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Major contributions to this report were made by the following persons:

Name	Area of Contribution
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SUMMARY

NASA-Langley initiated this "Study of Space Shuttle Environmental Control and Life Support Problems" to specifically investigate four significant problem areas that require special attention in parallel with the NASA Phase B study efforts. These four tasks are: (1) Cargo Module EC/LSS Definition; (2) Space Shuttle/Space Station Interfaces; (3) Shuttle/Payload Thermal Control and (4) System Reusability. The primary reasons for their selection are discussed in the following subsections.

Tasks Definition

Cargo Module EC/LSS Definition.- During the course of the Shuttle vehicle studies major emphasis was placed on definition and description of basic orbiter and booster configurations. Payload emphasis was limited to broad definition of the various payloads and the effects of payload volume and gross weight on the booster/orbiter design configurations and aerodynamic performance. However, the wide variety of manned and unmanned missions established for the overall Shuttle program require environmental control and life support in varying degrees. The magnitude of this support, the design approaches and effects on the basic vehicle were all areas in need of more comprehensive examination.

Space Shuttle/Space Station Interfaces.- Shuttle and Station studies were conducted independently of each other. Since one of the primary objectives of the Shuttle is to provide logistics support and personnel transfer between these vehicles in a docked operational mode and since each vehicle has an environmental control and life support system (EC/LSS), it was apparent that a more detailed examination of the interaction between the systems was needed.

Shuttle/Payload Thermal Control.- In addition to the EC/LSS definition needed for cargo modules, it was also recognized that special thermal control considerations for the payloads could have significant influence on the orbiter. The payloads can be classified into the following four categories: (1) those which are manned and require active control for personnel comfort; (2) those which must be kept warm to maintain a minimum temperature level; (3) those which are cryogenic and require high thermal isolation, and (4) those which are not thermally sensitive. This span of payload requirements and the large number of payloads emphasized the need for more detailed examination of the design needed to provide adequate support.

System Reusability.- The Shuttle vehicles will be subjected to multiple launches and re-entries over a ten year period. Turnaround time from recovery to launch is less than two weeks. It is important, therefore, to have maintenance concepts for servicing and repair of onboard equipment which minimizes delays caused by extensive fault isolation or preflight testing. The feasibility of improving system maintenance characteristics and long term life through added redundancy is another aspect which could influence system design. These areas, therefore, were identified as desirable for further examination.

Study Results

The significant results of this study are summarized below:

Cargo Module EC/LSS Definition.-

- o The cargo module EC/LSS must be sized for various passenger loads. This is based on supporting the 6-man and 12-man modular Space Station, and upon a wide variety of manned and unmanned missions. A two-man, six-man, and a ten-man cargo module EC/LSS will satisfy the support requirements. The ten-man unit meets the 12 passenger transfer requirement since two of the passengers can be housed in the Shuttle forward crew compartment, and are supported by the Shuttle EC/LSS. This clarification and division between the Shuttle's forward crew compartment and the cargo module's EC/LS systems is considered an important aspect in the cargo module definition, and all subsequent analyses, conclusions, and recommendations are based on this decision.

- o Comparisons made between the basic Shuttle orbiter four man EC/LS system and alternate systems customized to accommodate various passenger loads in the cargo module conclude that modular units of the basic system should be used for the cargo module on all manned missions. This conclusion is supported by the fact that Design, Development, Test and Evaluation (DDT & E) costs for customized systems are too high. These costs override the transportation to orbit costs penalty imposed by non-weight optimized systems. Multiple units of the basic EC/LSS have sufficient flexibility to meet the requirements of 20-ten man, 24-six man and 49-two man flights and are readily extendable to 30 days with additional expendables.

- o Crew size does not alter the basic decision to use non-regenerative EC/LS systems. This choice is primarily determined from mission duration and type of power source. Since the 6 and 10 man flights are Station personnel transfer flights and the actual passenger time spent in the Shuttle is approximately two days, it is concluded that the shorter duration stay time further substantiates usage of non-regenerative systems.

- o Missions of 30-days duration are too few to alter the decision to use the basic 7-day EC/LSS with additional expendables for extended capability.

- o Because of the need for autonomous payload operations either in a free-flying mode or with the payload module attached to the Station, payload module EC/LS system should be designed to operate independently of the Shuttle. This conclusion is of particular importance for the thermal control system.

- Only manned missions require EC/LSS provisioning. All other payloads require either thermal control only or no protection at all.

- Flight planning is such that 2-man flights are continuous throughout the flight program. Six-man flights occur until the Space Station acquires 12-man capability; thereafter, 10-man flights replace the 6-man flights. To accommodate this pattern, a manned module sized for the 2-man flights can be dedicated for this mission. A second module sized for the initial 6-man flights and later converted to accommodate 10-man flights can be dedicated for this purpose. Each module uses singular or multiple 4-man basic EC/LS systems because of the cost effectiveness previously discussed. This approach negates continuous removal and replacement of EC/LSS packages in the payload modules to convert from one mission to another.

Space Shuttle/Space Station Interfaces.-

- It is concluded that during the docked mode, Station and Shuttle personnel should rely on the Station for EC/LS. This recommendation is based on the fact that the Station has regenerative systems, greater volume than the Shuttle for habitability and is designed to accommodate a crew overload condition.

- The Shuttle EC/LSS can be placed in a quiescent operating mode subsequent to personnel transfer to the Station. This mode allows for rapid escape capability by eliminating extensive reactivation or warmup. Since thermal lag is considerable, this system remains operative at a reduced level throughout the docked period. All other EC/LSS elements can be readily activated on demand and can, therefore, be shut off during normal docked operations.

- To allow for brief intra-vehicular movement which may be desirable for cargo transfer, system checkout or for escape, the hatches between the vehicles should be open when the vehicles are docked. Total pressure control for both vehicles is provided by the Station. If personnel enter the Shuttle during the interim for any moderate amount of time, the Shuttle EC/LSS is activated.

- During the docked mode, EC/LSS interconnects between the Space Station and Shuttle are not required. Status monitoring and electrical power supply by umbilical connection should be provided. It may be desirable to transfer Shuttle fuel cell product water to the Station during the docked mode.

- The analysis of the Station feeding capability shows it to be inadequate to handle the 24 crew overload condition. A solution to the problem is to use the Shuttle cargo module feeding facilities to augment the Station. The disadvantages to this approach are that the module is not as habitable as the Station, that activation of the EC/LSS is required, and that personnel traffic between vehicles is increased. It is recommended, therefore, that increased feeding capability be provided in the Station.

Shuttle/Payload Thermal Control.-

- Shuttle payloads can be divided into three categories with respect to thermal control requirements; (1) an active system capable of maintaining temperatures within a human comfort range for manned missions; (2) a system which provides heat for satellite servicing, maintenance, placement and retrieval missions; and (3) a system which provides protection from excessive heat leak and condensation for propellant payloads. No thermal control is required for propulsion stages and quiescent satellites which have been designed for the environmental extremes.

- Approximately 150 ft^2 of radiator surface is required for manned missions. The requirement for operation both independent of the Shuttle and while contained within the Shuttle establishes the need for radiator locations on the upper surface of the module. During orbital operation, therefore, the Shuttle cargo bay doors must be opened to allow heat rejection. To minimize radiator area, an optical solar reflective (OSR) finish providing an α/ϵ of 0.05/.8 is recommended. The higher cost of the OSR surface is easily offset by higher performance characteristics and surface stability with respect to ultraviolet degradation.

- The mission payload analysis indicates that there are a total of 244 satellite payloads. Thermal control for approximately half can be satisfied with insulation blankets and relatively small electrical heaters. The remainder can be satisfied with passive thermal control techniques and require no Shuttle support. It should be noted that the thermal control requirements for a large number of satellite payloads have not yet been established. The percentages cited, therefore, are estimates based on limited current information. One of the more significant influences on the payload compartment environmemt is provided by onboard Shuttle cryogenics. Design of the compartment walls, insulating blankets and passive control methods should minimize heat loss from the satellites to the surrounding environment.

- Propellant payloads require no auxiliary thermal protection for orbital operation. During atmospheric operation, condensation and freezing would occur with unprotected tanks. To prevent this from occurring, an insulative bag, enclosing the payload with dry helium is recommended.

System Reusability.-

- Airlines do not utilize extensive fault isolation, status or trend monitoring for subsystems because it is too complex and costly.

- Airlines generally have the philosophy of waiting until parts fail before replacement. This approach eliminates failures from wear out caused by extensive checkout or scheduled maintenance. Airlines predicate this practice on the basis that no single failure can occur which jeopardizes passenger or aircraft safety. To adopt this policy for spacecraft, additional redundancy would be required to provide high initial reliability and safety and to compensate for failed components not discovered by automatic checkout.

- Reliability analyses show that the freon thermal control equipment of the heat transport loop subsystem and the humidity control subsystem have difficulty in meeting the 100 mission lifetime mean time to failure. The analysis concludes that repair, replacement and servicing for these subsystems are more likely than for the remainder of the EC/LSS equipment. It is also concluded that attempted improvements in these areas through redundancy adds critically to system weight while affording little gain in reliability.

- The analysis indicates that it is both practical and desirable to provide instrumentation for dynamic fluid mechanical components (coolant pumps), rotating and non-rotating electromechanical components (motors, relays, switches), electrical devices (resistors, capacitors), and electrochemical items (batteries). It is recommended that direct or indirect instrumentation be applied in accordance with subsystem criticality for each mission phase for these component categories. This analysis identified the O_2/N_2 supply and pressure control and heat transfer loop (freon thermal control) subsystems as the most critical for instrumentation.

- Grouping of EC/LSS components into modules for replacement is not a means of achieving quick turnaround. Modular replacement for maintenance results in a number of good, proven components being removed along with the faulty item. Because there is a high infant mortality rate on newly installed components, the trend is to avoid this kind of modularization. Airline recommendations are to break modules into smaller line replaceable units for ready replacement.

- This study conceptually defined a simple, practical fault isolation system that is a compromise between the two extremes of fully automated computerized checkout and gross status monitoring. A go-, no-go, green, red light display indicates components that do not demonstrate satisfactory operation. A single failure may result in indication of failure of several components (even though the system design tends to eliminate this by integrating sensor signals at the source) which would then be diagnosed by the ground crew during postflight maintenance. Airline experience shows that an experienced ground crew with minimum aids for trouble location can efficiently and rapidly perform repair service, thereby minimizing the need for complex automatic checkout systems.

INTRODUCTION

NASA identified specific problems associated with the design and development of a Space Shuttle orbiter vehicle environmental control/life support system (EC/LSS). This report investigated those problem areas. Results of the investigation are contained in five technical sections of the report covering: (1) mission/vehicle definition, (2) cargo module EC/LSS definition, (3) Space Shuttle/Space Station interfaces for determining the division of the EC/LSS responsibility, (4) Shuttle/Payload thermal control interfaces, and (5) EC/LSS system reusability. A New Technology section lists areas for further investigation.

The initial section (Mission/Vehicle Definition) furnishes a general description of the Space Shuttle mission. The crew and passenger complement associated with each of the missions is identified. The data presented provides the necessary background information to execute the study tasks. A standard traffic model showing number of flights, distribution, schedule, and frequency is established as a guideline. This model is consistent with the NASA schedule.

The succeeding sections of this report covers the four study tasks defined and discussed in the Summary. The cargo module EC/LSS which will support the wide variety of payloads and missions of the Space Shuttle Program is defined. The Shuttle baseline and alternate candidate EC/LSS concepts were selected, defined, and evaluated. The evaluation criteria applied were cost, weight, flight frequency, number of crew, modularity, and performance. Module performance characteristics, and technology advancement requirements are identified.

An investigation was carried out regarding the influence of the Shuttle and Station on each other. Results of docking, crew passenger/cargo transfer, system deactivation, and EVA/IVA were the major areas investigated. A Shuttle and Station interaction analysis was performed to determine division of EC/LSS responsibility. A combined Shuttle/Station EC/LS baseline system and alternate candidates were formulated and studied.

An evaluation was conducted regarding the requirements for the Space Shuttle payload thermal control as related to the Shuttle interfaces from pre-launch to orbit; and/or orbit through landing. Representative payloads were selected which present a cross-section of the thermal control problems which will be encountered during the various mission phases.

An EC/LS system reusability analysis was performed which examined sequentially; (1) all facets of ground test, maintenance, and refurbishment as being practiced by airlines, military services, suppliers, and manufacturers, (2) pre-flight refurbishment and post-flight checkout requirements at the EC/LSS component level, (3) mean-time-to-failure using system redundancy and weight limitations as applied to variable mission periods, (4) fault isolation feasibility, including EC/LS subsystems criticality by mission phases, instrumentation practicality level, and difficulty of implementation, and (5) line replaceable unit requirements. A conceptual fault isolation approach that minimizes inspection, maintenance, and turnaround time is also presented.

The report concludes with two appendices, one contains the data resulting from the detailed operation and mission analysis. The mission objectives and characteristics that have an effect on the Shuttle EC/LSS have been tabulated. Sensor/equipment, environmental protection and Shuttle support requirements are delineated. The second appendix contains a copy of a computer printout performed for the System Effectiveness Program in determining the design life and reliability of the cargo module EC/LSS.

MISSION/VEHICLE DEFINITION

This section contains a general description of the Space Shuttle mission. Emphasis is placed on the salient characteristics that have a direct effect on environmental control and life support. The crew and passenger complement associated with each of the missions is identified. The data presented herein provides the necessary background information to execute the study tasks.

A typical seven day mission flight profile is described which forms the basis for selecting a baseline EC/LSS. A standard traffic model showing number of flights, distribution, schedule, and frequency is established as a guideline. This model is consistent with the NASA master schedule.

The latest Shuttle and Station configurations as established for NASA by the Phase B contractors are described. Payload descriptions and general crew/passenger/cargo arrangements are presented as an input to the EC/LS system analysis. A mission operations analysis is performed that presents an overall mission sequence of events.

Mission Definition

Many potential uses for the Space Shuttle have been identified in terms of overall space program objectives. In this study the missions considered for their effect on the selection of a Shuttle EC/LSS, are the following six: (1) Space Station Resupply, (2) Satellite Placement and Retrieval, (3) Satellite Servicing and Maintenance, (4) Propellant Delivery, (5) Propulsion Stages and Payload Delivery, and (6) Short Duration Orbital Missions. Table 1 summarizes the major characteristics associated with each of these missions.

The primary mission of the Space Shuttle is to transport cargo and/or personnel to and from the Space Station/Base. In addition to cargo and personnel, five other basic missions support the overall space program. The following sections will give a brief summary as to the six missions objectives and requirements.

Space Station Resupply.- Logistic support to the Space Station is provided in the Space Shuttle. In addition to cargo and personnel, the Space Shuttle will be required to place at the Station (both attached and free-flying), Earth orbital experiment modules which would operate in conjunction with a Space Station/Base.

The 1981 Space Station will normally be inserted into a 270 nm, 55 deg inclined orbit and eventually grow into the Space Base. Support of the Space Station/Base in this orbit has been selected as the Space Shuttle design reference mission. Alternate orbits being considered for the Space Station include geo-synchronous and polar orbits. For support of the Space Station in a geo-synchronous orbit, the Space Shuttle will be required to rendezvous in a low earth orbit with a space tug for passenger/cargo transfer and eventual delivery to the Space Station. For polar orbits, Saturn V payload launch capability limits Space Station operations to about 200 nm.

Satellite Placement and Retrieval.- Mission objectives are to place a number of self contained satellites into a variety of orbits up to a maximum altitude of 800 nm for independent operation. Return to earth operation will include retrieval of high cost, high-priority, satellites and wherever pracical, space debris caused by U. S. and foreign "dead" satellites, expended upper stages, transtages, etc. For the missions being considered, payload weights will range between 200 and 20,000 lbs, allowing, in most cases, multiple payload delivery. Because orbital plane changes of more than a few degrees results in excessive propellant usage, multiple payload delivery missions will require satellite groupings by orbital inclination commonalities. Two orbit inclinations of major interest are a due east ETR launch (orbit inclination = 28.5°) and sun synchronous orbits (orbit inclination = 97°).

To retrieve satellites, the Shuttle must be capable of performing rendezvous and docking with passive satellites. In addition, the target satellites will require retrieval mechanisms which are compatible with those of the Space Shuttle. This requirement, it is expected, will be incorporated into future satellite designs.

Normal operations will be to deliver and retrieve satellites by remote controlled mechanical devices with EVA operations performed only as required. An example of EVA operations would be removal of protuberances, such as extendable booms or space erected panels prior to satellite retrieval.

Satellite Servicing and Maintenance.- The purpose of this mission category is to provide service and maintenance to large experiment modules and satellites operating in orbits at altitudes of up to 800 nm and inclinations ranging from 28.5° to sun synchronous. (There is the possibility of orbits at inclinations lower than 28.5° as well). While many of these modules or satellites may be operating in conjunction with a Space Station or Base, others may be in orbits that would be more readily accessible from the ground. These modules or satellites are logical candidates to be serviced and maintained by the Space Shuttle. The Shuttle would have the capability to revisit modules and satellites and bring them into an onboard facility where a service and maintenance crew could conduct these operations in a shirt sleeve environment.

Operating modes being considered include delivery of a satellite service module along with a logistics payload to the Space Station where a tug would transfer the module to the satellite to be serviced. The Shuttle service and maintenance facility will contain equipment, instruments, and supplies that will provide trained personnel the capability to conduct servicing, maintenance and repair operations. The servicing functions would be conducted on a periodic basis and would include such items as film changing and replenishment of attitude-control propellants.

Although highly automated satellites are designed for long term operations, the capability to visit such satellites in case of malfunctions is highly desirable. The Shuttle could provide the capability for on-orbit replacement of instruments and components. In cases where extensive repair might be required, the Shuttle could either return the satellite or experiment module to the ground or transport it to a Station or Base (depending on the satellite orbit inclination). Satellites that operate for long durations would be designed to accept updated instruments and sensors to enhance their operational capability. This replacement function would be accomplished by the Shuttle.

Propellant Delivery.- In this mission category the Space Shuttle is required to deliver LH_2 and LO_2 propellants to an Orbital Propellant Storage facility (OPS) in low earth orbit. The OPS facility has the function of providing propellant for unmanned planetary missions, the space based nuclear lunar shuttle, and for the space tug operations required for lunar surface and geosynchronous Station support. The Space Shuttle will be required to operate in three distinct tanker configurations to support this mission: (1) as an LO_2 or LH_2 tanker, (2) as a combined LH_2/LO_2 tanker, and (3) as a partial tanker to be used in conjunction with the normal delivery of supplies to the Space Station. During the high traffic periods of the program and for initial filling of the OPS, a dedicated tanker will be used for this purpose. For the dedicated vehicle the tankage and propellant transfer system is an integral part of the orbiter stage. Desirable operational orbits for the OPS range from 28.5° to 55° with an altitude sufficient to provide long orbital life time characteristics and to facilitate delivery to the Space Station (at higher inclination orbits).

The OPS facility itself is comprised of structurally connected cylindrical tanks capable of long duration orbital storage of LH_2 and LO_2 with a tanked mass approaching 1.2×10^6 lbs. The OPS receiver is a passive system maintaining a referenced stability and providing a docking capability with the Space Shuttle tanker. During orbital storage, operation and checkout of the OPS is remotely controlled from the ground through the Manned Space Flight Network (MSFN).

Propulsion Stages and Payload Delivery.- This mission category is concerned with the delivery of payloads and propulsion stages to low Earth orbit. The payload and propulsion stages are then placed on a high altitude Earth orbit or launched as interplanetary unmanned probes. The following principal operational modes will be required to deliver payloads and stages to the OPS facility:

- o Mode 1. The payload and stages are delivered to orbit in separate launches with assembly, fueling and launch performed in-orbit. Mode 1 operation implies the use of an orbital facility to assemble, checkout and fuel the stage and payload. The Space Shuttle would only deliver stages and payloads to the orbital assembly facility.

- o Mode 2. The fully assembled dry stage and payload is delivered to the OPS facility for fueling. Mode 2 will require the Space Shuttle to dock at the OPS facility for the purpose of fueling the propulsion stages. This would be followed by checkout and deployment of the payload and stages by the Space Shuttle. Once deployed, the stage and payload would revert to the control of the Space Station or ground control for final countdown and launch.

- o Mode 3. The fully loaded stage with payload attached is delivered to orbit, with subsequent checkout and deployment. Final checkout and launch is the same as Mode 2.

Short Duration Orbital Mission.- As a spacecraft, the Space Shuttle will have the capability of conducting Earth sensing surveys for up to 30 days stay time. Although many of the surveys will be conducted by the Space Base and unmanned satellites, the Space Shuttle will complement their activities by providing in-depth coverage of selected areas. Surveys proposed to be performed with the short duration mission mode include investigations in the areas of cultural resources, natural resources, and Earth sciences. Two mission modes are considered for this mission.

- o Mode 1. In this mode, the Space Shuttle performs as a dedicated mission vehicle conducting Earth sensing surveys. The orbital characteristics and mission requirements for this mission are generally the same as for the baseline mission, therefore, the mission profile will be similar. Normally, prelaunch activity will not be urgent and a launch response of about five hours will be sufficient. Launch will be in a southerly direction to an orbit having a perigee of about 100 nm and an apogee of 200 to 300 nm. Perigee will be located at the latitude which is of primary interest from the viewpoint of Earth resources evaluations. The altitude and inclination will be selected to provide a ground track with a constant local sun time (sun synchronous orbit).

Remote sensing of the Earth's surface involves use of high and low resolution imaging sensors over a wide range of the electromagnetic spectrum from the ultra-violet region and into the microwave bands. On-orbit operations will consist of activating these sensors over the areas of interest and storing the data for transmission to ground stations via electronic readout. Mission durations will range between 7 and 30 days, depending on the coverage requirements over the areas of interest.

Under normal operating conditions, there will be no urgency for the return to Earth phase. At least one return opportunity per 24 hour period to a prime landing site will be available. Because of its proximity to the WTR launch complex, scheduled return will likely be to Edwards AFB.

- o Mode II. In response to a need for a "quick" evaluation and detailed observations in a given area (such as a natural disaster) an on request surveillance capability will be required. To accommodate quick evaluation, it is desirable that the orbiter return to the launch site within one orbit revolution. Because this mission will be performed in response to an urgent situation the capability of being launched within two hours from a standby status is required.

Individual NASA payloads which are included in the NASA Mission Model have been reviewed as to the requirements imposed on the vehicle systems. Table 2 shows the relationship of the previously discussed missions with the NASA scientific categories and their included loads. Some payloads require multi-mission capability. It should be noted that 80 payloads are included within the eight scientific categories. These payloads are described in appendix A. The type of payload, mission objectives, characteristics and initial operational capability (IOC) are listed. Special emphasis is placed on those requirements that affect EC/LSS design. The types of sensors and the thermal and environmental protection that must be supplied by the Shuttle are also shown.

Mission Profile

For the Space Station/Base logistics mission, the Space Shuttle is to rendezvous with the Space Station to transfer passengers and cargo. The inplane profile (See fig. 1) is as follows: The boost-powered phase terminates with the Space Shuttle injecting into a 45 nm perigee ellipse with apogee altitude of 100 nm. At apogee a velocity increment is added which places the Space Shuttle in a parking orbit in order to time synchronize with the target. After proper synchronization, transfer is made to the terminal phasing orbit which is approximately 10 nm below the target orbit. Closure is performed by the Space Shuttle using a low acceleration system of 0.03 to 0.05g.

The remaining five mission profiles are either identical to the above or are dependent on the orbital characteristics of the target vehicle.

The Mission Frequency Profile (See fig. 2) has been established primarily to furnish guidelines for subsequent analysis concerned with EC/LSS selection, reusability, and overall cost/effectiveness analysis.

TABLE 2

MISSION/SCIENTIFIC CATEGORY RELATIONSHIP

SCIENTIFIC CATEGORY \ MISSION	Space Station Resupply	Satellite Placement and Retrieval	Satellite Servicing and Maintenance	Propellant Delivery	Propulsion Stages and Payload Delivery	Short Duration Orbital Mission
NASA Astronomy (NAS)						
Large Free-Flying Observations		X	X			X
Small Free-Flying Observations		X	X			
NASA Space Physics (NSP)						
Large Payload Module		X	X			X
Small Payload Module		X				
NASA Space Applications (NSA)						
Medium Satellite	X		X			X
Small Satellite		X				
Non-NASA Operational (NNO)						
Small Satellites		X	X			
NASA Biosciences (NBI)						
Bioscience Modules	X	X				
NASA Lunar Option 2 (NL 2)*						
Station to Lunar Base Supply						
NASA Support (NSU)						
Logistics	X			X		
Personnel Transfer	X					
NASA Planetary (NPL)						
Payload Modules					X	

* This scientific category is not related to any specific mission due to its current scheduled flight time.

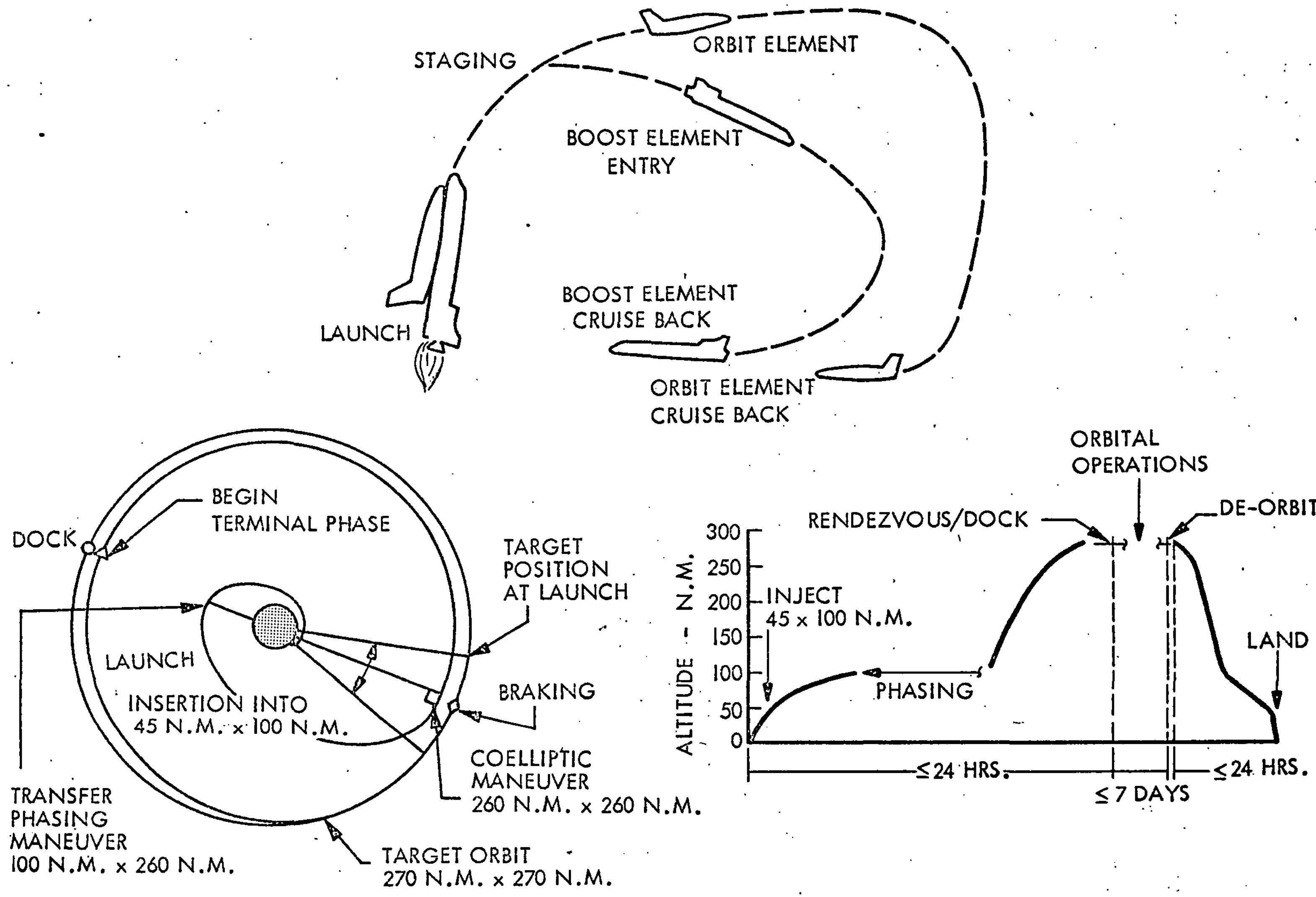

Figure 1.- Space Shuttle Operational Concept.

Space Shuttle traffic models have been generated by the NASA Space Shuttle Task Group based on Space Station/Base development, Shuttle schedules and on projected space programs. A basic assumption is that the Shuttle must be capable of operating over a 100 mission lifetime. The initial nominal traffic model established by NASA called for a total of 536 flights with the initial 16 flights originating in 1975 and reaching an average rate of 60 flights through 1985. The latest traffic model (ref. 1) updates this model and forms the baseline to establish the EC/LSS requirements.

Figure 2 shows five general classifications of flights with the type of flight identified by coded boxes. The table on the upper right portion of figure 2 shows the number and location of the crew, cargo handlers, and passengers for each of the missions. The final column shows the number of flights associated with each mission and the total number of flights for each crew passenger complement, as well as the total number of flights during the 1979 through 1990 time period.

A cumulative total of each type of flight is indicated at the right of the coded box in the body of the figure. This number indicates the cumulative total flights for the particular crew passenger complement through the successive years.

First flight, as shown in the figure, is initiated in 1979 with a total of 21 flights scheduled for that year. Flight frequency increases to a maximum of 47 flights during the fourth year from the first Month of Flight (MOF) and reaches an average of approximately 34 flights for the remainder of the program.

Figure 3 shows the distribution of manned payloads (2, 6, and 10 passengers flights) on a monthly basis. Also shown are the cargo flights involving either 2 or 4 crew members in the Shuttle forward cabin.

Examining the total flight schedule indicates that a maximum of five flights occur in a single month period. This is consistent with the NASA planned development schedule of five Shuttles as shown in figure 4.

Shuttle Configuration

The Phase B System Studies have defined a Space Shuttle configuration consisting of a combined booster/orbiter. This EC/LSS study is concerned primarily with the orbiter and the emphasis of this section will be on the general arrangements, internal location of equipment, and other aspects that directly affect selection of the EC/LS system.

Booster and Orbiter.- The Booster does not have any interface with the Orbiter EC/LSS. The major function it performs is placement of the Orbiter on the desired trajectory. It then reenters and cruises back to the base.

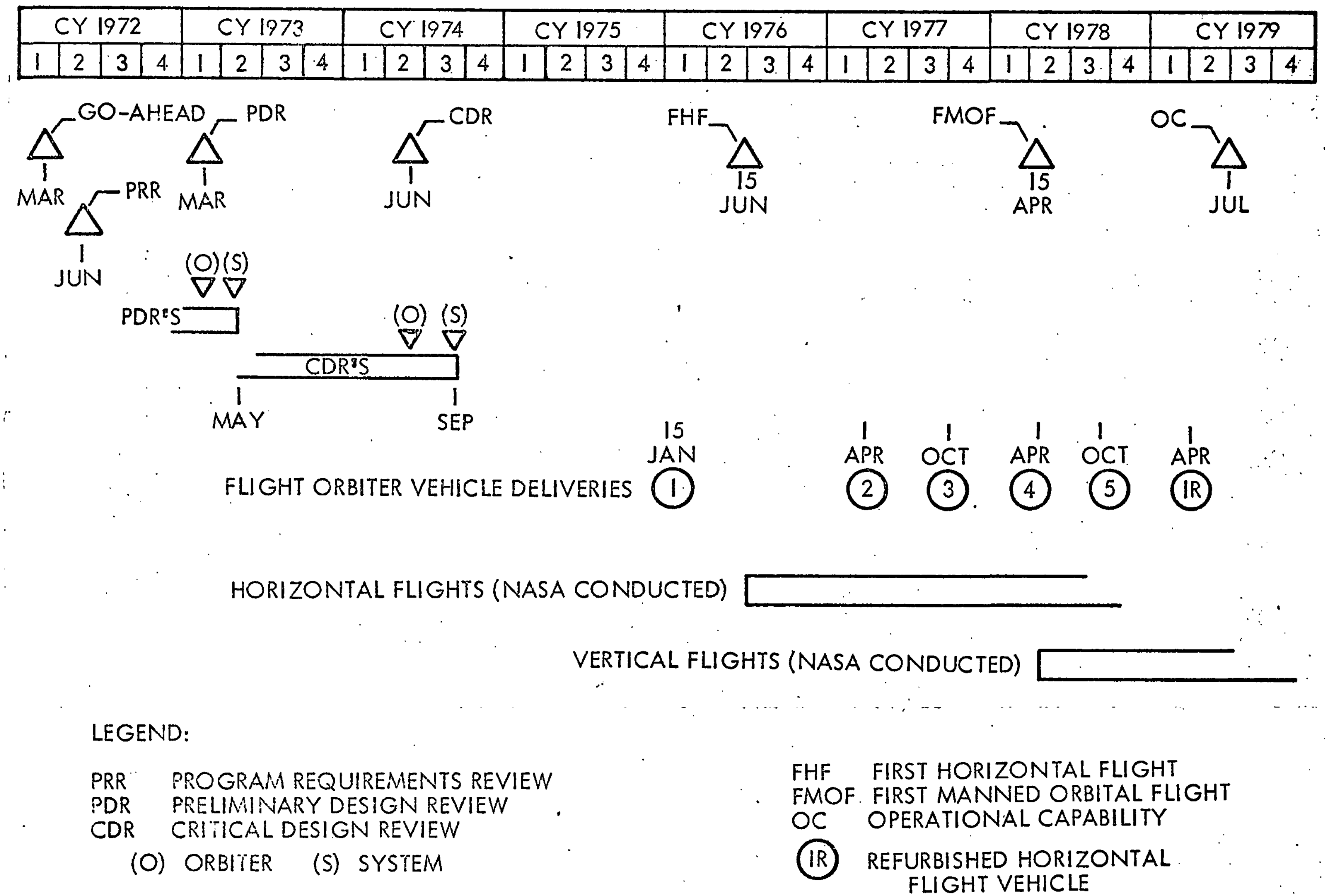

Figure 4. Space Shuttle Program - Orbiter Master Schedule.

Two basic Orbiter configurations were evaluated by the prime contractors: (1) Delta Wing Orbiter, and (2) Straight Wing Orbiter. The Delta Wing was selected (see fig. 5). The geometry of the selected configuration affects the location of the EC/LS system components, while aerodynamic considerations affect the thermal profile and the total mission duration.

The Orbiter consists of the following sections: (1) Crew Station, (2) Airlock, (3) Passenger/Cargo Compartment. The general accommodations of crew and passengers is illustrated in figure 5, which shows the forward crew compartment, passenger compartment, and cargo bay doors located in the cargo module. The passenger/cargo module is shown in figure 6 and illustrates a seating arrangement for the various passenger complements, EC/LSS, and consummable storage complements located under the passenger compartment. The baseline EC/LSS for the pilot/co-pilot and two cargo handlers will be located in the forward cabin and will be a separate system.

Forward locations of the fuel cells, battery power conversion equipment and water boilers minimizes the effects of transmission losses to the electrical/avionics usage systems and also effects plumbing weight savings to the EC/LSS.

Space radiators located on the cargo bay doors utilize an existing deployable break in the main structure of sufficient size to accommodate the required radiator for operation of the EC/LSS. Choice of location affects the weight of the fluid distribution network.

The design thermal profile during reentry is shown for the Delta Wing configuration in figure 7. This profile was used as the design reference for this study.

The orbiter electrical power supply consists of 120 VDC. The EC/LSS will connect to a common power distribution unit. The power distribution unit shall provide for inversion, conversion, and distribution.

Station Configuration

NASA Phase B Space Station definition studies (ref. 2, 3, and 4) provide the basis for most of the information presented in this section.

The general structural configuration of one design, shown in figure 8, is typical of all those evolved in the Phase B definition studies. The four-deck space proposed is about 33 ft. in diameter and 35 ft. long. Decks 1 and 2 have one pressure volume and 3 and 4 another. Six docking ports and hatches are shown for accepting experiment modules or for transferring passengers and cargo. A major interface with the Shuttle occurs with Attached Experimental Modules that operate outside the mold line of the pressurized core module but are physically attached to the core module. A secondary interface occurs with detached free flying modules (subsatellites) which can be retrieved within the core module during nonoperating periods.

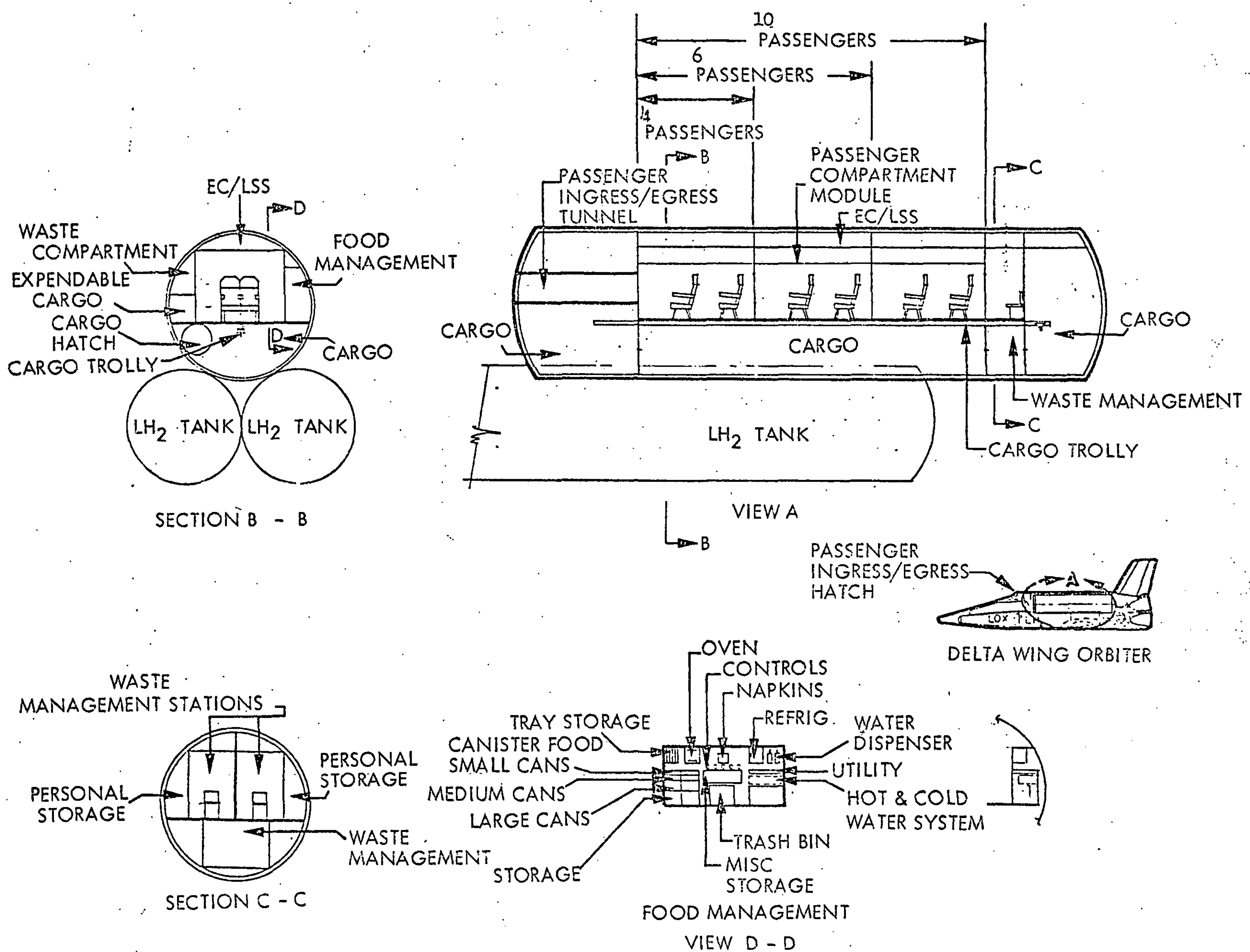

Figure 6.- Cargo Module Passenger Accommodations.

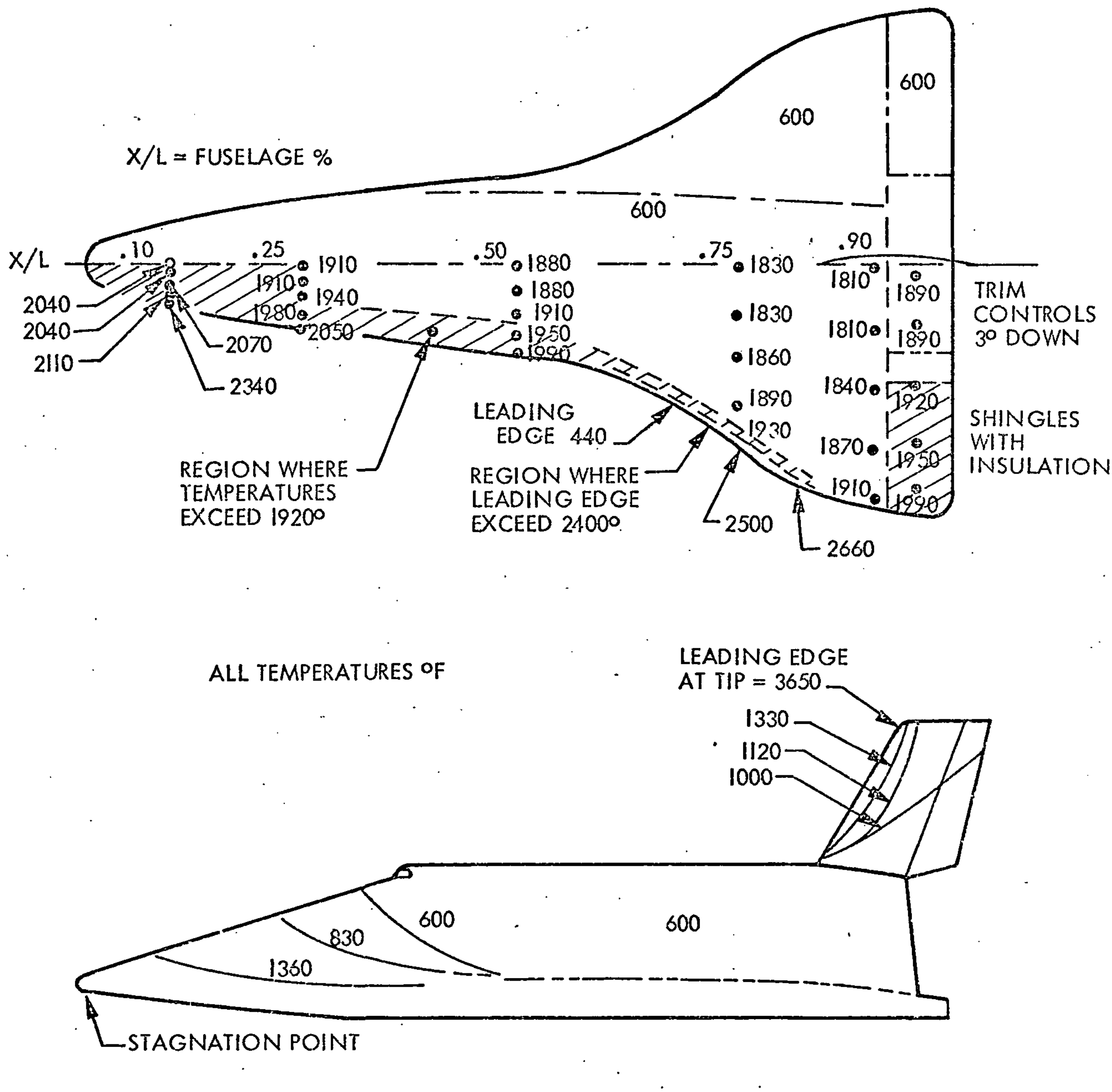

Figure 7.- Orbiter Design Temperature/Delta Wing.

The six docking ports are assigned as follows: Cargo (2), Detached Experiment Module (1), Attached Experiment Modules (2), and Power (1). The most critical subsystem of the Station that interfaces directly with the Shuttle is the EC/LSS.

Mission Operations Analysis

This analysis is organized around the six missions defined earlier. This section presents the typical sequence of events from pre-launch through station docking or insertion in a detached "free flying" mode and subsequent deorbit, entry approach and landing. Special considerations involving crew locations and activities are discussed.

The following assumptions were applied in development of the mission analysis:

(1) The initial Space Station will be occupied by a six man crew.

(2) The growth Space Station will be operated by a 12 man crew.

(3) All payload modules fall within the 20,000 lb guideline and consist of satellites, resupplies, personnel, and experiments depending on the flight requirements.

(4) The Space Station orbit is 270 nm and 55° inclination. The orbit for the free-flying payloads and the sortie payloads are not specified at this time.

(5) Six operational missions will be executed by the Shuttle.

(6) All low Earth orbital experiments that do not require unique orbital parameters or specific other conditions that cannot be practically met by the Station are considered to be incorporated into the Station laboratory facilities.

(7) The Shuttle/Payload combination provides an important capability prior to Station Initial Operational Capability (IOC) and also provides a complimentary function after IOC when unique requirements exist that favor that mode. For example, a portion of the Earth Observation research and application activity will be conducted on Shuttle/Payload sorties in order to obtain different Earth viewing coverage while the major portion of the Function Program Element (FPE) will be carried in a dedicated Research and Application Modules (RAM) attached to the Station.

(8) Three operational modes will be utilized to carry out the six missions: (1) Dedicated (attached to Station), (2) Detached (free flying RAM), and (3) Shuttle/RAM Sortie (Short Duration Orbital Flight).

The next section presents the sequence of events for the Space Station Resupply mission. All other missions with the exception of the Short Duration Orbital mission follow the same sequence of events, and therefore, are not repeated. However, personnel, location and activity for these missions are discussed. The sequence of events differences for the Short Duration Orbital mission follows this discussion.

Space Station Resupply Sequence of Events.- Figure 6 presented the crew/passenger complement and their nominal location for the Space Station Resupply missions. These missions involve logistics supply, delivery of experimental payloads, and passenger transportation to the Station.

The flight phase sequence of events from prelaunch through reentry, approach and landing is as follows: (See glossary of terms at the end of the sequence of events).

Prelaunch

Time Hr Min Sec	Duration Hr Min Sec	Event
-5:00:00	3:20:00	Chill down and slow fill Booster/Orbiter
-1:40:00	1:26:00	Fast fill Booster/Orbiter
-1:35:00	- -	Transfer power to Space Shuttle
-1:30:00	1:00:00	Activate fuel cells
-1:12:00	1:12:00	Replenish LH_2/LO_2
-1:00:00	00:20:00	Replace ground crews with flight crew
-00:50:00	00:40:00	Begin launch readiness checks
		Actuate on-board computers; check atmosphere
		Target guidance system activated
-00:10:00	00:10:00	Begin terminal countdown
0	- -	Liftoff

Launch & Ascent

Time Hr Min Sec	Duration Hr Min Sec	Event
0	- -	Liftoff vertical rise
00:00:00	00:00:00	Critical abort period
00:13:00		End vertical rise; start pitch maneuver
00:01:03	- -	Peak dynamic pressure
00:02:10		Release angle of attack constraint; initiate optimum attitude profile
00:02:24	00:00:52	3g acceleration
00:03:16	- -	Stage booster shutdown; Orbiter start booster separation; Control from Orbiter.

Orbital Transfer

Time Hr Min Sec	Duration Hr Min Sec	Event
5:00:00	- -	Update rendezvous flight plan
5:05:00		Stabilize altitude
		Determine ΔV vector for transfer
18:21:00	- -	Orient for thrusting
18:23:00	00:01:00	Countdown
18:24:00	- -	Begin thrust

Rendezvous

19:10:00		Shuttle 10 nm below and behind Station
		Begin long-range search pattern. Acquire station; visual confirmation; update rendezvous flight plan.
19:15:00		Switch to short range tracking mode.
		Switch to small thrust mode.
19:57:00		Orient and begin thrusting.
21:30:00		Adjust range rate as necessary. Continue thrusting until terminal velocity = 4 fps.
21:55:00		Rendezvous ends 100 ft. from station.
		Switch to docking mode.
22:00:00		Hard docked; begin transfer sequence.

Docked with Space Station

22:00:00	- -	Shuttle docked to Space Station
		Verify integrity of Station
22:00:00	122:00:00	Dormant storage conditions for all subsystems;
		Shuttle provides own power; monitor subsystem status from Station; establish standby condition for critical subsystems.
	00:48:00	Crew transfer
	04:00:00	Cargo transfer
	00:30:00	Return cargo transfer

Deorbit

Time Hr Min Sec	Duration Hr Min Sec	Event
144:00:00	- -	Decision to return
144:00:00	24:00:00	Activate all subsystems
		Initiate computer descent phase
		Select landing site
		Computer return trajectory
		Check local landing site weather
		Compute deorbit point (time)
		Notify ground stations
		Crew ingress to shuttle
		Perform systems checks
165:40:00		Separate from Space Station; initial ΔV = 1.0 fps until distance from Station $\geq$ 100 ft.
165:40:00	00:38:45	Loiter
166:10:00	00:03:00	Distance from station 1 nm; rotate from deorbit 180°
166:15:00		Stabilize attitude; deorbit guidance update; stow nose cap; retract RCS thrusters; start APU - verify hydraulic pressure
166:17:45	00:01:00	Begin deorbit countdown
		Position main engine
166:18:45		Deorbit thrusting ΔV = 437 fps T/W $\geq$ .15
166:19:45		Jettison liquid propellant (committed to reentry); stabilize attitude
166:20:00		Stow engine nozzle
166:21:00		Reentry guidance update
166:25:00		Determine entry footprint
166:30:00		Correct reentry trajectory

Entry

		Monitor entry point, entry attitude, and touchdown footprint; last GNC state vector sampling and update.
166:40:00		Retrack horizon sensors, star sensors, antennas, etc.
166:45:00		Rotate orbiter to entry attitude
166:51:45		Enter atmosphere 400,000 ft
166:53:00		Pre-communication blackout transmissions
166:55:00	00:37:00	Enter blackout region
166:55:15		300,000 ft, L_{max}, α , 55°
166:56:25		Aero control begins 270,000 ft; initiate pull-out at α = 55°
166:58:05		Pull 255,000 ft α = 23° Modulate ϕ
167:05:05	00:05:00	Constant altitude 250,000 ft α = 25°; fly temp. profile
167:21:00		220,000 ft full aero control

Entry (Cont.)

167:28:25	170,000 ft L/D_{max}, $\alpha = 15^\circ$ $\phi = 30^\circ$
167:32:00	Exit communication blackout
167:37:25	115,000 ft wings level $\alpha = 15^\circ$ $\phi = 0^\circ$
167:41:30	90,000 ft

Approach/Landing

167:41:30	00:02:05	Mach 6 to 1.2 range
167:43:00		Retract window shields; adjust glide angle
167:43:35		Begin transonic flight Mach 1.2 - 0.8
		Establish subsonic glide
		Perform emergency check (unpowered mode)
167:44:50		Deploy turbojets h = 45,000 ft $M < 0.9$
167:45:10	00:11:50	Ignite turbojets (idle thrust)
		Reduce airspeed to 250 KIAS; establish shallow glide and track; inbound aligned with runway; open nose landing gear doors (expose navigation aids)
167:45:30	00:03:20	Bank and begin 360° descending turn; energize landing aids
167:47:00	00:02:00	Perform final landing check; check computer landing profile
167:48:45		Reduce airspeed 200 KIAS
167:49:00	00:00:15	Lower landing gear
167:49:30		Adjust power setting and reduce bank angle
167:49:50		Final approach rollout 5 nm from threshold; lock on ILS
167:49:50	00:01:30	Verify outer marker; track on glide slope (3 degree)
167:50:00		Set final approach power
167:51:20	00:00:03	Initiate flare
167:51:30	00:00:30	Touchdown V = 165 - 180 knots
167:51:45	00:00:15	Reverse thrust
167:57:00	00:00:03	End landing, initiate vehicle shut down; crew egress; cool vehicle
168:00:00		Transport vehicle

During all phases of the mission except when docked to the station, the crew and cargo handlers remain in the forward compartment and the passengers remain in the passenger module. The flight crew duties during these periods are primarily restricted to vehicle maneuvers. The cargo handlers and passengers have no special activities except for routine sustaining functions.

A more descriptive analysis of personnel activities during the docked phase and impact on the EC/LSS is discussed in the Shuttle/Space Station Interface section of this report.

Alternate Missions.- The remaining missions which include Satellite Placement and Retrieval, Satellite Servicing and Maintenance, Propellant Delivery, and Propulsion Stages and Payload Delivery but exclude the Short Duration Orbital missions are either experimental or logistical.

In the case of the logistical flight, the cargo handlers effect transfer of cargo (propellants, propulsion stages, etc.) between the Shuttle and on-orbit facilities.

For experimental flights, the cargo handlers perform technical duties unique to the individual experiment.

Crew complement and location for these flights are also shown in figure 6.

Short Duration Orbital Missions.- This mission normally requires two technicians located in the cargo module. Mission duration varies depending on the payload. These missions involve Earth sensing surveys of 7 to 30 day duration, and surveillance sorties of less than 7 days. The most critical mode from an operational standpoint is the surveillance sortie which requires that the orbiter return to the launch site within one orbit revolution. However, from an EC/LSS standpoint, the 7 to 30 day mission for the manned Earth sensing survey is the most critical and the sequence of events for that mode will be given. As indicated by the sequence of events, a maximum mission period of 30 days is required. This mission involves an Earth survey payload where the crew size is dictated by the crew support function. Five investigative areas dealing with agriculture/forestry, geography, hydrology, oceanography and geology require target acquisition, measurement operation, data monitoring, maintenance and servicing. Two crew members services are required for a period of 30 minutes per orbit for target acquisition, measurement operation and data monitoring. Four manhours per day are required for maintenance and servicing. The prelaunch, launch and ascent sequences are the same as described for the Space Station Resupply mission. The remaining sequences are as follows:

Orbital Transfer

Time Hr Min Sec	Duration Hr Min Sec	Event
00:03:16	- -	Orbiter start
00:07:00	00:00:16	3g acceleration
00:07:16	00:43:45	Orbit inject hp/ha = 45/100 nm
		Coast to 100 nm
00:10:00	00:03:00	Roll 180° to on-orbit altitude
		Verify orbit
		Establish real-time communication with Space Station
		Computer flight plan for placement of Shuttle into desired orbital altitude

Orbital Transfer (Cont)

Time Hr Min Sec	Duration Hr Min Sec	Event
00:25:00		Determine 1st transfer velocity vector for desired altitude
00:48:00		Orient for transfer thrusting
00:51:00		Initiate thrusting
19:10:00		Acquire operational altitude
Orbital Operation		
20:00:00	00:30:00	Activate payload power supply
20:00:00	00:30:00	Checkout crew/cargo payload module and equalize pressure between front compartment and payload module
20:30:00	01:00:00	Perform general housekeeping functions and checkout EC/LS thermal, pressurization, etc. elements: transfer two crew members from forward compartment to payload module.
21:30:00	02:30:00	Initiate checkout and calibration of experiment equipment
23:00:00	04:00:00	Calibration of 20 Earth observation sensors
Time Day		
2-30	00:30:00/orbit	Target acquisition measurement operation and data monitoring.
Deorbit		
696:00:00	- -	Decision
696:00:00	24:00:00	Deactivate all sensors, initiate computer descent phase, select landing site. Crew ingress into forward compartment. Actuate power from Shuttle to payload.
720:00:00	- -	Deorbit thrusting
720:01:00		Jettison liquid propellant
720:13:00		Correct reentry trajectory
Entry		
720:38:00		Rotate orbiter to entry altitude
720:44:00		Enter atmosphere 400,000 ft.
720:45:00		Aero control begins 270,000 ft.
720:47:00	00:05:00	Control altitude 250,000 ft. fly temperature profile

Approach/Landing

Time Hr Min Sec	Duration Hr Min Sec	Event
721:13:00	00:02:00	Mach 6 to 1.2 range
		Begin transonic flight Mach 1.2 - 0.8
		Establish subsonic glide
		Perform emergency check (unpowered mode)
721:16:00	00:11:00	Deploy turbojets, ignite
721:27:00		End landing, initiate vehicle shutdown, crew egress

Glossary of Terms Used in Sequence of Events

α	Angle of attack
APU	Auxiliary Propulsion Unit
fps	feet per second
GNC	Guidance and Navigational Control
h	altitude
h_p/h_a	perigee/apogee altitude
ILS	Instrument Landing System
L/D_{max}	maximum Lift/Drag
LH_2	Liquid Hydrogen
LO_2	Liquid Oxygen
L_{max}	Maximum Lift
M	Mach number
ϕ	Reentry angle
RCS	Reaction Control System
T/W	Thrust/Weight
V	Velocity

Summary

This section has identified Shuttle missions and their effects on the selection of an EC/LSS. The most demanding missions with respect to EC/LSS requirements are the 10 passenger transfer flights and the 30-day Short Duration Orbital Mission.

A review and correlation of the NASA Scientific categories and their payloads with the mission has revealed definite requirements for EC/LSS support.

Payload characteristics that have a direct effect on EC/LSS are lifetime/revisit requirements. Many payloads will be designed for a ten (10) year life time which implies refurbishment/remove/replace implementation. Payload volumetric and weight requirements also effect the EC/LSS packaging and technique. The most significant factor is that many of the flights will require Shuttle support for EC/LSS maintenance. A detailed payload mission analysis with operational criteria included is presented in appendix A. Mission objectives and characteristics of the payload are also listed in this Appendix.

CARGO MODULE EC/LSS DEFINITION

This section of the report defines the cargo module EC/LSS which will support the wide variety of payloads and missions of the Space Shuttle program. The passenger, cargo and mission data developed in the Mission/Vehicle Definition section of this report forms the basis for this analysis. Performance, design and interface requirements relevant for identification of EC/LSS problems were selected from the mission data.

The Shuttle baseline and alternate candidate EC/LSS concepts were selected, defined and evaluated. The evaluation considered personnel complement and mission feasibility, and minimization of impact on basic vehicle design, transportation to orbit weights, maintainability and turnaround time.

The evaluation criteria applied were cost/effectiveness, weight, flight frequency, number of crew, modularity, and performance. Volumetric considerations that influence packaging criteria were examined. Module turnaround considerations involving method and operational procedures are discussed. A summary concludes this section with module performance characteristics and technology advancement requirements.

Performance and Design Requirements

The basic requirement for the cargo module EC/LSS is to support a variety of manned flights that differ in number of crew and flight duration. The Mission/Vehicle Definition section has established that the crew size variation due to the various mission requirements is as follows:

- 20 - ten passenger transfer flights
- 24 - six passenger transfer flights
- 49 - two passenger cargo/experiment flights

The mission duration is normally 7 days; however, there is one 28-day mission, High Energy Astronomical Observatory (HEAO), that requires crew calibration work during the initial portion of the mission. Potential Earth survey missions will require manned occupancy from 2 to 30 days.

EVA requirements are minimized in most transfer maneuvers, however, in the Satellite Servicing and Maintenance missions, there may be many unique situations where the most efficient method of refurbishing, removing, and replacing equipment is via EVA. Photographic supplies located externally on the satellite is a typical situation where an EVA might expedite removing and replacing film.

IVA activity occurs during the crew transfer period. The requirement exists to transfer six to ten passengers effectively. This infers that direct exit of the crew through the cargo module airlock directly attached to the Station might be preferred. There is an additional possibility that the Shuttle, after docking the cargo module to the Station, might have to disengage which would necessitate direct transfer capability from the cargo module.

The following requirements which are particularly pertinent to identifying problems in the EC/LSS have been selected from the mission data. These requirements will form the basis for the cargo module EC/LSS design.

Preflight Phase.- On-board vehicle checkout, system test, and functional system analysis will be performed. An integrated launch, loading, and refurbishment facility will be provided for logistics and servicing functions. Critical systems will have provisions for safing the system. Single point failures having potential mission abort implications will be minimized.

Launch Through Orbit Phase.- The flight crew and on-board systems will have the capability of performing all tasks during launch. Design conservatism and system redundancies will be utilized to eliminate system failures having potential mission abort implications. A shirtsleeve environment will be provided for both crew and passengers. Cargo transfer will be automated as much as possible and require little, if any, EVA.

Return Phase.- The cargo module and its systems will be self-sustaining for the missions. This requirement imposes passenger life support and cargo thermal support from deorbit through landing.

Post Flight.- On-board checkout and module replacement are required to achieve a turnaround time (from landing to launch) of less than two weeks. Maximum usage of standard aircraft type maintenance is required.

General.- The vehicle will have the following capabilities:

- Up to 20,000 lb. up/down cargo (quarterly cargo requirements for 12 man Space Station is 12,000 lbs.)
- Seven days on orbit life.
- 2 man minimum cargo module occupancy.
- 10 man maximum cargo module occupancy.
- 3g trajectory load factor - passengers.
- 4g trajectory load factor - cargo and crew only.

- Subsystems designed to fail operational after failure of the most critical components and fail safe after second failure.

- Displays will be all electronic (digital readout).

- The crew and passenger environment is "shirtsleeve".

- Cargo elements containing hazardous material will have self-contained protective devices or provisions.

- The cargo module and its systems shall be capable of use for 100 mission cycles with minimum of maintenance.

- Electronic interface systems will interface with a standardized redundant multiplex data bus system.

- For missions other than logistics, EVA capability will be provided at the expense of the allocated payload weight.

The following requirements are considered directly applicable to the EC/LS system design:

- The cargo module atmosphere and total pressure will be the same as the Space Station/Base (10 to 14.7 psia and O_2/N_2 mixture).

- Personnel/cargo transfer will normally be IVA.

- Provisions for deployment and retrieval of maximum cylindrical payloads is required. Normal operation will not include EVA.

- The vehicle design and its critical subsystems will include proper on-board provisions to quickly and easily place the vehicle in a safe condition following landing.

- Total vehicle turn around time from landing to launch readiness will be less than two weeks. The removal and replacement time will be minimized, by providing accessibility to modules.

- The vehicle will have design characteristics and reentry flight parameters that will provide low heating rate profiles necessary for maximum utilization of refurbishment thermal protection materials.

- Limited cargo transfer is possible through the personnel transfer hatch. More than one transfer interface with the Space Station/Base may be required.

- Provision for deployment and boarding of a maximum cylindrical payload is desired. Normal operations will not include EVA.

The payload variety and crew sizes are the major differences in requirements between the Shuttle and the cargo module EC/LSS. Both EC/LS systems must be compatible with the Space Station and/or independent free-flying modules.

Other differences which must be taken into consideration are; the large volumetric differences between the crew forward compartment and the cargo module, and the pressurized and unpressurized sections of the cargo module. Turnaround requirements also impose different design constraints on the Shuttle and cargo module EC/LS systems. The cargo module will be detached from the Shuttle for refurbishment, necessitating an independent method of ingress for removal and replacement of expendables and system maintenance.

Shuttle/Cargo Module Interface Requirements

Interface requirements (ref. 5) particularly pertinent to the cargo module EC/LSS design are listed below:

Cargo Bay Envelope.- The Shuttle shall provide a cargo bay with volume provisions for a payload envelope fifteen (15) feet in diameter and sixty (60) feet in length.

Crew/Access Tunnels.- The Shuttle shall provide two extendable tunnels: one for crew/passenger pressurized access to the stowed payload; and one for pressurized access if the payload is extended from the cargo module.

Cargo Module Access.- Prelaunch access shall be provided by the large cargo bay door(s). On-pad access shall be provided by a 15 in. by 15 in. access door on the L.H. side. On-orbit access shall be provided by a crew/passenger transfer hatch.

Cargo Bay Venting Provisions.- Provisions for purging and venting the cargo bay for all mission phases shall be provided on the cargo modules.

Electrical Interface.- The Shuttle shall provide the connectors and junction boxes to supply power to the cargo module. Shuttle power will only be available during the on-orbit phase. The cargo module shall supply the electrical power, including ground power, required by the payload during all other mission phases.

Avionics.- The Shuttle electronic system shall provide data transmission, command, display and control, checkout, data bus, guidance and navigation, and status monitoring services for the payload. Provisions for securing data and communications shall be cargo module supplied.

Hardware Communication Interfaces.- A two-way transfer of voice data via hardware shall be provided during the Shuttle/cargo module attached mode. When cargo module is detached or free-flying, an RF link shall be provided.

Shuttle Baseline EC/LSS

The NASA contractor studies (ref. 6 and 7) were reviewed and adopted as representative of the EC/LSS baseline model. The following summarizes the principal characteristics of the baseline model and will offer a point of departure for the selection of a cargo module EC/LSS. Figure 9 shows the two block diagrams which are the basic concepts for the EC/LSS, as defined by the NASA phase B contractors. For purposes of this study, concept 2 was used as the baseline. This concept was selected because it included all EC/LSS prime elements requisite to the evaluation and more comprehensive data in terms of equipment, components, and weight was available.

Waste Management.- An integrated vacuum drying waste management subsystem is selected. This system uses a static iris type urinal which is a combined water separator, fan, and water pump. This unit is a modification of the isolating bowl-type phase separator/water pump used for the Apollo Lunar Module. Fecal material is handled with a motorized slinger and waste collector/vacuum dryer.

A mission requirement exists to transport female passengers in the Shuttle, therefore, the waste management element must be compatible for both male and female use. It is assumed that the fecal collection system can be used by both sexes. The urine collection system will require a separate collector for female use. An adapter is required which can quickly replace or attach to the male urinal. This adapter will be easily fitted to the urinal and either will permit repositioning the funnel or permitting the female passenger to be positioned on the urinal in such a way as to capture the urine stream efficiently.

A 30-day mission extension is accomplished by periodic replacement of waste collectors, bacteria filters and activated charcoal filters, and a 30-day complement of biocide.

Food Management.- The food management system uses various food can sizes. The cans are aluminum with "pull-out" lids. These cans, together with dehydratables and drink packages, are stored in a canister designed to house 11 food serving cans. The drink package is of the Apollo type which uses a hermetically sealed flexible plastic container with a one-way valve for water insertion. Foods are categorized as follows:

- o Thermostabilized (sandwich spreads, tuna, puddings, etc.)
- o Rehydratable (entrees, desserts, salads and vegetables)
- o Wafer (ready to eat snacks)
- o Beverages (lemonade, wine, tang, etc.)

A locker compartment is used which is sized for the baseline mission. Ambient storage items can be stored in available space.

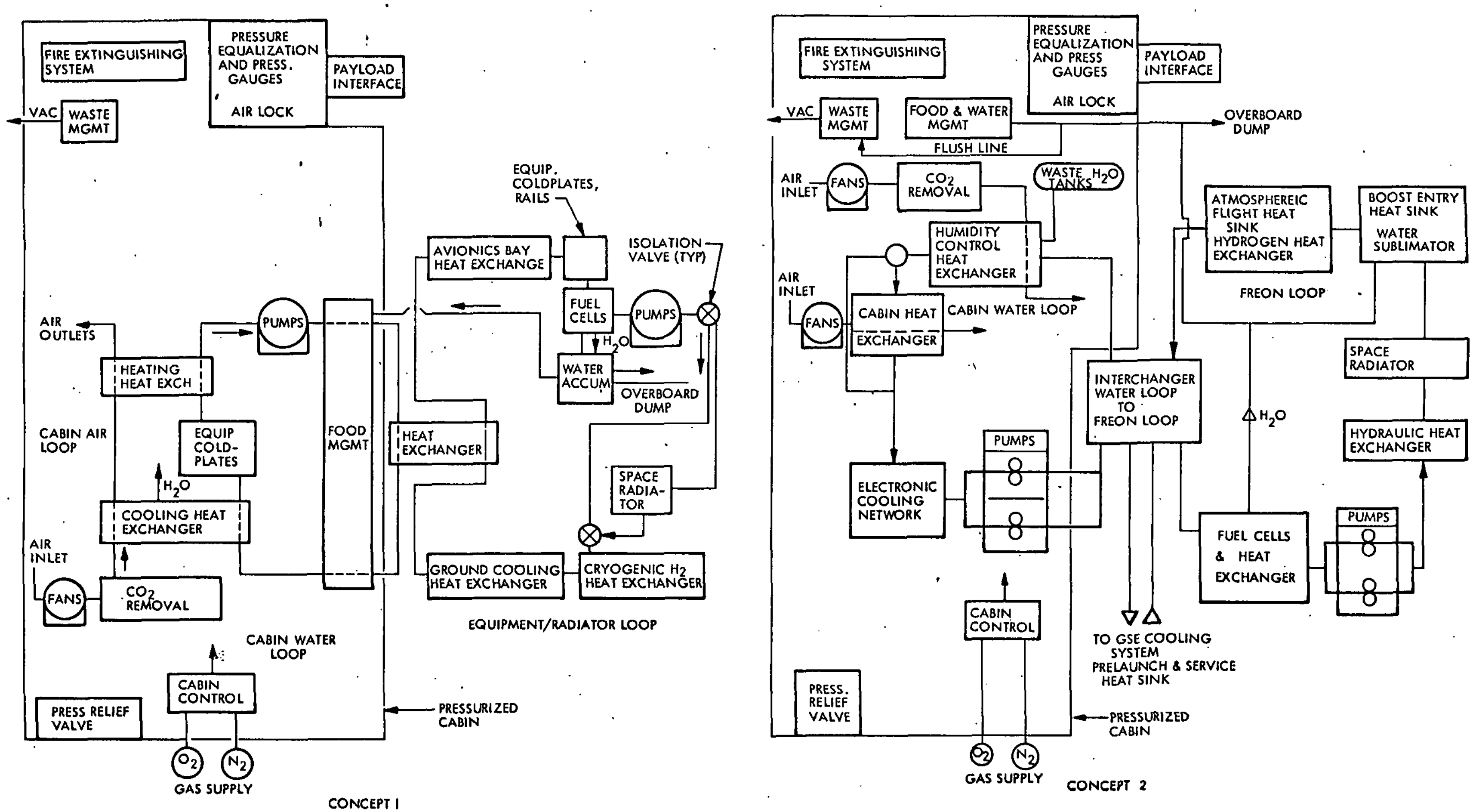

Figure 9.- EC/LSS Block Diagrams.

Food preparation is normally simplified by using ready prepared stored food. A service station is provided which contains a food tray compartment, oven, napkin dispenser, utensil drawers, hot and cold water dispenser and drain, and can opener. Trash is collected in a plastic bag.

Carbon Dioxide Removal and Odor Control.- Granular Lithium Hydroxide (LiOH and charcoal) is packaged in beds through which cabin air is processed. Three idential LiOH cartridges are installed in parallel, and each is sized for 24 hour operation. A total of nine cartridges are required for the 7 day mission which allows two cartridge failures within the subsystem fail safe requirements.

The system is based on supporting a four-man crew and operating continuously for a seven-day period. The subsystem consists of fixed hardware including fans, valves, and structure. A 30-day mission would require a minimum of 23 cartridges.

Humidity Control.- A three fluid condensing heat exchanger is used which contains two coolant passages and one air passage. One coolant passage is attached to the primary cabin coolant loop and the other to the secondary cabin coolant loop. The air passage receives cabin air flow from one of three process flow fans, each having its own check valve. The water separator is similar to the Apollo Portable Life Support System (PLSS) unit. The separator consists of an elbow separator with wicking built into a duct downstream of the elbow section. Three wicks (refrasil) each with its own water transfer discs are built into the duct. The transfer disc is a hydrophilic membrane. Alternate concepts using rotary or centrifugal separators were not considered for the baseline.

Cabin Temperature Control.- Electronic/electrical generated heat is coldplate cooled. A cold wall configuration which has coolant flow passages attached to the wall close enough together to meet wall temperature requirements (62° to $68^\circ F$ for cooling and 75° to $85^\circ F$ for heating) is provided. The cold wall in the ceiling and around the flight deck reduces the heat load on the cabin crew during space flight and after entry. The cold wall in the floor reduces heat loss through the floor to the cryogenic tankage located directly beneath.

A heat/exchanger fan is required for cabin air conditioning and removal of any heat not handled by the cold plate. A three fluid heat exchanger is used to reject the sensible heat from the cabin atmosphere to a coolant circuit. It is used in conjunction with a separate condensing heat exchanger which controls cabin humidity.

Heat Transport Loop.- A dual loop heat transport (radiator) subsystem is used which employs a water loop in the crew or pressurized compartment and a Freon-21 loop for the remainder of the vehicle. The primary water loop provides cooling to a food station water chiller, the humidity control heat exchanger, the coldplate mounted electronic equipment, and the cabin heat exchanger. A secondary water coolant loop provides the fail-safe requirement.

The Freon-21 loop picks up the heat rejected from the water loop in the interchanger and rejects it either through the space radiator, the sublimator, or a hydrogen heat exchanger. For ground operation, the heat rejected in the interchanger is removed by a GSE cooling system. A secondary Freon loop provides for the fail-safe requirement.

Auxiliary Heat Sinks.- A vehicle installed GSE heat exchanger is used to reject heat from the vehicle coolant loop during prelaunch while docked to the Station and after landing.

A water sublimator may be used for handling those peak heat loads which exceed the radiator capacity during orbit. When the sublimator inlet coolant temperature rises above the desired value, a temperature sensor activates the automatic switchover controllers opening the shutoff valves on the water inlet lines, admitting water to the sublimator. Sublimation of water removes heat from the coolant. The water vapor thus produced is dumped overboard. Alternate methods which employ evaporators or cryogenic heat exchangers were not considered for the baseline.

The sublimator is designed to reject 70,000 Btu/Hr.

O_2/N_2 Supply and Pressure Control.- This subsystem consists of plumbing, controls and regulators. The primary and secondary systems are identical to each other and completely redundant. The supply systems receive O_2 and N_2 from the storage systems at 900 psia. Total pressure is 14.7 psia with oxygen and partial pressure of 3.1 psia.

An emergency oxygen flow up to 55 lbs/hr for a period up to ten minutes is provided. The high pressure nitrogen is stored in two tanks. Pressure regulators, shutoff valves and fill connectors make up the subsystem. Redundant regulators and valves are provided. Operating pressure is 3000 psi and a capacity of 20 lb. of N_2 is in each container. Carbon filament composite tanks are used.

Fire Control.- An Apollo type fire extinguisher is used. It is a domed stainless steel cylinder 10 inches high with a seven inch nozzle and handle. The cylinder contains a polyethylene bladder capable of expelling two cu. ft. of foam (hydroxymethyl cellulose) in 30 seconds.

Water Management.- The sublimator water supply is initially provided at launch and subsequently replenished by excess water production from the fuel cells. The water capacity of the sublimator is sized for 12 hr. operation under emergency flight loads.

The potable water subsystem consists of a water tank with pressurized bladder, valving, plumbing, water chiller, and a heater. It will provide either hot or cold water for drinking and food reconstitution. The unit is sized for a 12 hour supply and also receives its water from the fuel cells.

Cargo Module Alternate EC/LSS Concepts

The mission variety, crew size, and variable duration allocated to the cargo module suggests reviewing alternate subsystem concepts. LMSC and NASA Shuttle contractors have conducted extensive analyses, both in the past and currently that examined tradeoff regarding mission duration and size of crew. For the baseline mission, all of these analyses have shown conclusively that based on first-flight cost, state-of-the-art, design complexity, and availability, the preferred approach is utilization of existing well proven concepts. For example, out of three CO_2, humidity, and thermal control subsystem concepts, LiOH with a condensing heat exchanger is recommended, even though there are other techniques that have competing advantages. However, as will be shown, the development cost factor far outweighs any of these advantages.

Heat rejection methods employing space radiators require further investigation. Alternate concepts must consider not only the orbital phase but the reentry, atmospheric and ground cooling periods.

Adding radiator panels to the Shuttle is one approach to handling the excess cargo module heat load. However, this will not allow for autonomous operation. Current radiator concepts include installation on the back of the cargo module doors or deployment from a storage area immediately below the cargo doors. Radiators sized to dissipate the projected basic shuttle heat loads use the major portion of the available area in these locations. However, additional radiator area may be prohibitive because of the limitations of available space. The problem is further compounded by the fact that there will be a considerable increase in requirements for low temperature heat rejection in the condensing humidity control heat exchanger. Humidity control system air flow rates must increase in proportion to crew size. Limiting the cabin dew point to acceptable levels requires the air outlet temperature of the heat exchanger to approach the lowest possible coolant temperature. This is only possible if the air to coolant mass flow ratio in this heat exchanger is maintained. This will result in a significant increase in coolant flow requirements. When the increased coolant flow along with the slight increase in heat load is combined, a significantly lower average radiator temperature results. The higher heat load and lower average radiator temperature combine to yield radiator areas which will be far larger than might be expected for the slight increase in the required dissipated load.

Alternate concepts that appear promising are: (1) adding radiator panel area to the existing Shuttle, (2) use of a deployable folded radiator, (3) use of auxiliary heat sinks such as water or hydrogen, and (4) reduction of the low temperature heat sink requirement through the use of a dessicant humidity control system.

The baseline EC/LSS designed for the Shuttle forward crew compartment forms a model for the cargo module EC/LSS. For the purpose of the cargo module EC/LSS design task, radiator panels will be an integral part of the cargo module which does not preclude autonomous operation.

Cargo Module EC/LSS Configuration Selection

Major considerations in the evaluation and selection of an EC/LSS configuration are: mission flexibility, module turnaround, and cost effectiveness.

Mission Flexibility.- Personnel complement, mission frequency, and missions up to 30 days duration were examined to determine their impact on EC/LSS selection. The mission frequency profile (fig. 2) illustrates the feasibility of scheduling 420 flights; however, careful master scheduling and adherence to the two week turn around is mandatory. As shown on figure 3, an early cargo module EC/LSS is required for the two passenger level, shifting to a six passenger level near the end of 1971, and continuing through 1985. The remaining flights alternate between the 2 and 10 passenger level. Using a modular approach, three unique cargo modules outfitted with one, two, and three basic Shuttle EC/LSS can be envisioned. Another approach is to establish only two cargo modules outfitted with one and two basic EC/LSS repectively, for the two and six passenger flights. When the six passenger flights end in 1985, a third EC/LSS is added to take care of the 10 passenger flights. This latter approach would have the decided advantage of lower cost and is recommended.

An examination of the NASA payloads and planned missions has indicated that a maximum mission duration for a free-flying payload module will be 30 days. Reviewing the manning requirements such as initial alignment, calibration, and start-up, it appears that a maximum of two technical crewmen could accomplish these tasks. Revisits to satellites for the purpose of servicing and maintenance will normally only require two technicians for two to three days. The most severe cases in terms of demand on the EC/LSS therefore are the 10 passenger transfer flights and the two man 30-day Short Duration Orbital Missions. The 10 passenger transfer flight might require a mission duration of seven days, therefore, the total man-days of life support will be 70. The two man 30-day mission requires a total life support requirement for 60 man-days. Therefore, if the selected EC/LSS is designed for the maximum, 10 passenger transfer flight, it will easily accommodate the less demanding flights. The excess capability can be retained or the modules can be off-loaded to meet the specific time requirement.

Additional mission flexibility is provided by personnel accommodations suitable for both male and female passengers, as previously discussed.

The EC/LSS flexible packaging criteria requires functional and centralized packaging within the cargo module. Fluid and electrical lines utilize a common entry and exit into and out of the package. Redundant paths, such as fluid lines, electrical wiring, etc., shall be located to ensure that an event which damages one line is not likely to damage the other.

A large proportion of available volume will be utilized by the ten passenger food, rest and waste management areas. In addition, there will be cargo space allocated for expendable supplies. Total volume availability is a 15 ft. dia. cylinder by 60 ft. length. Based on maximum utilization (75% of total volume), 8,000 cu. ft. of space is available. A summary of major equipment/personnel volumetric requirements shows:

	Vol. (cu. ft.)
10 Passengers (100 cu. ft. each for habitability and mobility)	1,000
Waste Compartment	190
Food Management Area	200
EC/LSS (180 cu. ft. for 4-man system)	540
Expendable Cargo (12,000 lb.)	400
Total Volume Requirements	2,330

The excess volume available of 5,670 cu. ft. would be utilized for payloads, propellants, and the remaining subsystems.

This analysis clearly shows that the cargo module EC/LSS with maximum cargo and passenger loading does not impose any serious volumetric constraint. A thirty-day mission extension capability by addition of consummables also appears within the volumetric capability of the cargo module.

Module Turnaround.- An analysis was made of all the scheduled tasks which must be accomplished from the time of landing until the subsequent relaunch. The rapid turnaround time-line requires four major phases: (1) post landing operations, (2) maintenance operations, (3) pre-launch operations, and (4) launch operations. Methods and operational procedures for changing the EC/LSS modules must be consistent with these four phases and be conducted within the two week turnaround cycle. The System Reusability section further discusses maintainability guidelines and other pertinent aspects of this area.

During the post landing operation, the Shuttle's crew and passengers egress from the vehicle, permitting shutdown of the EC/LSS; with the exception of the thermal control element. Vehicle cool down is accomplished to dissipate the surface heat and to cool the internal structure and subsystems. Ground Support Equipment (GSE) is utilized during this phase. The cargo module is next removed and transported to the maintenance facility. The EC/LSS located in the cargo module is scheduled for both scheduled preventative maintenance and unscheduled corrective maintenance.

Operational procedures for changing the EC/LSS modules is initiated by reviewing theperformance characteristics of the subsystem as it progressed through the various mission phases. Performance data is collected in a routine periodic basis during flight and if warranted can be requested at any time by a crew member. Trend analysis is performed with the raw data and decisions are made as to maintenance and/or remove/replacement needs. A sequential fault isolation procedure is initiated for a complete subsystem checkout. As discussed more fully in the System Reusability section of this report, a fault isolation technique which incorporates checkout circuitry integrated into the EC/LSS components will permit rapid evaluation as to the performance level and indicates which components are below level and are to be replaced.

Commonality of fittings, hose connections, quick-release couplings, etc. permits the rapid removal of an EC/LSS module. Wherever possible, the mechanical type modules such as the heat exchangers will have quick-release inlet-outlet couplings accessible so that the removal process is easily initiated. Electrical/electronic modules will have plug-in commonality, which only requires insertion/extraction procedures. Both types of modules, wherever practical, will employ hinged covers so sub-modules can be accessible. Modules will be mounted on slide rails, permitting extraction and/or complete removal when required.

Cost/Effectiveness Analysis.- This section presents results of a study to develop a methodology for determining the life cycle cost of the EC/LSS for the crew/cargo module of the Shuttle. It is concerned with all the Shuttle flights that involve manned support.

Four candidates were selected for evaluation consistent with the current mission definition: (1) a basic four man unit, (2) a two man customized unit, (3) a six man customized unit, and (4) a ten man customized unit. The weight breakdown for these four candidates are shown on table 3. Each candidate was evaluated using a cost effectiveness technique that examined critical cost/weight factors. It should be noted that the table reflects dry weights only based on conservative estimates and judgment.

The EC/LSS subsystems were examined to identify fixed and variable weight items. A varying percentage was assessed against the variable weight item ranging from a low percentage change for the 2 man unit to a higher percentage increase for the 10 man unit. The rationale for the varying percentage was that the 2 man unit could utilize many of the basic 4 man unit equipment with minimum modification, whereas the 10 man unit would require additional units plus supporting plumbing. A detailed weight analysis would probably show some variance in the weight estimates for the candidate units. The difference, however, as will be shown later in this section, would not significantly alter the results of the analysis.

The total cost can be summarized as

$$C_{Total} = \text{DDT\&E} + \text{Unit Cost} + K \times n \times W_{unit} \times \lambda$$

TABLE 3

CARGO MODULE EC/LSS WEIGHT SUMMARY*

Subsystems	Candidate EC/LS Systems			
	2 Man	4 Man	6 Man	10 Man
Waste Management	250	307	465	720
Food Management	25	25	25	25
Carbon Dioxide Removal and Odor Control	75	163	252	390
Humidity Control	75	110	170	260
Cabin Temperature Control	450	636	990	1500
Heat Transport Loop (Inc. Plumbing Fluid)	1500	1967	3090	4774
Auxiliary Heat Sinks	472	517	800	1240
O_2/N_2 Supply and Press. Control	80	170	284	410
Fire Control	32	32	32	32
Water Management	150	292	450	700
Total Weight	3,109	4,217	6,536	10,121

* Dry Weight Only.

where:
DDT&E (Design, Development, Test and Evaluation) estimates for variable passenger EC/LS systems were based on current Shuttle data and are as follows:

2 man DDT&E	= \$25 Million
6 and 10 man DDT&E	= \$35 Million

Unit cost varies as a function of size:

2 Man Unit	= \$1.0 Million
4 Man Unit	= \$1.5 Million
6 Man Unit	= \$2.0 Million
10 Man Unit	= \$2.5 Million

K = Transportation to Orbit Cost Factor

\$281/Lb maximum
\$160/Lb minimum

n = Number of Units required

W_{Unit} = Unit Dry Weight

Four Man Modular Unit	= 4,217 lb
Two Man Unit	= 3,109 lb
Six Man Unit	= 6,536 lb
Ten Man Unit	= 10,121 lb

λ = Flight Frequency (based on Current Traffic Model)

20 - Ten Passenger Transfer Flights
24 - Six Passenger Transfer Flights
49 - Two Passenger Cargo/Experiment Flights

A basic assumption is that a baseline EC/LSS will be utilized for the Shuttle forward crew compartment and that development cost for that unit is contained within the Shuttle total development cost.

Customized 2, 6, and 10 man EC/LSS candidates were established; sized to meet the specific support requirements of individual flights. Multiple or single based modular (4 man) EC/LSS candidates were established that either met or exceeded the specific flight requirements.

Table 4 summarizes the individual weights and costs associated with the candidate configurations. Under Cargo Module Passenger Transfer - 10 is shown the four possible groupings of EC/LSS units that will satisfy the support requirement. The ten-man unit exactly meets the 12-man Space Station transfer requirements since two of the passengers can be housed in the Shuttle forward crew compartment, and are supported by the Shuttle EC/LSS. The next group utilizes two basic modules and a customized two man unit to exactly meet the

TABLE 4
EC/LSS COST SUMMARY

MISSION	FLIGHT FREQUENCY (λ)	EC/LSS DRY WEIGHT (LB)	TRANSPORTATION TO ORBIT COSTS (MILLIONS OF $)		DEVELOPMENT COSTS (MILLIONS OF $)	UNIT COSTS (MILLIONS OF $)	TOTAL COSTS (MILLIONS OF $)	
			$281/LB	$160/LB			$281/LB	$160/LB
CARGO MODULE PASSENGER TRANSFER - 10								
Payload Module EC/LSS Groupings								
10	20	10,121	56.9	32.4	35.0	2.5	94.4	69.9
4+4+2	20	11,543	64.9	36.9	25.0	4.0	93.9	65.9
4+4+4	20	12,651	71.1	40.5	0	4.5	75.6	45.0
6+4	20	10,753	60.4	34.4	35.0	3.5	98.9	72.9
CARGO MODULE PASSENGER TRANSFER - 6								
Payload Module EC/LSS Groupings								
6	24	6,536	44.1	25.1	35.0	2.0	81.1	62.1
4+2	24	7,326	49.4	28.1	25.0	2.5	76.9	55.6
4+4	24	8,434	56.9	32.4	0	3.0	59.9	35.4
CARGO TRANSFER/ EXPERIMENT SUPPORT								
Payload Module EC/LSS Groupings								
2	49	3,109	42.8	24.4	25.0	1.0	68.8	50.4
4	49	4,217	58.1	33.1	0	1.5	59.6	34.6

requirement. The third group utilizes 3 basic modules and provides an excess of two man support. The last group shows a customized 6 man unit plus a basic module. The other flight candidates follow a similar grouping pattern, employing either customized and/or modular units. The second column gives the flight frequency, the third the EC/LSS dry weight, and the fourth gives transporation to orbit cost (i.e., the product of the second and third columns and the respective cost factor). This column illustrates the penalty associated with excess weight since the 3 basic modules appear at a cost disadvantage of $14.2 millions when compared against the 10 man unit at the $281/lb cost factor. However, when the next two columns are added into the total cost, this situation is completely reversed with the three modules showing a distinct advantage of $18.8 millions over the customized 10 man unit. This illustrates that the DDT&E costs outweighs the transportation costs penalty. This same result occurs in each of the manned flights tabulated.

Figures 10, 11, and 12 show the total cost associated with the various candidates. For the 10 passenger transfer flight, figure 10 shows the minimum cost for both the $281/lb and $160/lb in the group that includes three basic EC/LSS modules. Also shown is the effect of increasing the flight frequency. For example, using the cost factor of $281/lb shows that the flight frequency for the ten passenger transfer must increase to approximately 47 flights before a crossover occurs, which would lead to the selection of a customized EC/LSS. At the lower cost factor ($160/lb), the flight frequency would have to increase to 72 flights before the crossover point could be reached.

Figure 11 for the six passenger transfer flights shows the same trend. The total cost is substantially less than the ten crew passenger transfer because of the lighter weight of the EC/LSS and the launch cost has been reduced, even though the number of flights increased to 24. The frequency crossover is substantially greater with over 50 flights required to reverse the modular approach for the $281/lb cost factor. At the lower cost factor, more than 100 flights would be required to meet the crossover point.

For the cargo transfer/experiment support flights, figure 12 follows the general trend. Even though the flight frequency has increased to 49 flights, the frequency crossover occurs at the 83 flight load for the $281/lb cost factor and over 150 flights for the $160/lb cost factor. An increase in the cost factor would be accordingly reflected in a significant increase in flight frequency before crossover is reached.

The variables in the cost equation most subject to change are the DDT&E and the EC/LSS Unit Weight. The DDT&E could be substantially reduced (50%) without changing the conclusions of this study. The unit weight change is a critical factor, however, if one of the candidates weight changed, one would anticipate a corresponding change in the other candidates weights and the net result differences would be minimized.

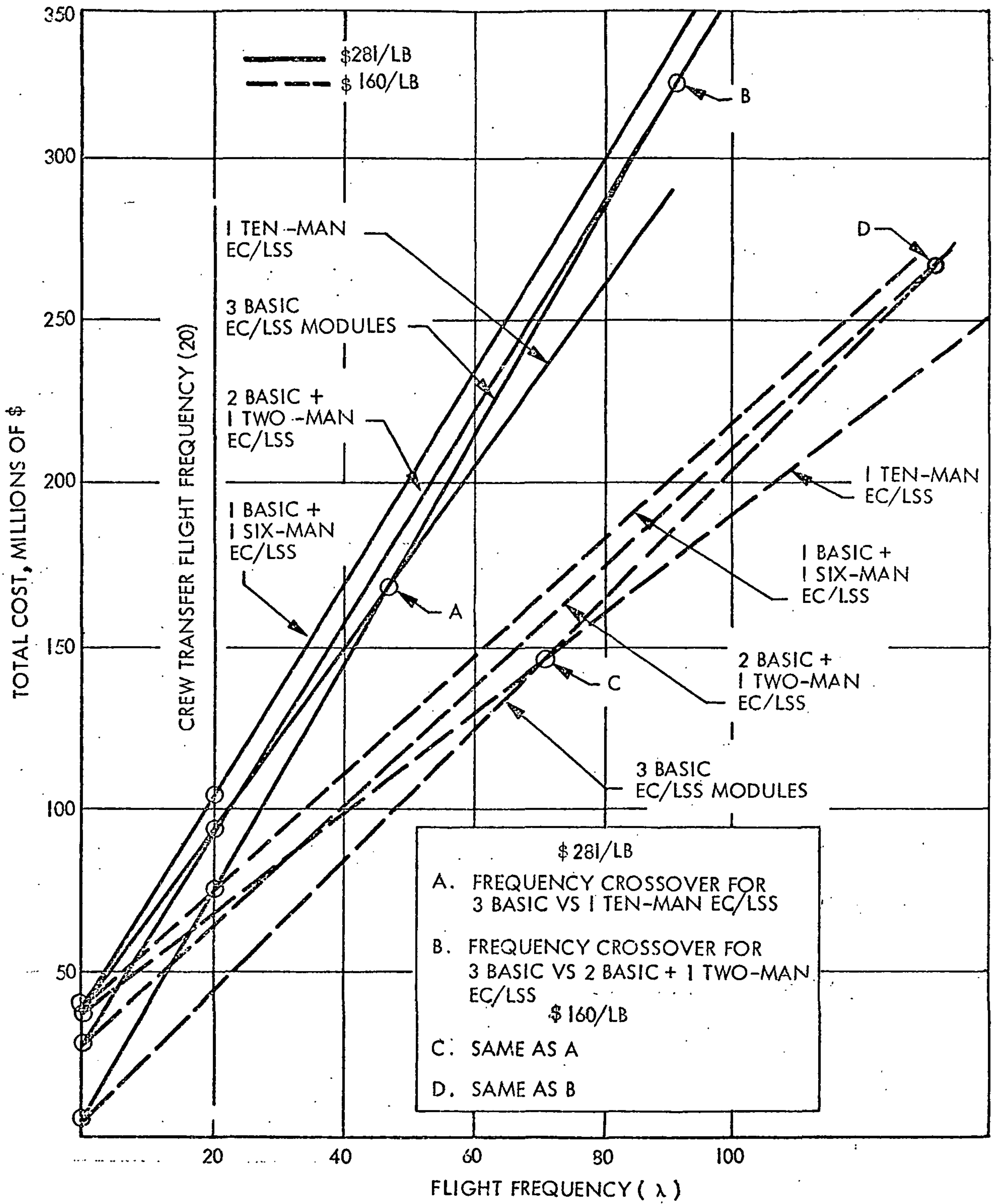

Figure 10.- Total Cost Versus Flight Frequency for Passenger Transfer - 10.

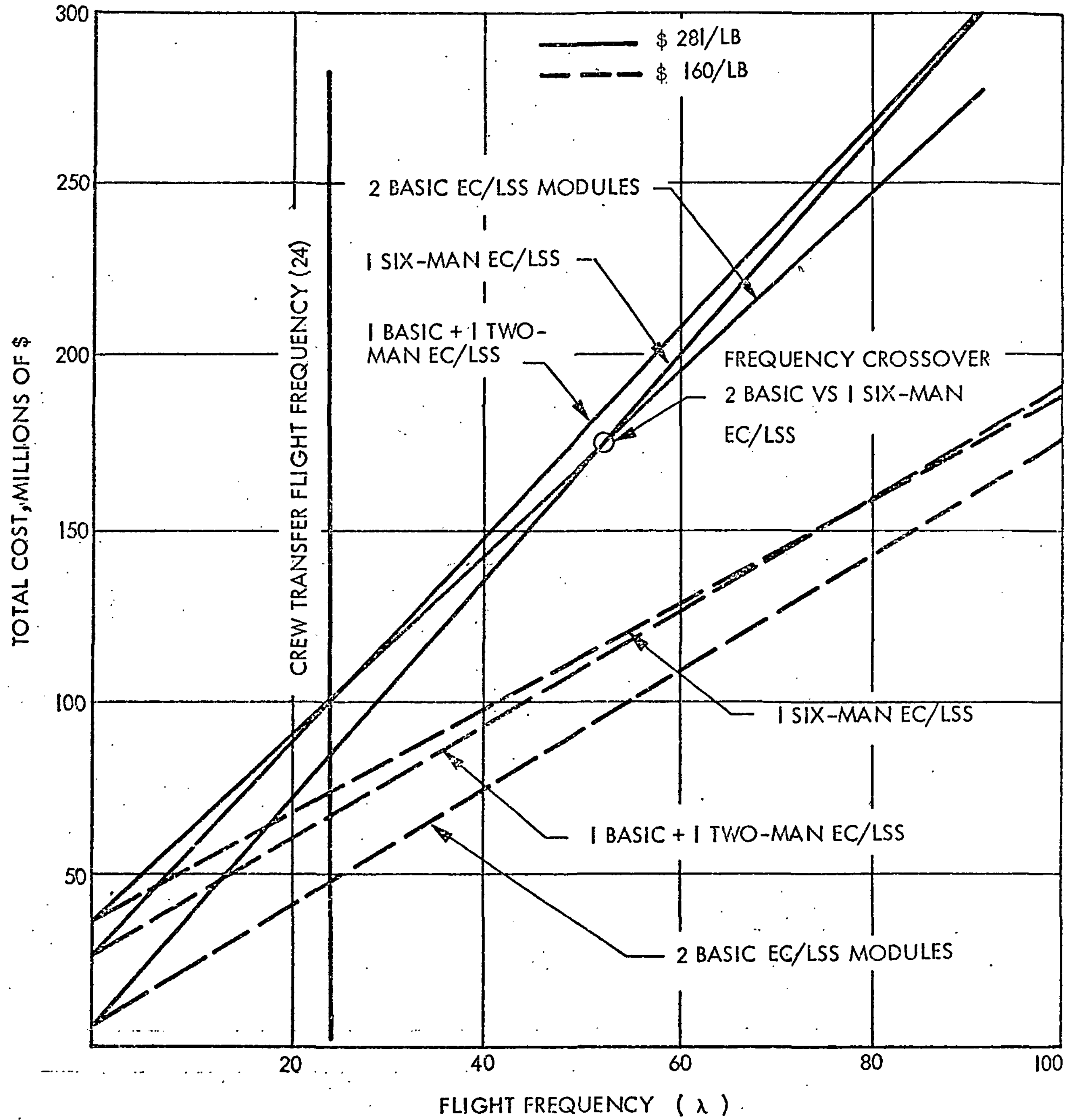

Figure 11.- Total Cost Versus Flight Frequency for Passenger Transfer - 6.

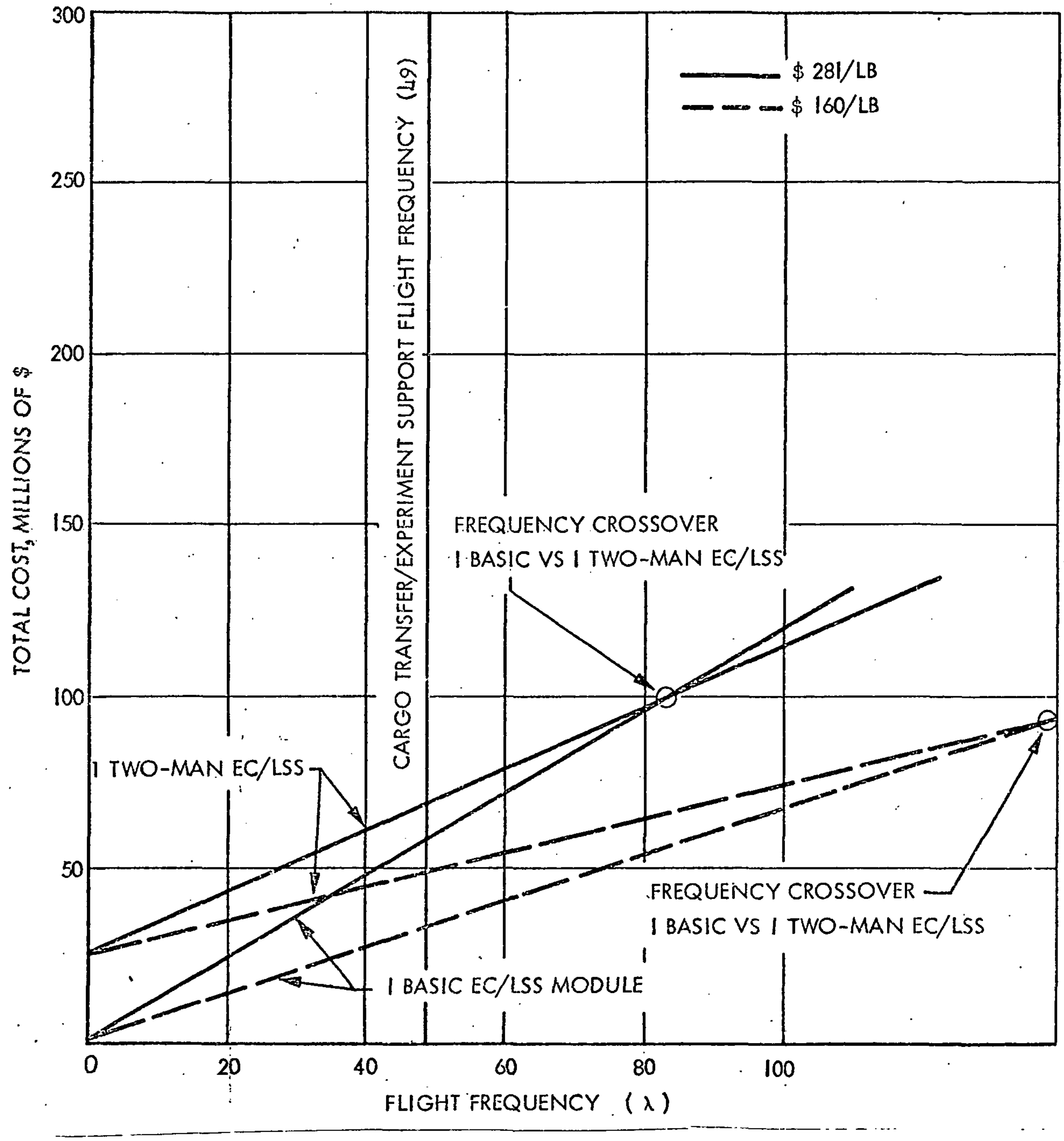

Figure 12.- Total Cost Versus Flight Frequency for Cargo Transfer/Experiment Support (Two-Man).

For example, the dry weights of most subsystems shown in table 3 are made up of certain common fixed weight items plus variable equipment weight that is a function of the number of crew. In some cases, the complete subsystem remains the same regardless of the number of men involved, e.g., Food Management and Fire Control. Because of this fixed common weight, such as plumbing, brackets, supporting structure, etc., differences between 2, 4, 6, and 10 man modules cannot be linear. It is assumed that a certain percentage of the basic module subsystems weight is fixed except for two of the subsystems which remains the same regardless of the number of men involved, e.g., Food Management and Fire Control (57 lb). The weight remainder is a function of the number of men involved. Data generated based on a 25% and 50% Fixed Weight assessment under the assumption listed above resulted in the following comparison:

Configuration	2 Man Unit	4 Man Unit	6 Man Unit	10 Man Unit
25% Fixed Weight (lb)	2664	4217	5770	8876
50% Fixed Weight (lb)	3191	4217	5241	7291
Table 3 Data (lb)	3109	4217	6536	10121

Using these revised weights and calculating the transportation to orbit costs for the total flights (93) in accordance with the cost effectiveness equation results in the total costs as shown below.

	Total Costs (Millions of $)			
	$281/lb		$160/lb	
Configuration	Customized	Modularized	Customized	Modularized
25% Fixed Weight	$226.0	$195.1	$174.0	$115.0
50% Fixed Weight	$220.7	$195.1	$168.9	$115.0
Table 3 Data	$244.3	$195.1	$182.4	$115.0

Note that the minimum customized cost for both the $281/lb and $160/lb cost factor is for the 50% fixed weight case. This occurs since the change in weight between the various candidates is minimized. The total minimum cost of the customized configuration is substantially above the modularized total cost ($49.2 millions for the $281/lb case and $67.4 millions for the $160/lb case). This confirms that the development costs associated with the customized version far outweigh the transportation to orbit costs.

Summary

Based on this analysis and consideration of major factors such as weight, development cost, transportation to orbit cost, and flight frequency, the modularized four-man units are preferred, with some modification in the thermal control area to reduce the 70,000 Btu/hr capability to a level commensurate with the passenger loading. There are logistic parameters that could be evaluated quantitatively which would influence this selection such as; packaging, spare parts cost, maintenance and training costs, and lot buys of initial equipment.

Modularization reduces the unit cost by reducing range of parts, quantity of spares, cost of repair, range of test equipment and reduced procedures. Supply stocking is reduced with a resultant decrease in storage areas. Less packaging engineering design hours are required since modularization requires a "one time only" approach.

Ease of maintenance is achievable with a modular approach since turnaround refurbishment cycle is minimized. Attachment and plumbing is standardized for modular installations. Less maintenance documentation is required due to fewer parts (fewer technical manuals, etc.). Training costs are reduced since a simpler modular system involves a less complex training program. For each parameter, experience shows that the modular approach results in a lower total cost.

The four-man modular unit concept permits flexibility in terms of variable number of personnel and cargo or combination cargo/passenger loads. Extension to 30 day mission is easily handled with addition of expendables. The oversize capability (2 four-man units for 6 passenger load, etc.) provides a degree of redundancy and increase in reliability or will permit changing mission requirements in terms of additional passengers.

The cost effectiveness section has demonstrated that minimization of total launch weight in a Shuttle type of mission is not as important as development costs associated with unique custom-built EC/LSS.

The independence of the cargo module EC/LSS from the Shuttle forward cabin EC/LSS enhances achievement of autonomous payload module operation. In fact, the real possibility that the Shuttle may have to detach from the Space Station due to dynamic considerations and leave the cargo module attached demands cargo module EC/LSS independence.

The cargo module EC/LSS will have a minimum impact on the basic vehicle design because of this independent capability.

Minimization of turnaround time to change modules is assured by incorporating commonality of components, built-in test equipment for fault isolation, and recording of flight and test data for trend analysis.

Technological achievements are required particularly in the packaging, fault isolation techniques, modularity, and reliability to meet the 100 mission requirement.

SPACE SHUTTLE/SPACE STATION INTERFACES

This section investigates the influence of the Shuttle and Station on each other resulting from docking, crew/passenger/cargo transfer, system deactivation, and EVA/IVA. The study identifies the mission operational sequence during the Station and Shuttle docked modes. The major EC/LSS capabilities for the Station were examined for normal, emergency, and overload operating modes. The interface requirements between the Station and the Shuttle Cargo Module were defined.

A Shuttle and Station interaction analysis was performed to determine division of EC/LSS support responsibility. This analysis indicates the basic designs, as currently defined, are adequate to provide the necessary support with minimum or no revision.

A combined Shuttle/Station baseline system EC/LS and alternate candidates were formulated and studied. These candidates considered (1) reducing Shuttle capabilities with more dependence on the Station, and (2) complete Shuttle independence.

Mission and Operation Analysis

Various manned missions have been proposed for the Space Shuttle, the most recent being the establishment of an initial six-man Space Station by assembly of individual crew/cargo modules with subsequent growth to a 12-man Station by attaching additional crew/cargo modules. The evolvement from a six-man to a 12-man Space Station necessitates numerous resupply flights. The initial flight series transports Station modules with a basic supporting crew of pilot and copilot and two cargo handlers. All four are in the forward compartment. The second flight series transports payload modules which are either integrally located within or attached to the Space Station. Two additional cargo handlers are transported in the cargo module. The third flight series provides passenger transport for establishing the six-man occupancy of the Station. Subsequent flights of this series provide personnel interchange and logistic support on a scheduled basis. The final flight series provides passenger transport for establishing the 12-man occupancy of the Station. Subsequent flights of this series provide personnel interchange and logistic support.

These manned missions follow the same operational sequence for docking and transferring men and/or cargo to the Space Station. The prelaunch, launch, orbit placing, reentry and landing sequences have been discussed in the Mission/Vehicle section of this report. The most significant problem is the amount of overload that the different missions impose on the Station. The initial Space Station is occupied by a six-man crew; however, its life support system is designed for 12 men. Therefore, the overload associated with the six man transfer will be non-existent. The 12-man overload (total of 24 crew) imposes the greatest hardship. This is accommodated by designing a 12-man overload capability for the critical elements of the EC/LSS such as a redundant standby unit for CO_2 removal, circulation ducts sized for a total of 24 men, etc.

Following docking and pressure equalization, there will be a short period of activating the airlock; then the crew and passengers egress to the Station.

It is estimated that a maximum time period of five days is involved in a normal crew/cargo transfer with the Shuttle/Cargo Module attached to the Station. This time period is occupied with verification of Station/Shuttle subsystems, deactivation of non-critical Shuttle subsystems, establishing standby condition for critical subsystems, and transferring crew and cargo. Crew transfer will be scheduled early in the docking phase and will require approximately one hour. Cargo transfer into and out of the Space Station using rails and motor driven cargo pallets will require approximately 20 hours.

If the orbit phasing time period is increased, then the maximum time period must be reduced. The effect would be a shorter stay time at the Station for the Shuttle.

Principal crew activities during the Shuttle stay period consist of crew changeover. The Station provides individual crew private quarters with provision for an extra fold-down bunk to accommodate a second crewmen during crew overlap. The Station's sleeping compartments provide 50 sq. ft. of floor area by 79 inch height for each crew member (3,950 cu. ft. - 12 men). This is significantly larger than that afforded by the cargo module (1,800 cu. ft. - 10 men) and suggests that only minimum utilization of the cargo module be made during the crew transfer.

Food and food management will normally be provided by the Station which will have the capability to support 12 crewmen for six months. Both normal resupply and adequate supplies for the crew overlap period will be transported by the Shuttle. Cooking facilities are only aboard the Station with the Shuttle crew utilizing ready-prepared foods when in a detached mode. Hygiene facilities will be provided by the Station during the Shuttle stay period. A garment washer and dryer is provided to accommodate the cleaning of garments on a seven day wash cycle. There are adequate waste management areas aboard the Station for the overload condition (four toilets - 12 man accommodation/toilet). Housekeeping and trash aboard the Shuttle will be packed and stowed for return since deactivating, compacting, and packaging of trash will also be added by the Station for Shuttle return.

IVA and EVA activity will normally be provided by operations personnel aboard the Station, using individually fitted suits and backpacks. The Shuttle will also have a limited EVA capability through non-customized suits with an independent EVA EC/LSS available during unusually hazardous conditions.

Typical crew activities during the five day stay period are summarized in figure 13. Due to volumetric limitations aboard the cargo module and the nature of the crew transfer cycle, it is advisable that the more critical 12-passenger transfer takes place early in the transfer period. It is assumed that the cargo module EC/LSS is operative during this period with limited crew attendance supporting primarily subsystem checkout, cargo interchange, and habitability. A total of 240 manhours per day would be available for all crew related operations assuming a 10-hour daily work shift for the crew overlap complement of 24 men. The Routine Experiment Operation time period can be reduced during this transfer period to permit familiarization for the replacement crew, as well as time to set up, calibrate, and checkout new experiments.

Scheduling of meal periods must be considered during the overload condition since one crew module is designated for this function. The particular module also serves for recreation, and for the galley. Assuming that a meal period occupies a total time period of one hour, then for three meals a day for 24 men, a total of 72 manhours will be involved. Room occupancy is limited to six men for comfort. Consequently, there will be four shifts required for each meal. This implies that as one meal shift, such as breakfast, ends, lunch immediately commences. This crowded schedule can be relieved by scheduling individual meal periods further apart, shortening the meal period, or by designating an additional eating area either aboard the Station or aboard the cargo module. The latter suggestion appears as the most reasonable solution, since there is adequate space aboard the cargo module to serve four persons on a three shift basis. There is adequate food supplied aboard the cargo module, although it is limited to ready prepared food.

Space Station EC/LSS Capabilities

The Station EC/LSS capabilities include: environmental control, oxygen and water recovery, utility provisions for the crew (water, waste, hygiene, food preparation, Station thermal control, experiment thermal control, and water recovery), and special life support (EVA/IVA water and oxygen, emergency oxygen, food, water, and fire detection and control).

Table 5 lists the major EC/LSS subassembly capabilities for the Station. Normal, emergency, and overload operating modes are delineated. The load capability column shows the division of EC/LSS support for each of the three operating modes. For example, in the nitrogen distribution subassembly, one-half of the normal operating support is provided by the lower decks (1 and 2) and the other half by the upper decks (3 and 4). For the emergency modes, total operational support is provided by either upper or lower decks.

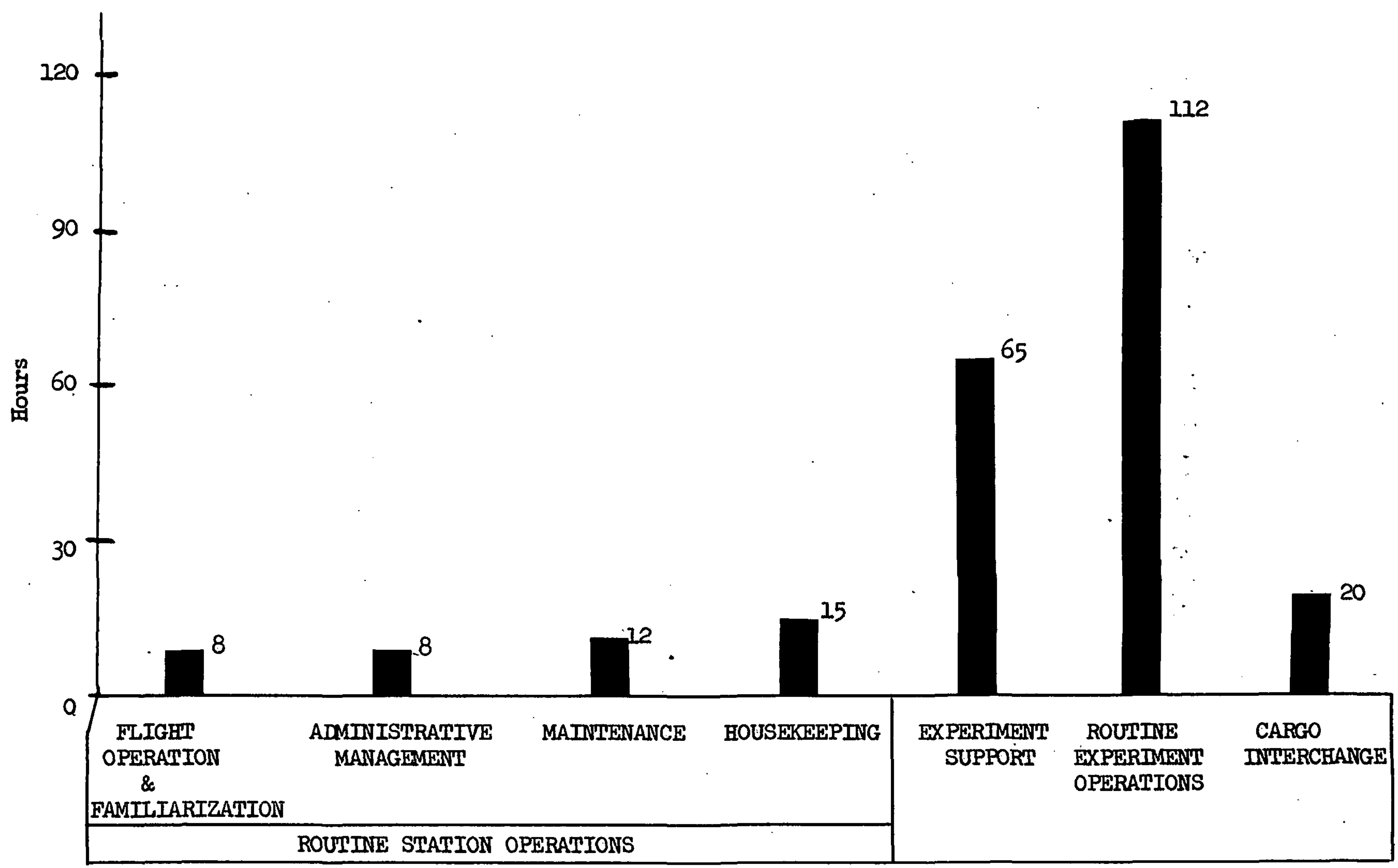

Figure 13.- Typical Daily Work Schedule - 24 Crew (Five Day Crew Transfer Period).

For the oxygen supply subassembly, it should be noted that during the overload operating mode, there is a 12-man load capability in both the lower and upper decks. The overload capability will support 24 men during the overload passenger transfer period.

This philosophy applies for contaminant control, circulation, and waste management. Non-critical elements such as wash and condensate water recovery overload periods can be handled by re-scheduling functions on a more frequent time basis.

The Station modular design has necessitated building in the capacity of utilizing modules simultaneously or under emergency conditions using individual modules. This, in turn, requires redundancy in the critical elements of the EC/LSS. The redundancy is there for the emergency mode; however, it could also be utilized for the crew overlap period.

Shuttle EC/LSS Capabilities

The Shuttle EC/LSS capabilities include all the elements involved in the Shuttle forward crew compartment as well as the cargo module EC/LSS which operates as an independent system. Table 6 lists the major EC/LSS capabilities for the Shuttle forward crew compartment. For the majority of the elements complete redundancy affords fail operational, fail safe failure modes. One-half of the system operates during the normal mode. This is especially true for the critical subsystems such as O_2/N_2 supply and pressure control.

The Shuttle cargo module EC/LSS is similar to the crew compartment EC/LSS except that it employs modular units of the baseline EC/LSS to size the system for four, six, or ten passengers. An important aspect of the Shuttle EC/LSS capability is that there is a possibility that the Shuttle vehicle will detach from the Space Station after attaching the cargo module. This means that the cargo module EC/LSS must be completely independent, particularly the thermal control. Some subsystem support can be provided by the Station. For example, power can be shifted from the Shuttle vehicle to the Space Station for cargo module support.

Space Shuttle/Station Interfaces

Interfaces exist between the Space Station and Shuttle Cargo Module during passenger transfer and/or cargo interchange, and between the Space Station and Experiment Attached Modules. The following data summarizes these interfaces, and presents a general description of the major docking requirements between the Station and the Shuttle modules. Specific interface requirements that influence the Shuttle/Payload EC/LSS design concepts will be delineated. These data will be limited to the following critical aspects: Environmental Protection, Electrical Power, Crew/Habitability, and EC/LSS (ref. 8, 9, 10 and 11).

Alternate candidates for this combined Shuttle/Station baseline system EC/LS were examined which included: (1) off-loading of the Shuttle's expendables with more dependence on the Station, and (2) complete Shuttle independence. These two alternate candidates afford some effectiveness in logistic support; however, the overriding fact that the Station employs a regenerative life support system, redundant subsystems to meet the fail-safe requirement, and incorporates special design features to meet overload conditions, tends to support the recommendation of the combined Shuttle/Station baseline system EC/LS.

Station Docking/Cargo Module Interfaces.- A 20,000 lb. cargo module containing consummables, experiment modules, and/or passengers, will be hard docked to the Station. The cargo module will be extended from the cargo bay. Closing rates of the Shuttle will be controlled such that RCS failure combinations will not result in collision with the Station. Closing rates will be one fps or less. Docking alignment requirements are $\pm$ 5 in. from the centerline, $\pm$ 4° centerline angular misalignment, and $\pm$ 4° in rotational misalignment. Cargo modules will be docked primarily at the deck farthest from the solar arrays.

Docking aids include artificial lighting of the Station and cargo module docking mechanism, two-way closed circuit TV of each docking mechanism, two-way "spoiled" laser transmitters and receivers, and possibly visual targets compatible with TV.

Station Environmental Protection/Cargo Module Interfaces.- The Shuttle environmental protection system will limit the cargo module interior walls and equipment surface temperature to 57°F min. and 105°F max. The Station EC/LSS will provide a nominal 70°F environment within the interior of the cargo module.

Station Electrical Power/Cargo Module Interfaces.- Electrical control cables/umbilicals will be required. These cables will provide control/monitor connection between the Station and the cargo module. The umbilicals will provide valve control signals, monitor circuits for critical parameters (total pressure, pO_2, CO_2 levels, etc.), audio intercommunication, and caution and warning signals.

Electrical power cable/umbilicals will be required to supply the power connection between the Station and cargo module. The umbilicals will provide power for air-circulation fans, and lights, and for cargo module quiescent storage prior to departure for Earth return.

Station Crew/Habitability/Cargo Module Interfaces.- The Station will provide guide rails and powered trolleys for transport of crew and/or cargo between the cargo module and the Station. Guide rail attachments will be required in the cargo module to facilitate installation of the rails which are stored in the Station. The cargo module will provide crew mobility aids in the form of handholds, guide rails, and other devices to facilitate crew locomotion, stabilization, and bracing.

The cargo module will provide storage capacity for the Station resupply expendables and spares.

Station EC/LSS/Cargo Module Interfaces.- N_2/O_2 gas pressurization is required. The supply will be N_2 or O_2 or a mixture thereof, from the Station or the cargo module.

The Station will provide a pressure equalization valve which will be controllable from either the Station or the cargo module.

An air duct for pumpdown of the docking port interface area will be provided by the Station. The pumpdown assembly will be controllable from the Station or the cargo module. Two flexible air ducts, one for supply and one for return of conditioned air will be provided by the Station. The ducts will be used only when the hatch is open and will require implacement by the crew. The ducts normally will be used to provide air for the crewmen in the cargo module when performing cargo transfers or maintenance tasks.

Internal air circulation fans will be required of the cargo module to circulate air between 15 and 100 ft/min for crew comfort.

Station Docking/Attached Modules Interfaces.- Attached modules will be limited to a maximum number of three at any one time. Attached modules will have docking aids similar to those on cargo modules.

Station Environmental Protection/Attached Modules Interfaces.- The attached modules will provide radiation shielding to protect the crew during service and maintenance.

Station Electrical Power/Attached Modules Interfaces.- Average and peak power demand for all experiments will be provided by the Station.

Station Crew/Habitability/Attached Modules Interfaces. - Interfaces are identical to the Station Crew/Habitability/Cargo Module interfaces.

Station EC/LSS/Attached Module Interfaces.- Interfaces between attached modules and Station are as follows:

1) Provide ducting at the docking port to accept and discharge atmosphere from and to the Station EC/LSS.

2) Flow velocity between the Station and attached modules will be between 15 and 100 ft/min. Blowers will be incorporated in the mobile duct-work as required.

3) Atmospheric heat discharged to the Station will not exceed 3415 Btu/hr.

4) Attached modules depressurization will be designed such that any atmospheric dump to space will not exceed allowable angular momentum levels for the Station (60 lb. ft/sec.)

5) The core module will provide the attached modules with an atmosphere having a total pressure control the same as the Station (14.7 psia nominal, with variation to 10 psia). Humidity control will provide an atmosphere at 8 to 12 mm Hg partial pressure of H_2O. CO_2 partial pressure will be maintained below 7.6 mm Hg.

6) A gas transmission system will be provided for feeding resupply gases from the core module to a depressurized attached module.

Station Regenerative Systems/Shuttle Interfaces.- The Space Station EC/LSS provides for a nominal 12-man, 10-year mission capability with resupply every 180 days. A 24-man crew can be accommodated for five days. This latter capability offers potential utilization of the Station EC/LS system by the Shuttle and would result in minimizing expendables storage. The advisability of the Shuttle relying on the Station regenerative system or storing waste products to transfer to the Station for regeneration or disposal is dependent on mission operational considerations and on the level of demand placed on the regenerative systems.

During normal mission operations involving the passenger transfer, the Shuttle will remain docked, however, there is the distinct possibility that the Shuttle might have to disengage due to stability limitations and leave the passenger/cargo module attached to the Station. This would imply that the passenger/cargo module would then either require Station EC/LSS support or have its own independent system. Therefore, from mission operational considerations, support from the Station EC/LSS is considered feasible and desirable for the passenger/cargo module.

The characteristics and capabilities of the Space Station regenerative systems must be assessed to determine the impact during the Shuttle overload condition (24 passengers aboard the Station for five days). Station EC/LS subsystems which are regenerative in nature include: CO_2 management, trace contaminant control, atmospheric composition, water management, and waste management. The molecular sieve CO_2 removal subassembly, which consists of CO_2 absorber and desiccant beds operated on a regenerative cycle, provides for removal of CO_2 from the cabin atmosphere and for concentration of the CO_2 for oxygen reclamation. The collected CO_2 is transferred to a Sabatier reactor where it is combined with hydrogen to form methane and water. The product water plus a suffient amount of makeup water is electrolyzed to provide the metabolic and leakage oxygen requirements. Water electrolysis produces 28.3 lb/day of oxygen for the normal crew complement of 12 men; however, redundant electrolysis cell stacks are provided for the emergency condition. A redundant molecular sieve is provided for the overload condition, with the additional CO_2 removed from the atmosphere and dumped to space.

Regenerable charcoal beds are used in the trace contaminant control subassembly and are sized for the normal crew complement. Each bed absorbs contaminants from the cabin for a period of 10 days and is then desorbed by heating to 572°F and exposing to space vacuum for 10 days. For the overload condition, several approaches might be considered: (1) size the charcoal cartridge for the overload condition initially, (2) carry an extra spare charcoal cartridge (75 lbs.), and (3) utilize an accelerated desorption cycle.

In normal operation, the dual atmospheric control assemblies are operating at half (six-man) capacity in each of the two Station's decks. For periods of crew overload during crew exchange, the additional 12-men can be handled by operating both assemblies at full 12-man capacity.

The Space Station will incorporate basically two water systems: (1) a potable water system operating from a vapor compression reclamation subassembly, and (2) a wash water system operating from a reverse osmosis reclamation subassembly. The primary potable water recovery subassembly recovers water from all urine, experiments, and the dishwater. A redundant subassembly is provided for utilization during crew exchange when twice the processing rate is required. Waste water consisting of wash water, humidity, and Sabatier condensate is received and stored in holding tanks. It is then inserted into the reverse osmosis circulation loop and the reclaimed water is continuously removed and pumped through a series of charcoal and bacteria filters. The subassembly is designed to operate at the nominal process rate for 18 hours per day. The remaining six hours are allocated to maintenance and/or extended operation to account for greater than nominal usage rates.

The waste processing subassembly includes provisions for collection of trash, drying and sterilizing, compaction, and storage prior to Earth return by the Shuttle. The waste processing center consists of a drying chamber, a compactor, and a storage chamber. Separate processing equipment is installed on separate decks, each being capable of servicing the entire Station. Utilizing both subassemblies will accommodate the 12-man overload. Compacted waste is removed and placed in remote storage every seven days.

Based on the characteristics and capabilities of the Space Station regenerative systems, it appears advisable for the Shuttle to utilize their facilities. Complete dependency is not advisable due to possible emergency modes or curtailed capacity due to a maintenance problem.

The major advantages that accrue from storing waste products, such as, urine, excess water, carbon dioxide, and solid waste on the Shuttle for subsequent transfer to the Station for regeneration or disposal is the reduction in returned weight to Earth (even though minimum), reducing the complexity of permanent wet waste storage on the Shuttle, and providing potable water to the Station from both the Shuttle stored urine and excess water sources.

Station/Shuttle EC/LSS Utilization.- Table 7 lists the subsystems making up the EC/LSS and notes whether the subsystem's function is to be carried out by the Shuttle or the Station.

Waste management is best carried out by the Station's subsystem since the crew will all be located in the Station during the majority of the transfer period. Food management is relegated to the Shuttle since the Station's feeding capability is inadequate to handle the 24 crew overload condition. CO_2 removal and odor control and humidity control is best handled by the Station's regenerative subsystems. Cabin temperature control can be monitored and controlled by the Station's subsystem. Heat transport loop is provided by the Station overload capability. The auxiliary heat sinks are only operable during prelaunch, while docked to the Station during peak heat loads, and after landing. If peak heat loads occur during docked mode, Shuttle subsystems can be readily activated. The regenerative O_2/N_2 supply and pressure control subsystem of the Station is designed to handle the overload condition. Fire control capability of the Shuttle (Apollo type fire extinguisher) provides backup. Water management is best handled by Station's subsystem since supply is adequate and Shuttle's supplies can be used as logistical support.

Summary

It is concluded that during the docked mode, Station and Shuttle personnel should rely on the Station for EC/LS. This recommendation is based on the fact that the Station has regenerative systems, greater volume than the Shuttle for habitability and is designed to accommodate a crew overload condition. The Shuttle EC/LSS can be placed in a quiescent operating mode subsequent to personnel transfer to the Station. This mode allows for rapid escape capability by eliminating extensive reactivation or warmup. Since thermal lag is considerable, this system remains operative at a reduced level throughout the docked period. All other EC/LSS elements can be readily activated on demand and can, therefore, be shut off during normal docked operations.

To allow for brief intravehicular movement which may be desirable for cargo transfer, system checkout or for escape, the hatches between the vehicles should be open when the vehicles are docked. Total pressure control for both vehicles is provided by the Station. If personnel enter the Shuttle during the interim for any moderate amount of time, the Shuttle EC/LSS is activated.

During the docked mode, EC/LSS interconnects between the Space Station and Shuttle are not required. Status monitoring and electrical power supply by umbilical connection should be provided. It may be desirable to transfer Shuttle fuel cell product water to the Station during the docked mode.

TABLE 7

STATION/SHUTTLE DESIGN INTERACTION

EC/LS Subsystems	Shuttle Furnished	Station Furnished	Remarks
Waste Management		X	Shuttle facilities will be available if required for personnel convenience.
Food Management	X		* Station's facilities are inadequate for overload transfer period.
Carbon Dioxide Removal and Odor Control plus Humidity Control		X	Operating Station's regenerative subsystems is cost effective.
Cabin Temperature Control Heat Transfer Loop		X	Quiescent operating mode of the Shuttle during the docked mode requires minimum Station support.
Auxiliary Heat Sinks	X		Normally not operable during docked mode.
O_2/N_2 Supply and Pressure Control		X	Regenerating Station subsystem is designed for overload transfer period.
Fire Control		X	Shuttle carries backup equipment.
Water Management		X	Crew's location on the Station during transfer period permits this function to be handled by the Station.

* Recommended that increased feeding capability be provided in the Station.

The analysis of the Station feeding capability shows it to be inadequate to handle the 24 crew overload condition. A solution to the problem is to use the Shuttle cargo module feeding facilities to augment the Station. The disadvantages to this approach are that the module is not as habitable as the Station, that activation of the EC/LSS is required, and that personnel traffic between vehicles is increased. It is recommended, therefore, that increased feeding capability be provided in the Station.

SHUTTLE/PAYLOAD THERMAL CONTROL

The objective of this task was to evaluate requirements for Space Shuttle payload thermal control as related to Shuttle interfaces from prelaunch to orbit; and/or orbit through landing. The thermal interfaces which exist between the Shuttle and its payloads during the various mission phases present significant design problems. Representative payloads were selected which present a cross-section of the thermal control problems which will be encountered during the various mission phases.

Payload Thermal Control Considerations

The factors associated with the payload thermal control requirements and concepts are influenced by conditions which change as a function of mission phase and are presented below.

Prelaunch.- The Shuttle Thermal Protection System (TPS) consists of a high thermal resistance shell which effectively isolates the ascent propellant tanks from the external environment. This results in a very low internal temperature unless high purge gas flow rates and temperatures are provided. For this reason, the payload compartment wall should be a high thermal resistance composite to minimize heat loss from the payload compartment and/or use purge gas flow. Figure 14 illustrates the effect of N_2 purge gas flow rates on the payload compartment wall temperature. Uninsulated walls may experience temperatures as low as 350°R with no Shuttle purge or with a non-flowing stagnant N_2 purge. Wall temperature can be increased to over 500°R by Shuttle purge rates on the order of 1500 lb/min (total Shuttle purge, excluding payload compartment). The wall temperature can also be increased with an ambient temperature N_2 purge of the compartment during doors closed periods. The payload compartment wall temperature can be significantly increased by use of insulation as illustrated in figure 14.

Limitations are necessarily imposed on cargo compartment door opening after propellant tanking to prevent water condensation on potentially cold wall surfaces. Propellant tankage payloads loaded before liftoff also will have restrictions imposed for purging and door closure to prevent air and water vapor condensation on cryogenic tank insulation and structural surfaces.

Ascent.- Payload/Shuttle ascent phase interface considerations include cooldown of the interior compartment wall as purge gases are vented and heating of external skins due to aerodynamic heating. Heating during ascent is slight because of the TPS so that the average compartment internal wall temperature does not increase significantly.

Orbit.- The payload/Shuttle interface during the orbital phase has a significant impact on thermal response and operational characteristics. Vehicle orientation and altitude and radiating surface properties effect heat rejection capability. The interface between Shuttle and payload radiator systems is affected by concepts which apply the radiators to Shuttle cargo compartment doors. The emerging concept resulting from Phase B contract studies suggests installation of radiators to the interior of doors which remain open on orbit. The open door concept also allows for installation of a radiator to a payload module within the Shuttle compartment. A second approach involves dual doors, one of which deploys on orbit and incorporates a radiator. A second door closes back over the payload compartment to protect the compartment on orbit. This concept will not allow for incorporation of a radiator system into a payload module remaining on the Shuttle.

Reentry.- Thermal protection requirements for payloads returning to Earth vary widely, depending on whether they are manned, inert, or contain cryogenic fluids (potential with tanker modules). Requirements pertinent to each category are discussed in subsequent sections.

Thermal Control of Space Shuttle Payloads

The Space Shuttle provides the capability for delivery, servicing, and retrieval of such a broad range of payload types that virtually all previously employed approaches to spacecraft thermal control will be involved. These missions impose relatively severe thermal constraints on payload components and Shuttle/payload interfaces. The flexibility required to handle these various classes of payloads is not always compatible with the type of specialized thermal control subsystems previously employed on launch vehicles.

Most payloads require consideration of only heat rejection and temperature control. Propellant tanking payloads (cryogenic) impose additional interface problems involving purging, prevention of condensation and thermal isolation to prevent propellant heating.

This section will highlight the critical areas of thermal control for specific payloads within selected mission categories as they relate to Space Shuttle operations. These payloads illustrate the typical problems associated with all other payloads in the mission category.

The missions selected were Space Station Resupply, Satellite Placement and Retrieval and Propellant Delivery. The Satellite Servicing and Maintenance mission has similar thermal control requirements to the Satellite Placement and Retrieval mission and the Short Duration Orbital mission has similar requirements to the Space Station Resupply mission. Consequently, the latter two missions are not highlighted in the following analysis. Table 8 shows the parameters having a significant effect on thermal control for the selected payloads.

TABLE 8
REPRESENTATIVE SPACE SHUTTLE MISSION AND PAYLOADS

Mission	Duration	Crew Size	Operative Mode	Altitude & Inclination	Heat Load	Remarks
<u>Space Station Resupply</u>						
(NSU-1) Personnel Transfer	7 Days	4 Plus 10 Pass.	Attached	270 nm $i = 55^{\circ}$	7000 Btu/hr	Complete EC/LSS.
<u>Satellite Placement and Retrieval</u>						
(NSP-1) Lower Magnetosphere	7 Days	2 - 4	Detached	100 x 2000 nm Elliptical $i = 90^{\circ}$ $i = 0^{\circ}$	40 watts (Experiment power)	Passive thermal design (with heater) equipment 0 to 35°C.
<u>Propellant Delivery</u>						
(NSU-3) Tug Fueling	7 Days	2	Orbital Propellant Storage Facility	270 nm $i = 55^{\circ}$	Large LH_2 tank	Multilayer insulation and purge bag.

Space Station Resupply Mission.- The primary mission of the Space Shuttle is to transport personnel, experiment modules and cargo to and from the Space Station. The life support requirements of the passengers present the most urgent thermal control problems for this type of mission.

A brief description of the Shuttle thermal control subsystem is presented below as a point of departure for payload module EC/LSS design. The subsystem shall have the capability to provide thermal control, as required, for electronic equipment in the personnel compartment, payload bay and remote equipment bays. The baseline heat rejection subsystem is composed of the following elements:

- An onboard heat exchanger for use during ground operations and which can also be utilized while docked to the Space Station.
- A hydrogen evaporator to provide heat sink capabilities during ascent and reentry.
- A combination radiator-sublimator to maintain required heat rejection during orbital operations.
- A vapor compression cycle for ferry flight.

The total heat rejection capability for this system is 70,000 Btu/hr. Water is selected as the coolant in the crew compartment due to its high specific heat and in order to eliminate toxic contaminant release into the crew compartment. Freon 21 is used in the external heat rejection loop because of its low freezing temperature and low viscosity.

The preliminary configuration and sizing requirements for the payload module EC/LSS heat rejection subsystem are based on the following considerations:

- A maximum payload module heat rejection capability of 7,000 Btu/hr is required. This rate will allow for the sensible and latent metabolic requirements of ten passengers (550 Btu/hr per passenger) and 1,500 Btu/hr for subsystem equipment heat dissipation (pumps, fans, etc.).
- The orbiter EC/LSS will provide the required heat sinks during the prelaunch, ascent, reentry and ferry flight portions of the mission.
- The payload module heat rejection subsystem must be autonomous for orbital operations.

The payload module heat rejection requirements are approximately ten percent of the orbiter EC/LSS design loads; therefore, the use of Shuttle heat sinks for other than orbital portions of the mission can be accomplished by the addition of expendables. A radiator will supply the payload module EC/LSS orbital heat rejection capability at the least cost and lowest weight. This system will also provide the greatest flexibility for meeting extended mission requirements. Since there is a requirement for orbital payload module operations while separated from the Shuttle, e.g., either docked to the Space Station or free flying, the addition of radiator area to the internal surfaces of the cargo bay doors to accommodate payload module heat rejection during this phase of the mission was not considered. The radiator would be located on the payload surface directly beneath the cargo module doors. During orbital operations while aboard the Shuttle, this concept would require that either the cargo bay doors remain open or that the orbiter radiators be sized for 77,000 Btu/hr (70,000 Btu/hr generated by the orbiter plus the 7,000 Btu/hr payload module heat rejection requirement).

As in the crew compartment, water is selected for the coolant within the payload module and Freon 21 is used in the external heat rejection loop. This arrangement provides a low temperature coolant in the radiator subsystem. An interchange heat exchanger is used to couple the two fluid loops.

An interchanger outlet temperature of approximately 40°F is required in the cabin loop to meet cabin humidity control requirements and efficiently accommodate metabolic heat loads. The Freon loop requires an interchanger inlet temperature of 35°F. Radiation sink temperatures are presented in figure 15 as a function of radiator α_S/ϵ ratio for representative thermal environments which will be encountered at an altitude of 270 nm. The thermal environment to which the radiator is exposed can be reduced by imposing α_S/ϵ attitude restrictions on the payload module. The calculated radiation sink temperatures do not include interaction with either the orbiter or the Space Station.

Surface-finish coatings developed as Optical Solar Reflectors (OSR) that offer a low α_S/ϵ ratio of .05/.8 are currently in use at LMSC. This value is twice as low as those obtainable from other surface coatings. The basic materials selected for OSR are vapor-deposited silver and fused-silica substrates. Silver is vapor deposited on the silica, and an overcoat of vapor-deposited Inconel is added to inhibit corrosion. An OSR is stable in the space radiation environment, and the basic materials can withstand temperatures from -320°F to over 800°F.

The cost of an OSR surface finish (approximately $700/ft^2) may dictate the selection of a coating with less desirable properties. Flexible optical solar reflectors (FOSR) are available with α_S/ϵ ratios of .10/.82 (silver deposition on FEP teflon) and .17/.81 (aluminum deposition on FEP teflon). The cost of these surfaces are approximately $25/ft^2, however, they are subject to ultraviolet degradation. The solar absorptance of the silver FOSR will increase from .08 to .1 after 4000 equivalent sun hours. The paint systems offer the most economical thermal control surfaces (approximately $5/ft^2); however, their resistance to ultraviolet degradation is very low. The α_S/ϵ ratio of typical white paint systems will degrade to .3/.9 after exposure to ultraviolet radiation. The effect of orientation on radiation performance can be demonstrated by a comparison of OSR sink temperatures for the following cases:

- o Full sun on radiator and no view of the Earth (Tsink = 355°R)
- o Full sun on radiator and edge on view of the Earth (Tsink = 405°R)
- o Full view of the Earth and maximum Earth albedo (Tsink = 430°F).

The radiator area required for a specified energy rejection rate at these various sink temperatures can be determined using figure 16. For example, at a Freon flow rate of 700 lb/hr and a heat rejection requirement of 7000 Btu/hr, the radiator inlet and outlet temperatures are 75°F and 35°F respectively. The flow rate in the water loop for this case would be 175 lb/hr and the interchanger inlet and outlet temperatures are 80°F and 40°F. Radiator area requirements corresponding to the above sink temperatures are 103 ft^2, 127 ft^2 and 200 ft^2, respectively.

The baseline payload module heat rejection subsystem would be sized for ten passengers and the lower rejection requirements imposed by two, four or six passengers could be accommodated using radiator bypass techniques. Minimum radiator sizes can be obtained by placing the most stringent attitude restrictions on the ten man module, and relaxing these restrictions as the passenger load is decreased. The resupply missions, for example, which typically require two passengers, could utilize the basic payload module energy rejection subsystem with no attitude restrictions.

Satellite Placement and Retrieval.- The Shuttle will be used to economically place satellites in Earth orbit. The cargo bay has the capability of handling payloads up to fifteen feet in diameter and sixty feet in length. Multiple deliveries of smaller satellites are also being planned.

Since orbit plane changes require significant amounts of propellant, the selection of a group of satellites for multiple delivery will probably be made on the basis of commonality with orbital parameters (e.g., inclination). These payload groupings must also take into account the possible conflicting temperature requirements of the satellites in order to minimize the weight and volume penalties associated with thermal isolation.

It is conceivable, for example, to fabricate a set of reusable, low weight bags of varying dimensions, to protect satellites which have temperature requirements that are incompatible with either the cargo bay environment or the balance of the payload. A low weight multilayer insulation system (ρ = 3 lbs/ft^3) which will provide excellent thermal isolation in a vacuum environment is an ideal candidate for this application.

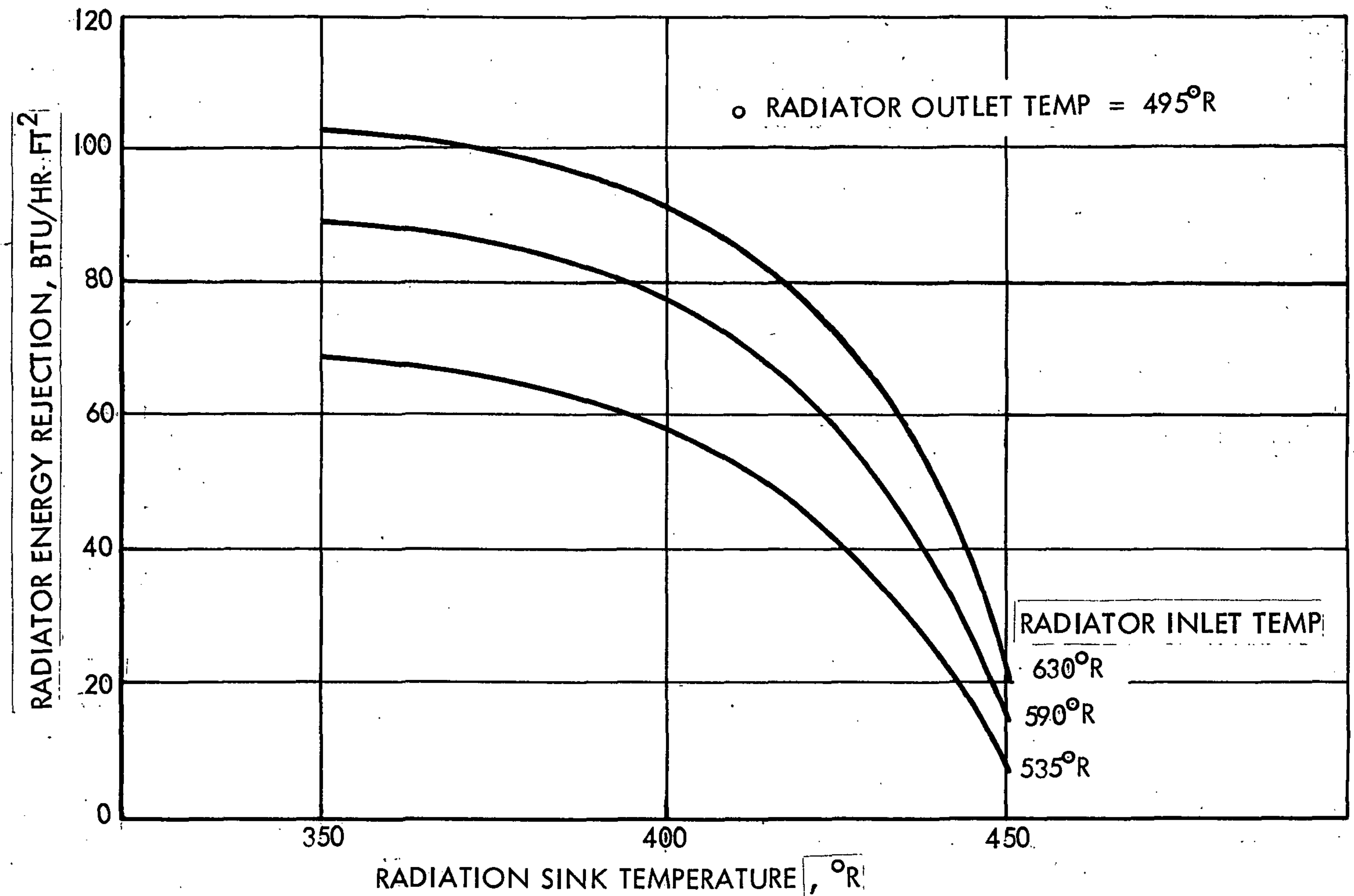

Figure 16 Radiator Energy Rejection vs Radiation Sink Temperature

The lower magnetosphere satellite was selected as an example of a payload which is a candidate for multiple delivery. Pertinent system characteristics and requirements in addition to the orbital parameters and power dissipation listed in table 8 are given below:

Satellite weight	=	1200 lbs.
Launch dimensions	=	2.5' dia. x 4'
Expected lifetime	=	3 years

The satellite is similar in general description to the ISIS-X system. Passive thermal design techniques are utilized to maintain the temperature levels of internal components in the range of 0^{o} to $35^{o}C$ on orbit. The spin stabilization system used for altitude control will have the additional effect of damping out satellite skin temperature gradients imposed by the external orbital environment.

During prelaunch periods, it is anticipated that the floor of the cargo bay will approach $350^{o}R$ as a lower limit due to its proximity to the ascent LOX tanks ($T = 162^{o}R$). This temperature level will, of course, be a function of tank and cargo bay interface insulation thicknesses as well as nitrogen purge rates, however, using the $350^{o}R$ floor temperature as a "worst case" assumption, it is reasonable to postulate an average cargo bay N_2 purge gas temperature of $400^{o}R$ during the prelaunch phase.

Multilayer insulation heat leak (watts/ft^2) versus insulation thickness is presented in figure 17 as a function of payload temperature level. The predicted heat leaks are based on a mean cargo bay N_2 purge gas temperature of $400^{o}R$ and assume the multilayer insulation is permeated with GN_2 ($k = .012$ Btu/hr ft ^{o}R). Energy must be supplied to the satellite system within the purge bag in order to maintain required temperature levels. The lower magnetosphere satellite, for example, would require on the order of 200 watts during the prelaunch phase for a multilayer bag thickness of 2 inches in order to maintain component temperatures in the range of 0^{o} to $350^{o}C$. A heater on a thermostat set to cut off at a temperature level of $20^{o}C$ would be used to supply the required makeup energy.

The effectiveness of the multilayer bags will increase markedly as the N_2 gas is vented during ascent. The thermal conductivity per unit insulation thickness (k/X) will approach .005 Btu/hr ^{o}R for boundary temperatures on the order of $0^{o}F$ under vacuum conditions. The selection of a multilayer bag surface finish with a low solar absorptance to infrared emittance ratio (i.e., white paint α_S/ϵ .3/.9) will reduce the possibility of overheating the payload once the cargo bay doors are opened on orbit.

The heat leak through the multilayer insulation bag during reentry is estimated at approximately 1000 Btu; assuming that all of this energy goes into the 150 lb. of mission equipment, the resultant equipment temperature rise is on the order of $35^{o}F$.

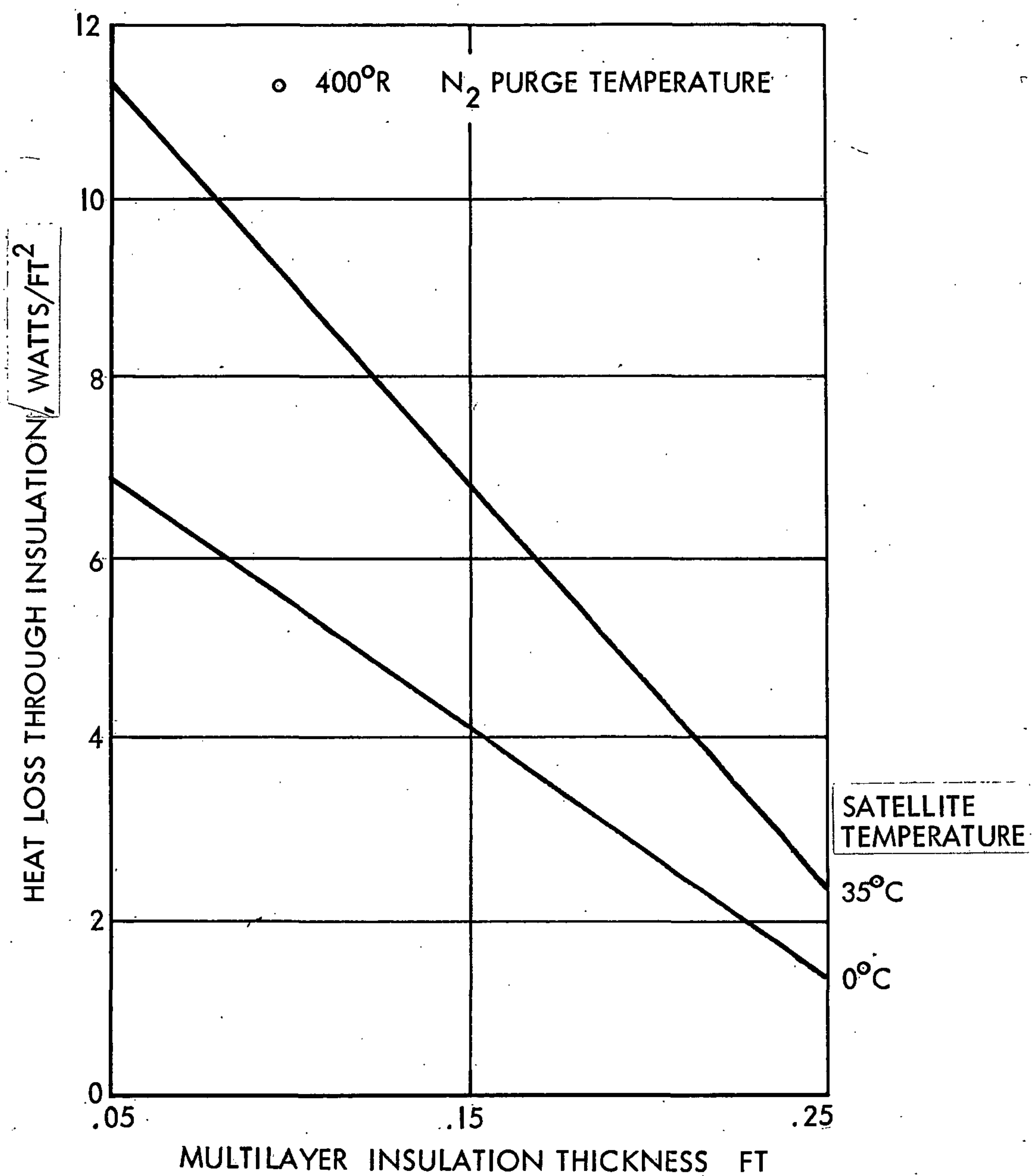

Figure 17 Multi-Layer Insulation Bag Prelaunch Heat Loss vs Insulation Thickness

Propellant Delivery.- The object of this section is to study the ground hold and reentry heat transfer that will occur to an LH_2 propellant tank in the payload bay of the Space Shuttle vehicle. One objective will be to predict the heat transfer rate that will occur into the LH_2 tank during the above flight phases, and the rate of LH_2 boiloff that may occur. The other objective of this study is to predict the purge bag temperature and determine if gas or water vapor condensation is possible on the purge bag during ground hold or reentry operations.

The LH_2 tank payload chosen for this study is the Reusable Nuclear Vehicle (RNV) Propellant Tank Module shown in the sketch of figure 18. The tank is 15 feet in diameter and about 60 feet long, with a surface area of 2815 ft^2 and an internal volume of approximately 9500 ft^3. As shown in the cross-sectional sketch of figure 18, the tank is surrounded by two meteoroid shields with a total thickness of 0.028 inches of aluminum. The tank wall has an average thickness of about 0.056 inches of aluminum, and is surrounded by a multilayer insulation system of up to 1.0 inch in thickness.

For the purposes of this study, the helium purge bag was assumed located at the outer meteoroid shield location, so that a helium purge bag thickness of 2.0 inches will exist between the LH_2 tank wall and the purge bag. The actual purge bag location could be anywhere from the inner to outer shield location. An inner shield purge bag location should result in lower heat leaks and purge bag temperatures than the model assumed here, but the inner purge bag model would be more difficult to analyze. With a full 1.0 inch thickness of multilayer insulation, the purge bag would most likely have to be located outside of the inner meteoroid shield to allow outgassing of the helium during ascent. For these reasons, the purge bag was assumed located just inside of the outer meteoroid shield.

Since this tank effectively fills the Space Shuttle payload bay, an annular thickness of 4.0 inches was assumed to exist between the structural walls of the payload bay and the purge bag outer surface of the LH_2 tank. This annular space would be filled with nitrogen gas (GN_2) during ground hold and ascent, while it would refill with atmospheric air during reentry. Two major forms of heat transfer will occur from the payload bay structure to the purge bag outer tank surface. One is radiation heat transfer across the annulus, while the other is conduction/free convection heat transfer across the gas annulus, which are designated as q''_T, the heat rate per unit tank area, where the nomenclature $q'' = \frac{q}{A}$ is used in this study.

Two major objectives were considered in this study. The first was to analyze the ground hold performance of a full RNV propellant tank loaded with LH_2 under steady state conditions. The results of this analysis would be a prediction of heat leak and boiloff rates, purge bag temperature, and an estimate of condensation problems. The second objective was to analyze the reentry performance of an empty RNV propellant tank under the transient reentry conditions that can be expected to occur on a medium cross-range reentry trajectory. This analysis should result in a transient prediction of the purge bag and propellant tank temperatures, and allow an estimation of the condensation problems that may exist on reentry.

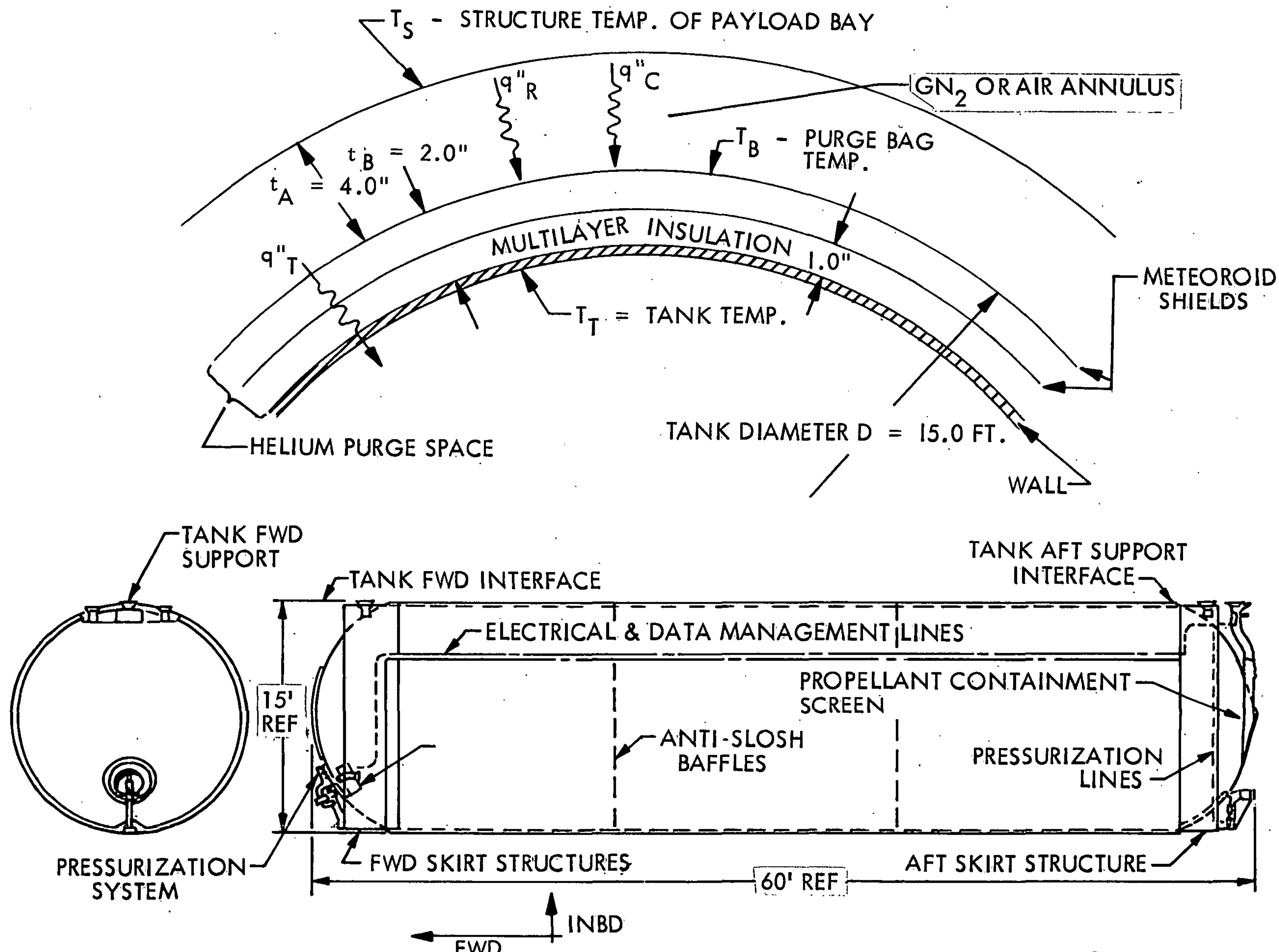

Figure 18.- RNV Propellant Module Structure.

For this steady-state ground hold analysis, referring to figure 18, helium gas is present in the purge bag and N_2 gas is assumed in the annular space between the tank and payload bay, both at about one atmosphere pressure. With a full LH_2 tank, the tank wall temperature is assumed to be $T_T = 40^oR$ and the average payload bay structure temperature $T_S = 400^oR$ should exist in the fully loaded Space Shuttle during ground hold. For steady-state operation, the total heat rate through the nitrogen gas annulus (q''_R + q''_C) must equal the conduction heat leak through the insulation into the tank, q''_T. These heat rates can be estimated as below:

$$q''_T = \frac{k_{He}}{t_B} (T_B - T_T)$$

where:

q''_T = heat rate per unit tank area

k_{He} = helium gas conductivity in the annulus

t_B = helium purge gas thickness = 2.0 in.

T_B = purge bag temperature

T_T = LH_2 tank wall temperature (fully loaded during ground hold).

$$q''_R = F_\epsilon \quad (T_S{}^4 - T_B{}^4)$$

where:

q''_R = radiation heat transfer across the annulus between the payload bay structure and the purge bag outer tank surface.

σ = Stefan-Boltzmann constant

F_ϵ = 0.1, assumed emissivity factor between purge bag and bay wall.

T_S = average payload bay structure temperature (fully loaded Shuttle during ground hold.

From reference 12, the heat transfer conductance k_c of the annular space dimensions shown on figure 18 can be estimated from the following relations:

$$q''_C = \frac{k_C}{t_A} (T_S - T_B)$$

where:

k_C = conduction/free convection heat transfer conductance.

t_A = annular thickness existing between the structural walls of the payload bay and the purge bag outer surface of the LH_2 tank.

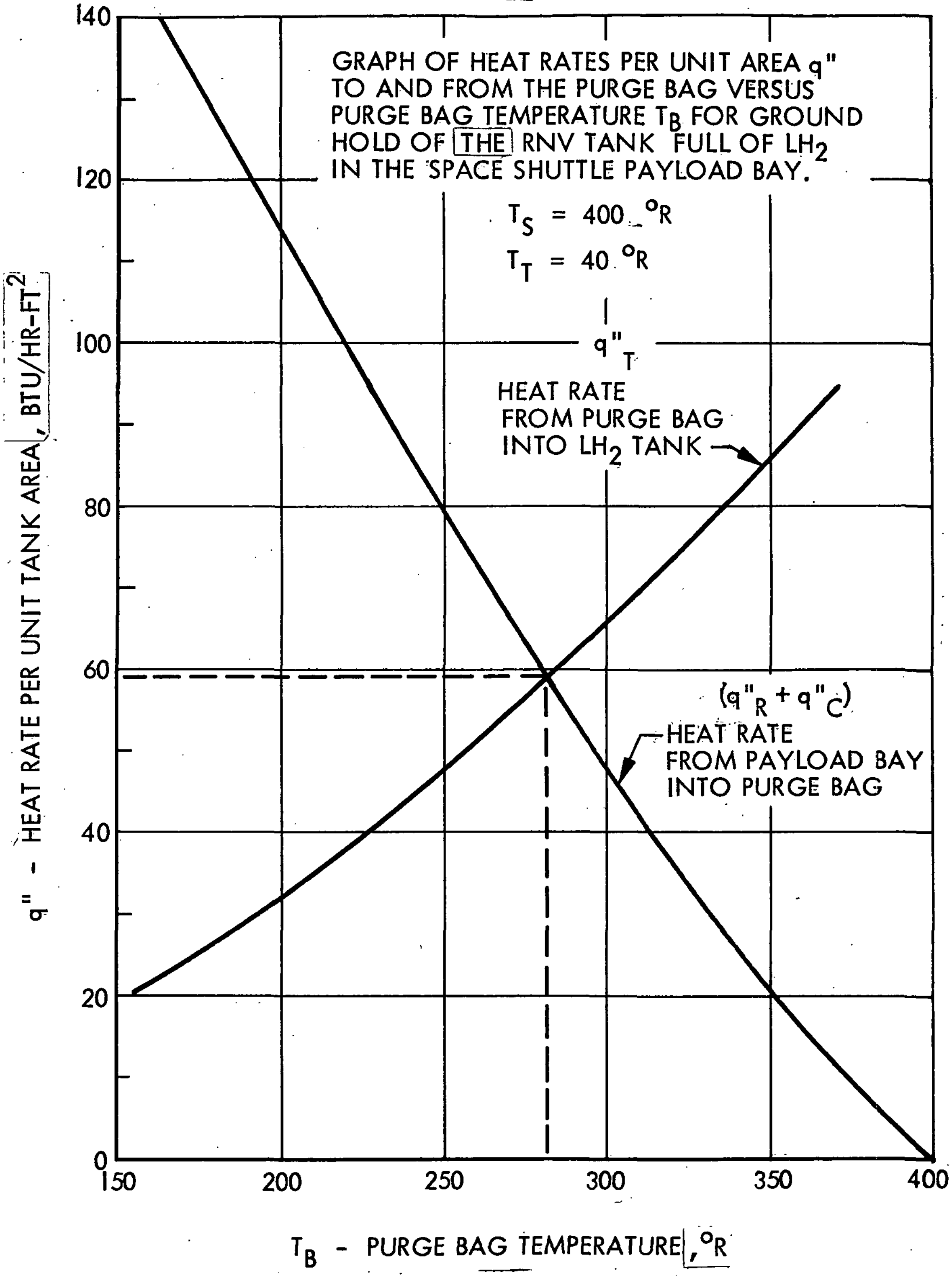

Figure 19 - Heat Rates vs Purge Bag Temperature - Ground Hold.

k_C = k, for Gr $\leq 10^8$

k_C = $0.01\, k\, (Gr)^{\frac{1}{4}}$, for Gr $> 10^8$

where:

k = gas conductivity in the annulus

Gr = Grashof number

$$Gr = \frac{\rho^2 g D^3}{\mu^2} = \frac{(T_S - T_B)}{T_S}$$

ρ = gas density

g = acceleration

D = purge bag diameter

μ = dynamic viscosity

Based upon the above relations, the total heat rate across the nitrogen gas annulus $(q''_R + q''_C)$ and the heat rate into the tank, q''_T are plotted as a function of the purge bag temperature T_B on figure 19 for the ground hold case. The intersection of these two heat rate curves, where $q''_T = (q''_R + q''_C)$ represents the steady state solution of the tank heat rate and purge bag temperature. From figure 19, this solution yields $T_B = 280^\circ R$ purge bag temperature and a tank heat rate of $q''_T = 60$ Btu/hr ft^2.

The ground hold purge bag temperature of $T_B = 280^\circ R$ will not result in any gas condensation in a dry GN_2 environment in the annular space during ground hold. The tank heat leak q''_T of approximately 60 Btu/hr ft^2 will result in LH_2 boiloff rate at one atmosphere pressure of 885 lb/hr for the RNV propellant tank module payload under ground hold conditions.

During the reentry of an empty RNV propellant tank in the Space Shuttle, an estimate is made of the coldest temperatures that could be expected of the propellant tank and the purge bag. For this analysis, it was assumed that the payload bay structure temperature T_S was initially $T_S = 530^\circ R$ at the start of reentry. The tank was assumed to be emptied of LH_2 just prior to reentry, so that the tank temperature was $T_T = 40^\circ R$ prior to pressurization. The meteoroid shields and outer surface of the multilayer insulation was assumed to be $T_B = 500^\circ R$ before pressurization.

At 12 minutes before entering the Earth's atmosphere, it is assumed that nitrogen gas pressurant is applied to the inside of the tank to prevent collapse, and the helium purge gas is applied to the multilayer insulation. These gases within the tank and purge bag will result in temperature equilibrium of the purge bag contents, over a time period of about 10 to 15 minutes. This equilibrium temperature for the purge bag and contents would approach $T = 326^{\circ}R$, if the nitrogen and helium gases were added to the tank at $T = 500^{\circ}R$. Approximately a $20^{\circ}R$ differential is required between the tank temperature and the purge bag temperature to account for initial radiation heat transfer effects. Therefore, the initial temperature of the tank contents was assumed to be $T_T = 320^{\circ}R$, and the initial temperature of the purge bag was assumed to be $T_B = 340^{\circ}R$ at reentry. These start temperatures for the tank contents and purge bag should represent the minimum possible values, assuming the tank wall was at LH_2 temperature just prior to reentry.

With the start temperatures T_T and T_B fixed, it is possible to solve for T_T and T_B as a function of time t during the reentry phase. The heat rates q''_R, q''_C, and q''_T during reentry are computed from the same relations given in the ground hold analysis. The total heat rate to the purge bag q''_{TOT} is then found from:

$$q''_{TOT} = q''_R + q''_C$$

The heat rate that will raise the tank contents temperature is then q''_T, whereas the heat rate that will raise the purge bag temperature is:

$$q''_B = q''_{TOT} - q''_T$$

The temperature rise ΔT_T and ΔT_B over a time period Δt is:

$$\Delta T_T = \frac{q''_T}{C''_T} \Delta t$$

$$\Delta T_B = \frac{q''_B}{C''_B} \Delta t$$

where $C''_T = 0.30 \text{ Btu}/^{\circ}R \text{ ft}^2$ = tank, gas contents, and half of multilayer insulation capacity per unit area.

$C''_B = 0.10 \text{ Btu}/^{\circ}R \text{ ft}^2$ = purge bag, meteoroid shields, and half of multilayer insulation capacity per unit area.

The results of this transient solution are shown on the graph of figure 20. The assumed aluminum payload bay structure temperature T_S, which was used in this analysis, is shown in the top curve, with $T_S = 530^{\circ}R$ ($70^{\circ}F$) at the start of reentry and rising to $T_S = 660^{\circ}R$ ($200^{\circ}F$) at 10 minutes after landing, when structural cooling would begin in the Space Shuttle. The dew or freezing temperature T_F of the atmosphere (ref. 13) is shown on this graph for the air that will surround the purge bag during atmospheric entry. This T_F curve assumes a dewpoint temperature of $70^{\circ}F$ at the sea level landing field.

The computed RNV tank contents temperature T_T rises from an assumed value of $T_T = 320^oR$ at the time of helium purge to $T_T = 420^oR$ at 10 minutes after landing. The purge bag temperature T_B was found to rise from an assumed initial temperature of $T_B = 340^oR$ to a final $T_B = 540^oR$ at 10 minutes after landing. As can roughly be seen from the T_T and T_B curves, only radiation heat transfer occurs to the purge bag from the payload bay structure during the first 15 minutes, followed by radiation and conduction heat transfer up to about t = 55 minutes. After that time, free convection heat transfer dominates in the gas annular space, and both the purge bag and tank contents temperatures rise rapidly. At the time of landing, t = 78 minutes, the purge bag temperature is $T_B = 500^oR$ (40^oF).

The hashed area between the T_F and T_B curves on figure 20 shows the region where water vapor condensation and/or freezing can occur on the purge bag. This condensation time would exist for a period of about 12 minutes, and would be most active at the time of landing, where T_B is about 30^oF lower than the dewpoint. At all other times during reentry, the purge bag temperature is greater than the dewpoint, and is always greater than the GN_2 or GO_2 liquification temperature of air.

There are two rather simple methods that might be used to obtain $T_B > T_F >$ at landing, so that no water condensation would occur on the purge bag. One method would be to raise the tank and purge bag temperature at the start of reentry, so that the purge bag temperature would be well above the dewpoint temperature during the landing phase. This would easily occur in this model if reentry were to start with an average purge bag-tank temperature of $\geq$ $T \geq 400^oR$, which would exist with a warm tank on orbit. However, the minimum temperature tank assumed in this analysis could be warmed to that level in orbit by pressurizing the helium purge bag with He and the tank with GN_2 about two hours before reentry. With no helium gas in the purge bag, the tank temperature will rise about one ^oR/hr due to heat leaks. With helium gas in the purge bag and radiation from the payload bay, the tank temperature will rise at about 35^oR/hr. If the purge bag-helium leak rates are not too excessive over a two hour time span, this would appear to be the best way to raise the average tank temperature and avoid water vapor condensation on the purge bag during reentry.

A second method of raising the purge bag temperature during reentry would be to raise the emissivity factor F_ϵ between the purge bag and payload bay. With $F_\epsilon = 0.1$, as assumed in this analysis, radiation heat transfer raises the average tank temperature by about 35^oF. If F_ϵ were increased to 0.2, the final purge bag temperature should rise by about 35^oF, which would just eliminate water vapor condensation from the analysis shown on figure 20. An increase of the emissivity factor F_ϵ from 0.1 to 0.2 would result in a very small increase in the heat leak during orbital flight with evacuated insulation and only a small increase in the heat leak to the tank during ground hold operations.

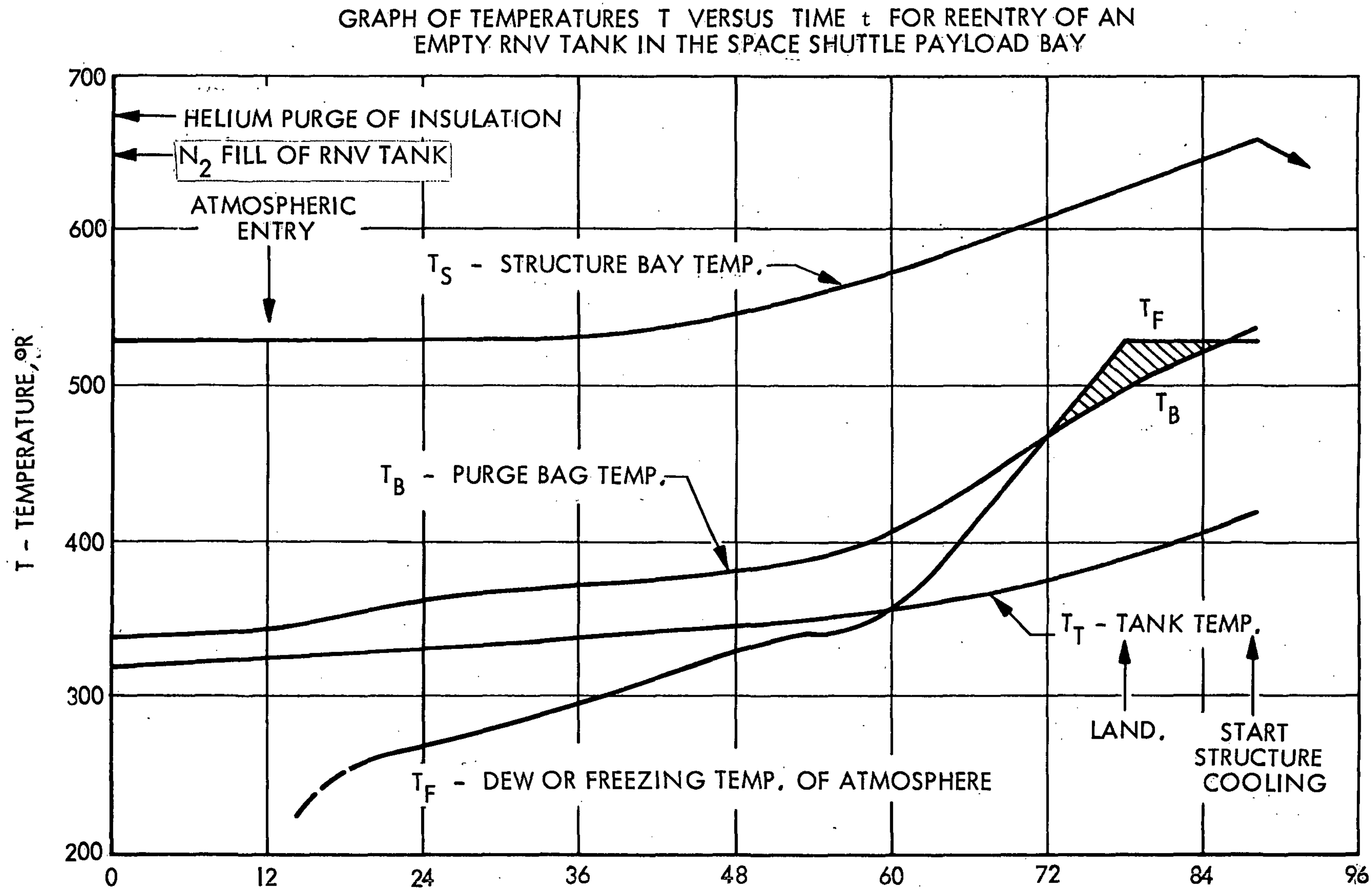

Figure 20.- Temperature vs. Reentry Time - Empty RNV Tank.

Based upon the above ground hold and reentry analyses of the RNV propellant tank in the Space Shuttle payload bay, it appears possible to design a closed, non-circulating gas, purge bag system that could eliminate all water vapor and gaseous condensation on the purge bag under normal modes of operation. The weight, power, and complexity advantages of a simple closed helium purge bag compared to a more complex hot circulating helium gas purge bag design should be strived for if possible in the system design.

One area that should be examined with this tank system is the possibility of GN_2 condensation on the purge bag during the ascent phase of flight. During ascent, the heat rate to the LH_2 tank will decrease from the ground hold value of $q''_T = 60$ Btu/hr ft^2 to a radiation/conduction heat rate through the gas annulus of about 10 Btu/hr ft^2. As can be seen from the q''_T curve of figure 19, this heat rate would result in a steady state purge bag temperature of $T_B \approx 100°R < 114°R < 140°R$, the last two temperatures being the freezing and condensing temperatures of nitrogen gas. This ascent phase with radiation/conduction heat transfer through the purge bag/payload bay nitrogen gas annulus will occur over a time period of only one or two minutes, and the thermal capacity of the purge bag should not allow the purge bag to reach the condensation temperature of GN_2 during ascent. This should be checked in a transient analysis to see if a problem may exist with GN_2 condensation on the purge bag during ascent flight.

This tank system should also be investigated for reusability if an emergency reentry with LH_2 in the tank is required. For this case, GN_2, GO_2 and water vapor condensation would occur during the early phases of reentry with only water vapor condensation occuring in the lower atmosphere when the purge bag temperature would approach the ground hold temperature of 280°R. This severe condensation might damage the purge bag, and require replacement of the purge bag before the tank could be reflown.

The above analysis was performed for the purge bag located at the outer meteoroid shield. Locating the purge bag at the inner meteoroid shield would probably decrease the heat leak and decrease the purge bag temperature. This system would result in lower ground hold boiloff rates, but would yield lower heating rates and lower purge bag temperatures on reentry than those computed above.

Summary

Specific payloads within selected mission categories have been examined to illustrate the type of thermal control problems which will exist at the Space Shuttle/payload interface. The missions selected were the Space Station Resupply, Satellite Placement and Retrieval, and Propellant Delivery.

The results of the Space Station Resupply Study indicate payload module radiator areas for autonomous operation on the order of 150 ft^2 are necessary to meet a 7,000 Btu/hr EC/LSS heat rejection requirement without imposing significant attitude restriction on the payload module.

The requirement for thermal isolation between small payloads selected for multiple delivery was identified in the Satellite Placement and Retrieval section. The feasibility of using a low weight multilayer insulation bag to meet the thermal requirements of the lower magnetosphere experiment during the various mission phases was demonstrated.

The potential for condensation on large LH_2 tanks (typical of propellant delivery missions) during prelaunch and reentry has been evaluated. Purge bag temperatures for the RNV configuration are on the order of 280°R during prelaunch; resultant propellant boiloff rates are on the order of 885 lb/hr.

Further emphasis should be placed on design requirements for the payload compartment wall since the thermal isolation it provides from low temperature propellant tanks will have an impact on operational requirements (i.e., purge rates, door open and closure, etc.) as well as the payload thermal response.

SYSTEM REUSABILITY

The reusable nature of the Shuttle and the need for quick turnaround are requirements for a spacecraft which must be met with new concepts to reduce inspection and maintenance times to an absolute minimum.

The System Reusability analysis performed in this task examined sequentially: (1) all facets of ground test, maintenance, and refurbishment as being practiced by airlines, military services, suppliers, and manufacturers, (2) pre-checkout, refurbishment, and post-flight checkout requirements at the EC/LSS component level, (3) mean-time-to-failure using system redundancy and weight limitations as applied to variable mission periods, (4) fault isolation feasability including EC/LSS elements criticality by mission phases, instrumentation practicality level, and difficulty of implementation, and (5) line replaceable unit requirements.

This analysis established the maintainability guidelines for:

- Location, accessibility and arrangements of EC/LSS modules and components
- Incorporating design facilities which minimize inadvertent wear or damage and servicing
- Detecting and/or precluding potential trouble

This section also presents detailed results of these reusability analyses and discusses a conceptual fault isolation approach that minimizes inspection, maintenance, and turnaround time.

LMSC Survey of Aircraft Test, Inspection, and Maintenance Practices

A survey of standard inspection and servicing practices of representative airlines and suppliers was conducted to provide reference information for determination of the applicability of these practices to the Shuttle EC/LSS for achievement of rapid turnaround. This survey consisted of discussions with responsible maintenance representatives from Eastern Airlines, Trans World Airlines, United Airlines and Garrett Corporation. In particular, subsystems relating to environmental control and pressurization were discussed. The survey also included a review of the activities and philosophies on the subjects of the Aircraft Integrated Data System (AIDS) and Automatic Data Acquisition System (ADAS) as used by TWA, Malfunction Analysis, Detection and Repair (MADAR) as used by the C-5A, the results of LMSC Shuttle fault isolation trade studies and practices anticipated for the Lockheed L-1011 as recommended by the Lockheed/CALAC chairman of the review board for environmental control and pneumatic system maintenance. The significant information abstracted from these sources follows.

Eastern Airlines.- In the past, overhauls were scheduled on the basis of a specific number of operating hours. The current trend is to flexibly schedule maintenance on the basis of the accumulation of minor failures rather than time. Upgrading of equipment to incorporate improved models is on the basis of safety or attrition only. A quantitative system checkout is performed prior to major overhaul at approximately 10,000 flying hours. Intermediate checks are performed approximately every 1000 flying hours. Lubrication levels in refrigeration unit sumps are checked at 300-400 hours with oil changed at 1000 hours. Vehicle failure modes and effects analysis reveals that no failure mode causes a major problem and equipment fails "safe". Each component is evaluated for the possibility of failure detection, the result being that some components are monitored for degradation while others are allowed to operate until failure.

It has been found that the actual life of equipment exceeds the supplier estimates. For example, plate fin heat exchangers have a life of 30,000 to 40,000 hours, bootstrap compressors exceed 3000 hours, temperature controls are good for approximately 10,000 hours, valves in the engine bleed area operating at 800°F are good for about 5000 hours, and ordinary shut-off valves last indefinitely. In most cases, degradation rates are short, and failures can be considered as sharp edged. Valves and fans are in this category. Exceptions are heat exchangers and air cycle systems which degrade slowly. Some items, such as pressure regulators, go out of limits. This condition can be determined by test and compensating adjustments made to bring the item back within tolerance. It can also be predicted that something will fail every 1000 to 1500 hours, but the type of failure is not predictable.

As far as the Environmental Control System is concerned, only the freon system on the 707 and the 720 are modular. Air cycle systems are not modularized. When a component has failed, it is examined to determine if it is to be refurbished or scrapped. If the component is overhauled, some parts are automatically replaced ("O" rings, springs, bushings, bearings, etc.).

Diagnostic instrumentation is on the increase. This instrumentation contains fault isolation provisions for ground checkout similar to the TWA flight system (AIDS).

Trans World Airlines.- This survey included a review of a series of papers describing the activities and philosophies on the subject of automatic data recording such as AIDS and ADAS. The installation of multiparameter recording systems on TWA's aircraft has provided an important tool for advancing the maintenance state of the art on modern airlines. These systems are computer controlled data systems designed to continuously sample and record data from various aircraft systems sensors throughout the flight duration. Signal parameters relative to problems within any of the monitored aircraft systems and/or trend data are permanently recorded for subsequent trouble detection and analysis by ground facilities. From the information recorded, trends deviating from standard are determined and the aircraft are investigated as to the causes for not performing up to the prescribed level. Once the causes are determined and corrective action taken, subsequent data processed through the program shows the improvement obtained as a result of these efforts. This type of correction can be established for analyses of the following air conditioning parameters; air cycle peak pressure, cabin inlet duct temperature, bleed manifold pressure, cabin temperature and cabin altitude.

Very little additional aircraft maintenance has been needed in order to keep the sensed parameters operational. This is due, primarily, to the fact that the signals are obtained from the same sources which supply information for the cockpit display.

United Airlines.- This airline has little information on fault isolation. In the case of a failure in which the bad part can't be determined, all suspects are removed and replaced to eliminate testing and false repair delays, thereby minimizing turnaround time. The removed parts are then bench tested and good items are placed back on the shelf without rework of any kind. Faulty items are repaired as necessary. No procedures employing regular replacement of parts are used. Although some trend analysis is used, parts don't generally wear out; they fail unexpectedly.

It has been found that even dynamic units have higher reliability than suggested by suppliers. For example, on the DC-8, experience shows that the freon unit has an actual operating life of 14,000 hours as opposed to the supplier recommended 2500 hours. Experience has also shown that because of the very high equipment reliability, attempts to incorporate later improved equipment models meets with increased probability of failure. Aside from black box functions, which are examined critically, United prefers keeping system functions separate when possible, since sharing can be the cause of problems.

NASA Survey of Aircraft Test, Inspection, and Maintenance Practices

The NASA survey (ref. 14) included information derived from United Airlines, Lockheed Missiles & Space Corporation, North American Rockwell Corporation, Flight Research Center (Edwards Air Force Base), Trans-World Airlines, and the Boeing Company. The various aircraft and experimental vehicle programs discussed encompassed the aspects of maintenance, refurbishment, test, and checkout of the following vehicles: C5A, 747, 720, 707, 880, DC9, and L-1011 aircraft, and X-15, HL-10, and X-24A and F3 (F2-M2) flight research vehicles. The maintenance concepts and experiences associated with these vehicles are described in the following paragraphs.

Commercial Aircraft.- The major commercial airlines provide their own personnel, facilities, and equipment for the following inspection and repair replacement activities:

"A" Check - A simple walk around visual inspection at every stop.

"B" Check - Includes all "A" check items plus manipulation of flight controls, steering gear, etc., with visual check of satisfactory operation, every 25 hours at any stop.

"C" Check - Some tools and bench simulation and measuring equipment, open inspection ports, remove cowls, etc.; example is pressurizing altimeter parts, measuring pressure and observing display accuracy.

"D" Check - Accomplished in line maintenance facilities, an Inspection and Repair as Necessary (IRAN) concept is utilized to predict the remaining life (predicted by system expert, tools used if required) of components or replacement if it cannot be removed and replaced (R/R) in line during a lower maintenance level check (such as "C" check). R/R if required is performed at 350-500 hours.

"E" Check - Overhaul maintenance performed in home-base overhaul facilities.

The time involved in the "D" check activities is usually 72 hours and performed mainly on weekends. The time involved in "E" checks is usually scheduled to be 7-14 days, depending on any engineering modifications or updates to be incorporated. When a new model aircraft is added to a fleet, the inspection periods are more frequent and once operating and inspection history has been accumulated, the time between inspection periods (after negotiation with the FAA) is lengthened. The airlines perform sample "D" and "E" inspections on a limited number of aircraft in order to lengthen these time intervals as much as practical.

When the 747 was put into service, the inspection and maintenance policies were changed by the incorporation of the activities as normally performed in the "E" checks into the "D" checks (such as auto-pilot removal and replacement). This was done so the aircraft would not be tied up in the maintenance dock for any extended period of time.

The scheduled maintenance checks for the Lockheed L-1011 are being scheduled as follows:

- o 200 hour service checks (overnight or 6 hours duration).
- o 800 hour periodic check (overnight or 8 hours duration).
- o 8000 hour major check (a maximum of 5 days).

In order to achieve this rapid turnaround capability, Lockheed emphasizes that the biggest time consuming factor involved is in fault isolation and troubleshooting times, and thus they have installed Built-In Test Equipment (BITE) wherever possible. The BITE allows troubleshooting down to the remove-replaceable black box called the Line Replaceable Unit (LRU).

The airlines almost without exception utilize the "canned system" concept. The systems, engines for example, are given extensive testing using stands and then sealed in airtight containers until use, at which time they are installed, given an operational test and then flown. Some of the biggest benefits of the "canned system" concepts are:

- The detailed testing does not interfere with the aircraft flight schedule.
- Testing and maintenance can be much more detailed and extensive.
- A more production line type of operation, resulting in the reduction of new problem areas.
- Overall ground support equipment requirements are reduced.
- The test and inspection time for the total aircraft is greatly reduced.

The airlines for the 747 and L-1011 aircraft are planning to use an onboard parameter monitoring and recording system on a trial basis. Its function is not checkout per se, but is basically an anomaly identification and recording scheme that minimizes ground troubleshooting and maintenance time.

In general, the commercial airlines have rejected real time-telemetry transmission and reduction systems for the following reasons:

- Cost of maintenance
- Initial cost
- Cost of personnel for data reduction and evaluation
- Reliability of the instrumentation and monitoring equipment itself
- Belief that operational characteristics of the aircraft systems themselves are sufficient.

Airlines maintenance personnel have been given opportunities to have direct inputs into the design and manufacture of the new large aircraft such as the L-1011 and the 747, so that when operational, inspection, and maintenance problems should be considerably reduced. As an example, each 747 engine

(P&W-JT9D) breaks down into 8 modular sections that can be removed and replaced in far shorter time than previous jets, and the complete engine can be replaced in one-half the time required for the P&W JT3D-3B engine on the 707-320B airplane. The only checks required after engine installation are leak checks. The 747 Inertial Navigation System (INS) utilizes built-in testing principles which are from 95 to 98 percent self monitoring and are modularized for easy flight line removal and replacement.

The biggest breakthrough on the 747 aircraft influencing minimum downtime is that Boeing and the customer airlines formed a maintenance organization with a six point program:

- o The education of subcontractors and designers on the principles of practical maintainability.
- o Review of all components for type of maintenance required, i.e., seal replacement, pump assembly replacement, etc.
- o Verification of maintainability design during the mockup first article inspection and during the flight test program.
- o Surveillance of engineering drawings and all vendor proposals.
- o Emphasis on maintenance of significant systems and components, i.e., engine, main gear shaft seals, etc.
- o Documentation of detailed maintainability analyses, maintenance action (on the job) studies and accumulation of in-service history.

The commercial users generally utilize the Inspect and Repair as Necessary (IRAN) concept for largely established components/systems and work towards making newer, less proven components/systems acceptable to the FAA on the same basis. The checks made in the IRAN concept are generally operational, utilizing operational crew displays in most cases. Some very special component/systems (those that are absolutely necessary to safe aircraft operation between liftoff and 100 ft.) such as the altimeter and altitude indicators, are checked with small handcarried equipment.

Airlines maintenance engineering personnel state that other data taking means, such as onboard monitoring/recording telemetry, and the use of automatic ground checkout equipment on line, is investigated each time a concept change or new equipment is introduced, but the costs, complexity, weight, and requirements for trained personnel have always previously prevented adoption.

Military Aircraft.- The current military trends are exemplified to a large degree by the C-5A aircraft now undergoing flight testing at the various Air Force installations. One of the major departures from the commercial aircraft is in the utilization of the MADAR installed on the initial procurement of 58 aircraft.

The MADAR concept is one of monitoring continuously and automatically 960 parameters, and allowing the manual monitoring of an additional 360 parameters via an oscilloscope, and comparing the wave forms against pre-determined waveforms presented on the MADAR display console.

The MADAR allows fault isolation of 1927 out of 3324 LRU's on the C-5A. The LRU's themselves incorporate BITE and with this, the MADAR interrogates each BITE indicator.

Another basic concept that is practiced by both the commercial aircraft maintenance personnel and the military for the C-5A is the "hands off" concept, which is reducing component and subsystem operational checks on the aircraft to an absolute minimum to preclude premature failures due to overchecking and human error.

The scheduled inspection program for the C-5A is:

- A preflight inspection which is valid for 24 flying hours which is prime or to the "B" check for commercial aircraft.
- A basic postflight inspection usually performed after each enroute stop and at the final destination.
- A phased inspection which is divided into 6 "work packages" performed at 100 flying hours time. As a rule, different systems and areas are inspected during performance of each work package.
- A depot level IRAN inspection every 3650 flying hours or every two years.

Current planning calls for no scheduled replacement (except for engines and wiring) of C-5A components on any pre-determined time basis.

The principal use of the MADAR is to provide the aircrew with a greater degree of decision-making information and to enable the ground crew to determine the C-5A functional status through reduction of the package of flight tapes by a ground data reduction facility.

In some instances, the onboard Avionics accounts for 50 percent of the maintenance down time as opposed to 10-15 percent for the propulsion system, so commercial users would rather let the military qualify any new onboard Avionics system.

Another Avionics system investigated was the Integrated Light Attach Avionics System (ILAAS). The ILAAS is currently undergoing evaluation in a modified Navy A-6 Intruder Aircraft. The basic concepts involved in the ILAAS:

- Extensive use of Built-In Test Equipment within each LRU.

- Display of status information to the aircrew such that they can make a decision-action relationship, i.e., the information displayed is to a level indicating failed paths and redundant paths still available for use.

- Display of LRU status information available to the carrier flight line maintenance crew such that they can remove and replace failed LRU's and get an operational status check after replacement.

- The BITE monitors the LRU status continually (dependent on system mode) and a latching indicator stays "on" after power removal to retain failure indication and to provide a rapid, direct fault indication to the maintenance crew.

The ILAAS is required to detect at least 95 percent of all failures that occur and must reduce the ratio of maintenance manhours to flight hours less than 1.0. Preliminary data shows that this ratio is on the order of 0.43, and thus is exceeding initial expectation.

Another integrated Avionics system under development is the Air Force Mark II, in which each LRU contains BITE and self-test circuitry. BITE has a high confidence level, i.e., 95 percent fault detection, but requires interruption of the LRU operation, whereas the self-test usually has a lower confidence level but does not require operational interruption. A central special purpose computer formats the BITE and self-test outputs and drives an aircrew display which shows decision-action level status.

Experimental Aircraft.- The programs investigated concerning the experimental aircraft/lifting bodies were the X-15, X-24A, HL-10 and the F3 (F2-M2) at the Flight Research Center. The X-15 will be discussed more thoroughly than the other aircraft since it has more flying experience (199 flights).

The philosophies that were inherent in the X-15 program were:

- Minimum GSE to make the vehicle ground activities expeditious and less complex. The electronics were packaged into a removable bay which allowed fast and easy removal and replacement of the electronic assemblies which, if faulty, were troubleshot using bench test equipment.

- Flight operational instrumentation was minimal in comparison with present Block II Apollo Command Module practices (86 Pulse Code Modulated channels as opposed to over 300 Pulse Code Modulated channels). The instrumentation for the X-15 encompassed both the operational or housekeeping information as well as experimental data.

- Instrumentation for the pilot as well as the operational telemetry was only to the level where the pilot could make a decision-action, i.e., decision to switch to a redundant, degraded, or shut-off mode.

- The majority of the ground test and servicing functions were simplified and minimized, thus keeping the complexity of electronics to a minimum as well as preparation time and operations. For example, the entire preflight checkout and servicing operations usually took four and one-half hours and required approximately twenty-five personnel including the B-52 flight and ground crews. The lifting body (HL-10, X-24A and F3 (F2-M2)) requirements involve approximately the same number of personnel.

- The nominal ground turnaround time for the X-15 was approximately 10 days for the vehicle itself. In practice, this ground time has been longer due to the unavailability of flight experiment equipment and/or its installation into the vehicle.

- The detailed test and checkout activities for the X-15 flight systems were performed prior to installation into the vehicle. The electronic systems (located in the removable electronics bay) were given (if indicated to be faulty by functional operating checks while installed on the vehicle) extensive testing and inspection on the bench. Experience indicated that the detailed testing on the bench or engine test stand resulted in a better checkout than could be accomplished with complex GSE after installation on the vehicle, and also resulted in decreased vehicle turnaround time.

At the start of the X-15 program, the engine (YRL99, burning NH_3 and LOX) was changed out after every flight and put on the test stand, but as the program progressed, the engines averaged 2-1/2 flights per change out but underwent a functional check after and prior to each flight. In terms of cost for refurbishment for each flight, between $200,000 and $300,000 was expended (ref. 15). This includes all personnel materials and hardware costs. The typical annual operating costs for the three X-15 aircraft were approximately 13 million dollars, which included both manpower and facilities.

System Refurbishment

The unique requirements placed on a spacecraft to be reusable and have a quick turnaround capability can only be met with unit replaceability, component accessibility, and establishment of effective component and subsystem maintainability guidelines.

Past space vehicles have been designed either with equipment buried in inaccessible locations or with limited improvement by modularization. The problems that have been associated with these practices have included:

- o Removal of major installations to repair or replace a faulty component

- o Disconnecting or decoupling of fluid or gas lines to remove EC/LSS modules. Opening of these lines has imposed the necessity to perform extensive leak checking, draining, purging, flushing and refilling of systems when reconnected.

- o Stringent cleanliness requirements relative to particulate size and count for the cabin equipment and atmosphere.

The system refurbishment analysis was initiated by establishing definitive maintainability guidelines based on the airline survey; defining and listing critical EC/LSS components that require special handling during the major phases of activity ranging from landing to relaunch.

Maintainability Guidelines.- The primary maintainability guidelines are concerned with accessibility, rapid and accurate fault identification, and minimal post-installation functional checkout requirements. Two major maintainability concepts, "designed-in" and "canned" are recommended for incorporation into the guidelines.

A "designed-in" maintainability concept is initiated by providing the designer with operation and maintenance requirements early in the vehicle definition phase. An application of this technique is the "changing out" of engines on the DC-10 aircraft. The specification states that it is a requirement to demonstrate that any engine can be removed in two hours and a replacement engine installed in a second two hour period. This concept is coupled with the "canned" concept, i.e., the system will have to be thoroughly checked out prior to "canning", such that upon need of use, it can be easily installed, interfaces verified and functional performance verified using an absolute minimum of time, personnel and support equipment requirements. The individual "canned" system approach allows parallel test and maintenance activity, performance by highly experienced and knowledgeable personnel as opposed to sequential testing by relatively inexperienced personnel.

EC/LSS requirements for maintainability guidelines have been established for the system reusability analysis. The guidelines are arranged in order of safety, design, accessibility, and replaceability criteria as follows:

- o Data acquisition devices such as sensors and transducers will be protected against damage during routine servicing.

- o Critical components/elements subject to contamination will be accessible and removable for sterilization.

- o Critical and/or sensitive adjustments will be protected by covers, locking devices, etc.

- All control valves will incorporate external position indicators.
- Shutoff valves will incorporate manual override provisions for emergency or test operation.
- Critical measurements such as pressure and temperature, will be displayed in the forward compartment. Pressure ports will be available at key maintenance points.
- Liquid spillage will be precluded by incorporating self-sealing disconnect or shutoff valves for all liquid-cooled units.
- Material subjected to wear due to frequent replacement or inspection will be carefully selected. For example, the use of aluminum threads will be avoided in filler caps, drain plugs, etc., where use is frequent.
- The EC/LSS components will be designed so that repair on the vehicle or on site at a lower maintenance level can be performed.
- Low-pressure lines will have a minimum number of connections to alleviate leak detection requirements.
- Alignment and positioning of equipment will be accomplished using positive keying techniques.
- Liquid-cooled units will be provided with a means for detecting internal leakage without removal of components or unusual tooling.
- Sectionalized ducting will be provided to permit replacement of an individual section without disturbing structure of adjacent components.
- All duct connections will be readily accessible for inspection and maintenance.
- All components with screen ports open to cabin environment will be accessible and removable for cleaning and inspection.
- Test connections will be provided for troubleshooting critical parameters such as temperature, flow, pressure, etc.
- Components will be removable without disturbing adjacent components.
- Standard tooling will be required to insure interchangeability.
- Packaging of the various elements making up the EC/LSS will be arranged to facilitate replacement.

Turnaround Analysis.- Determination of turnaround activity required investigating three major phases; post-flight checkout, refurbishment, and preflight checkout. Components of the EC/LS subsystems were examined as to their specific needs as far as remove/replace, cleaning, and disposition. Table 9 summarizes the postflight/refurbishment/preflight checkout procedures at the component level. Actuation and disposition activities during the post-flight checkout period are listed. The refurbishment cycle would normally occur prior to flight and consists primarily of replacing plug-in modules, replacing cartridges, and installing "canned" flight units. The preflight checkout consists of the actuation mode involving calibration signals of power application, and the monitor/verification mode requiring visual verification.

Several critical problems became apparent through this analysis:

- o Preflight checkout requires an integrated test unit capable of inputting a range of electrical signals, measuring a wide range of downstream pressures, and detecting low leakage rates.

- o Postflight checkout and refurbishment requires incorporating quick disassembly and breakdown of many elements to a size suitable for applying sterilization techniques such as autoclaving.

- o Postflight checkout includes a general clean-up and must be scheduled early in the refurbishment sequence.

- o Refurbishment is, in many cases, a matter of bringing flight storage units to the payload module and installing them. Easy accessibility must be guaranteed since this action will be scheduled late in the refurbishment sequence.

- o Visual inspection is required in many cases which will necessitate having the equipment accessible or having indicators located in such a way that they can be clearly observed.

Table 9 also lists the EC/LS subsystems replacement probability based on MTF for each subsystem. This value was determined through an LMSC computer program which is discussed under the following reliability analysis.

Reliability Analysis

The computerized reliability program used for the analysis of this system has been selected with respect to system configuration and analysis requirements. A system can either be a multithread system or a single thread system. A typical multithread system consists of a single thread system for a certain function which is backed up by a different system for redundancy. This type of redundancy is typical for information subsystems or navigation subsystems. This analysis is based on a single thread system which is considered more representative for the EC/LSS.

In order to increase the reliability, each item of equipment has to be made individually redundant, or an identical system has to be used as backup (multithread system). The analysis requirement is to optimize the system with respect to reliability and weight.

The most suitable reliability program chosen for this type of analysis is the computerized System Effectiveness Program (SYEFF) available at LMSC (ref. 16). The SYEFF program optimizes a system by computing the reliability for the basic system configuration, and then computing the reliability for each item considered redundant, one at a time. That item which gives the greatest increase in reliability for the least weight, cost, and volume, is added and the system reliability recalculated with the first redundant item, the result is called state 1. Another item is added and the process is repeated, resulting in a new state.

The iteration process stops as soon as one of the optimization constraints, weight, volume, or cost, is reached or the system cost goes through a minimum inflection point. Instead of all three constraints, either one or two constraints only can be used for the system optimization.

The flexibility of the program provides the user with the following options for making each component redundant:

- An item or piece of equipment can start as a single item and one or more items can be added at a time.

- The item can occur in many locations and one or more of the items may be added at a time.

- The item functionally called a one-shot item, which must be treated differently mathematically, can be identified by putting the proper code in the right location, and then the options as described above apply.

- Redundancy can be suppressed for any particular item if no redundancy is desired or is feasible.

When the optimizing process approaches the constraint limit of weight, volume, or cost, it is possible that the next item to be made redundant would exceed the constraint limit. For example, if 2 lbs. are left for optimization, and the item selected provides the highest increase of the remaining items in the system but at the same time its weight is 4 lbs., then this item is not made redundant but the one with the next highest increase in reliability, which will not go over the constraint limit, is selected.

The output of the program provides tables for:

- The initial subsystem reliabilities, weights, costs, and volumes.
- The data as have been inputted, listings of assigned item number and item name.
- State by state decision calculations and resulting reliabilities.
- The final redundancy configuration.

Appendix B provides a typical computer printout for the humidity control subsystem. In addition, the various outputs include cumulative weight and volume, total cost, total R&D cost, total operational cost, expected number of systems required to meet a mission requirement, reliability and system mean life. For better accuracy of the optimization process, it is advisable to optimize a total system, not only a subsystem, because a weight increment added to any one subsystem under separate investigation can probably be added with better results somewhere else with respect to the total system.

Additionally, the program takes the computer data and provides a plot to a microfilm recorder which produces a plot output in terms of reliability vs. time, reliability sensitivity to incremental weight and if cost data is included - expected cost vs. weight, and cost vs. design life. Numbers are shown on the plots which represent the state numbers. The same numbers are also shown in a table containing a listing of item numbers (Kappa's) and item names. Also, the reliability block diagram can be used as cross reference between state number, item number, and item name. For this application and because of the preliminary nature of the block diagram and design concepts of the EC/LSS, the runs have been limited to a weight constraint only. The program takes into consideration a target weight limitation that does not exceed 100% of the initial subsystem weight, and MTF based on mission time. The eight EC/LSS subsystems that were considered consisted of single components only (no redundancy). The complete EC/LSS that was considered consisted of single subsystems and then as two or more completely redundant subsystems for the maximum reliability model. The final system weight could not exceed 2500 lb.

The procedure for the reliability analysis was as follows:

- Review existing NASA contractors EC/LSS block diagrams.
- Develop single element reliability block diagrams composed of only selected critical components.
- Establish weight and failure rate data for these selected critical components.

- Prepare/run computer program.
- Prepare summary matrix showing weight and reliability on subsystem/system basis.
- Printout curves showing relationship between Reliability and Mission Time, and Reliability and Weight.

NASA contractors EC/LSS block diagrams were reviewed with respect to mission and functional requirements and redundancy. Individual subsystems and interfaces were identified, necessary for the preparation of reliability block diagrams. Equipment Lists were prepared for each individual subsystem.

Based on the EC/LSS block diagrams and subsystems Equipment Lists, the reliability single thread block diagrams were developed. The total EC/LSS reliability block diagram is shown in figure 21. The single thread system and associated reliability block diagrams do not imply a direction of a signal or functional flow of the system. They deviate from the conventional nomenclature and block diagram presentation. A reliability block diagram drawn to represent a single thread system merely indicates and identifies which items in the system are critical to system performance and have to perform simultaneously for the required function of that particular system/subsystem. It also means that any one item shown in the single thread causes the system to fail, if the item fails. Any item used for monitoring equipment only or used for a function not vital for mission performance is not shown in the single thread reliability block diagram. If a circuit contains more than one item performing the same function, these items are shown as redundant items. The presentation of such a configuration is the same as the presentation of a parallel circuit in a design.

The single thread reliability block diagram identifies the system/subsystem and any item in the subsystem by name and executive item numbers. It also shows the failure rate and weight for each item.

Using the equipment lists and reliability block diagrams of the subsystems, the failure rates for each item in the subsystem were determined. In order to select an adequate failure rate, three sources for failure rates were reviewed; Apollo actual data, Hamilton Standard data, and FARADA (Failure Rate Data). The most suitable failure rate of a part or equipment was assigned to it after evaluating the selected failure rate with respect to performance, weight, and applicability.

Based on the reliability block diagram, equipment lists and special conditions as stated by the Design or Systems Engineer, the computer input deck for the SYEFF program was prepared. Computer runs were made on each subsystem for mission periods of 168 hours (7 days), 720 hours (30 days), 8,400 hours (50 missions) and 16,800 hours (100 missions). The computer output tables were used and a summary matrix prepared.

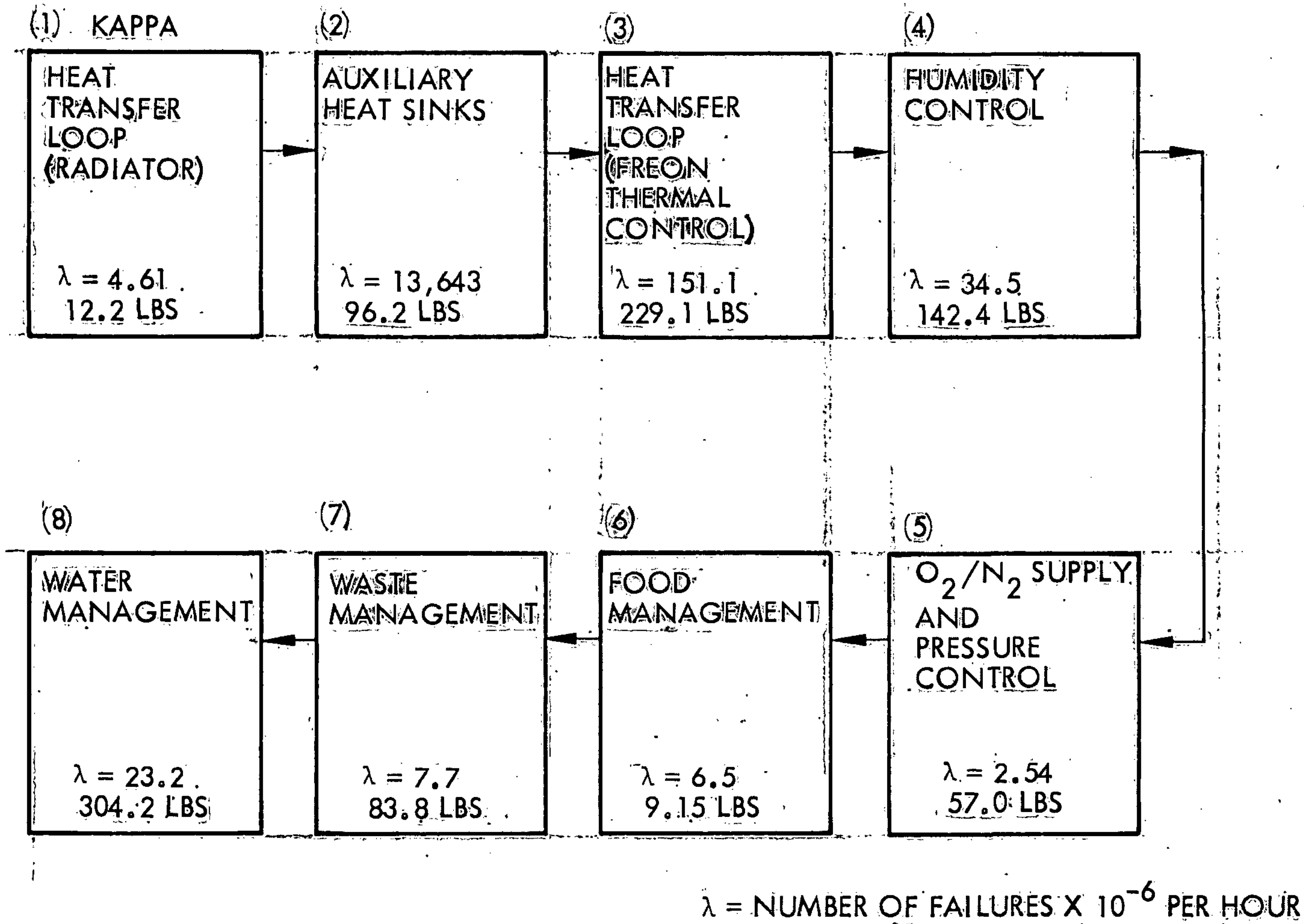

Figure 21.- Single Thread Reliability Block Diagram (EC/LSS).

Table 10 shows the subsystem/system weight and reliability data for the four mission periods, both for the initial weight case (I) and the final maximum weight case (F). For the short-term missions (168 and 720 hours), increasing the reliability of the subsystems by adding critical elements achieves the required MTF and produces a high degree of reliability. For the 50 missions (8,400 hr.), the majority of subsystems show the same result; however, as noted on the table, the Heat Transport Loop (Freon Thermal Control) and Humidity Control subsystems both experience difficulty meeting the MTF requirement and both have marginal reliability. The Fire Control subsystem was not analyzed based on its inherent high reliability.

For the 100 missions (16,800 hr.), the majority of subsystems meet the required MTF; however, as noted on the table, the Heat Transport Loop (Freon Thermal Control) and Humidity Control subsystems both experience difficulty meeting the MTF requirement and both have marginal reliability.

The most significant of the total EC/LSS reliability results are that the 50 mission MTF can be met by complete redundancy; however, the 100 mission requirement is not met.

The computer also provided two plots for each subsystem and for the total EC/LSS showing the relationship between reliability and mission time, and reliability and weight. The first curve, figure 22, showing the total EC/LSS is of interest with respect to the various mission durations. It can be used as a decision making tool for system refurbishment and determination of mission duration. It shows the slope of reliability decreasing with respect to time. The second curve, figure 23, showing the total EC/LSS reliability vs. weight by states, is very useful as a decision making tool for the designer or systems engineer with respect to finalizing redundancy configurations. Using figure 23 in conjunction with the computer printout (See appendix B), one can identify the item added at each state, what reliability gain was achieved, and what the weight increase was. A flattening of the curve indicates that the weight penalty is too high for the gain in reliability. Optimization is feasible where the rate of change is still high, whereas the point where the slope of the curve drastically flattens is the point where adding redundancy is no longer justified.

Results of this analysis identifies two critical subsystems: (1) heat transport loop (freon thermal control), and (2) humidity control. The results of the computer analysis and the computer outputs for the humidity control subsystem are discussed below, as an example.

The step-by-step procedure discussed earlier is exemplified by table 11, which lists the Humidity Control subsystem critical equipment developed from reviewing the subsystem block diagram. The single element reliability block diagram is next developed, as shown by figure 24. The item number (Kappa) is shown above the component box and the failure rate data and weight is listed below the box.

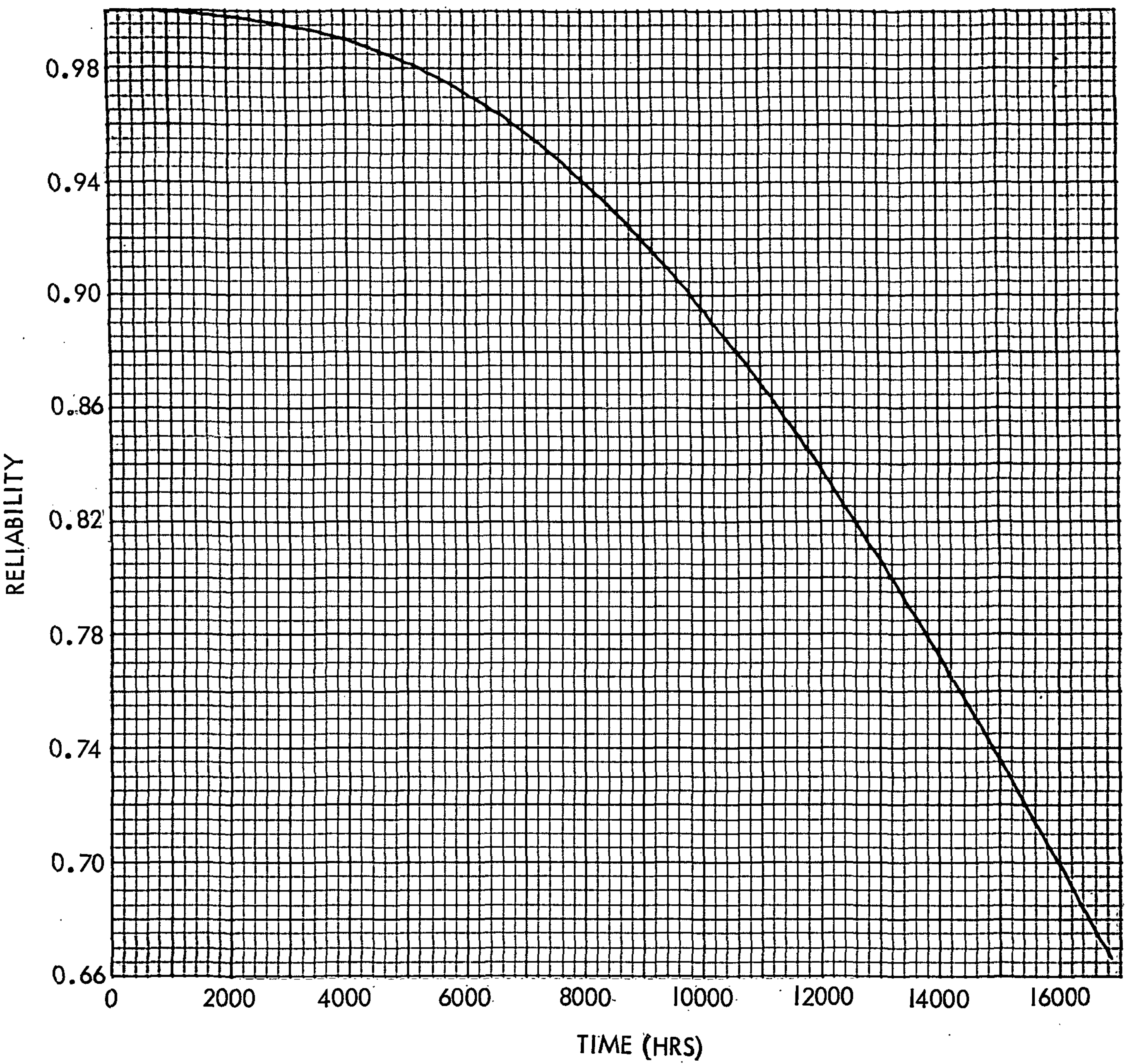

Figure 22.- Reliability Versus Mission Time (EC/LSS - 100 Missions).

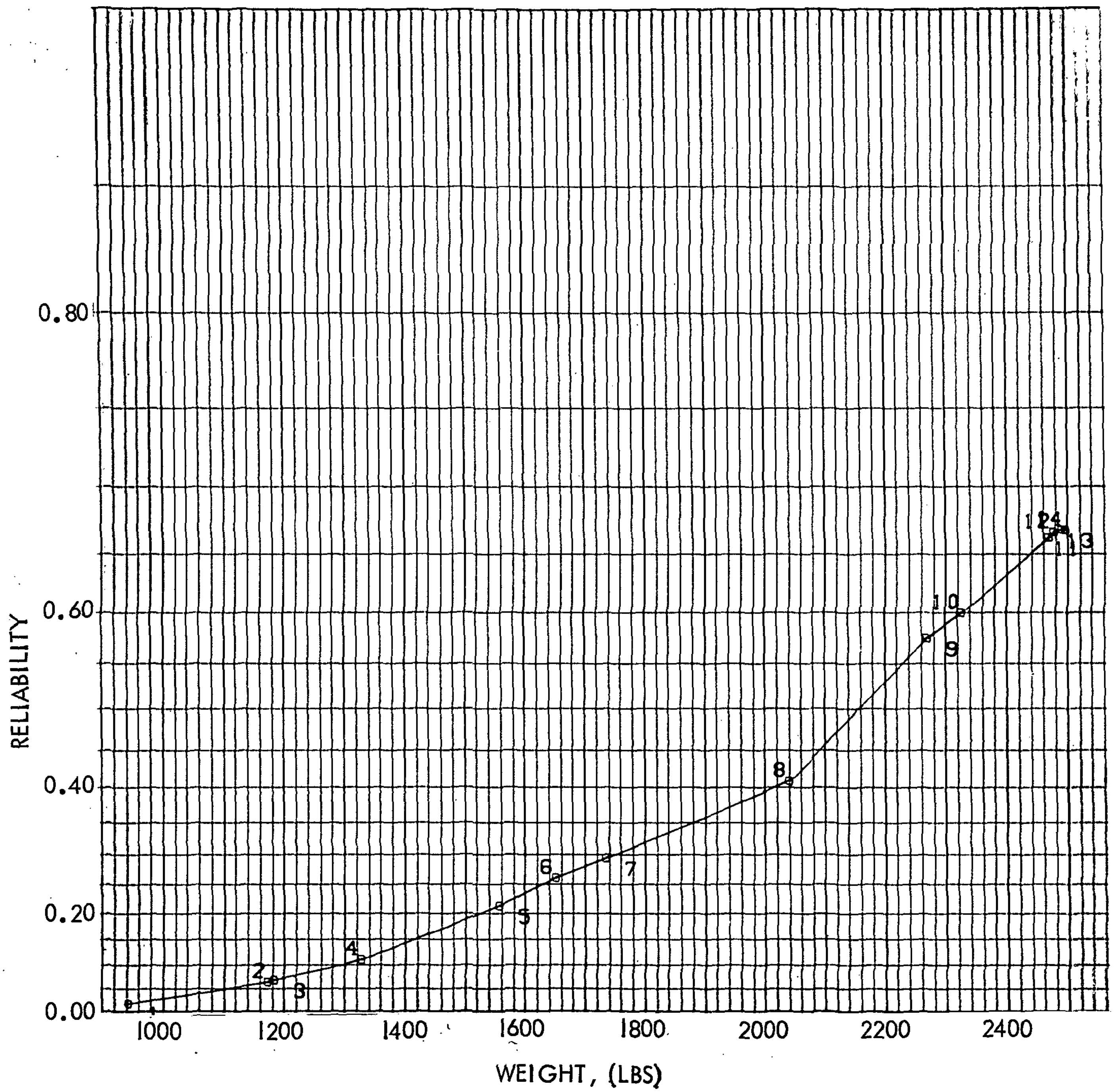

Figure 23.- Reliability Versus Weight (EC/LSS - 100 Missions).

TABLE 11

HUMIDITY CONTROL SUBSYSTEM CRITICAL EQUIPMENT*

Item	Quantity
Disposable Filter	1
Relief Valve	1
Debris Trap	1
Fan	3
Check Valve	12
Regulator	3
CO_2 Absorber Canister	3
CO_2 Absorber Assembly	6
Heat Exchanger	3
Wick Separator	9
Cooler	1
Water Separator	3
Water Shutoff Valve	3
	Total 49

* Equipment critical to subsystem performance and failure of which causes subsystem to fail.

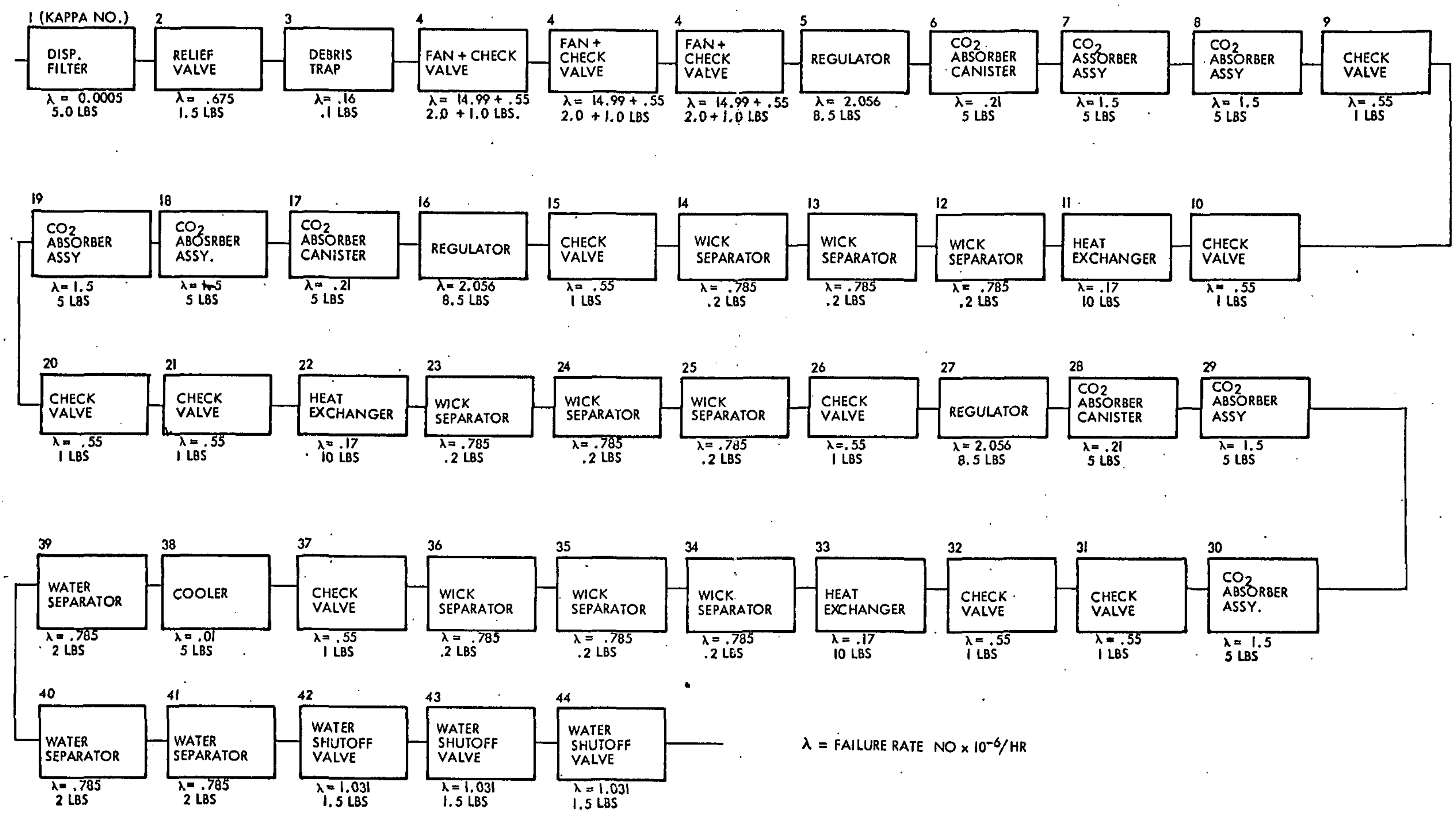

Figure 24.- Single Thread Reliability Block Diagram
(Humidity Control Subsystem)

Figure 25 shows the plot output of reliability vs. time. Note the negative slope of the curve as the mission time increases, resulting in a reliability of 0.7208454 at the desired mission time of 16,800 hours (100 missions). Figure 26 shows the reliability vs. weight for 100 missions. As can be seen from the curve, the first few redundancies added to the system at states 1 through 9 cause a significant increase in reliability for a small addition of weight. The slope of the curve drastically flattens, starting with added redundancies at state 10 and up. The reliability increase for the first 9 redundancies is much higher than for the items added at and after state 10. The weight increase for the first 9 redundancies is much lower than for the items added after state 10. Another change in slope is noted in state 24. From then on, the reliability increase is negligible. As the target weight is approached, the weight increased from 144.2 lbs at state 9 to 160 lbs. at state 27, compared to the small reliability increase from 0.6213582 at state 9 to 0.7208454 at state 27, which represents a weight penalty of 15.8 lbs. for a negligible reliability increase. Corresponding change in MTF is from 13,407 to 14,371 hours.

Based on this analysis, it is recommended that additional computer runs be scheduled during a later phase when realistic weights, volumes, and cost data can be incorporated.

Fault Isolation Analysis

The Space Shuttle EC/LSS was analyzed to identify the elements requiring fault isolation and subsequently evaluated as to the applicability of fault isolation techniques.

Step-by-step procedures involved: (1) critical element identification by analysis of existing aircraft data to isolate the elements into those which display operational problems, and those which do not, (2) three dimensioned array preparation relating the identified Shuttle EC/LSS into the fault isolation priorities by mission phase, (3) element analysis by concentration on subsystems which impact crew and vehicle safety during ascent to touchdown, and determination of instrumentation practicality, and (4) conceptual fault isolation approach.

Critical Element Identification.- Use was made of Air Force accumulative data from 75,000 flight hours of operational use on the C-141 fleet which is equivalent to approximately four and one-half (4.5) orbiter lifetimes based on an average of 100 seven-day missions per vehicle per life. The Air Force utilizes a Work Unit Code (WUC) to identify the elements and maintains data banks which permit recovery of data such as actual replacement, average replacement time, actual failure rates, etc. for each WUC designator. The number of these elements are reflected under the quantity column in table 12.

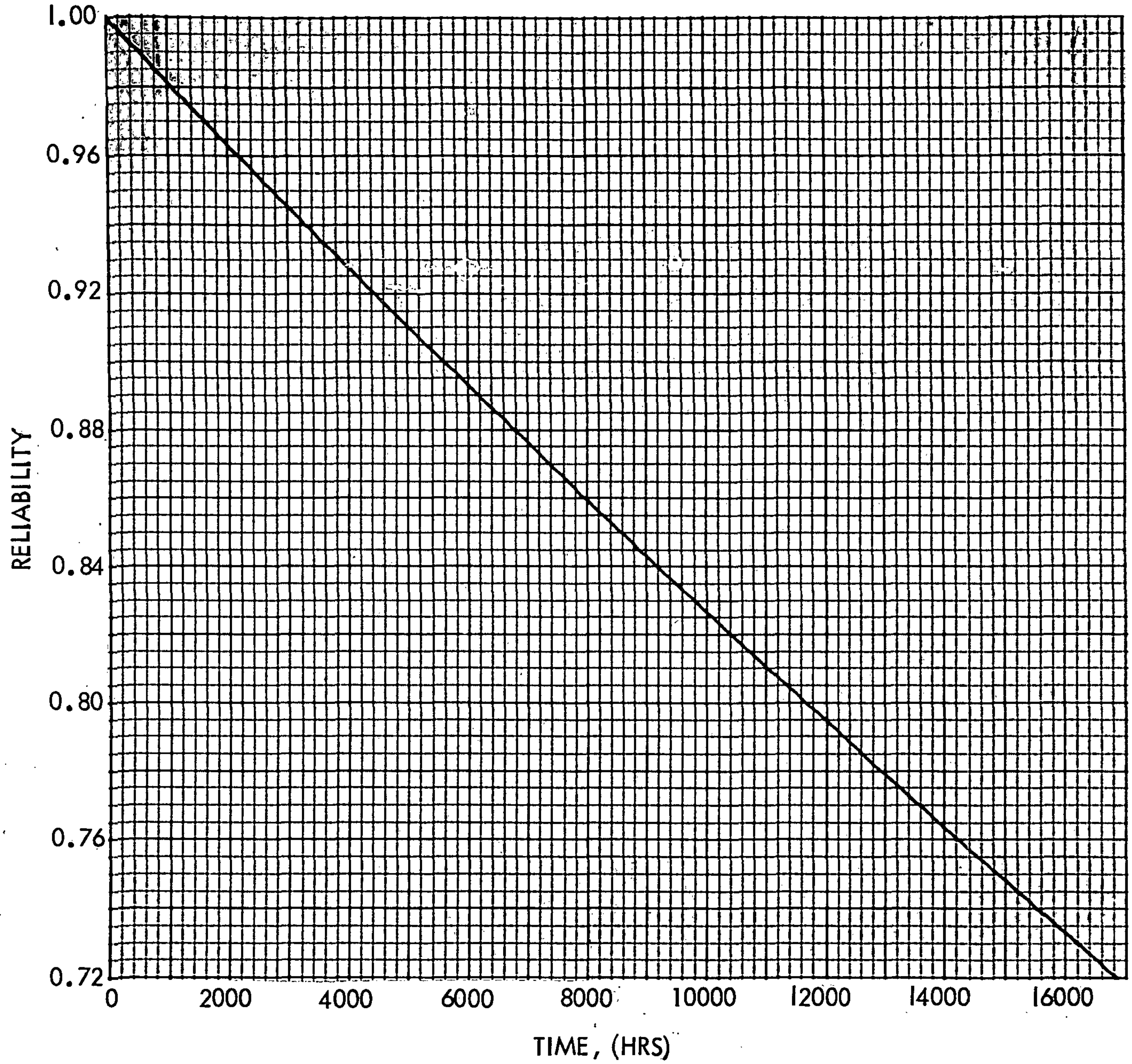

Figure 25.- Reliability versus Mission Time
(Humidity Control - 100 Missions)

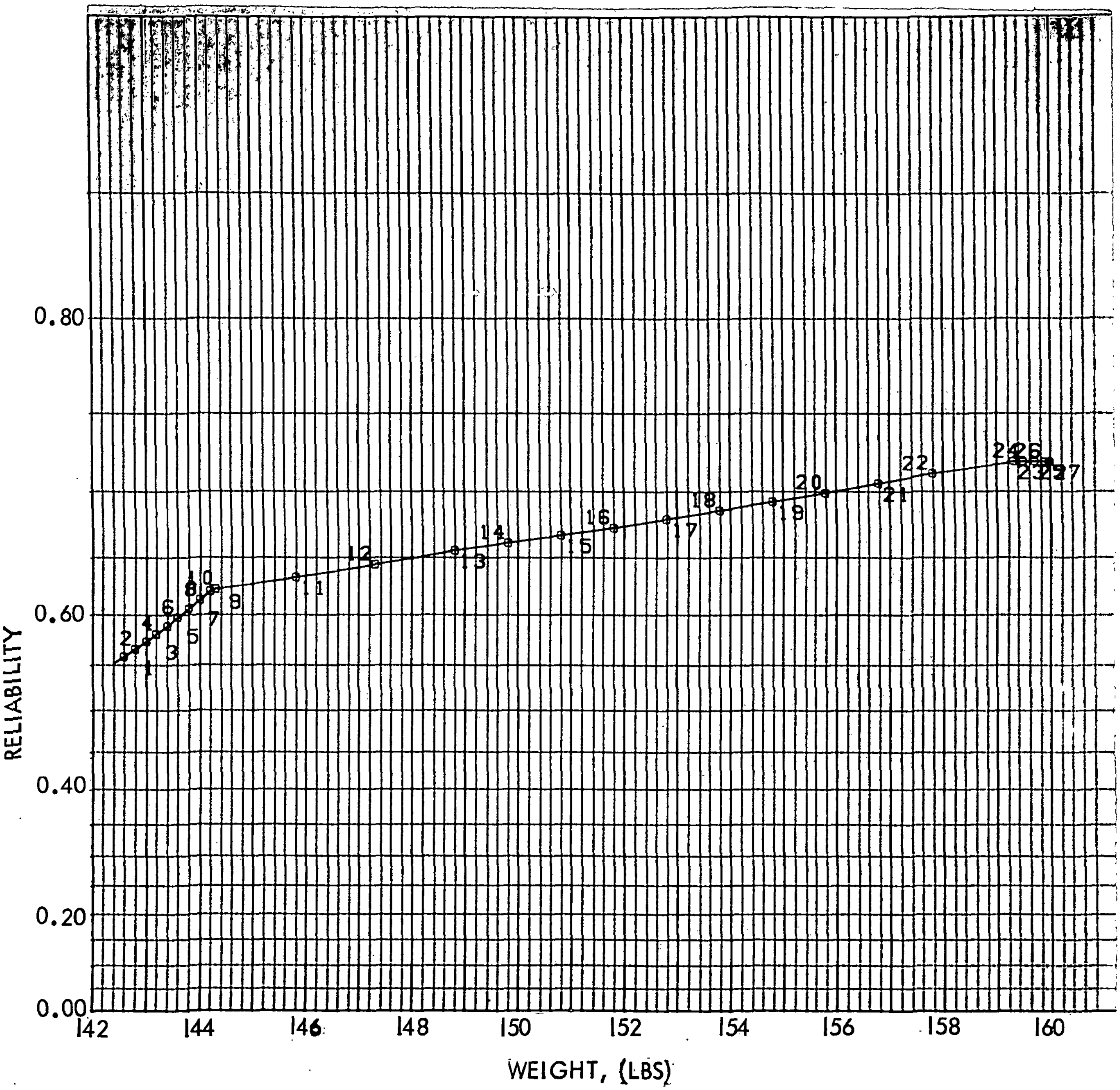

Figure 26.- Reliability versus Weight
(Humidity Control - 100 Missions)

TABLE 12

WORK UNIT CODE CLASSIFICATIONS BY POPULATION DENSITIES

Classification	Example	Designator	Quantity	% Of Total Quantity
Fluid Mechanical Static	Hydraulic Lines	FMS	100	4.64
Fluid Mechanical Dynamic	Hydraulic Actuator	FMD	263	12.20
Metallic Mechanical Static	Skin and Frames	MMS	365	16.94
Metallic Mechanical Dynamic	Spring	MMD	397	18.42
Nonmetallic Mechanical Static	Windscreen	NMS	116	5.38
Nonmetallic Mechanical Dynamic	Tires and O Ring Seals	NMD	45	2.09
Electromechanical Rotating	Electric Motors & Gyros	EMR	112	5.20
Electromechanical Nonrotating	Relay, Switch	EMN	408	18.93
Electrical	Resistor, Capacitor	ELT	235	10.90
Electrochemical	Battery	ELC	2	0.09
Electronic	Receiver, Transmitter	ELN	112	5.20
			2155	99.99

The Space Shuttle will not contain units identical to an airplane, however, the work units can be classified into groupings (pressure, mechanical, electrical and electronic) which logically can be expected to display similar characteristics in wearout/failure and in instrumentation requirements. The classifications are shown in table 12. The majority of elements occur in the electromechanical non-rotating, and metallic mechanical areas; however, the majority of the problems as far as maintenance are concerned with the electronics area.

A detailed investigation of the WUC classification in terms of practical level of instrumentability and relative difficulty of fault isolation, resulted in establishing table 13. The higher the practicality index, the easier it is to carry out fault isolation.

Table 13 lists the eleven classifications. Column A is the estimated fraction of the population which can be instrumented directly for fault isolation and B is the fraction which can be deduced but not directly instrumented (1 -low, 10 -high). The sum of these columns is the level to which each classification can be fault isolated through instrumentation techniques.

Column C is the relative difficulty of accomplishing the fault isolation to a meaningful decision including considerations of software development and general state of the art in supporting technology (1 -hard, 10 -easy).

Column D is the normalized product of C with the sum of A and B

$$\left[(A + B)\ (C)\right] \Big/ \left[(A + B)\ (C)\right]_{C_{max}}.$$

Results of this analysis indicates that it is practical and desirable to provide fault isolation instrumentation for components of EC/LSS subsystems incorporating parts pertaining to: fluid mechanical dynamic (FMD), electromechanical rotating (EMR), electromechanical non-rotating (EMN), electrical (ELT), electrochemical (ELC) and electronic (ELN). For example, there will be approximately 25 major components in a critical EC/LSS subsystem such as the Heat Transport Loop. Applying the population density and practicality index to the critical classifications would indicate a requirement for monitoring a majority of these components. This approach could apply and is practical for the entire EC/LSS if the number of components and the practicality of instrumentation warrants it.

Subsystems Criticality by Mission.- Since the need for fault isolation is identified with the mission criticality factor, the next step is identifying critical subsystems of the EC/LSS and evaluating their relationship with the mission phases. A three dimensional matrix resulting from this evaluation is presented in table 14. Important assumptions employed in the analysis are:

TABLE 13

INSTRUMENTATION PRACTICALITY

Class	Practical Level of Instrumentability		Relative Difficulty of Fault Isolation	Practicality Index
	Direct A	Indirect B	C	D
FMS	1	5	2	0.148
FMD	7	1	8	0.790
MMS	1	2	3	0.111
MMD	3	4	4	0.345
NMS	0	1	2	0.025
NMD	0	5	1	0.062
EMR	6	2	6	0.592
EMN	7	2	9	1.000*
ELT	6	2	7	0.691
ELC	8	1	6	0.667
ELN	8	1	5	0.556

* EMN is used as reference for normalization because it has the highest normalized product.

TABLE 14

EC/LSS CRITICALITY BY MISSION PHASE

EC/LS Subsystems \ Mission Phase	Pre-Launch	Launch Ascent	Orbit Oper'n.	De-Orbit	Re-Entry	Approach Landing	Ferry
O_2/N_2 Supply and Pressure Control	1	1	1	1	1	1	4
Water Management	4	4	3	4	4	4	4
Waste Management	4	4	4	4	4	4	4
Heat Transfer Loop (Freon Thermal Control)	2	2	2	2	2	2	2
Food Management*	4	4	4	4	4	4	4
Humidity Control	4	4	4	4	4	4	4
Heat Transfer Loop (Radiator)	2	2	2	2	2	2	N/A
Auxiliary Heat Sink (Sublimator)	4	4	4	4	4	4	N/A

Key: 1 Critical to Crew Safety
2 Critical to Vehicle Safety
3 Necessary for completion of Mission
4 Non-critical for Mission success.

* Ready-prepared food backup available.

- o Each subsystem was evaluated for the effect of a single component failure in a single thread system
- o The subsystem failure was evaluated for the direct effect on crew safety, vehicle safety, et al; not on secondary effects which could result from an uncorrected failure being propagated into another subsystem.
- o Those subsystem failures which are not critical to life/vehicle at the time of occurrence, but on a succeeding mission segment with the failure present is critical, are defined as mission critical. The assumption is that the crew will not exedute the critical segment but will make a safe abort, by rescue, if necessary.

As noted on table 14, water management does not become critical until the orbit operation phase when drinking water must be provided for the 7-day mission. The heat transport loop subsystem involves vehicle safety since a failure could jeopardize vehicle critical electronic equipment. Humidity control is less sensitive since a high humidity level would prove uncomfortable, however, it could be tolerated. The radiator is an integral part of the heat transport loop. The sublimator of the auxiliary heat sink subsystem is required for overload peaks during orbit operation is considered non-critical.

EC/LS Subsystem Analysis.- The two subsystems identified as critical by the criticality analysis and selected for examination (O_2/N_2 supply and pressure control and heat transport loop subsystems) feature outputs of pressure and temperature respectively. Fault isolation involves establishing conditions of valves, regulators, heat exchangers, temperature and switches, etc. Since some of the equipment and supplies are Shuttle located, such as Freon and water supplies for the thermal loop, it is necessary to establish failure location (Shuttle or within the EC/LSS). The heat transport loop subsystem is the most complex since the single subsystem (no redundancy) contains at least 75 major components. Pressure and temperature sensing comprises the majority of the fault isolation instrumentation. Upstream and/or downstream pressure sensing at a few critical packaging points with suitable logic information would pinpoint the operational malfunction of values as well as fluid loops. Temperature sensors optimally placed and tied into an appropriate display network would assist the initial effort. The high pressure gas system by virtue of fail-safe operations and adequate instrumentation would sense a rise or loss of pressure in the system.

Line Replaceable Units.- Fault isolation is being recommended to the critical line replaceable unit (LRU) level since this approach is included in the new generation carriers. Identification of the LRU's of the EC/LSS is a direct outgrowth of the system reusability analysis. Analysis of the Shuttle EC/LSS block diagrams gives the first indication as to critical items and to the practicality of removing and replacing elements. A listing of LRU's has been compiled on table 15, relating the LRU's to the EC/LSS subsystems. This listing will furnish a design reference point to ensure that the LRU's will comply with performance and design requirements oriented around quick accessibility, remove and replace criteria, good maintenance and reliability characteristics, and proven safety procedures and techniques.

Conceptual Fault Isolation Approach.- It has been estimated that module removal and replacement without fault isolation can result in a 50% incorrect diagnosis, therefore, a major requirement exists to develop an integral fault location detection technique that provides a high confidence in locating faults down to the critical LRU and/or module package level.

As a result of the fault isolation analysis, the following preliminary requirements have been established for a Fault Location Indicator Test Equipment (FLITE) system:

- The EC/LSS is to be 100% independent of FLITE.
- All sensors are to have the most basic output = go, no-go.
- FLITE is to be versatile - it can be used on any EC/LSS in the Shuttle by simply connecting it into any central connector panel.
- It should do something the technician cannot - in pinning down the failed LRU.
- FLITE is to be compatible with the data management system.
- All FLITE indicators are to be extremely simple, go, no-go.
- FLITE is to recognize missing sensor connections and turned-off LRU's so as not to confuse these conditions with fault identifications.
- FLITE is to have no operating controls, only on/off switching.
- FLITE is to use digital logic for maximum simplicity of design and most reliable performance.
- FLITE is to use logic operations as a first level computer technique in fault isolation (FLITE is a semi-computer -- does some decision making).

TABLE 15

LRU'S FOR SPACE SHUTTLE EC/LSS

WASTE MANAGEMENT

- Shutoff valves (2)
- Check valves (2)
- Biocide dispenser
- Biocide tank
- Urinal
- Isolation valve
- Water separator
- Bacteria filter
- Charcoal filter
- Dump valve
- Dump nozzle
- Vacuum vent valve
- Isolation valve
- Waste collection

FOOD MANAGEMENT

- Oven and controls
- Calrods (2)
- Motor
- Fan (2)
- Light circuit
- Control switch
- Contact switch
- Solenoid switch
- Water heater
- Timer

HUMIDITY CONTROL

- Disposable filters
- Relief valve
- Debris trap
- Fan (2)
- Check valve (2)
- Regulator
- CO_2 absorber canister (3)
- Check valves
- Wick separator (3)

TABLE 15.- Continued.

HEAT TRANSFER LOOP

Heat exchanger
Cabin blowers (2)
Check valves (2)
Temperature control valve (2)
Cabin temperature sensor (2)
Cabin temperature anticipator
Cabin temperature controller
Cabin temperature selector
Cabin temperature signal amplifier
Freon bypass valve (2)
Hydrogen heat exchanger
Hydrogen flow control valve (2)
Hydrogen shut-off valve (2)
Freon temperature controller (2)
Override switch (2)
Temperature sensor (4)
Heat exchanger
Pump (2)
Check valves (2)
Filter
Accumulator
Accumulator isolation valve
Sublimator
Temperature sensors (2)
Fuel cell heat exchanger
Pump outlet pressure transducer
Accumulator quantity transducer
Water valve flow controller
Water shut-off valve
Radiator control valve
Radiator isolation valve (2)
Proportionate valve
Proportionate controller
Temperature sensors (2)
Check valve
Heat exchanger - Radiator
Water bypass valve - Radiator
Water chiller - Radiator

TABLE 15.- Concluded

AUXILIARY HEAT SINKS

Pumps (2)
Check valve (2)
Accumulator
Filter
Accumulator isolation valve
Accumulator quantity transducer
Pump outlet pressure transducer
Fluid exchanger
Water temperature controller
Interchange bypass valve

O_2/N_2 SUPPLY AND PRESSURE CONTROL

Gas tank (as required by mission)
Check valve (per assembly)
Main shut-off valve

WATER MANAGEMENT

Water tank
Waste water tank
Check valve (2)
Water shut-off valve (2)
Silver-ion generator
Water heater
Water chiller
Water dump nozzle

The Fault Location Indicator Test Equipment (FLITE) design is a new concept made possible by recent developments in integrated circuits and light emissive semiconductors. Although superficially similar to the Built In Test Equipment (BITE) concept, FLITE is uniquely suited to fault isolation in environmental control and life support as well as hydraulic/pneumatic/mechanical systems, while BITE was conceived for systems which were primarily electronic in nature.

FLITE consists of two equipment groups, one group being a series of individual go, no-go sensor/signal conditioner modules integrally combined with the Line Replaceable Units (LRU's) of the EC/LSS, the other group being a flexible logic and display module (see fig. 27) located at a convenient access or service point near the EC/LSS or carried out at maintenance depots as a plug-in test instrument. It is intended that FLITE be used primarily by maintenance technicians at permanent maintenance stations; however, the continued availability of FLITE on board the operational spacecraft may be desirable in the event of emergency in-flight maintenance requirements.

The sensor/signal conditioner modules located throughout the EC/LSS are designed around approximately five simple sensing functions. It is presently believed that it is desirable to monitor critical temperatures, pressures, flow rates, motor functions and valve actions. Each sensor module requires about one cubic inch of space. It screws into or onto the element being monitored and a small connector provides tie-in to a light cable leading to the central connector panel near the logic and display module. There are four main connectors, each connector receiving the lines from two of the eight EC/LS subsystems. The sensors do not all send outputs directly to the central connector panel. At some of the LRU's where there are several sensors each, the outputs of the sensors go to a common AND gate which does not produce an output until all of the inputs are in a no-go condition. In no case does an LRU have more than one output cable as a result of this logic design. Not all identified LRU's have sensors, as they do not have functions which permit easy instrumentation, but may require some method of indirect detection, such as a thermistor attached to a duct.

The logic and display module is placed near the central connector panel. It is readily removable and may be left out of the spacecraft if so desired for any flight. As presently conceived, it will consist of four "pages" and a power converter module. Each page is about 4" x 12" x 1" thick, and carries the logic and light emitting diode (LED) displays for two of the eight EC/LS subsystems. The total package of 4 pages and a power conditioner is 4" x 4" x 16". Each page has a row of red, green, and amber LED groups down the left side, each LED group labeled for the LRU which it represents. During normal operation of the LRU, only the green LED is active. In the event that an LRU goes into a significantly abnormal operating mode, the red LED comes on and the green goes off. This indication will also exist in the event that a sensor module connector has not been attached. In the event that an LRU is taken out of service by removing power, the FLITE logic will display this fact by activating the amber LED and extinguishing the red and green.

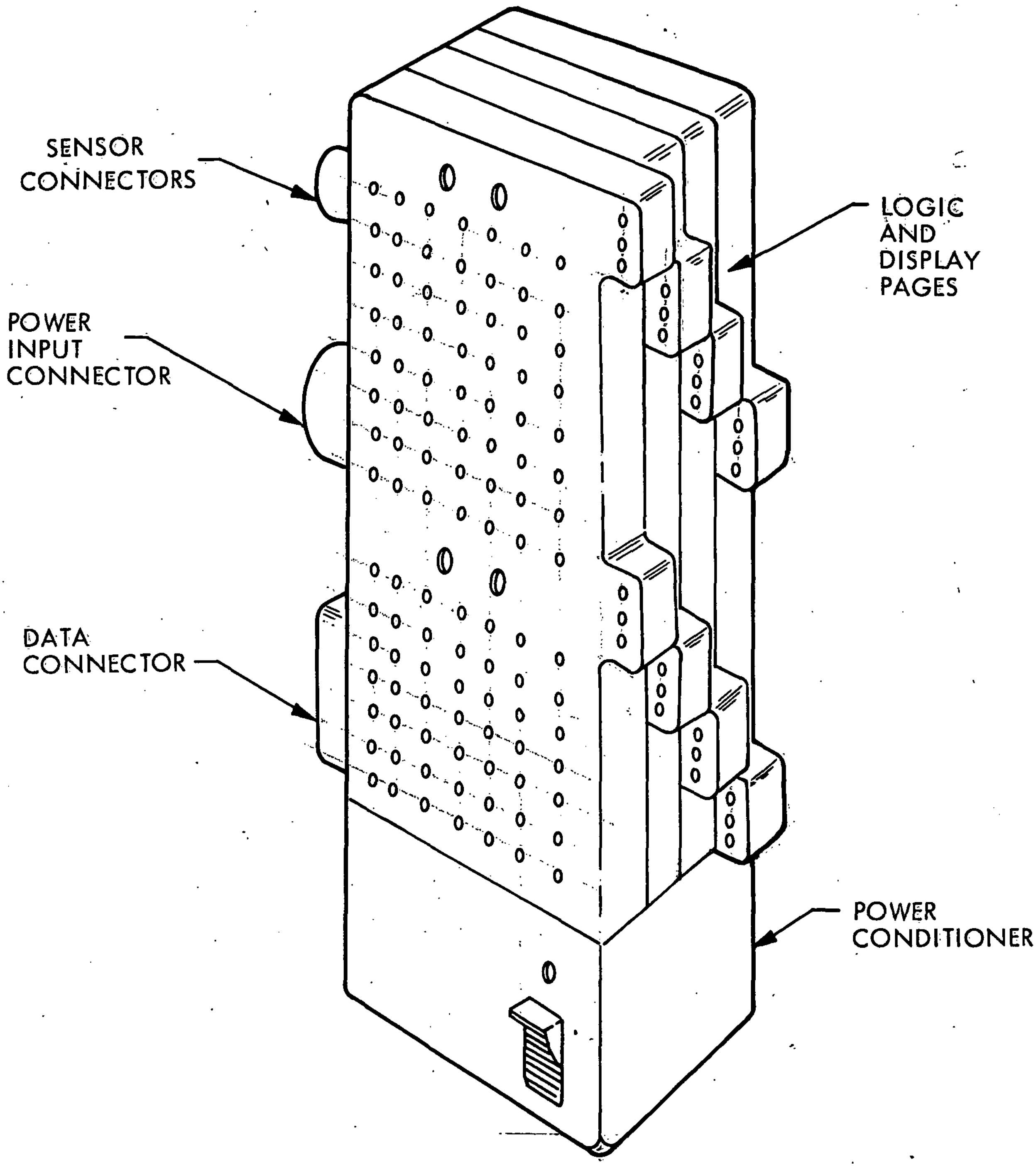

Figure 27. FLITE Logic and Display Module.

Since the pages of logic physically cover one another with only the front page fully exposed, each page has been provided with two tabs which project to the right. These tabs are in successively lower positions on each page so that all tabs show from the front. Each tab carries red, green and amber summation indicators for each EC/LS subsystem. A green indication will be displayed only if all powered LRU's are operating normally, red if one or more LRU's are down. In the event that one or more LRU's are taken out of service by removing power, the amber LED will come on. In this case, the red or green indicator may also be on. Each logic and display page has a single connector going to the central connector panel, and power to the page is interlocked through the connector so that the page displays are all out when the connector is removed.

The logic and display module has a small power conditioning unit located at its base. This power conditioner includes an indicator and an on-off control. The FLITE system, with sensor modules, will be turned on only at those times when needed for fault isolation, thus keeping power consumption to a minimum.

The logic and display module offers the possibility of a remote summation indicator at the pilot's position, however, this is considered minimally appropriate to the prime function of FLITE which is fault location, not status monitoring.

Consideration was given to the design of a totally redundant FLITE system in which there would be two logic trees operating in identical and parallel modes with a parity checking circuit at the output. Analysis suggested that this was probably the introduction of a needless complication since the FLITE will be used almost entirely by technical maintenance personnel at such times as the spacecraft is either docked or on the ground.

The FLITE logic and display assembly includes a multi-pin connector which provides a digital output for each LRU. This connector may be used with a data management system or a flight recorder, if one is used aboard the spacecraft.

The use of a mini-computer for fault analysis was considered. Again, the additional complexity caused by the inclusion of a computer which itself becomes a maintenance problem, is considered to be unnecessary, in view of the significant fault isolation capability of the basic FLITE. The data connector from FLITE leaves open the possibility that a mini-computer could be used at a later time.

Summary

The survey of standard inspection and servicing practices of representative airlines resulted in establishing basic maintainability guidelines for the EC/LSS. Major concepts, "designed-in" and "canned" are recommended.

Critical problems during postflight, refurbishment, and preflight checkout were identified and are primarily concerned with testing requirements, cleaning-up criteria, and storage of flight units at the ground facility.

The reliability analysis identified two potentially critical subsystems (humidity control and heat transport loop) as far as MTF and probability of replacement. The results of the analysis give a good indication as to where the design emphasis must be placed.

The fault isolation analysis indicates that it is practical and desirable to provide fault location detection instrumentation for critical components of the EC/LSS. Failure of the O_2/N_2 Pressure and Control subsystem is considered critical to crew safety, while failure of the Heat Transport Loop subsystem is considered critical to vehicle safety. Line replaceable units (LRU's) of the EC/LSS were identified and listed to assist in determining the best approach to implementing a design concept for fault isolation.

Preliminary requirements and a conceptual approach evolved from the fault isolation analysis. The Fault Location Indicator Test Equipment (FLITE) system offers a simple method of instrumenting LRU's and displaying faults with a light emitting diode (LED) as a basic display element.

NEW TECHNOLOGY

As a result of this study, three areas of further investigation are recommended. These are: (1) development of a fault isolation system for the EC/LSS, (2) development of thermal insulating techniques (purge bags) for cryogenic and non-cryogenic payloads, and (3) investigation of effects on equipment resulting from quiescent storage and development of techniques to compensate for any detrimental effects.

Fault Isolation

It is recommended that a simple method of EC/LSS fault identifying and locating be developed. The method should be designed primarily as an aid for trained ground maintenance technicians, but onboard usage by astronauts should not be precluded. The system suggested for this concept is described under "System Reusability" in this report. The following steps are suggested for the development of this concept:

- o Prepare a laboratory model of a representative EC/LS system or subsystem which might be used on the Shuttle. Usage of flight hardware in the model, however, would not be required.
- o Instrument the model, at the component level, in accordance with the system concept.
- o Simulate component failures and analyze accuracy of system fault identifying and locating.
- o Compare the advantages and disadvantages of this system against a fully computerized technique.

Payload Thermal Insulation

To effectively isolate cryogenic and non-cryogenic payloads from the compartment environment, it is recommended that insulation techniques as described under "Shuttle/Payload Thermal Control" be initiated. This development activity would examine two kinds of payloads: (1) those which must be maintained above a minimum temperature level, and (2) those which are cryogenic and must be protected from excessive heat leak and condensation or freezing. To develop this concept, the following steps are suggested:

- Prepare small scale laboratory models which represent the thermal characteristics of the payloads in question and of the surrounding payload compartment.

- Fabricate insulation bags for the models for both cases. The non-cryogenic payloads will also have heaters inside the insulating bags. The cryogenic insulating bags will include provisioning for helium purge.

- Subject the laboratory setup to the thermal conditions experienced by the Shuttle during prelaunch, launch, ascent and reentry. Operations will include nitrogen and helium purging as described in the report.

- Measure characteristics of heat transfer between the payload and surrounding environment.

- Establish best method of insulating the payloads considering composition and fabrication of multilayer material and purge techniques.

Quiescent Storage

Since the Shuttle will rely principally on the Space Station during the docked mode and its onboard EC/LS system will either be inactive or at a minimum operating level, further examination of the effects of this kind of operation on the equipment is warranted. Of specific interest is the effects of extended deactivation on components which are in a space vacuum environment. The lag times associated with reactivating system elements and their effects on operations are also essential considerations for investigation. Corrective action needed to compensate for undesirable effects should be considered for further development activity.

APPENDIX A

MISSION AND PAYLOAD ANALYSIS

The Mission/Vehicle Definition section presented a summary of one NASA spaceflight program and indicated the requirements that manned missions would impose on the Shuttle EC/LSS. These data resulted from a detailed operation and mission analysis presented in this appendix that examined eight NASA scientific categories which included eighty (80) payload configurations. The mission objectives and characteristics (those having an effect on the Shuttle EC/LSS) have been tabulated and are listed in table A-1. Interfaces between the payload and the EC/LSS are summarized in table A-2. Sensor/equipment, environmental protection and Shuttle support requirements are delineated.

NASA Astronomy (NAS)

These flights place large and small free-flying observatories at altitudes ranging from 230 nm to 1 astronaumical unit (A.U.). Man's role in the low orbit satellites is placement, retrieval, servicing and maintenance. One group of payloads will occupy the entire volume of the cargo module and require the maximum payload weight capability of the Shuttle. Typical of these are (1) Large Stellar Telescope, (2) Large Solar Observatory, (3) Large Radio Observatory, and (4) High Energy Astronautical Observatory (HEAO). The latter payload (HEAO) is unique in that it requires extensive calibration and alignment prior to operation. It is estimated that a 28 day start-up period is required with subsequent revisits for the purpose of refurbishing, pickup of films, etc.

It is envisioned that the above type of payloads will require initial placement in orbit of the payload module, and initiation of start-up procedures. This could be accomplished minimally with a Shuttle crew of four (pilot/co-pilot plus two technicians) or maximally with the Shuttle crew of four plus two additional technicians carried in the cargo module. The latter case would require placement of an EC/LSS module in the cargo module. Additional logistics support for this class of payload is required. Carrying out servicing could be accomplished either by EVA or IVA. The payload design layouts indicate a fairly dense packaging with telescope, antennas, cameras, etc. However, it appears feasible to design the interior of the payload module in such a way that one or more technicians could enter and have adequate working space. Making all the equipment accessible to EVA does not appear practical except for film storage accessibility. A combination EVA/IVA approach appears optimum at this time.

The procedure would consist of docking the cargo module to the free-flying stabilized module with personnel entering the payload module through the air-lock. Environmental Control and Life Support functions would be carried out by the cargo module EC/LSS. Since the payload module normally does not require environmental protection, it will be necessary to pressurize the module and provide thermal control prior to the technicians entry.

NASA Space Physics (NSP)

These flights place large and small payload modules in orbit at low circular altitudes, highly elliptical, synchronous, 1 A.U. and aboard the Space Station. But of the twelve configurations studied, only one appears to require a EC/LSS interface (High Energy Cosmic Ray Laboratory). The rest either have orbits that are not practical for logistics support or are placed aboard the station which supplies any EC/LSS requirement. The High Energy Cosmic Ray Laboratory does particle counting and would normally be unmanned. Logistic support is required for picking up and replacing exposed plates, servicing cosmic ray counters and range energy detectors. This observatory will occupy the entire volume of a payload module. The same logistic support techniques envisioned for the Astronomy payload modules would apply.

NASA Space Applications (NSA)

These flights place small to medium satellites into low to synchronous orbits. These satellites are unmanned except for one possible exception and do not require manned servicing. Their lifetimes are normally 1 to 2 years and will be replaced by later series satellites. The one exception is the Earth Observatory Station, which will either be manned for short duration missions (2 to 30 days) or will require manned servicing on a routine basis. An alternative to this type of mission is installation of an Earth Observation module in the Space Station which then could be maintained and serviced by Station personnel. If the observatory becomes a free-flying module, then the same technique employed for other detached modules will apply.

Non-NASA Operational (NNO)

These flights place small satellites (less than 3,500 lb) into high or synchronous orbits. Revisits with EC/LSS support might prove practical since some of their lifetimes are as long as 7 years. Cameras, spectrometers and radiometers are equipment that require logistic support. Because of the small size of the satellites (6 ft. diameter spheres), it is feasible to retrieve the satellites into the cargo module and have technicians refurbish them. This would require cargo module EC/LSS support.

NASA Bioscience (NBI)

These flights place bioscience modules either aboard the Space Station, or on a high orbit, or in a low circular orbit. The Space Station bioscience module will be serviced by Station personnel and will be supplied by routine logistic support. The other two classes of flights will be operated independently and will not require EC/LSS support.

NASA Lunar Option 2 (NL2)

These flights are planned for lunar support. Transportation of personnel and equipment between the Space Station and Lunar Station and from an orbiting Lunar Station to the Lunar Base Station and back are typical missions. Science cargos, consumables, and crew rotation will be supported. This program is beyond the scope of the present study. The EC/LSS support required for this type of mission will evolve from the Space Shuttle EC/LSS. Consumables requirements for lunar crewmen will be approximately 3 men, 180-days which calls for a considerable advancement over the presently contemplated Shuttle EC/LSS.

NASA Support (NSU)

The Space Station support flights places the largest demand on the Shuttle EC/LSS. They provide logistics support to the Space Station. A total of 44 flights are required over the ten year period. These flights transport 6 to 10 passengers in the cargo module for crew replacement.

The Shuttle will dock the cargo module to the Station. Egress from the Shuttle will take place through an air-lock. The majority of EC/LSS support during the five (5) day transfer period will be provided by the Station.

Other NSU flights will require minimal Shuttle EC/LSS support. For example, propellant flights for Tug Support is a sortie mission with the Shuttle crew monitoring an automatic transfer of the propellant either into a Propellant Facility or into the Tugs.

NASA Planetary (NPL)

These flights are designed to furnish planetary data. They will be unmanned and will be launched either from a low earth orbit or from the Space Station. The major function of the Shuttle will be placement of both small and large payload modules into the proper orbit. Automatic devices will be provided that extend and separate the payload module from the Shuttle. Independent propulsion systems will be activated to place the satellite on the planetary transfer maneuver.

APPENDIX B

SYSTEM EFFECTIVENESS PROGRAM (SYEFF) COMPUTER PRINTOUT

A copy of a typical computer printout is shown in table B-1 to illustrate the optimizing process used in SYEFF. The example selected represents the 100 mission analysis performed for the humidity control subsystem. Similar analysis and computer printout data was developed for each subsystem and for the total EC/LSS. The computer data contained in the table is consistent with the single thread reliability block diagram (fig. 24) and has been provided as a plot output in terms of Reliability versus Time and Reliability versus Weight on figures 25 and 26 respectively.

The first column on table B-1 is entitled STATE and represents the number of state by state decision calculations made on this run. There were 28 state decisions made as illustrated on the table.

The second column is entitled CODE and is used to identify an acceptable (2) or an unacceptable (4) decision. After a 4 code is listed, the computer selects an item with the next highest increase in reliability which will not go over the weight limitation. If the item selected meets the constraints, then a code 2 is listed. As noted on the table, the 27th state decision resulted in 160.0 lb (the preselected limit) while the 28th state decision resulted in exceeding the weight limit. The search ended since the maximum weight was exceeded and computer time allowance was used up.

The third column, KAPPA, is an assigned item number. In this example, there were 44 critical items. As noted on the table under state 1, decision Kappa 2 (relief valve as called out on figure 24) is added to the subsystem as a redundant item. For identification and weight of all the Kappas listed on table B-1, refer to figure 24.

Column 4 lists the initial weight (lb) of the humidity control subsystem (142.4 lb) and the subsequent weight increase as redundant items are added. Note that the end term following the weight indicates the unit places in the weight column.

The mean or average life as shown in column 5 is based on the failure rate of the subsystem under investigation. The failure rate is taking into account the added redundant items.

The reliability numeric shown in column 6 is based on the design life, which is the specified mission time in this case, 100 missions or 16,800 hours.

The mean life of the subsystem will always be lower than the design life, since the reliability is never one, but the mean life approaches the specified mission time as redundancy is added and the reliability approaches one. At the reliability of 1.0, the mean life would be equal to the design life. A reliability of less than one, for instance R = .995 at 16,800 hours, means that given 1000 items, 5 of them would fail before the required time period of 16,800 hours is completed. Therefore, the mean or average life of 1000 items has to be less than the 16,800 hours.

For the subsystem selected, table B-1 shows that a total of 28 state by state decisions were executed. At state 27, the maximum allowable weight (160 lb) was reached by adding Kappa 3 (debris trap). MTF calculated was 14,371 hours which is substantially below the design life of 16,800 hours (100 missions). The resultant subsystem reliability is listed as 0.720454.

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ABSTRACT

This report identifies and examines significant problems associated with the Space Shuttle environmental control and life support system (EC/LSS). Four problem areas were investigated: (1) Cargo Module EC/LSS Definition, (2) Space Shuttle/Space Station Interfaces, (3) Shuttle/Payload Thermal Control, and (4) System Reusability. Mission guidelines were established for 420 flights over a 12 year period. Ninety-three (93) of these flights requires a cargo module EC/LSS for passenger complements which vary from two to ten. The interacting Shuttle/Station EC/LSS responsibilities for docked and autonomous operating modes were established. The feasibility of providing thermal protection to payloads during launch and reentry by using a purge bag was investigated. A reliability and mission criticality analysis was performed to determine critical subsystem elements for crew/vehicle safety. Fault detection instrumentation was assigned on the basis of element practicality and safety. Turnaround operational procedures and maintainability guidelines were defined.