baseline for all pressurized volume requirements.

1.2 INITIAL SIZE AND GROWTH CONCEPT

Initial SCB crew size is dictated by the nature and amount of crew activities required in the initial station and by the work shift arrangement adopted for the initial station. These trades assume an initial concentration on construction activities, using a 3-man construction crew, and that a two-shift arrangement would be adopted. Thus the initial crew would be 7, consisting of 6 construction workers and a station commander.

With two shifts, it is advisable to isolate sleeping areas from eating and recreation areas, thus separate modules for sleeping and food management and for recreation are indicated. Each module is initially sized to accommodate the entire crew of 7. All 7 crewmen can sleep simultaneously in the Habitation Module and all 7 crewmen can eat simultaneously in the Crew Support Module.

Growth concept options considered were (1) add accommodations for additional crewmen as needed by rearranging facilities in the Habitation and Crew Support Modules, (2) add additional, differently configured, modules for increments of additional crew, configuration to be determined by the number of additional crewmen to be accommmodated, and (3) add identical Habitation and Crew Support modules to accommodate crew sizes in multiples of the initial crew complement (i. e., 14, 21, 28, etc.).

Growth concept option 3 was adopted as hving the least design impact because the added modules would be identical in design to the initial modules. In this growth concept, the crew size increase from 7 to 14 can be accommodated by the addition of an additional Habitation Module. (A Spartan approach would permit growth to 14 crewmen without adding the second Habitation Module if it is assumed that sleeping accommodations can be shared, i. e., "hot bunking.") Food management and recreational activities for the larger size crew can be accommodated by using shift arrangements in the initial Crew Support module. Crew size increases to 21, and then to 28, can be accommodated by adding one Crew Support Module and two additional Habitation Modules.

1.3 NUMBER OF CREW IN HABITATION MODULE

The NASA JSC baseline provided crew modules, each accommodating 3 crewmen for sleeping and personal hygiene. The initial 6-man station thus had two crew modules and the 12-man growth station had four crew modules. The present trades evaluated one module vs two modules for the initial SCB configuration, with an initial crew complement of 7. In the one module option the crew module would accommodate all 7 crew members, while in the two module option 3 of the crew would sleep in one module and 4 in the other.

Primary criteria used in the evaluation were (1) compatibility with growth concepts, (2) standardization of modules, (3) compatibility with different work shift arrangements, and (4) crew safety.

The single module option was selected as providing a standardized Habitation Module which would best accommodate the growth concepts previously selected (i.e., growth in multiples of the initial crew complement). This option does not dictate a particular shift arrangement but is compatible with either a one-shift or 2-shift or even a 3-shift arrangement. Compared with the JSC baseline, this module is 3.4m (11.3 ft) longer and has $53m^3$ more total volume.

1.4 LOW-COST MODULE DESIGN

The cost of the module structure is a small, but nevertheless significant, portion of the total module cost which includes engineering, materials, manufacturing, assembly, integration, testing, and changes. The engineering and system costs required to provide the part traceability and accountability records essential for system safety and reliability, can be minimized by minimizing the parts count through the use of large integrally machined sections. The materials costs can be minimized by using monocoque skins which require the addition of a considerable number of separately identificable parts to duplicate the integrally machined provisions. The results of a study to determine which of these two approaches is lower in cost indicated that the cost difference between integrally machined isogrid and monocoque cylinder configurations is too small to be used as the criterion for choosing between them. The cost savings provided by the low parts count with the inetgrally machined cylinder is balanced by the increased materials and manufacturing costs. Additional criteria must be reviewed to determine the superior approach.

The isogrid design provides a weight savings of about 1500 lb and eliminates huckbolt penetrations of the pressure shell and the attendant potential leak source. The monocoque skins provide improved radiation and meteoroid shielding. The isogrid cylinder is preferred, based on MDAC manufacturing experience on Saturn and Delta, coupled with unique in-house design and analysis capability. Others without this background of experience might prefer the monocoque configuration. Both appear to present highly viable low-cost approaches for the design of a Space Station module.

1.5 HANDLING OF LARGE STRUCTURAL ELEMENTS

The trade studies in manned vs automated methods of materials handling restricted the definition of materials handling to the process of moving and positioning SCB construction materials (large structural elements and large quantities of elements or components) from one place to another in orbital space, excluding consideration of assembly operations on those materials. The options considered were (1) fully automated with monitoring from a remote control station, (2) crane operations with operator in a pressurized compartment, (3) crane operations with manual assistance by EVA crewmen at the terminal end of the transfer or at both the initial and terminal end of the transfer, and (4) completely manual transfer. Criteria used to assess the alternatives included (1) cost and design complexity, (2) flexibility, (3) efficiency, and (4) crew safety. For handling of large structural elements it was concluded that Option 3, crane operations with manual assistance by EVA crewmen (at the terminal end of the transfer), provides the least cost, most efficient, and most flexible alternative. For large quantities of elements or components (which might include everything from antenna panels to small parts), it was concluded that Option 3 should again be the method of choice with the proviso that small elements or components would be packaged so that a large number of them could be handled as a unit.

1.6 MACHINERY ENVIRONMENTAL, CREW SUPPORT, MAINTENANCE

No specific trades were conducted in this area, but preliminary evaluations resulted in the following conclusions:

- Subsystems equipment area do not have to be manned for purposes of monitoring performance since the status of all subsystems will be continuously monitored and outputs integrated into the data management system for display at a control console.
- Since EVA is an acceptable method of maintenance for SCB, that machinery (e.g., RCS thruster) which is normally exposed to the space environment can be maintained by crewmen in the EVA mode.
- All areas of the SCB, including the Power Module, which shirt-sleeved crewmen will occupy, will be pressurized and remain pressurized except for emergencies.
- The only SCB areas which will normally be unpressurized at some times and pressurized at other times are the airlocks.

1.7 MACHINERY SELECTION

Since the cost of an orbital construction worker is on the order of $10,000 per hour, his use will be justified when the anticipated production is not sufficient to amortize the cost of developing fully automated assembly equipment. The degree of automation that will be practical is also a function of the configuration of the space product. A product configuration which permits use of fabrication processes that can be simply automated, such as the pultrusion for plastics and composites or the roll-forming of ductile metals, is clearly a candidate for fully automated production because the automation of these processes is already well developed. These processes are naturally suited for continuous flow production. The large radiometer and the multibeam lens are examples of configurations that are not amenable to space fabrication because of the degree of manual labor involved in current fabrication techniques for the composite antenna faces and the difficulty of automation. They are best suited for ground fabrication with manual space assembly. The SPS solar array is an example suitable for fully automated production.

1.8 ORBITER DOCKING LOCATION AND MODULE BERTHING

The selection of the concept for the delivery and controlled mating of the various SCB modules has an impact on the SCB configuration and design. Three basic alternatives exist: direct docking of modules to any port, Orbiter direct docking along the SCB X-axis followed by module berthing, and complete berthing operations using RMS and/or crane. In the concept with direct docking to any port, the Orbiter RMS erects a payload module to its docking module and the Orbiter drives the module into the available SCB docking port. In the case with direct Orbiter docking along the X-axis of the SCB, the Orbiter docks itself, and then the RMS and/or the SCB crane removes the module from the Orbiter and berths it into an available port. In the complete berthing mode, the Orbiter RMS is used to berth the SCB core to the Orbiter docking module. Following verification of berthed/docked interfaces, the Orbiter RMS and/or the SCB crane performs the module removal/mating operations.

Orbiter attitude stability and orbiter position accuracy error sources established a recommended module separation of five feet while berthed. The 5-ft module separation appears to be adequate if all modules being docked are of equal length. Direct docking of a shorter module, such as the logistics module, would not be possible with a full-length module adjacent. X-axis direct docking and module berthing using RMS and/or mobile crane minimizes the interference problem and permits module spacing to remain at 5 ft. As a result, all ports on the core module can be used to accommodate modules varying in length. Therefore, the X-axis direct docking of the Orbiter with each payload module berthed using the RMS and/or the SCB mobile crane will be retained as the baseline mode. The exception will be in emergency situations, where the Orbiter will dock to any of the berthed modules.

1.9 ORIENTATION OF SPACE BASE

The orientation of the space base affects several of its subsystems with regard to sizing and support resources. The major subsystems considered in the tradeoff were Stabilization and Control, Reaction Control, and Power. The configurations studied varied from a bare SCB to one with an Orbiter attached at one end and a 30m radiometer at the other. The high-gravity gradient/centripetal moments associated with the larger configurations indicate a strong preference for an orientation in which the principal inertia axes are oriented parallel and orthogonal to the center of the earth. This requires a rotation of vehicle attitude as much as 28 deg from a geometric axis alignment. Once the principal axis orientation is satisfied, the order of the principal axes relative to the orbit plane is relatively insensitive from the standpoint of Reaction Control and Power. The effect of solar angle (β = angle between sun vector and orbit plane) favors the higher β angles for both Reaction Control and Power subsystems. However, low β angles cannot be avoided, and will be the driving design cases.

1.10 BUILDUP OF JIGS AND FIXTURES

Space fabrication of the various jig and fixtures, as opposed to transport of the finished parts required, is justified if the transportation costs saved by the shipping of bulk materials (rather than finished parts) to orbit is greater than the increase in fabrication costs (in orbit over ground fabrication). Otherwise, it is less expensive to fabricate the fixtures on the ground and simply transport the finished parts to orbit. Space fabrication should therefore involve automation of the fabrication process to minimize the fabrication manhours. Since jigs and fixtures represent a very small volume of production, the automated equipment required for fabrication of their components should be required also in other objective element applications to present a practical alternative. A collapsible fixture may prove desireable where an examination of launch sequences and facility-phased requirements indicates that the benefit from the cargo bay volume gained outweighs the reduced cost of a rigid truss fixture. However, the primary element of the strongback selected for an orbital facility was a simple truss beam hinge folded near its center to fit in the bay; little benefit could be found for the added volume available with a collapsible truss for this particular application.

1.11 LEO VS GEO CONSTRUCTION

The effects of space construction at LEO or GEO were determined for four program options having operations at both locations. Four objective elements were of prime concern: space power system TA-3, Mark II radiotelescope, the the 27m multibeam lens, and the cross-phased array for personal communications. TA-3, being the largest system, was the dominant issue.

Seven major factors that influence the program cost of construction at LEO or at GEO were evaluated. Thse included the following:

1. Number of SCB elements needed.
2. Transportation requirements.
3. Orbit transfer techniques.
4. Orbit keeping.
5. Orbital forces and moments.
6. Plasma interactions.
7. Radiation.

The transportation requirement differences accounted for the largest cost influence. GEO construction required extensive crew operations at GEO and their needed support in terms of OTV flights, logistics, etc, accounted for a $2.6B cost increment for GEO construction. This increase includes the savings incurred by employing a growth Shuttle to reduce the number of total flights and the use of a high Isp electrical system to transfer TA-3 (295, 000kg) from LEO to GEO.

LEO construction was recommended for the program options analyzed primarily because of the cost difference. Other factors that also influenced the selction were that GEO constructions would require more SCB modules and would have to deal with a more severe radiation environment.

1.12 LOW-g ENVIRONMENT FOR SPACE PROCESSING

Space processing places demands on the SCB to maintain the microgravity environment experienced inside the processing module during critical periods of operation. The microgravity requirements, for example for processing shaped crystals, are 10^{-3}g for periods of 30 days. These requirements in this case stem from the need to levitate the melted material to avoid contamination from the furnace container and to prevent the growing crystal from forming discontinuities caused by gravity-driven forces such as convection. Preliminary tradeoff considerations of alternate methods to meet these requirements have included the following: (1) attachment of the space-processing modules to the SCB c.g., (2) free-flying space processing modules, (3) flying the SCB "around" the critical space processing apparatus, and (4) include special equipment within the space-processing module to negate the effects of SCB motion on space-processing operations. From the information at hand the study concluded that the first method is preferred for the following reasons: (1) lacking specific space-processing operating regimes the first methods appear to be the most straightforward approach, (2) initial R&D in space processing on Spacelab will use the first method, and (3) Skylab experience suggests that the first method is adequate.

1.13 ANTENNA CONSTRUCTION CONCEPTS

Analysis of material requirements for minimum deformation under adverse thermal conditions resulted in a recommendation for GY 70 graphite fiber prepreg as the facing material for the antenna segments. A fiberglass honeycomb dialectric core material was chosen on the basis of cost and availability. Epoxy adhesive and moldings would be used, and assembly segments would be cured in a heated platen press to 350°F at 100 psi. Finishing would be done by machining and inspection would be done by dimensional, and radiographic or ultrasonic techniques.

Due to the availability of people, equipment, and facilities on earth, fabrication in orbit cannot be justified. Volume production, which does not appear to be a valid hypothesis concerning antennas, is needed before on-orbit fabrication might become a competitive alternative. Assembly and test operations were found to be suitable for performance on orbit.

1.14 OTV PEFORMANCE OPTIMIZATION

The requirements for LEO to GEO transfer were determined for those program options containing GEO operations. The capabilities of various OTV types to meet them were compared and a reusable two-stage (common stage design) was selected. A reusable system was selected for low operating cost. Two stages were selected to further reduce operating cost (fewer propellant logistics flights) and to allow complete stage delivery in the Shuttle bay, and to allow more mission flexibility. The common stage design was used to reduce design costs and to maximize performance (Shuttle bay length limited). The stage sizes needed to satisfy the program options varied from 40, 000 to 55, 000 kg propellant (LH_2/LO_2) per stage depending upon the option selected. Since the upper level was near the maximum length that could be transported in the Shuttle bay, the maximum was used for concept design.

1.15 OTV PROPELLANT OPERATIONS

The OTV selected was space-based to achieve maximum performance and lowest cost. The flight schedule needed to satisfy the program options resulted in a system supplied by a Shuttle tanker or a growth Shuttle tanker. The mission operations of rendezvous, fueling, assembly, etc., were considered and found acceptable using the tanker mode only. Therefore, the added cost and complexity of using an orbital depot as a propellant storage facility was not felt warranted.