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LMSC
SPACE SYSTEMS
DIVISION

FINAL REPORT
INTEGRAL LAUNCH AND REENTRY VEHICLE
VOLUME I
CONFIGURATION DEFINITION
AND PLANNING

PART B - SECTIONS 10 THROUGH 17

FINAL REPORT

INTEGRAL LAUNCH AND
REENTRY VEHICLE
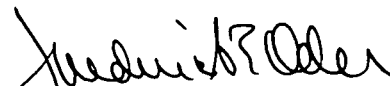
Volume I

Configuration Definition and Planning

Part B — Sections 10 through 17

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FOREWORD

This final report for the Integral Launch and Reentry Vehicle (ILRV) Study, conducted under Contract NAS9-9206 by Lockheed Missiles & Space Company under direction of the NASA Marshall Space Flight Center, is presented in three volumes. Volume I, Configuration Definition and Planning, contains results of the preliminary cost analyses, conceptual design, mission analyses, program planning, cost and schedule analyses, and sensitivity analyses, accomplished under Tasks 1 through 6. Volume II covers Task 7, Technology Identification; and Volume III contains results of the Special Studies conducted under Task 8.

Principal LMSC task leaders and contributors in performance of this study include:

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Test	R. W. Benninger	Aerodynamics	C. F. Ehrlich
Operations	K. Urbach	Weights	A. P. Tilley

The three volumes are organized as follows:

Volume I - Configuration Definition and Planning

Section

1	Introduction and Summary
2	System Requirements
3	Configuration Summary
4	Vehicle Design
5	Performance and Flight Mechanics
6	Aerodynamics
7	Aerothermodynamics
8	Structures and Materials
9	Propulsion

CONTENTS

v

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Section 10

AVIONICS

The electrical power; guidance, navigation, and control; communications; and data management subsystems are discussed in the following paragraphs. The study results for integrated avionics approaches are given in Volume III of this report.

10.1 ELECTRICAL POWER

Electrical power during all mission phases from liftoff through final landing is supplied by a system of fuel cells and emergency batteries on the orbiter and by primary batteries on the booster. The high mechanical power requirements of both stages associated with high-rate actuation of aerodynamic control surfaces are supplied by a chemical turbine-driven hydraulic system during the reentry and landing phases, as described in Section 9 - Propulsion.

The Space Shuttle has several requirements that impact electrical power system development technology. This technology, which is identified in Volume II, should be considered in the selection of the power system source and supporting equipment. The atmosphere entry period imposes a cooling problem on both the primary and standby power systems. Fuel cells, the leading candidate as the primary source for the orbiter, will require open-cycle operation or, possibly, evaporative cooled cells. Lightweight high-efficiency power control and distribution systems will be required, as will aircraft-type maintenance procedures and modular design to facilitate maintenance. Test plugs and instrumentation for ground checkout and diagnostics will be integrated into the power system hardware.

10.1.1 Recommended Configuration

Total power systems output requirements, shown in Table 10-1, for the booster and orbiter prime power source are based on subsystem power requirements as a function of specific mission phase.

Table 10-1

TWO STAGE PRIMARY POWER REQUIREMENTS SUMMARY

ORBITER				
Phase	Duration (hr)	Average Power (w)	Losses (w)	Energy (kw-hr)
Preflight	1.5	6,990	515	11.25
Launch and ascent	0.11	4,650	345	0.55
Orbit	168.0	4,875	350	876.0
Rendezvous and docking	3.0	7,980	578	25.68
Deorbit	0.66	5,260	386	3.71
Entry	0.66	4,930	368	3.50
Landing	0.27	15,035*	1,112	4.36
Total energy required				925.05
BOOSTER				
Preflight	1.5	6,914	496	11.10
Launch and ascent	0.05	4,659	336	0.25
Reentry	0.13	4,603	333	0.64
Cruise and landing	1.17	14,746*	1,091	18.53
Total energy required				30.52

*Includes 6,500 watts provided by cruise engine generators for cruise engine operation and fuel boost pumps.

System losses reflect 5 percent distribution losses and 85 percent inverter efficiency. Figure 10-1 shows the predicted power usage profiles for the booster and orbiter missions.

BOOSTER PROFILE

CRUISE AND LANDING

REENTRY

PREFLT

LAUNCH AND ASCENT

POWER (KW)

TIME (HR)

ORBITER PROFILE

PREFLIGHT

LAUNCH AND ASCENT

ENTRY

POWER (KW)

RENDEZVOUS
AND DOCKING

APPROACH
AND LANDING

DEORBIT

ORBIT

TIME (HR)
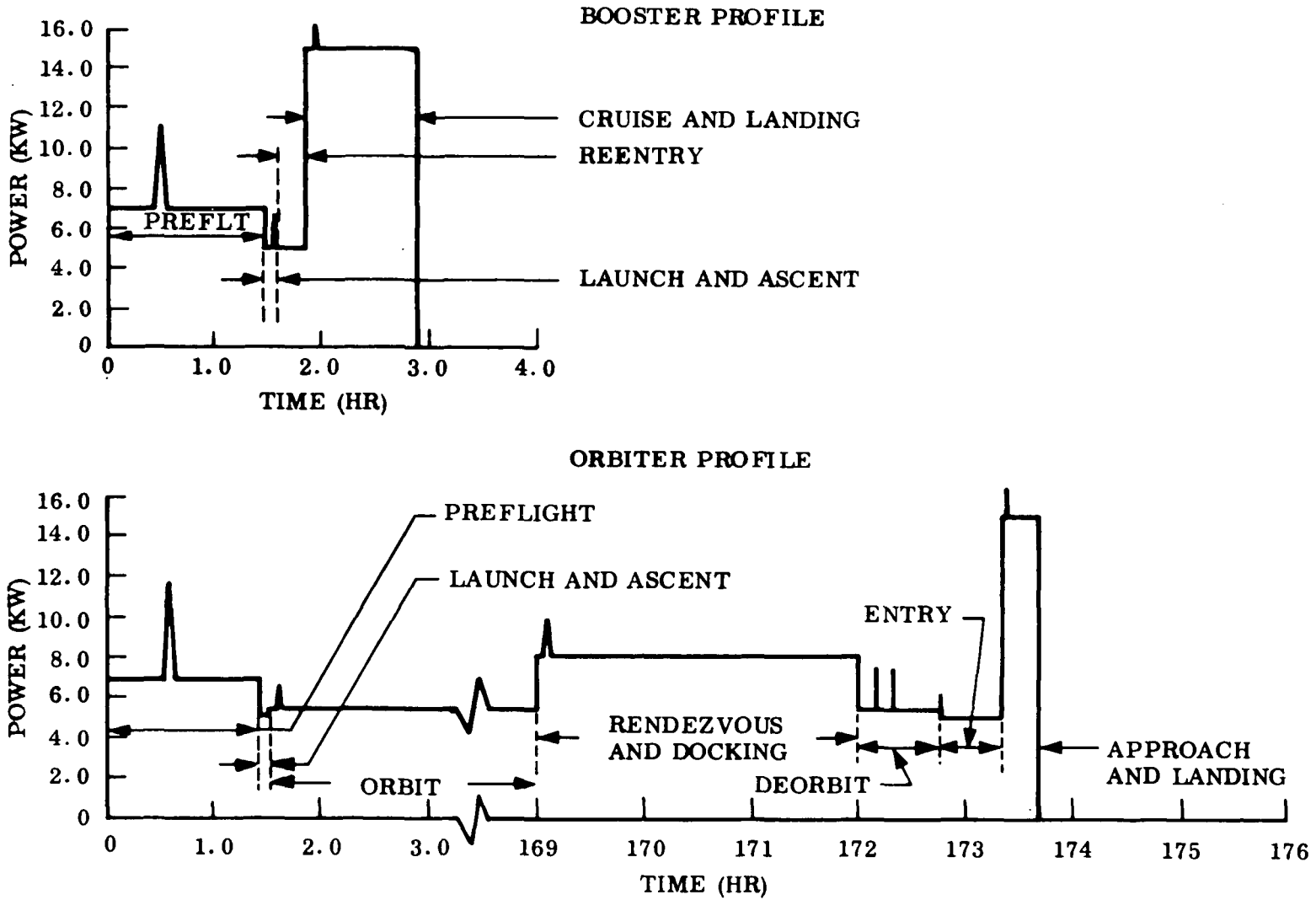
Fig. 10-1 Two-Stage Power Profile

The major components selected for the orbiter power system functions are shown in Table 10-2. A block diagram of the electrical power subsystem is shown in Fig. 10-2. Provisions is made for power to be supplied by alternators driven by the main turbojet thrusters during vehicle operation in the atmosphere.

Table 10-2

ORBITER EPS SELECTED COMPONENTS (BASELINE)

Functions	Components
Primary power generation	Capillary matrix 5-kw, liquid-cooled, fuel cell modules (3 ea)
Emergency power	Batteries, 86 AH, silver zinc (6 ea), 12,800 w-hr, 215 lb
Conversion	Solid state inverters, 3φ, 400 cps, 115 v, 4 ea
Distribution	DC, 28 ±3 volts, two buses, AC, 3φ, 400 cps, 115 v two buses
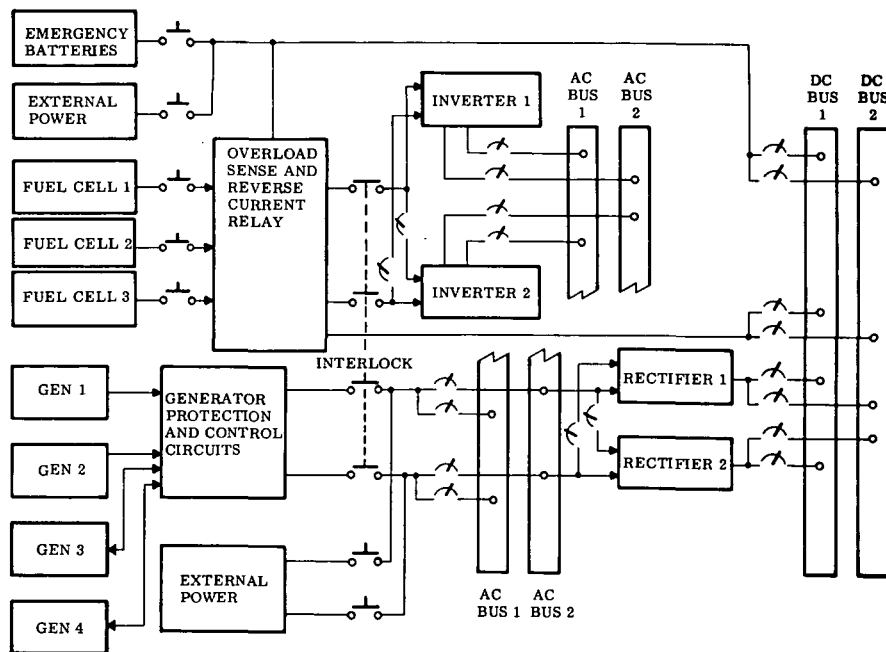
Fig. 10-2 Electrical Power Subsystem Block Diagram – Orbiter

10-4

The primary power source employs two high-performance capillary matrix fuel-cell modules with one set of supporting assemblies. The characteristics of one fuel-cell module is shown in Table 10-3 and schematically in Fig. 10-3.
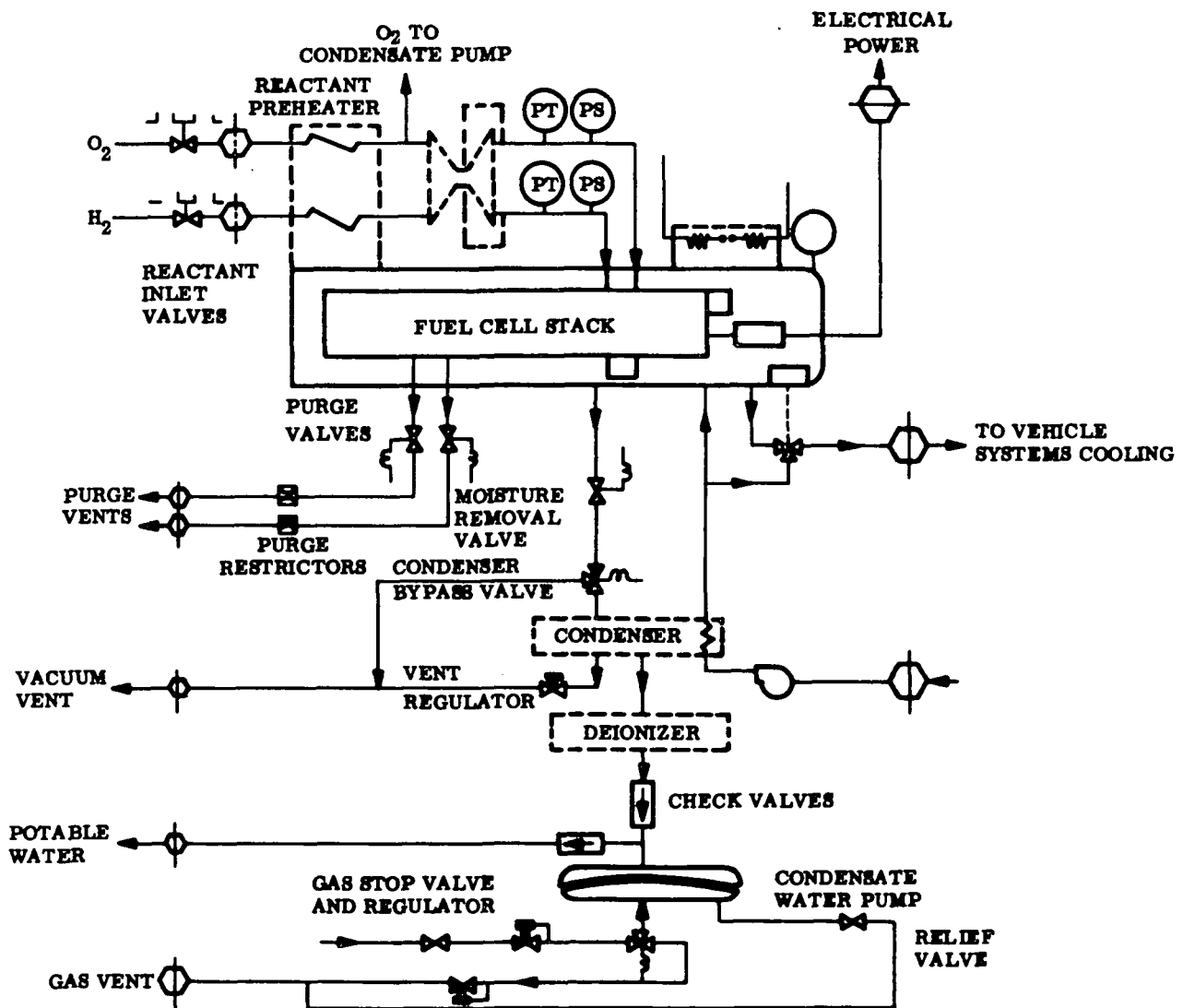
Fig. 10-3 Fuel Cell System Schematic

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Table 10-3

FUEL CELL MODULE CHARACTERISTICS

Description	Data
Type	Capillary matrix
Dimensions (in.)	12 x 12 x 38 (3.2 ft^3)
Weight (lb)	160
Power rating (kW)	5, nominal
Nominal voltage	28
Parasitic power (W)	150
Operating temperature (^{o}F)	195
Reactants	Oxygen and hydrogen
Operating life (hr)	3,000
Number of cell sections	30
Dual cooling mode	Liquid and evaporative

Three reactant storage tank sets are used to provide fail-operational capability with 50 percent reactant redundancy, as shown in Table 10-4.

Table 10-4

FUEL CELL REACTANT STORAGE DATA

Description	Data
Specific reactant consumption (SRC) (lb/kw-hr)	0.8
Weight H_2 required for power plus 50 percent (lb)	137.1
Weight O_2 required for power plus 50 percent (lb)	1096.7
Storage mode	Supercritical
Storage pressure (psia)	300 (H_2), 850 (O_2)
Outer diameter spherical tanks (3 ea) (in.)	39.8 (H_2), 32.5 (O_2)
Fueled weight tank system (3 ea) (in.)	109.6 (H_2), 490.4 (O_2)
Reactant storage system total weight (1 ea) (lb)	1800

Silver-zinc batteries were selected as the primary electrical power source for the booster stage. The EPS functions and selected components are shown in Table 10-5. Figure 10-4 is a block diagram showing the interfaces and connections of the major components of the booster power system.

Table 10-5

BOOSTER SELECTED COMPONENTS (BASELINE)

Function	Components
Prime power generation	Batteries, silver-zinc (12 ea)
Conversion	Solid-state inverters, 3φ, 400 cps 115 v (4 ea)
Distribution	DC, 28 ±3 V, two buses, AC, 3φ, 400 cps, 115 V, two buses
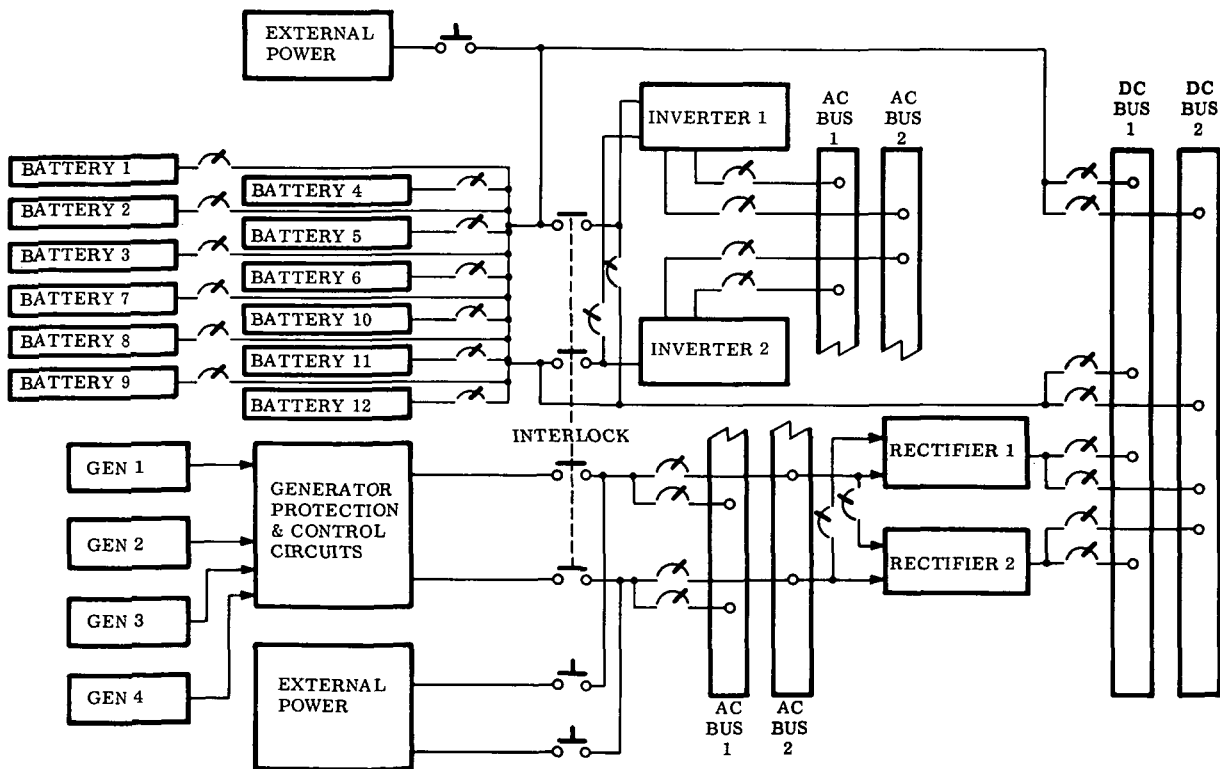
Fig. 10-4 Electrical Power Subsystem Block Diagram – Booster

Of the 12 batteries supplied, nine are required and three are redundant. The characteristics of the silver-zinc batteries are given in Table 10-6.

Table 10-6

SILVER-ZINC BATTERY CHARACTERISTICS

Description	Data
Energy required (w-hr)	24. 6 kw-hr (includes 6. 1 kw-hr supplied for redundancy)
Energy density (high-rate) (w-hr/lb)	60
Battery weight (lb)	410
Battery volume (ft^3)	7. 15

10. 1. 2 Alternate Approaches

Analysis of the electrical power system requirements for the orbiter and booster stages resulted in consideration of the following elements:

- Primary power sources

  Capillary matrix fuel cells
  Cryogenic chemical dynamic turbine
  Silver-zinc batteries

- Power distribution system

  Centralized or decentralized conditioning equipment
  Standard harnessing or lightweight flat conductor cable

10. 1. 3 Evaluation of Alternative Approaches

Silver-zinc batteries were ruled out as the orbiter power source because a nonredundant battery complement is seven times heavier than either of the other candidate sources (11, 000 vs. 1, 600 lb). Although the chemical dynamic turbine is competitive in terms of weight, it uses propellants less efficiently than a fuel cell in order to limit gas temperatures. Therefore it is not considered as a candidate at this time. Comparison of major fuel cell systems data, including module voltage vs. power and current density, led to the consideration of a high-rate, liquid-cooled capillary-

matrix fuel cell similar to the 4.5-kw Pratt & Whitney PC8B-3 or the 5.0 kw Allis-Chalmers unit. A potential exists for achieving much higher cooling and resultant power rates by the incorporation of evaporative cooling in these units.

The same candidate power sources evaluated for the orbiter were considered for the booster stage. In this instance, the total energy requirement of 18.4 kw-hr reduces the significance of the weight criterion, since the use of any of the three alternatives results in a weight of 410 pounds or less. Silver-zinc batteries were selected on the basis of highest energy density, less development risk, ease of vehicle refurbishment through battery replacement, and lower cost.

An emergency power source of 12.8 kw-hr capability is required for the deorbit, reentry, and landing phases of the orbiter vehicle. The similarity of this requirement to that of the primary power source for the booster led to the selection of the silver-zinc batteries for this application. However, to reduce flight-to-flight cost, rechargeable (limited) or remote activated Ag-Zn batteries should be considered.

Both flat conductor cabling and the more conventional round-wire cabling have been considered for power and signal distribution. Lightweight flat cable will be used throughout the orbiter and booster for power distribution. In addition to the in-flight weight saving they permit, flat conductor cabling harnesses have the following advantages over harnesses made with round-wire cabling:

- More strength and flexibility because of the collective strength of the conductors and their stronger insulation
- More efficient heat dissipation due to increased surface area
- Controlled and reproducible electrical characteristics
- Up to 40 percent less installed volume
- Lower production and assembly costs because of reduced errors, ease of inspection, and durability
- Uniform vehicle-to-vehicle wiring

The use of aluminum conductors instead of copper for high-rate power distribution can also result in significant weight savings.

10-9

The baseline configuration is shown for simplicity with two 28-volt unregulated dc buses and two 115-volt ac buses. Final decisions concerning power conditioning before and after distribution can only be made when subsystem requirements are defined in more specific detail. It is probable that dc power will be distributed at voltage higher than 28 and decentralized ac power conditioning will be incorporated in some areas.

10.2 GUIDANCE, NAVIGATION, AND FLIGHT CONTROL

The guidance, navigation, and flight control system must provide position determination, trajectory control, attitude control, and stability augmentation during all mission phases. It must be capable of completely autonomous operation after prelaunch data readin, except for a landing guidance data link. It must be capable of both fully automatic and manual control through all mission phases. Table 10-7 lists some of the constraints that the system imposes on the orbiter.

Table 10-7

GNC CONSTRAINTS ON VEHICLE DESIGN

	Mission Phase
External (Field of View)	
IMU-within optical view of external alignment reference	(prelaunch)
Horizon sensor − 170 deg cone, centerline to earth center	(orbit)
Star sensor − 6 deg cones within 36 deg overall cone, looking forward	(orbit)
Radar altimeter − earth-pointed dish antenna	(orbit)
Rendezvous sensor-forward pointing RF antenna or optical sensor head	(orbit)
Landing approach transponder − RF cone, forward-looking, depressed	(landing)
Radar altimeter − earth pointed dish antenna	(landing)
ILS navigational equipment antennas − RF cones, forward, and down pointing	(landing)
Air data sensors − freestream ram and static air pressures	(post-reentry)
Internal: (Installation and Orientation)	
IMU − within optical view of sensors, close as possible	(prelaunch, orbit)
Aligned to prime structure ±0.05°	(prelaunch)
Adjacent to computers for tap of heat dissipation	
Computer − Reasonably near data management computer high heat dissipation installation	
Horizon sensor − aligned to prime structure ±0.05°	(prelaunch)
protect from rapid temperature changes	
Star sensor − aligned to IMU block ±0.05°	(prelaunch)
Rate gyros − package located aft	

10.2.1 Subsystems

The system consists of seven subsystems having well defined interfaces. These are
illustrated in Fig. 10-5.

```
┌─────────────────────────────────┐      ┌─────────────────────────────────┐
│ TARGETING/GUIDANCE SUBSYSTEM    │      │ FLIGHT CONTROL SUBSYSTEM        │
│                                 │      │                                 │
│ ● TARGETING EQUATIONS           │      │ ● DRIVE ELECTRONICS             │
│     PRELAUNCH, PRETRANSFER      │      │   (TVC, RCS, ACS, TCS)          │
│     PREDESCENT                  │      │ ● ELECTROHYDRAULIC ACTUATORS    │
│                                 │      │ ● STABILITY AUGMENTATION        │
│ ● GUIDANCE EQUATIONS            │      │   EQUATIONS                     │
│     ASCENT, TRANSFER BURN,      │      │ ● ATTITUDE CONTROL EQUATIONS    │
│     REENTRY                     │      │                                 │
└─────────────────────────────────┘      └─────────────────────────────────┘

┌─────────────────────────────┐          ┌─────────────────────────────┐
│ STATE VECTOR SUBSYSTEM      │          │ RENDEZVOUS/DOCKING          │
│ ● IMU                       │          │        SUBSYSTEM            │
│ ● HORIZON SENSOR            │  ┌──────────────┐                      │
│ ● STAR SENSOR               │  │ COMPUTER     │ ● RENDEZVOUS SENSOR  │
│ ● RADAR ALTIMETER           │→ │ SUBSYSTEM    │                      │
│ ● RATE GYRO PACKAGE         │  │ ● COMPUTER   │← ● DOCKING SENSOR    │
│ ● AIR DATA SENSOR           │  │   SOFTWARE   │                      │
│ ● SKIN TEMPERATURE SENS     │  │ ● COMPUTER   │ ● DATA PROCESSING    │
│ ● DATA PROCESSING EQ.       │  │   HARDWARE   │   EQUATIONS          │
└─────────────────────────────┘  └──────────────┘                      │
                                                 └─────────────────────────┘

┌─────────────────────────────┐          ┌─────────────────────────────┐
│ NAVAID SUBSYSTEM            │          │ AUTOMATIC LANDING           │
│ ● VOR-DME                   │          │      SUBSYSTEM              │
│ ● ADF                       │          │ ● BEACON TRANSPONDER        │
│ ● ILS                       │          │                             │
│ ● RADAR ALTIMETER           │          │ ● DATA LINK RECEIVER        │
│ ● MULTIMODE RADAR           │          │ ● DATA PROCESSING           │
│ ● DISPLAY EQUATIONS         │          │   EQUATIONS                 │
└─────────────────────────────┘          └─────────────────────────────┘
```

Fig. 10-5 Guidance, Navigation, and Flight Control System Elements (Orbiter)

10.2.1.1 <u>State Vector Subsystem.</u> The state vector subsystem, composed of state-sensing instruments and associated data processing equations, provides a complete description of the vehicle state with respect to the outside world during all mission phases. Typical state parameters are vehicle location, velocity, attitude, dynamic pressure, and angle of attack. Since this subsystem is the source of all state data, it performs more than just the navigation function.

The baseline orbiter state vector subsystem is fully automatic and insensitive to cloud cover and sun position. It consists of an inertial measurement unit (IMU), strapdown star sensor, a horizon sensor, a radar altimeter, a rate gyro package, an air data sensor, skin temperature thermocouples, and associated filtering routines. The numbers of individual equipment packages will be determined by an analysis of system reliability and safety. A manual state update capability on orbit would employ a sextant and landmark tracker. The booster will require only the IMU, rate gyros, and air data sensor from this group.

10.2.1.2 <u>Targeting and Guidance Subsystem.</u> The targeting and guidance subsystem is an information system only; its hardware requirements are met by the other sub-systems. It consists of equation routines for processing state data to compute trajec-tory and guidance constants and in-flight steering and throttling orders, as well as ignition and cutoff discretes to the flight control subsystem and to the pilot's displays.

10.2.1.3 <u>Flight Control Subsystem.</u> The flight control subsystem processes state data and guidance commands to produce actuation signals and pilot display data. In addition to equations, it includes the drive electronics and electrohydraulic servo actuators.

10.2.1.4 <u>Rendezvous and Docking Subsystem.</u> The rendezvous and docking subsystem is composed of sensors (e.g., laser transmitter and receivers) and associated data-processing equations. This subsystem provides relative state data and steering and thrusting orders to the flight control subsystem and also pilot display data.

10.2.1.5 <u>Approach and Landing Subsystem.</u> The approach and landing subsystem is composed of sensors (e.g., radar transponders) and associated data processing equations. This subsystem provides relative state data and steering and thrusting orders to the flight control subsystem and pilot display data.

10.2.1.6 <u>Navaid Subsystem.</u> The Navaid subsystem is composed entirely of RF equipment used for navigation during aircraft operations — ADF, VOR-DME, ILS, radar altimeter, and possibly a multimode radar. ILS equipment is included on the assumption that it will be necessary for routine low-visibility flight operations at any commercial or military airfield. The approach and landing subsystem, on the other hand, is specifically oriented towards automatic Category 3 operations at designated sites. These two landing systems may be consolidated after further study.

10.2.1.7 <u>Computer Subsystem.</u> Each of the six elements of the system contains its own computations. The equation sets are combined and then programmed into the computer subsystem, which may be part of a federated complex or a dedicated unit totally within the guidance, navigation, and flight control system.

10.2.2 Operating Modes

The location of the equipment is indicated in Fig. 10-6, and the normal and backup operating modes for each flight phase are shown schematically in the block diagrams of Figs. 10-7 through 10-12. Not shown is equipment redundance, which reflects complete compliance with Space Shuttle reliability and safety criteria.

Normal modes are fully automatic, pilot monitored. Backup modes, used in the event of loss of a critical primary function, employ the pilot in an active role. To provide a true backup capability, it is assumed that an independent, emergency IMU and computer are used to generate critical ascent and reentry display information.

The NAVSAT data link (Fig. 10-8) may be used to replace any one or two orbit state sensors. The manually operated state sensors are used singly or in combinations as appropriate.

1 INERTIAL MEASUREMENT UNIT

2 GUIDANCE/NAVIGATION/FLIGHT
CONTROL COMPUTER

3 HORIZON SENSOR

4 RATE GYRO PACKAGE

5 RADAR ALTIMETERS

6 RENDEZVOUS AND DOCKING
SENSOR

7 STAR SENSOR

8 AIR DATA SENSORS

9 APPROACH/LANDING TRANSPONDER

10 THRUST VECTOR/FLOW CONTROL
DRIVE ELECTRONICS

11 CONTROL SURFACE DRIVE
ELECTRONICS

12 SERVO PACKAGE

13 REACTION CONTROL THRUSTERS
DRIVE ELECTRONICS

14 REENTRY TEMPERATURE SENSOR

15 TURBOJET CONTROL ELECTRONICS

16 MULTIMODE RADAR

17 VOR-DME RECEIVER

18 ILS LOCALIZER AND GLIDE SLOPE
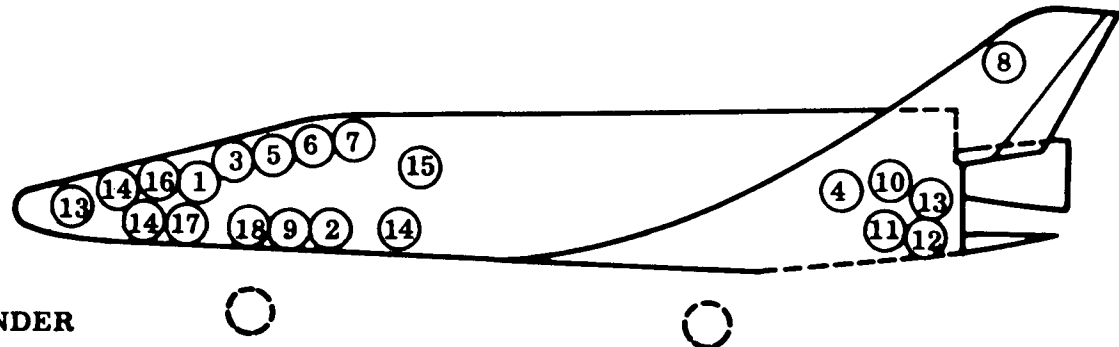
Fig. 10-6 Guidance, Navigation, and Control Equipment Installation

NORMAL
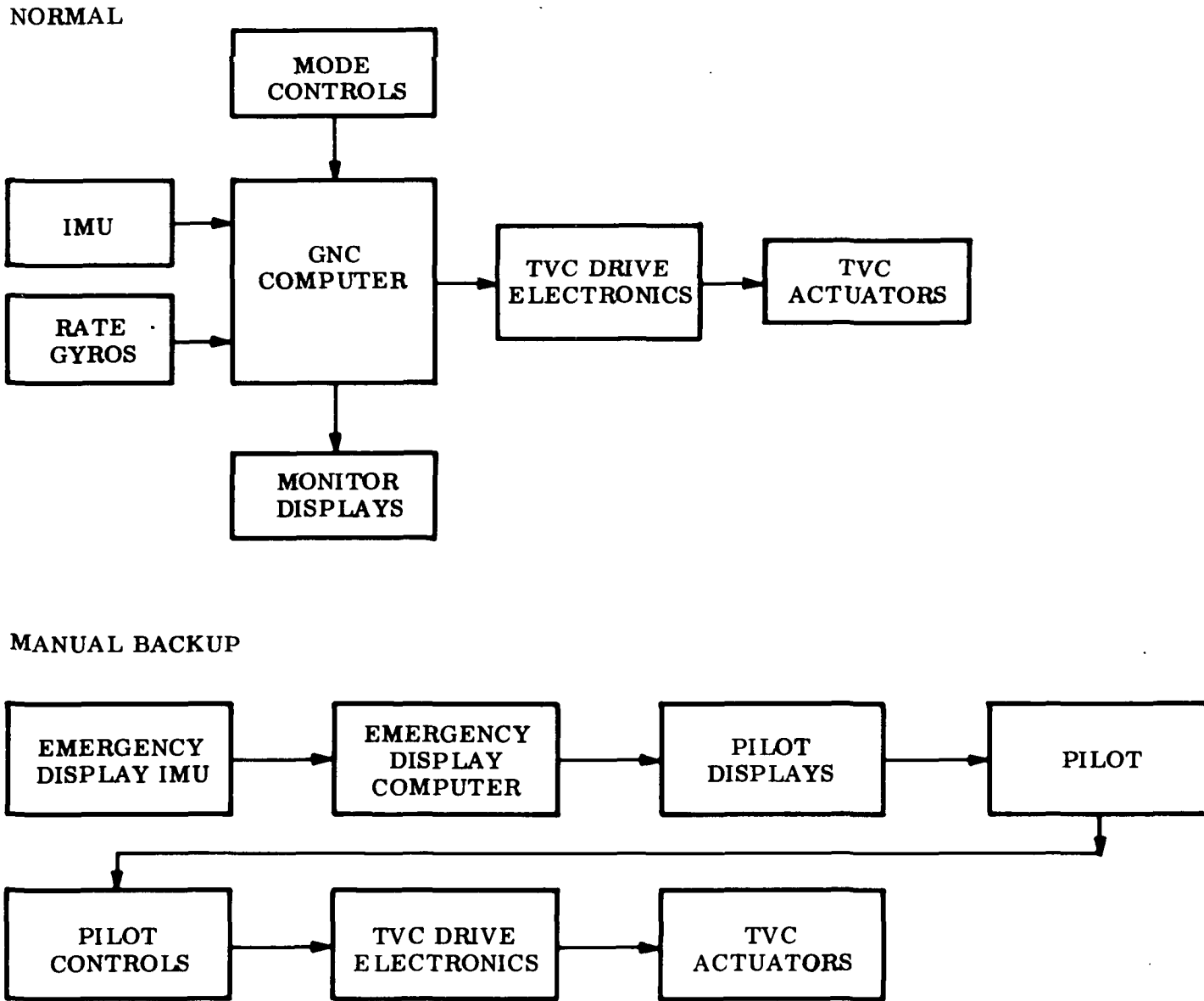
MANUAL BACKUP

Fig. 10-7 Ascent Phase Modes

NORMAL
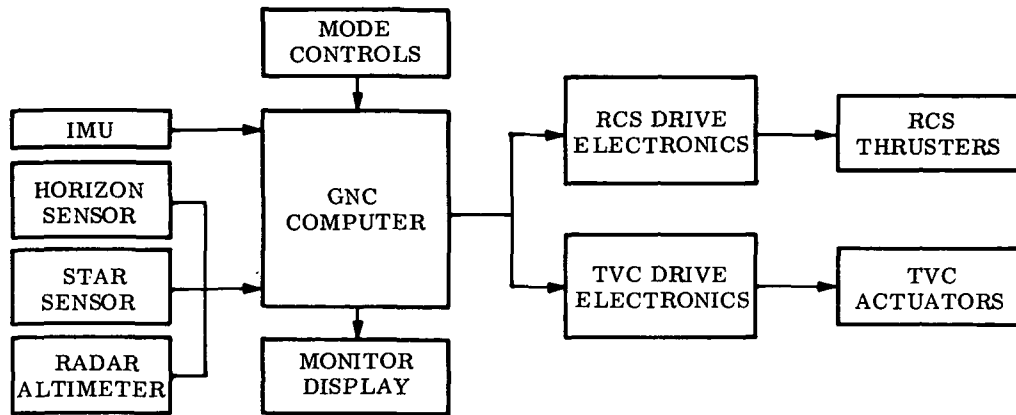
AUTOMATIC BACKUP (REPLACES ANY TWO SENSORS)
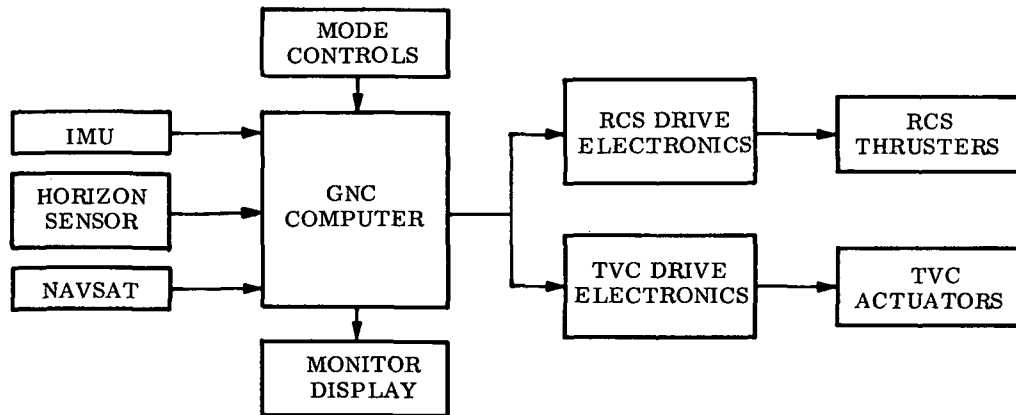
MANUAL BACKUP
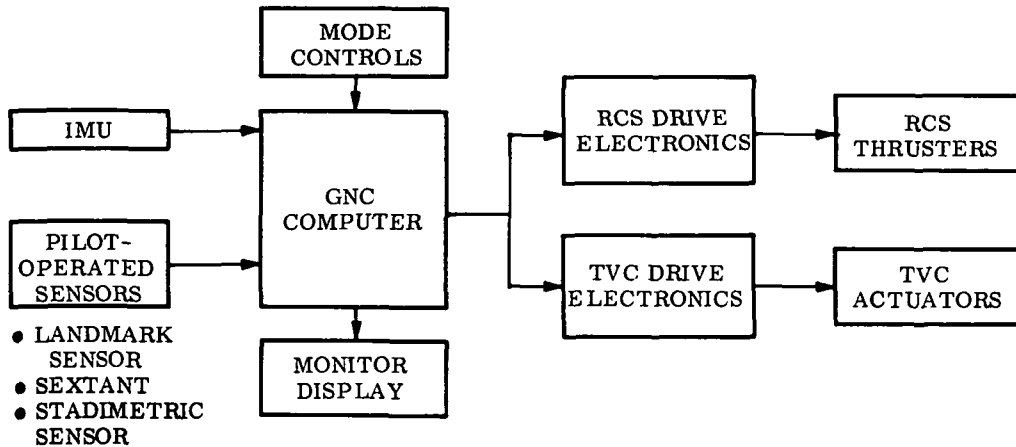
Fig. 10-8 Orbit Phase Modes

10-17

NORMAL

MODE
CONTROLS

IMU

RENDEZVOUS
SENSOR

GNC
COMPUTER

RCS DRIVE
ELECTRONICS

RCS
THRUSTERS

TVC DRIVE
ELECTRONICS

TVC
ACTUATORS

MONITOR
DISPLAY

AUTOMATIC BACKUP

MODE
CONTROLS

IMU

DATA LINK
FROM
STATION

GNC
COMPUTER

RCS DRIVE
ELECTRONICS

RCS
THRUSTERS

TVC DRIVE
ELECTRONICS

TVC
ACTUATORS

MONITOR
DISPLAY

MANUAL BACKUP

IMU

GNC
COMPUTER

PILOT
DISPLAYS

PILOT

PILOT
CONTROLS

RCS DRIVE
ELECTRONICS

RCS
THRUSTERS

TVC DRIVE
ELECTRONICS

TVC
ACTUATORS

WINDOW
RETICLE
AND
OPTICAL
FILTER

Fig. 10-9 Rendezvous Phase Modes

NORMAL

```
              ┌──────────────┐
              │     MODE     │
              │   CONTROLS   │
    ┌───────┐ └──────┬───────┘
    │  IMU  │        │
    └───┬───┘        ▼
┌──────────────┐ ┌──────────────┐   ┌──────────────┐   ┌──────────────┐
│   DOCKING    │→│     GNC      │──▶│  RCS DRIVE   │──▶│     RCS      │
│   SENSOR     │ │   COMPUTER   │   │ ELECTRONICS  │   │  THRUSTERS   │
└──────────────┘ └──────┬───────┘   └──────────────┘   └──────────────┘
┌──────────────┐        │
│    SPACE     │        ▼
│   STATION    │ ┌──────────────┐
│   DOCKING    │→│   MONITOR    │
│   SENSOR     │ │   DISPLAYS   │
└──────────────┘ └──────────────┘
```

MODE CONTROLS — IMU — DOCKING SENSOR — GNC COMPUTER — RCS DRIVE ELECTRONICS — RCS THRUSTERS — SPACE STATION DOCKING SENSOR — DATA LINK — MONITOR DISPLAYS

MANUAL BACKUP

```
┌───────┐
│  IMU  │
└───┬───┘
    ▼
┌──────────┐   ┌──────────┐   ┌────────┐   ┌──────────┐   ┌──────────────┐
│   GNC    │──▶│  PILOT   │──▶│ PILOT  │──▶│  PILOT   │──▶│  RCS DRIVE   │
│ COMPUTER │   │ DISPLAYS │   │        │   │ CONTROLS │   │ ELECTRONICS  │
└──────────┘   └──────────┘   └───┬────┘   └──────────┘   └──────┬───────┘
                                  ▲                               ▼
       ┌──────────────┐    ┌──────────┐              ┌──────────────┐
       │  ALIGNMENT   │    │  WINDOW  │              │     RCS      │
       │  MARKS ON    │──▶│ RETICLE  │              │  THRUSTERS   │
       │SPACE STATION │    └──────────┘              └──────────────┘
       └──────────────┘
```

Fig. 10-10 Docking Phase Modes

NORMAL

```
                    ┌──────────┐
                    │   MODE   │
                    │ CONTROLS │
                    └────┬─────┘
                         ▼
┌──────────┐        ┌──────────┐        ┌──────────────┐       ┌──────────┐
│   IMU    │        │          │        │  RCS DRIVE   │       │   RCS    │
└──────────┘        │          │───────▶│ ELECTRONICS  │──────▶│ THRUSTERS│
┌──────────┐        │   GNC    │        └──────────────┘       └──────────┘
│  RATE    │───────▶│ COMPUTER │
│  GYROS   │        │          │        ┌──────────────┐       ┌──────────┐
└──────────┘        │          │───────▶│  ACS DRIVE   │──────▶│   ACS    │
┌──────────┐        │          │        │ ELECTRONICS  │       │ ACTUATORS│
│   SKIN   │        └────┬─────┘        └──────────────┘       └──────────┘
│TEMPERATURE│─────       │
│THERMOCOUPLES│          ▼
└──────────┘        ┌──────────┐
                    │ MONITOR  │
                    │ DISPLAYS │
                    └──────────┘
```

MANUAL BACKUP

```
┌──────────┐   ┌──────────┐   ┌──────────┐   ┌──────┐   ┌──────────┐       ┌──────────────┐      ┌──────────┐
│EMERGENCY │   │EMERGENCY │   │  PILOT   │   │      │   │  PILOT   │       │  RCS DRIVE   │      │   RCS    │
│ DISPLAY  │──▶│ DISPLAY  │──▶│ DISPLAYS │──▶│PILOT │──▶│ CONTROLS │──────▶│ ELECTRONICS  │─────▶│ THRUSTERS│
│   IMU    │   │ COMPUTER │   └──────────┘   └──────┘   └──────────┘       └──────────────┘      └──────────┘
└──────────┘   └──────────┘                                                ┌──────────────┐      ┌──────────┐
                                                                           │  ACS DRIVE   │      │   ACS    │
                                                                           │ ELECTRONICS  │─────▶│ ACTUATORS│
                                                                           └──────────────┘      └──────────┘
```

Fig. 10-11 Reentry Phase Modes

NORMAL

```
                              ┌──────────────┐
                              │     MODE     │
                              │   CONTROLS   │
                              └──────┬───────┘
         ┌──────────────┐           │
         │     IMU      │           ▼
         └──────────────┘    ┌──────────────┐    ┌──────────────┐    ┌──────────────┐
         ┌──────────────┐    │     GNC      │    │  ACS DRIVE   │    │     ACS      │
         │  RATE GYROS  │───▶│   COMPUTER   │───▶│ ELECTRONICS  │───▶│  ACTUATORS   │
         └──────────────┘    └──────┬───────┘    └──────────────┘    └──────────────┘
┌──────────────┐                    │
│    RADAR     │  ┌──────────────┐  ▼
│ TRANSPONDER  │  │ RADAR DATA   │ ┌──────────────┐
└──────────────┘  │ LINK (UHF)   │ │    PILOT     │
                  └──────────────┘ │   DISPLAYS   │
                  ┌──────────────┐ └──────────────┘
                  │   AIR DATA   │
                  │    SENSOR    │
                  └──────────────┘
```

MANUAL BACKUP

```
┌──────────────┐
│     IMU      │
└──────────────┘
┌──────────────┐  ┌──────────────┐  ┌──────────────┐  ┌──────────┐  ┌──────────────┐  ┌──────────────┐
│     RATE     │  │     GNC      │  │    PILOT     │  │          │  │    PILOT     │  │  ACS DRIVE   │
│    GYROS     │─▶│   COMPUTER   │─▶│   DISPLAYS   │─▶│  PILOT   │─▶│   CONTROLS   │─▶│ ELECTRONICS  │
└──────────────┘  └──────────────┘  └──────────────┘  └────▲─────┘  └──────────────┘  └──────┬───────┘
┌──────────────┐                                          │                                  │
│   AIR DATA   │                                    ┌─────────┐                        ┌──────▼───────┐
│    SENSOR    │                                    │ WINDOW  │                        │     ACS      │
└──────────────┘                                    └─────────┘                        │  ACTUATORS   │
                                                                                       └──────────────┘
```
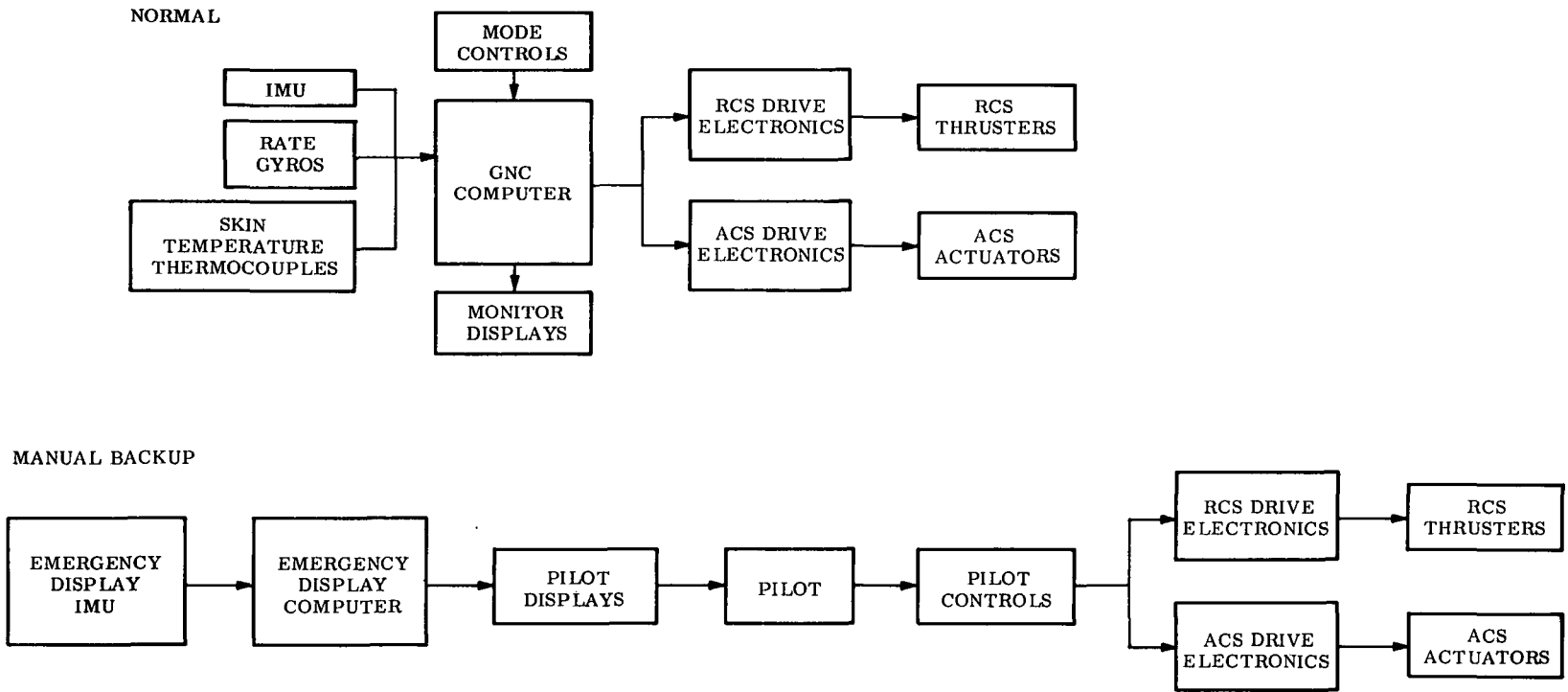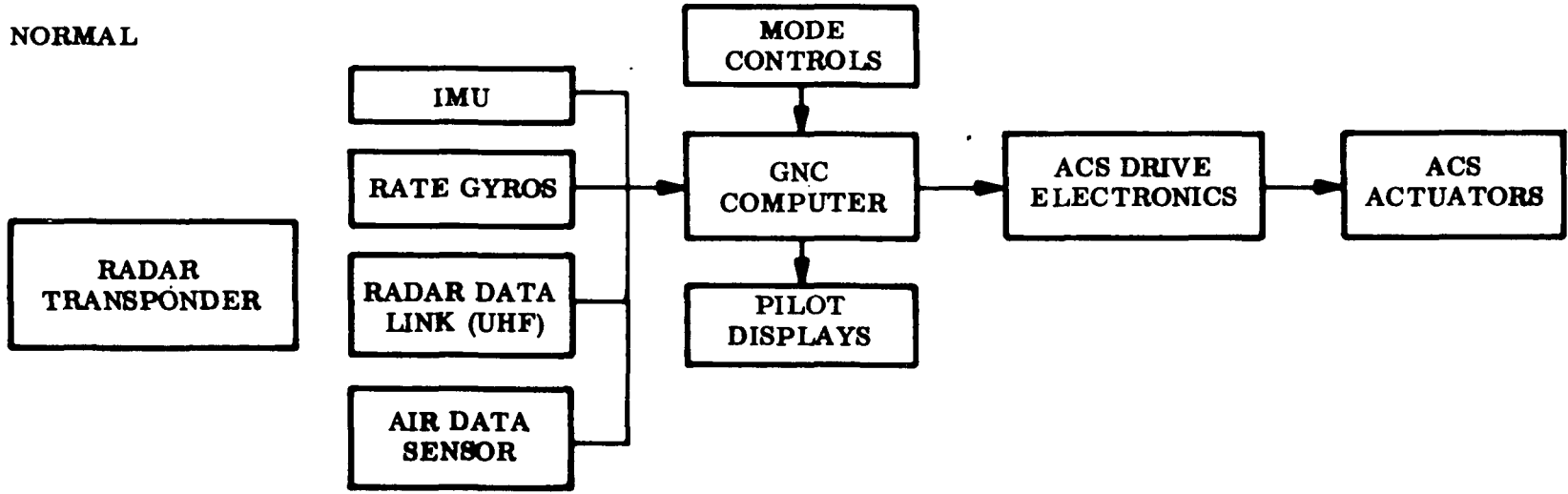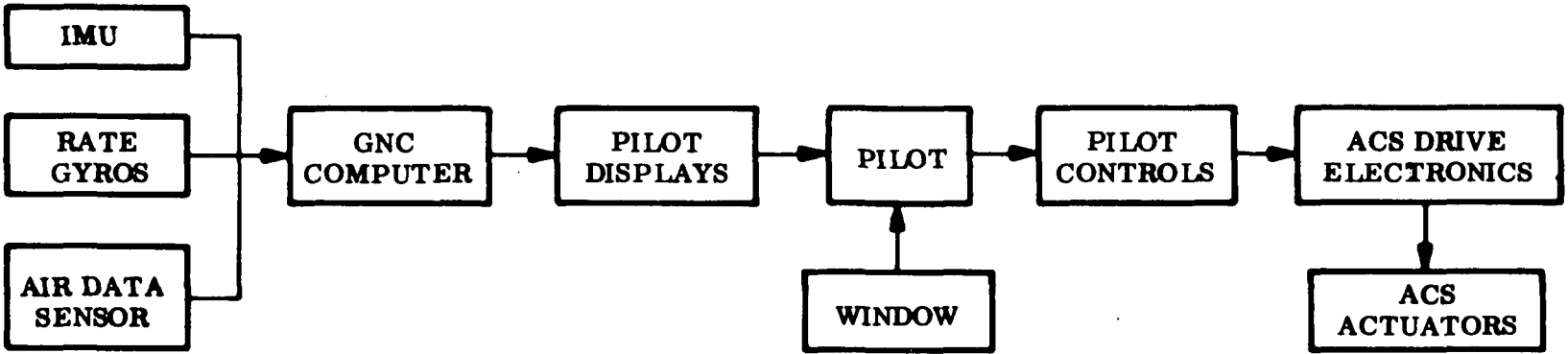
Fig. 10-12 Approach and Landing Phase Modes

10.2.3 Recommended Subsystem Configurations

Significant recommendations for each subsystem are presented in the following paragraphs, along with evaluations of alternate approaches.

10.2.3.1 <u>State Vector Subsystem</u>. The preliminary estimated 3σ accuracy required of the state vector calculation on orbit is as follows:

±0.5-nm position

±10-fps velocity magnitude

±0.05-deg velocity direction

±0.05-deg attitude

These values, which apply after boost into orbit and after an update on orbit, will degrade with time until updated again. They are chosen to be compatible with reentry initial condition accuracy requirements.

<u>Inertial Measurement Unit</u>. The IMU can be considered to be the source of all state data during all mission phases. Optical, RF, and ambient sensors are used periodically to update the state as a result of errors in the IMU data.

Both strapdown and gimbal systems have been compared as shown in Table 10-8. The strapdown system should be smaller, simpler, and more reliable than a gimbaled system. Maintenance is easier, adding to its cost advantage. Since it is sensitive to the angular as well as translational vibration environment, strapdown accuracy is less than with a comparable gimbal system. Introduction of the laser gyro should reverse this situation. A strapdown system has been selected with the provision that present accuracy is compatible with ascent and descent requirements.

Table 10-8

COMPARISON OF GIMBALED AND STRAPDOWN IMUs

Gimbaled	
Advantages	Disadvantages
Greater accuracy since IMU can be calibrated in vehicle just before flight and instruments are isolated from angular vibration environment	Slip rings and four gimbals required for all attitude capability Complicated maintenance, assembly, and internal alignment
Strapdown	
Advantages	Disadvantages
All-attitude reference capability Smaller, lighter, minimum mechanical complexity, and less power Provides attitude rate data directly Fewer thermal considerations More rugged for ascent and reentry environments Adaptibility to techniques of redundancy Ease of component replacement and aligning other sensors	High torque requirement for gyro Transformation matrix requires high-speed computation Sensitive to angular vibration environment More likely to require external optical alignment to achieve acceptable accuracy level Requires more knowledge of vehicle vibration environment for compensation equations Greater computation burden for IMU error compensation

Table 10-9 lists all known error sources in strapdown systems induced by motions of the system. Those gyro errors of greatest significance and their relative sizes are shown in Table 10-10. These errors are reduced to acceptable levels by detail design of the instruments and by programming compensating equations into the guidance computer data processing.

Table 10-9

ERROR SOURCES IN STRAPDOWN INERTIAL REFERENCE SYSTEMS

	Affected Components	
Source	Gyros	Accelerometers
Anisoinertia rectification	X	X
Angular acceleration	X	X
Rebalance loop		
Torque asymmetry	X	X
Quantization	X	X
Limit cycling	X	X
Bandwidth	X	X
Misalignment	X	X
Anisoelastic rectification	X	
Cross coupling	X	
Spin/input		
Spin/output		
Mass unbalance	X	
Input axis		
Output axis		
Rotor		
Inertia cross products	X	
Gimbal friction	X	
Coning	X	
Two-axis angular motion		
Nonidentical gyro loops		
Undetected coning		
Quantization		
Float fluid convection	X	
Random drift	X	X
Vibropendulous effect		X
Nonlinear g^2 effect		X
Sculling		X
Size effect		X

Table 10-10

GYRO ERROR RATES

Source	Normalized Effective Drift Rate
Cross coupling	
Spin/input	1.0000
Spin/output	0.0125
Anisoinertia	
Low frequency	0.0100
High frequency	0.1900
Mass imbalance	0.0063
Anisoelasticity	0.0013
Torque asymmetry	0.0325
Misalignment	0.0625
Inertia cross products	0.0100
Coning*	0.6250

*Pseudo-coning due to cumulative error in the computer cosine matrix algorithm

Prelaunch azimuth alignment by earth rate gyrocompassing has been chosen over an optical technique on the assumption of acceptable accuracy. The largest error contributor is the drift stability of the east-west gyro. For a launch into a 55-degree orbit, the sensitivity is 0.33 degree inclination for a 28-arc-min azimuth uncertainty, which corresponds to a gyro drift uncertainty of 0.1 deg/hr.

A conventional arrangement of three gyros and three accelerometers and duplication of assemblies for active redundancy is recommended, pending further study. A concept permitting less than 18 sensors for triple redundancy calls for six gyros and six accelerometers, arranged with their sensitive axes perpendicular to the faces of a regular

dodecahedron. This arrangement potentially offers much higher reliability, since only five of the six instruments need be operating for full active triple redundancy.

Orbit State Sensors. The Space Shuttle must be capable of determining its orbit ephemeris and body attitude with onboard instrumentation.

During intervals on orbit, the state vector subsystem gathers data with respect to the earth's surface (horizon angles and altitude) and the star field (times of appearance in a reticle) (Fig. 10-13). This information is processed in a recursive filter to yield best estimates of all parameters that can be modeled. Shown below is a typical list of 28 components of the state vector, which are simultaneously computed:

Item	Number of Components
Vehicle position	3
Vehicle velocity	3
Vehicle attitude	3
Attitude rates	3
Instrument biases	5
Instrument scale factors	5
Gyro drifts	3
Gyro torquer errors	3

By detecting the infrared radiation at the earth/space transition, horizon sensors can locate the local vertical. Scan techniques include edge tracking, conical scan, and dither. A conical or dither scan can be ground tested and should be capable of an accuracy better than ±0.05 deg, not including alignment errors. A candidate system, the LMSC-developed dither scan is shown in Fig. 10-14. Also evaluated were the Barnes Type 13-156 circular scan, Quantic Mod 4 edge tracker and a TRW edge tracker.

HORIZON SENSOR + IMU ESTABLISHES
VEHICLE-TO-ORBIT COORDINATE
ANGLES (ϕ, θ, ψ)

RADAR ALTIMETER MEASURES
ALTITUDE (a, e, tp, u)

DIHEDRAL ANGLE

S/S LOS

NOMINAL ORBIT

STAR SENSOR + STORED DISCRETE GIVES POSITION ALONG ORBIT AND DIHEDRAL
ANGLE WITH RESPECT TO DESIRED ORBIT (i, ωp, Ω_N, λ, τ)
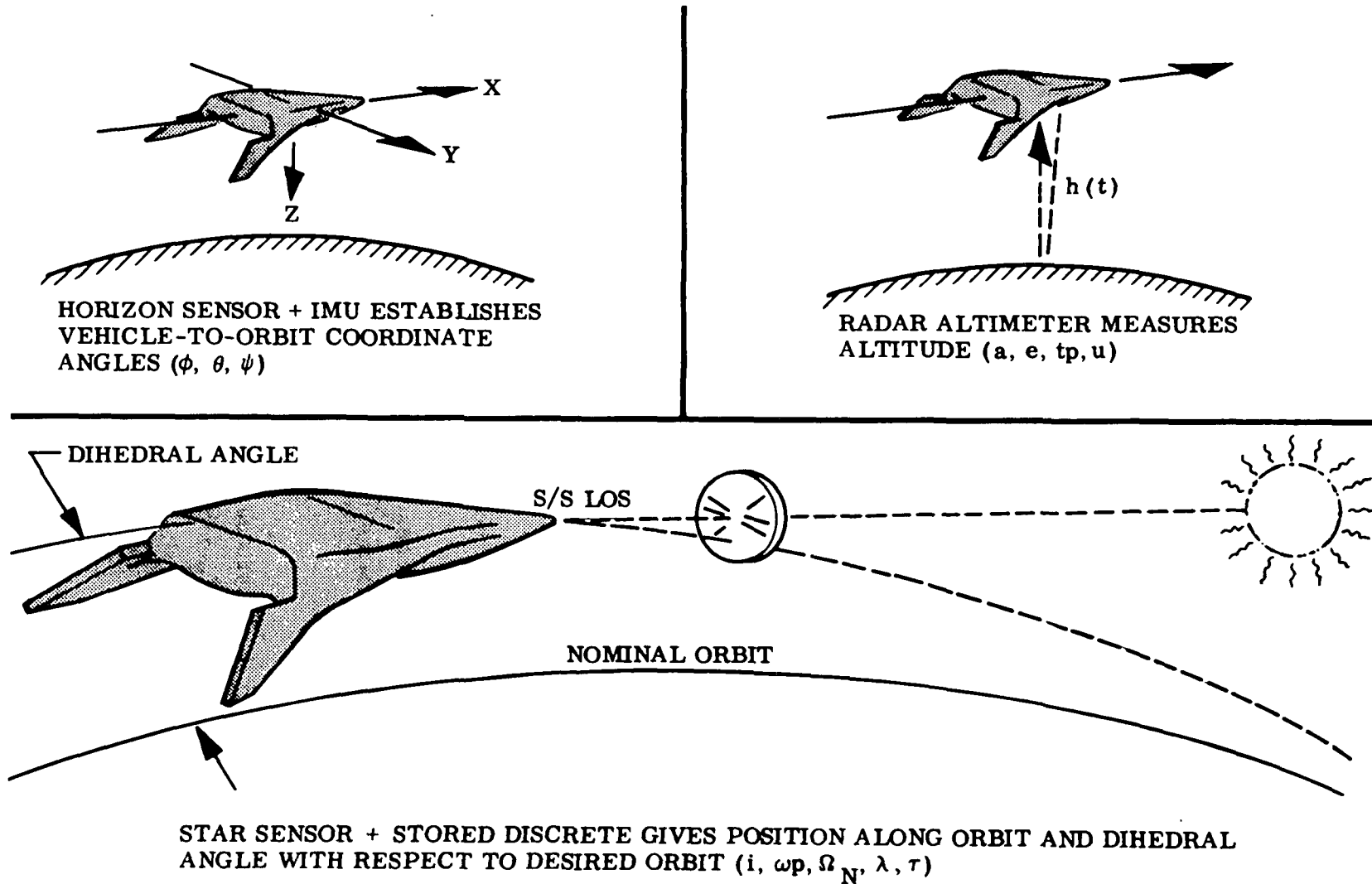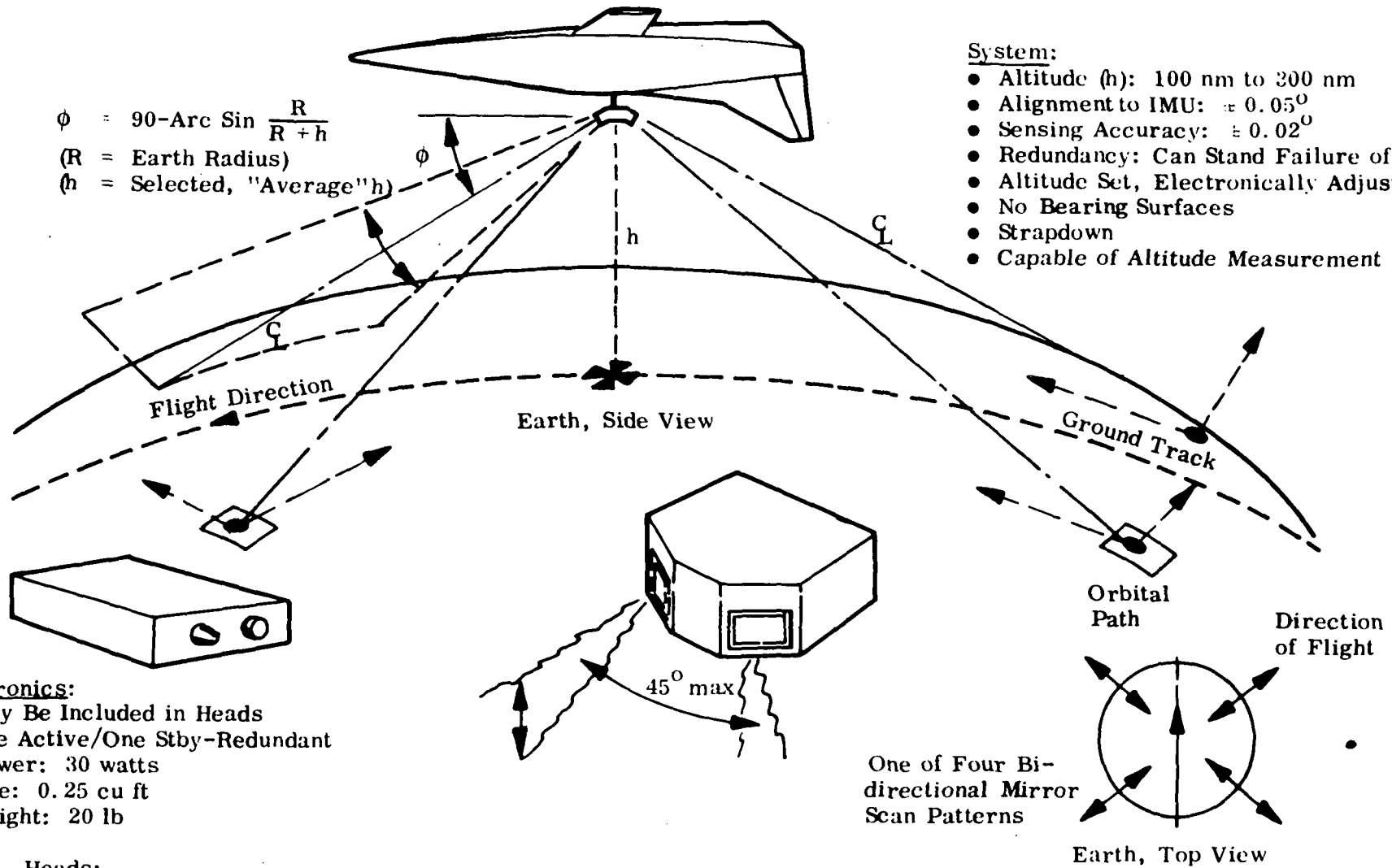
Fig. 10-13 Orbital Navigation Approach

$$\phi = 90 - \text{Arc Sin} \frac{R}{R+h}$$

(R = Earth Radius)
(h = Selected, "Average" h)

ϕ

h

Flight Direction

C_L

C_L

Earth, Side View

Ground Track

System:
- Altitude (h): 100 nm to 300 nm
- Alignment to IMU: $\pm 0.05^\circ$
- Sensing Accuracy: $\pm 0.02^\circ$
- Redundancy: Can Stand Failure of 1 Scan
- Altitude Set, Electronically Adjusted
- No Bearing Surfaces
- Strapdown
- Capable of Altitude Measurement

Orbital
Path

Direction
of Flight

Electronics:
- May Be Included in Heads
- One Active/One Stby-Redundant
- Power: 30 watts
- Size: 0.25 cu ft
- Weight: 20 lb

45° max

One of Four Bi-
directional Mirror
Scan Patterns

Earth, Top View

Heads:
- Scan Angle: 50°/Freq: 10 H_z; Two Active, Two Stby/Rend.
- Two Bolometers/Mirror-Scanner
- Deployed Through Payload Doors
- Power: 12 W, Size: 0.1 cu ft, Wt: 8 lb

Fig. 10-14 Horizon Sensor System

The strapdown star sensor selected has a solid state photo-detector to generate discrete pulses when preselected stars enter the field of view. A star field reference map storage is compared to the actual discretes for orbit angle update (Fig. 10-15). A system of this type is currently under development by a Lockheed/ Honeywell team. A star tracker senses a greater number of stars but requires gimbals (an added error source) and several units for coverage of the celestial sphere.
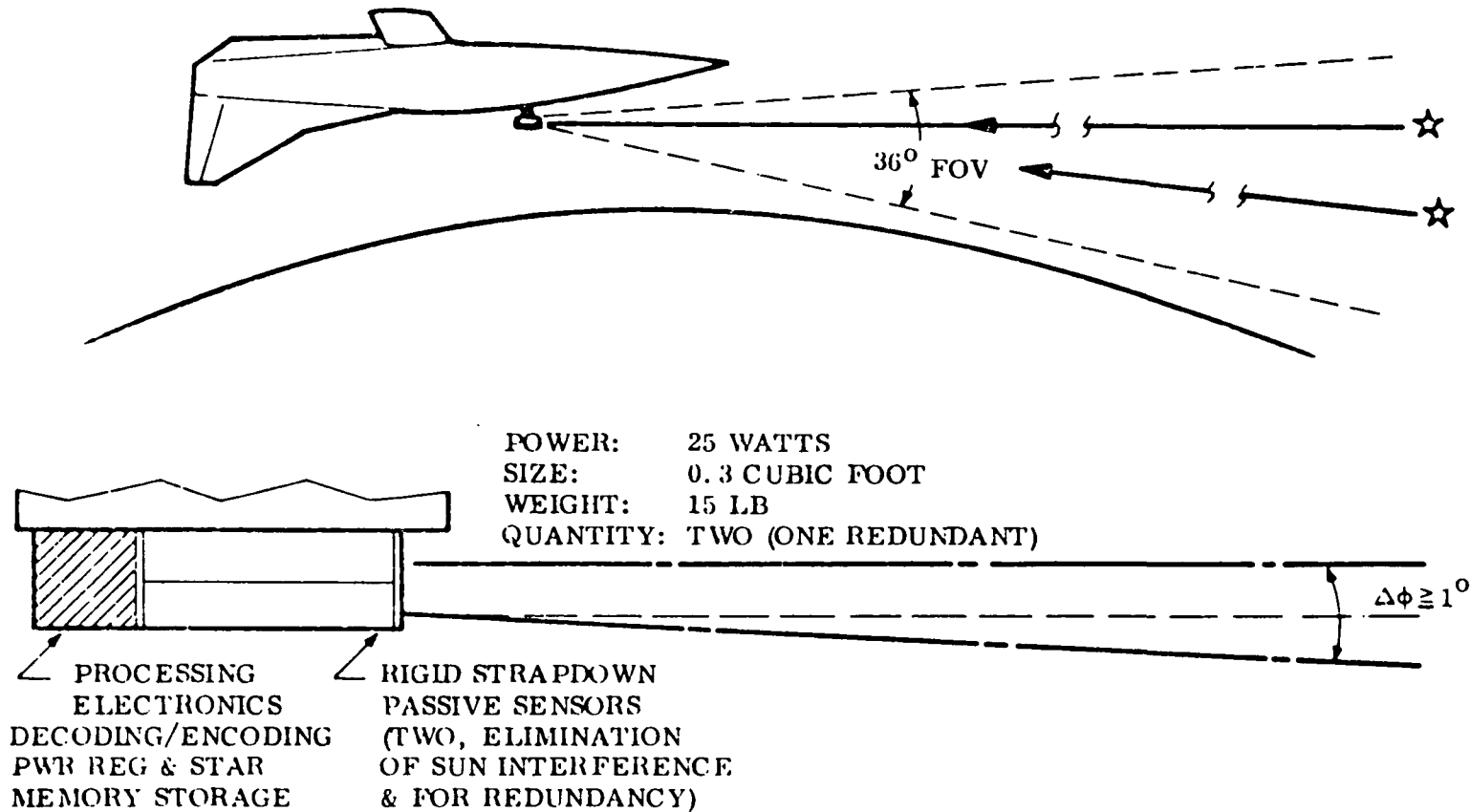
In addition to measuring altitude over the range of 50 to 300 nm, the radar altimeter can provide redundant measure of vehicle horizontal and vertical velocity and pitch, yaw, and roll attitude. Portions of the radar altimeter system can also be employed for glide slope extension during low-visibility approaches.

The recommended radar will be an outgrowth of RASS, the Bell Aerosystems/NASA phase monopulse system. It would use two 15-inch antennas, one on the vehicle top-side for orbit and one on the bottom for landing. The Bell system generates four skewed orthogonal difference beams and a sum beam to generate data for altitude, ground velocity, vertical velocity, pitch, yaw, and roll attitude.

Current on-orbit accuracies are ±150-foot altitude and ±0.1 degree local vertical. With longer integration times (in excess of 10 seconds) accuracy would be comparable to that of present horizon sensor systems.

10.2.3.2 Targeting and Guidance Subsystem. Targeting is performed during pre-launch for ascent, parking orbit for transfer burns, and station orbit for reentry.

36° FOV

POWER: 25 WATTS
SIZE: 0.3 CUBIC FOOT
WEIGHT: 15 LB
QUANTITY: TWO (ONE REDUNDANT)

$\Delta\phi \geqq 1°$

PROCESSING   RIGID STRAPDOWN
ELECTRONICS   PASSIVE SENSORS
DECODING/ENCODING   (TWO, ELIMINATION
PWR REG & STAR   OF SUN INTERFERENCE
MEMORY STORAGE   & FOR REDUNDANCY)

- MEASURES INERTIAL POSITION, GIVEN TIME, ALT AND LOCAL VERT
- SINGLE-FIX ACCURACY: $< \pm 0.005°$
- ALIGNMENT TO IMU: $\pm 0.1°$, TO HORIZ SENSOR: $\pm 0.05°$
- DEPLOYED THROUGH PAYLOAD DOORS: LOOKS FWD
- STARFIELD SCAN IS AT ORBITAL RATE & RESIDUAL ATT. RATES

- 30 (+) STAR LIBRARY STORED IN ERASABLE MEMORY
- INERTIAL POSITION FIXES RELY ON HORIZON SENSOR ACCURACY
- FIXES UNAFFECTED BY EARTH WEATHER
- NO BEARING SURFACES

Fig. 10-15 Star Sensor System

The trajectory control program settings, guidance constants, mission sequence of event discretes, and some flight control gains are computed with targeting equations.

Guidance computations will be performed during ascent, orbit transfer burns, and reentry.

For ascent, the subsystem must include the capability of realtime adjustment for out-of-plane launches due to intentional holds for rapid target phasing or for delays and the ability to adapt to large parameter changes, such as failure of one or more engines or emergency early separation. Saturn guidance should be adaptable to the Space Shuttle requirements.

Vernier throttling can be used to control the thrust acceleration to a nominal program to minimize dispersions due to propulsion and weight tolerances.

It is recommended that vehicle guidance start at liftoff for airload and trajectory orbit control rather than using an open-loop technique for this purpose. After exit from the atmosphere, the guidance equations would be switched to one of several possible forms of explicit algorithms with constrained terminal conditions.

During the boost phase, guidance signals can originate in either the booster or the orbiter. The tentative recommendations concerning this are as follows:

- Guidance should be from the booster stage during the boost phase. The orbiter stage guidance should compute from liftoff through staging and transmit active steering and throttling commands after separation and orbiter ignition.

- Display for the orbiter crew should include the vehicle/trajectory state, as computed in the booster, for direct comparison with the state computed in the orbiter and with the desired state as computed in the orbiter.

The opposite recommendation for the primary role is reached for the Triamese, since a master system must exist for the three stages. Thus, for Triamese, boost guidance commands should probably issue from the orbiter vehicle and be fed to each of the booster vehicle flight control systems.

10-31

This requires synchronization of navigation and vehicle state data as well as guidance commands to preclude spurious transients at separation.

The tradeoffs between guiding from the Two-Stage booster or from the orbiter during boost through the atmosphere are as follows:

- Orbiter primary

  Advantages — May have greater redundancy

  Drawbacks   — Requires signal interconnection
              — Must be synchronized with boost guidance
              — Requires switchover at staging

- Booster primary (recommended)

  Advantages — Continuous guidance for return
              — Booster TVC used for stabilization during boost
              — Smooth transition through staging
              — Eases booster pilot backup

  Drawbacks   — Adds orbit injection mode to booster computer

Transfer orbit targeting to Space Station position will approximate the two or three optimum cases in consideration of elapsed time to arrival, expected dispersions, and desired end conditions. Burn guidance will be based on velocity-to-be-gained with time-to-transfer constraints. Reentry targeting will place the footprint over the desired landing site by selection of a phasing orbit and the deorbit burn and establish the reentry trajectory by an angle of attack-bank angle program.

In addition to Phugoid damping, energy management, and terminal-point error control, reentry inertial guidance equations must constrain trajectory dispersions to limit skin temperatures during the peak heating regime. Direct skin temperature measurements would then serve as a backup.

10.2.3.3 Flight Control Subsystem. The flight control subsystem processes state data to generate control actuation signals in response to guidance and pilot commands and atmospheric disturbances to produce correct, stable attitude responses.

During ascent, air load and drift control is obtained with a simple attitude plus rate autopilot, plus a guidance signal that controls the error from a programmed acceleration vector induced by winds aloft. The commanded acceleration vector is derived from a weighted combination of path control and load relief.* The effect of the load relief weighting factor, λ , is shown in Table 10-11 and Fig. 10-16 for the wind profile shown in Fig. 10-17. Tight path control and no load relief are obtained when $\lambda = 0$, while no path control and maximum load relief result when $\lambda = 1$. Maximum loads ($q\alpha$) may be reduced from 3200 psf-deg ($\lambda = 0$) to 2300 psf-deg ($\lambda = 0.9$) at the expense of 800 pounds to orbit. This tradeoff is considered very worthwhile because 800 pounds, or 0.11 percent of orbiter propellant, does not significantly contribute to typical performance margins of 0.75 percent when root sum squared with the other contributors. Even if considered alone, 800 pounds of additional orbiter propellant increases launch weight by only about 24,000 pounds, but 900 psf-deg reduced $q\alpha$ reduces launch weight about 175,000 pounds.

The corresponding reduction in engine gimbal arc required for pitch control is 4.5 degree.

Wind Azimuth. Results show that $q\alpha$ max is essentially independent of wind azimuth when an appropriate roll control method is used. Figures 10-18, 10-19, 10-20, and 10-21 show time histories of $q\alpha$, δ_{pitch}, δ_{roll}, and $\dot{\delta}_{pitch}$ for four cases: no wind (N), headwind (H), sidewind (S), and tailwind (T).

The engine pitch gimbal deflections (δ_{pitch}) are in large part due to composite vehicle. cg tracking. Figure 10-22 shows this contribution isolated, permitting evaluation of the amount necessary for trajectory control. By trajectory reshaping of the no wind case, it is possible to reduce the positive engine deflection after liftoff to 1 degree and reduce the total required pitch gimbal arc about 2 degrees. Head- and tailwind loads can be exactly balanced if desired. Required propellant margins tend to become independent of wind azimuth for large λs.

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*Space Shuttle Data (Volume VI: Performance and Flight Mechanics), LMSC-A955317A, Sept 12, 1969, pp 7-1 to 7-54

Table 10-11

LOAD RELIEF RESULTS

λ	Wind	$q\alpha_{max}$ (deg-psf)	A_{z} (fps²)	δ_p (deg)	$\dot{\delta}_p$ (deg/sec)	δ_y (deg)	$\dot{\delta}_y$ deg/sec	ΔW (lb)
0	No	1145	0.8 / -22.4	2.9 / -13.5	1.2 / -0.4	0.2 / -0.2	0.1 / -0.2	–
	Head	3316	0.8 / -22.3	7.4 / -13.5	6.5 / -5.8	0.7 / -0.2	0.4 / -0.2	-1035
	Tail	2723	12.2 / -22.5	2.4 / -13.5	6.0 / -8.6	0.3 / -0.5	0.1 / -0.4	+775
	Side	3185	0.8 / -22.4	7.5 / -13.5	7.7 / -6.5	0.5 / -0.4	0.7 / -0.3	-164
0.5	No	1272	0.8 / -22.2	2.9 / -13.5	1.1 / -0.3	0.3 / -0.2	0.1 / -0.2	–
	Head	2947	0.8 / -21.8	6.1 / -13.5	5.9 / -5.6	0.7 / -0.2	0.3 / -0.2	-1118
	Tail	2463	11.2 / -22.5	2.4 / -13.5	6.2 / -8.2	0.3 / -0.4	0.1 / -0.4	-227
	Side	2648	0.8 / -22.3	5.3 / -13.5	8.0 / -7.0	0.5 / -0.5	0.8 / -0.5	-924
0.9	No	1301	0.8 / -22.2	2.9 / -13.5	1.1 / -0.4	0.3 / -0.2	0.1 / -0.2	–
	Head	2062	1.0 / -21.8	3.5 / -13.5	6.0 / -5.9	0.5 / -0.3	0.3 / -0.2	-1406
	Tail	2208	10.2 / -22.6	2.4 / -13.6	6.7 / -8.2	0.3 / -0.4	0.2 / -0.3	-1802
	Side	2278	2.1 / -22.1	3.0 / -13.5	7.3 / -6.8	0.6 / -1.1	0.5 / -1.2	-1582
1.0	No	1335	0.8 / -22.2	2.9 / -13.5	1.1 / -0.3	0.3 / -0.2	0.1 / -0.2	–
	Head	2134	1.9 / -21.6	3.5 / -13.5	5.7 / -6.0	0.5 / -0.3	0.3 / -0.2	-1882
	Tail	2132	9.8 / -22.6	2.3 / -13.5	6.9 / -7.8	0.4 / -0.3	0.2 / -0.4	-2148
	Side	3150	5.0 / -22.1	3.0 / -13.5	6.5 / -6.0	0.8 / -1.1	0.7 / -1.0	-2201
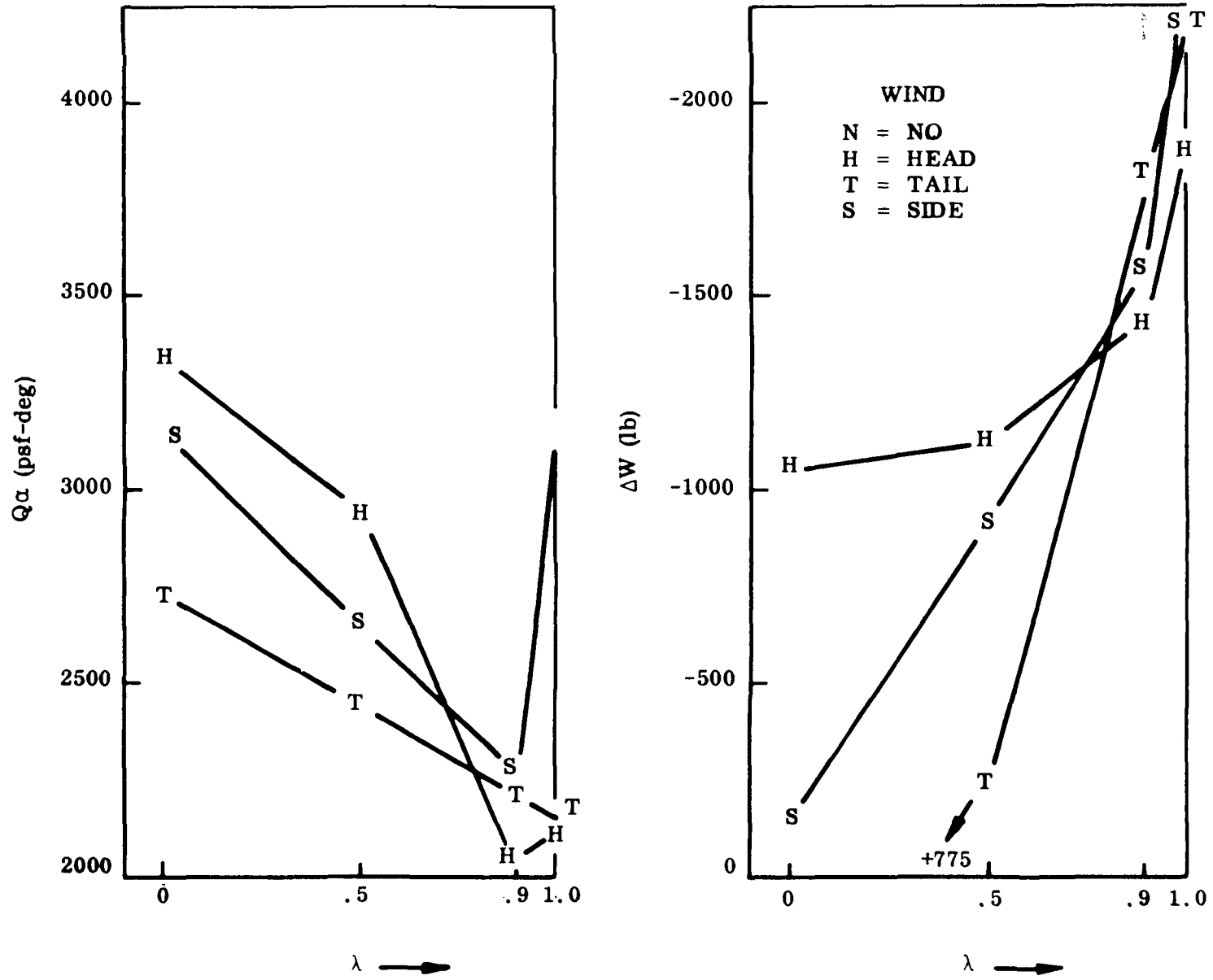
Fig. 10-16   Load-Relief Results

Fig. 10-17 Wind Profile

LAMBDA = .9

QALPHA (PSF-DEG)   VS   TIME (SEC)

Fig. 10-18

LOCKHEED MISSILES & SPACE COMPANY

LAMBDA = .9

PITCH DEFLECTION(DEG)   VS   TIME(SEC)

Fig. 10-19

LOCKHEED MISSILES & SPACE COMPANY
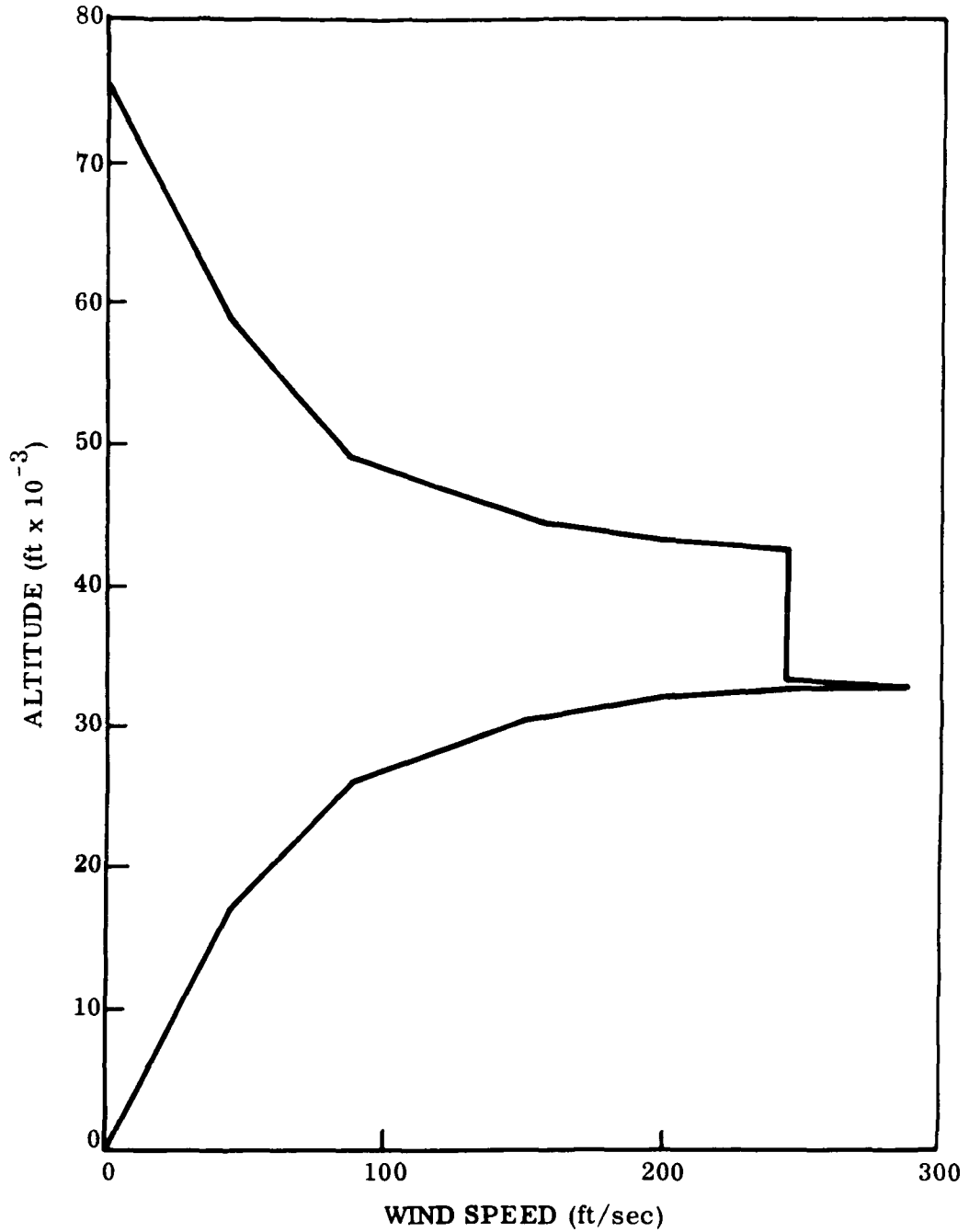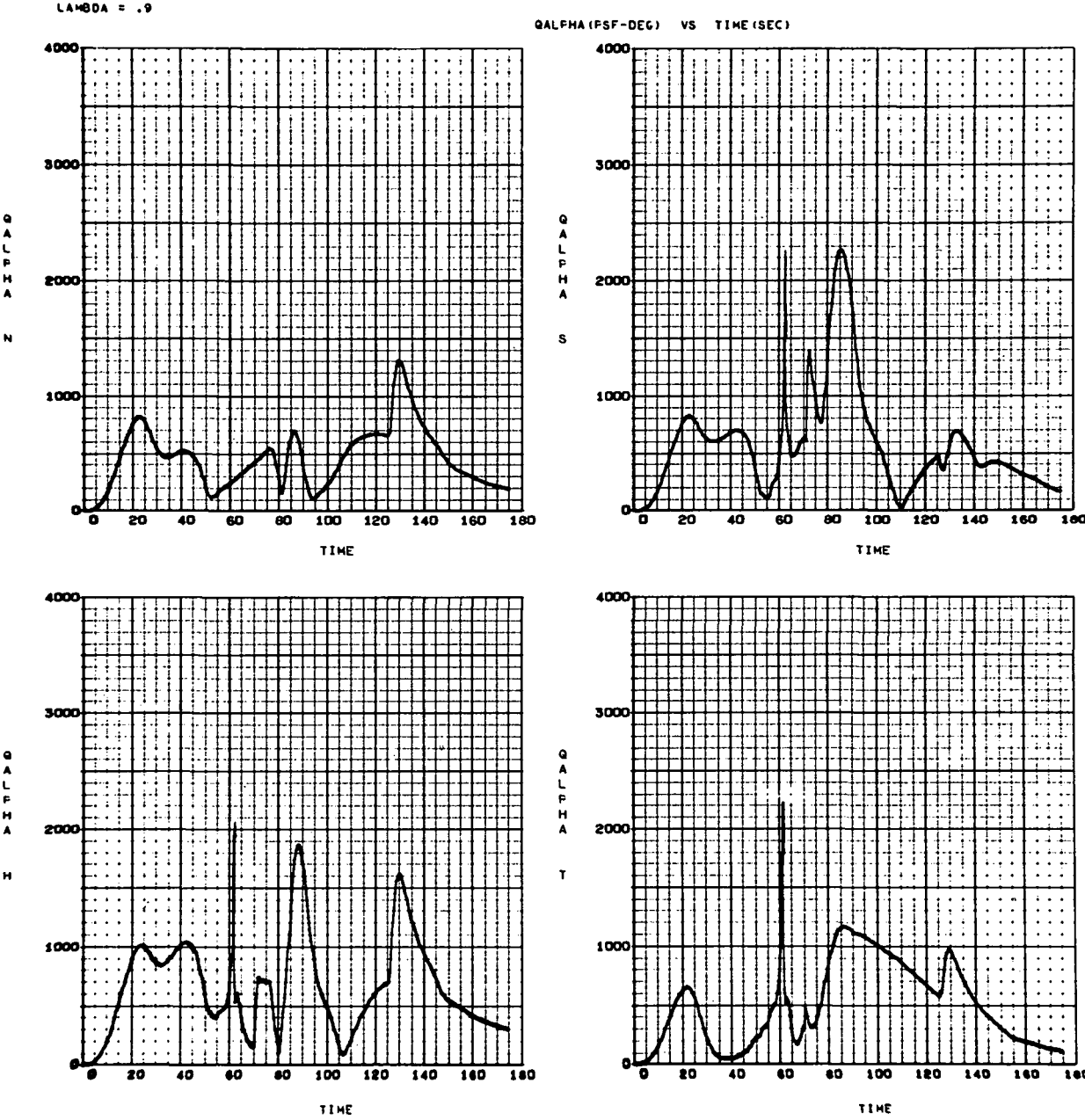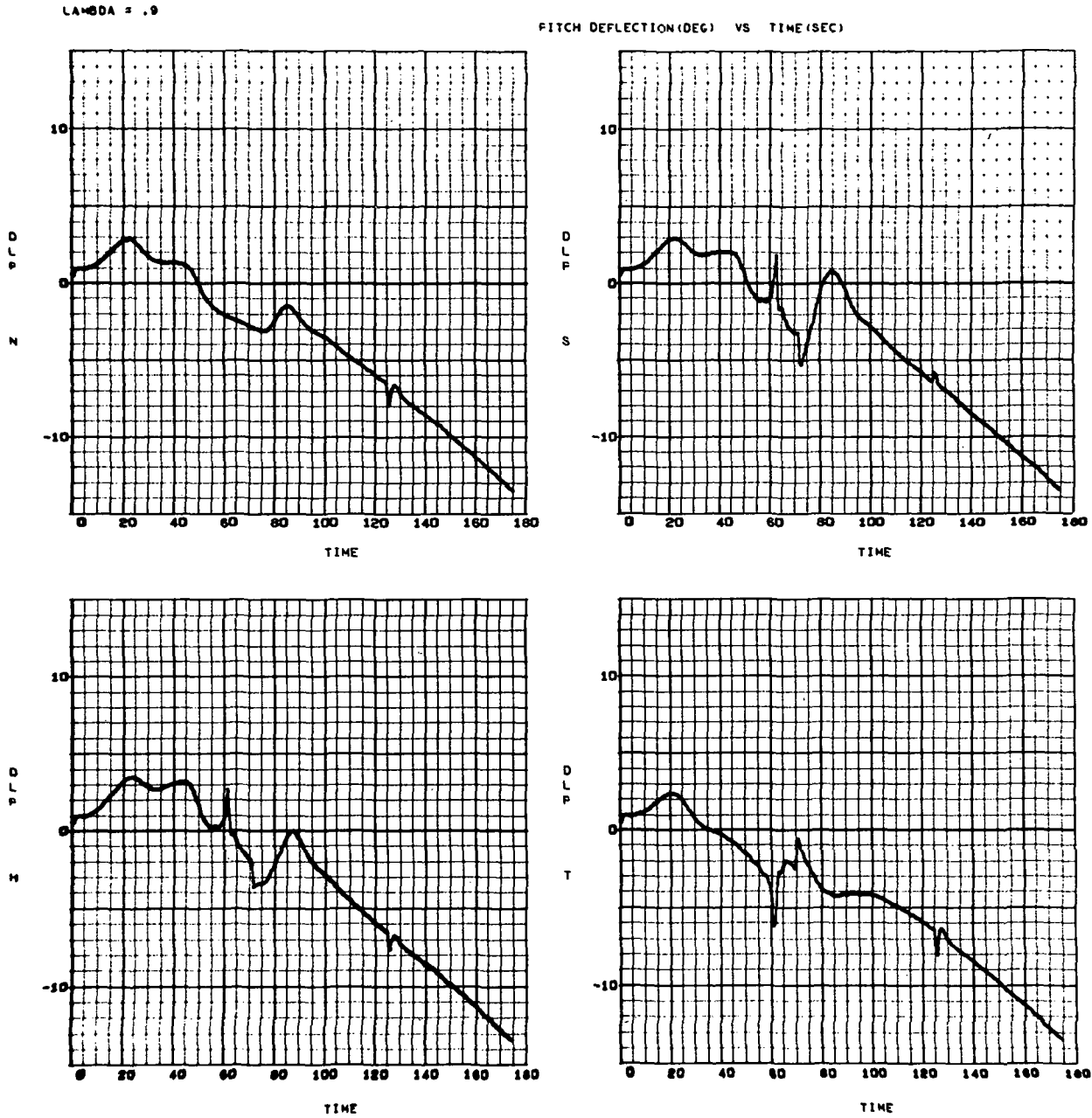
LAMBDA = .9

ROLL DEFLECTION(DEG) VS TIME(SEC)

Fig. 10-20

LOCKHEED MISSILES & SPACE COMPANY
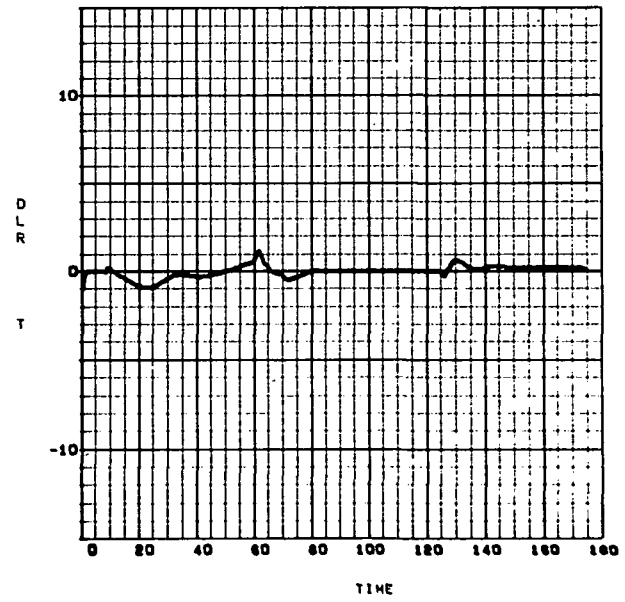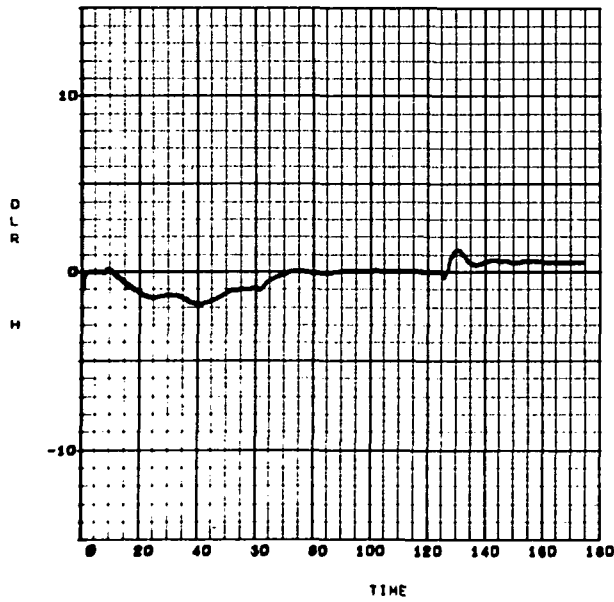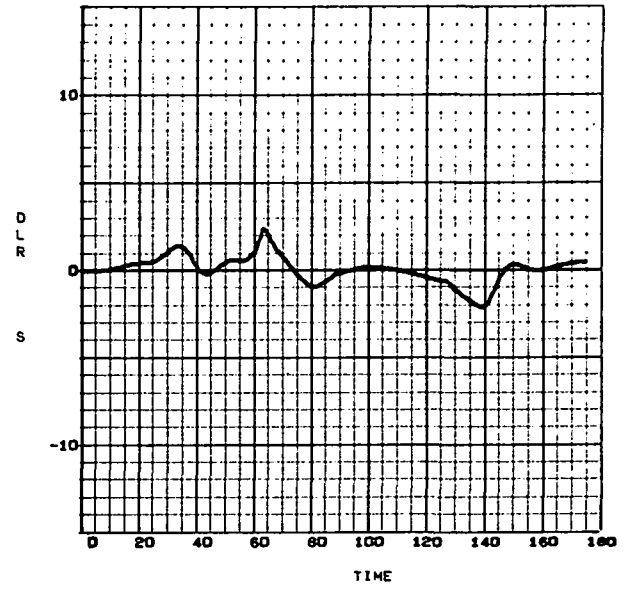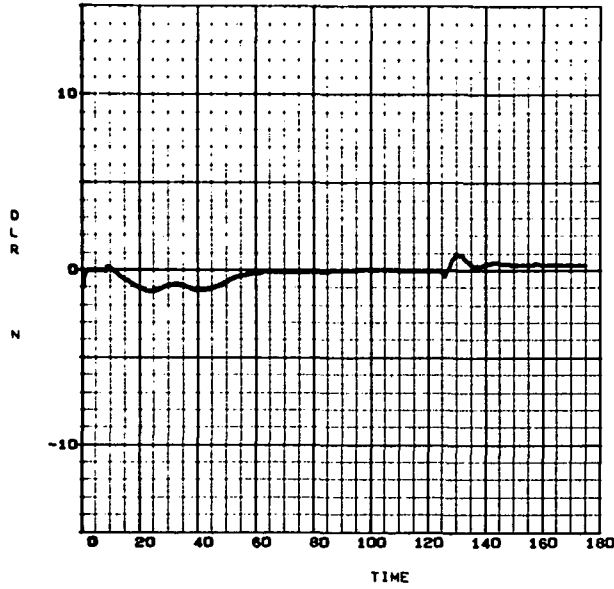
LAMBDA = .9

PITCH DEFLECTION RATE(DEG/SEC)   VS   TIME(SEC)

Fig. 10-21

LOCKHEED MISSILES & SPACE COMPANY
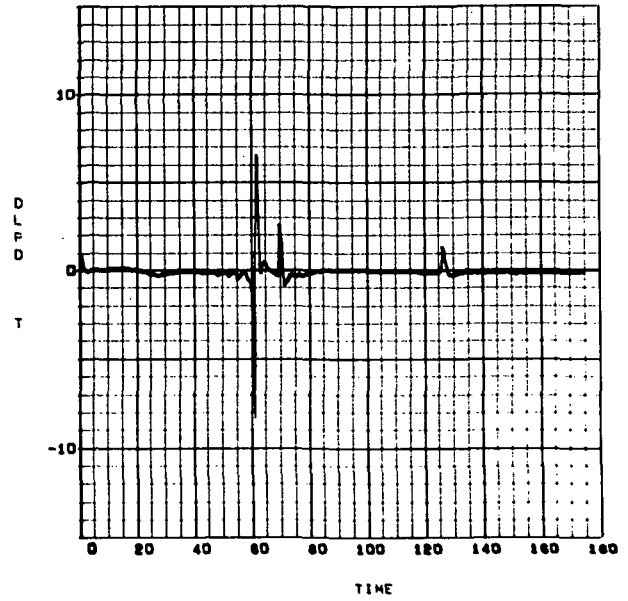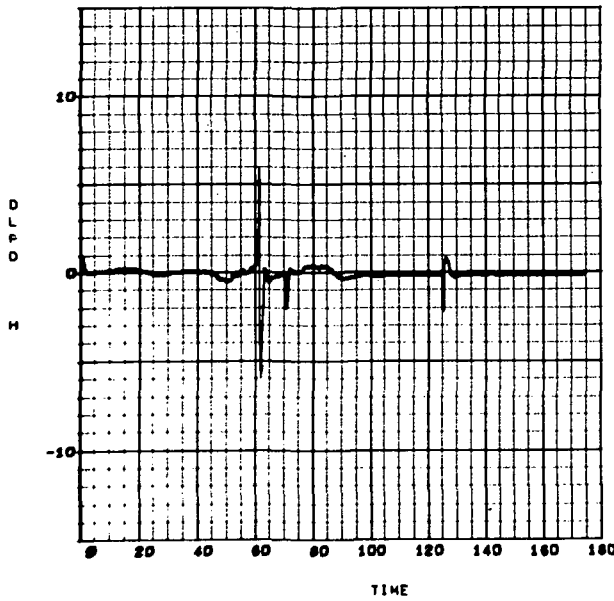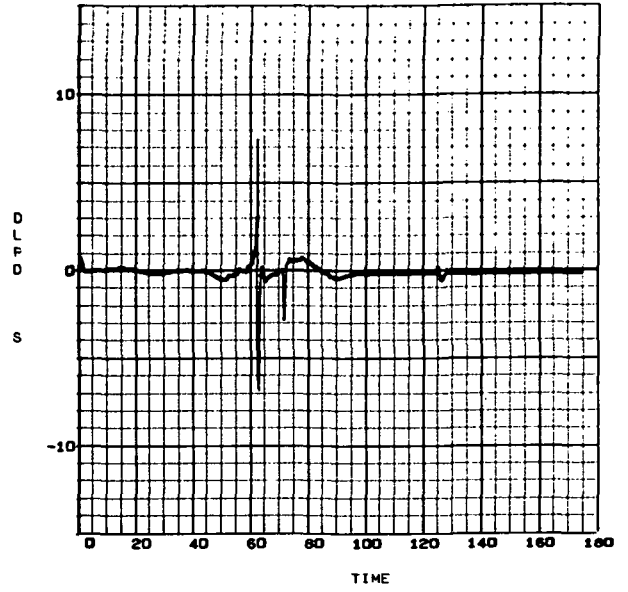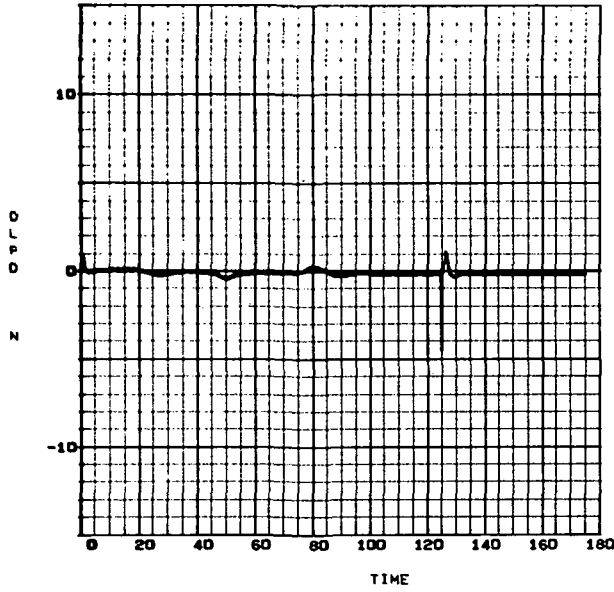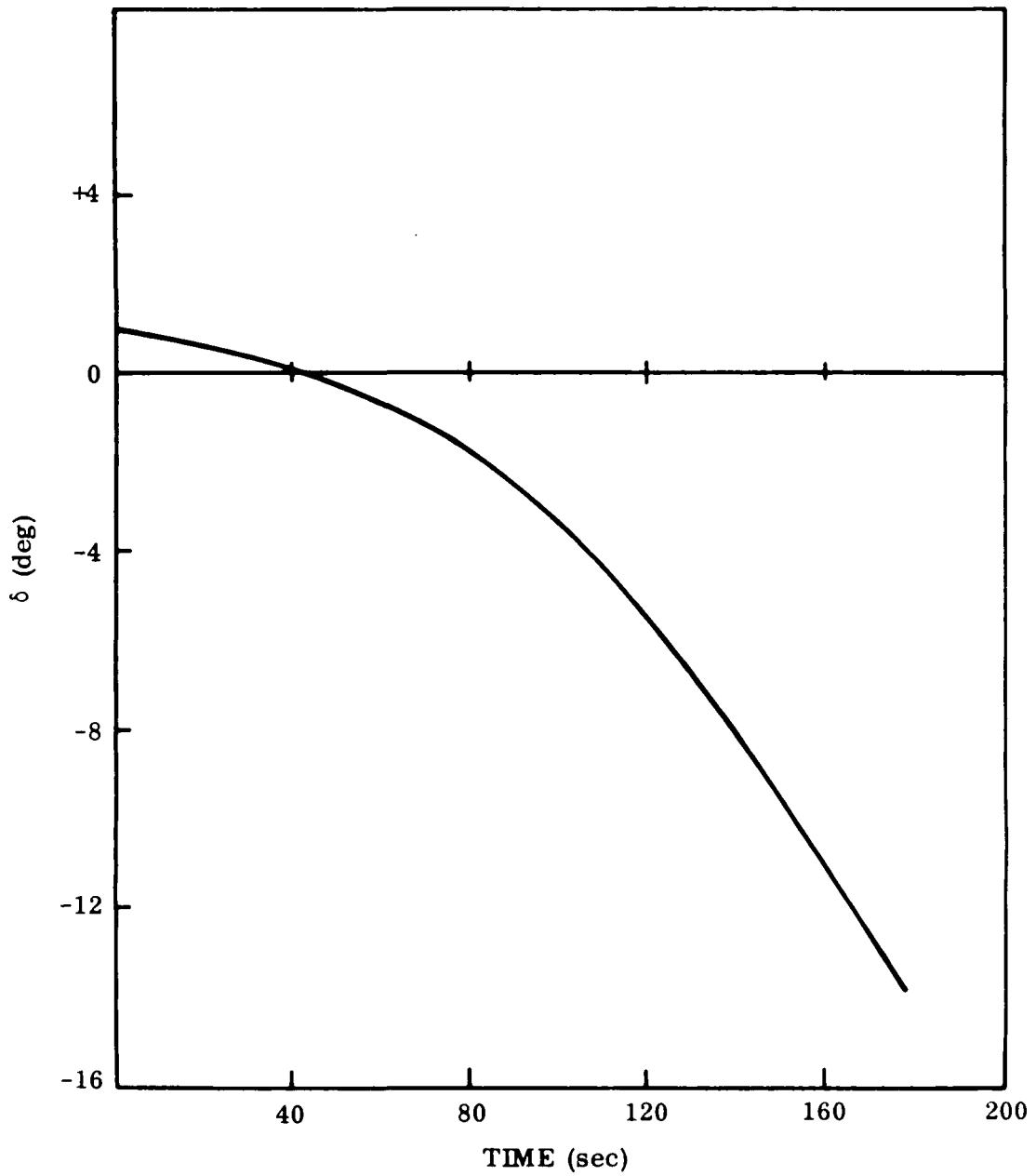
Fig. 10-22 Pitch Engine Deflection Required to Track Composite Vehicle cg

10-41

Fig. 10-23

LOCKHEED MISSILES & SPACE COMPANY
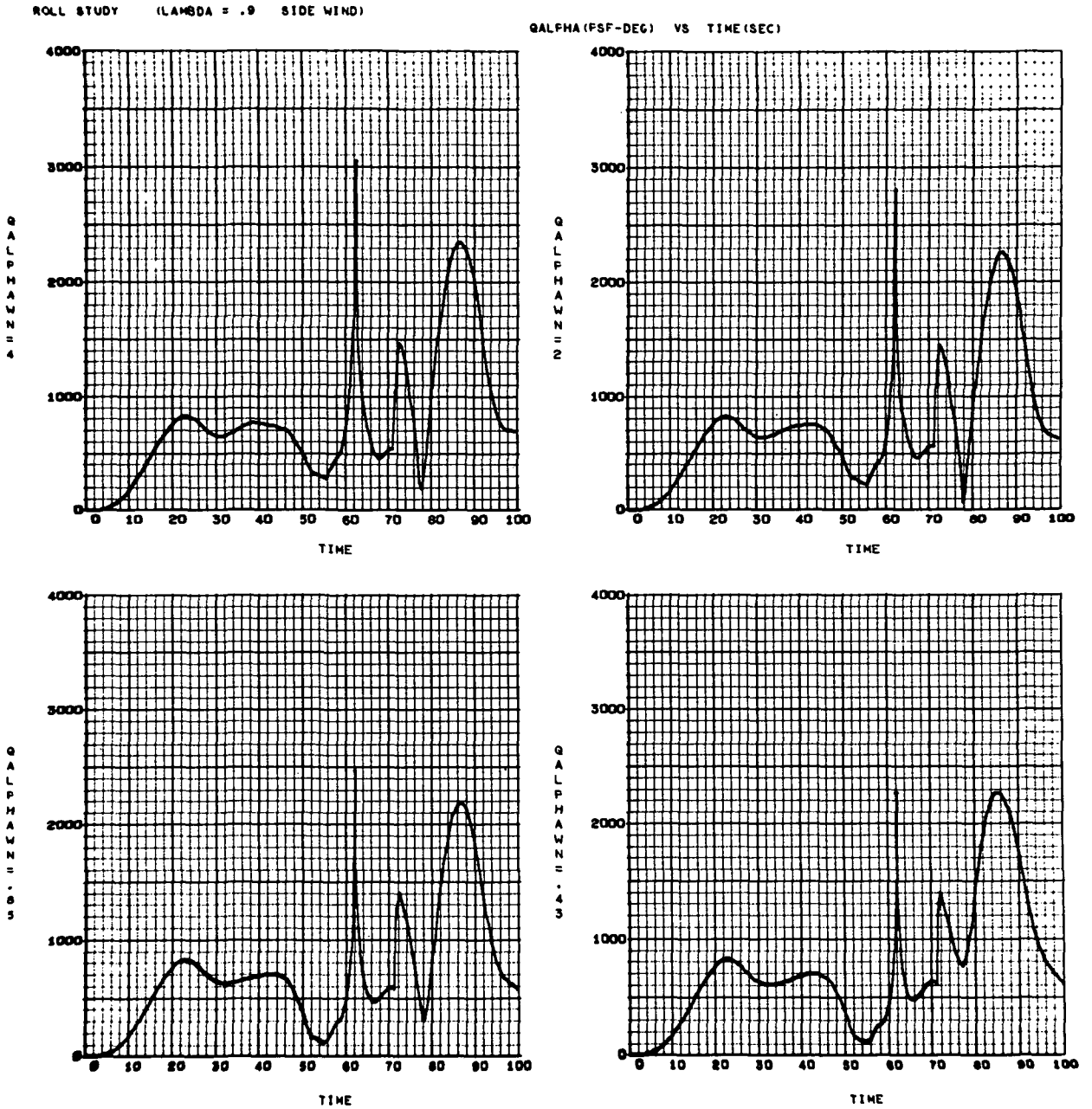
ROLL STUDY   (LAMBDA = .9   SIDE WIND)

PITCH DEFLECTION(DEG)   VS   TIME(SEC)

Fig. 10-24

LOCKHEED MISSILES & SPACE COMPANY

Fig. 10-25

LOCKHEED MISSILES & SPACE COMPANY
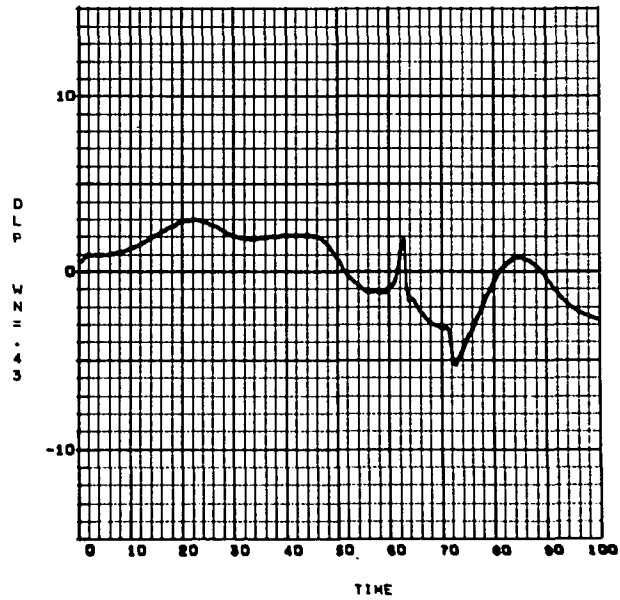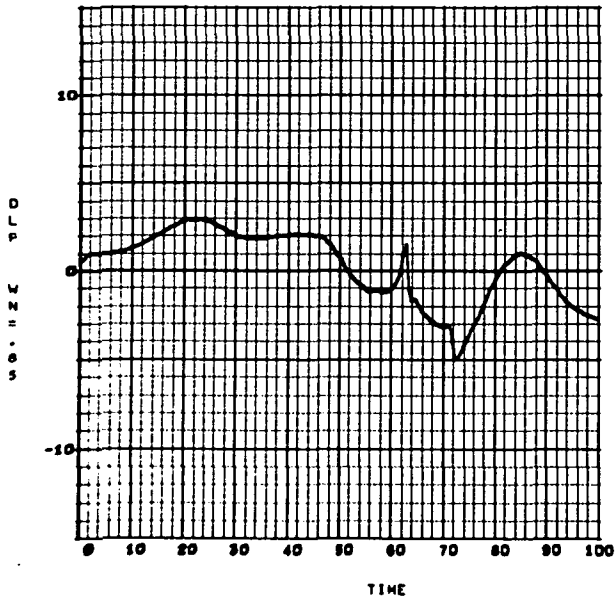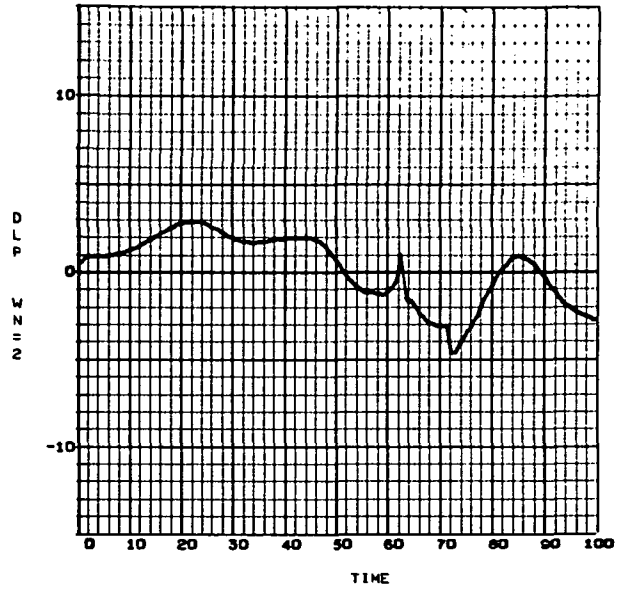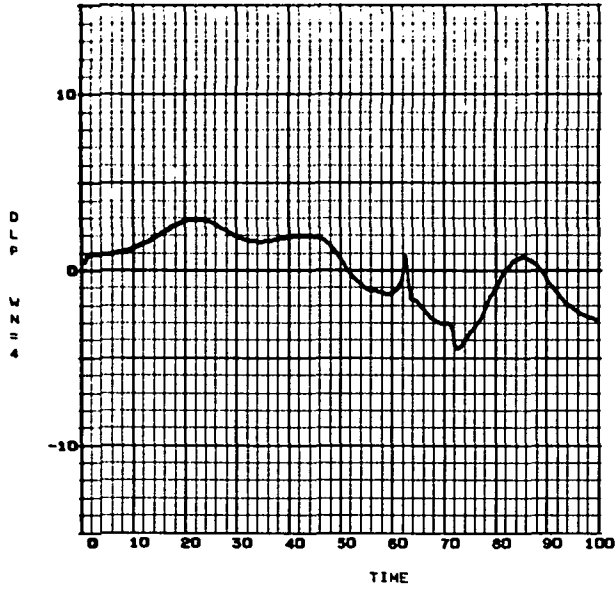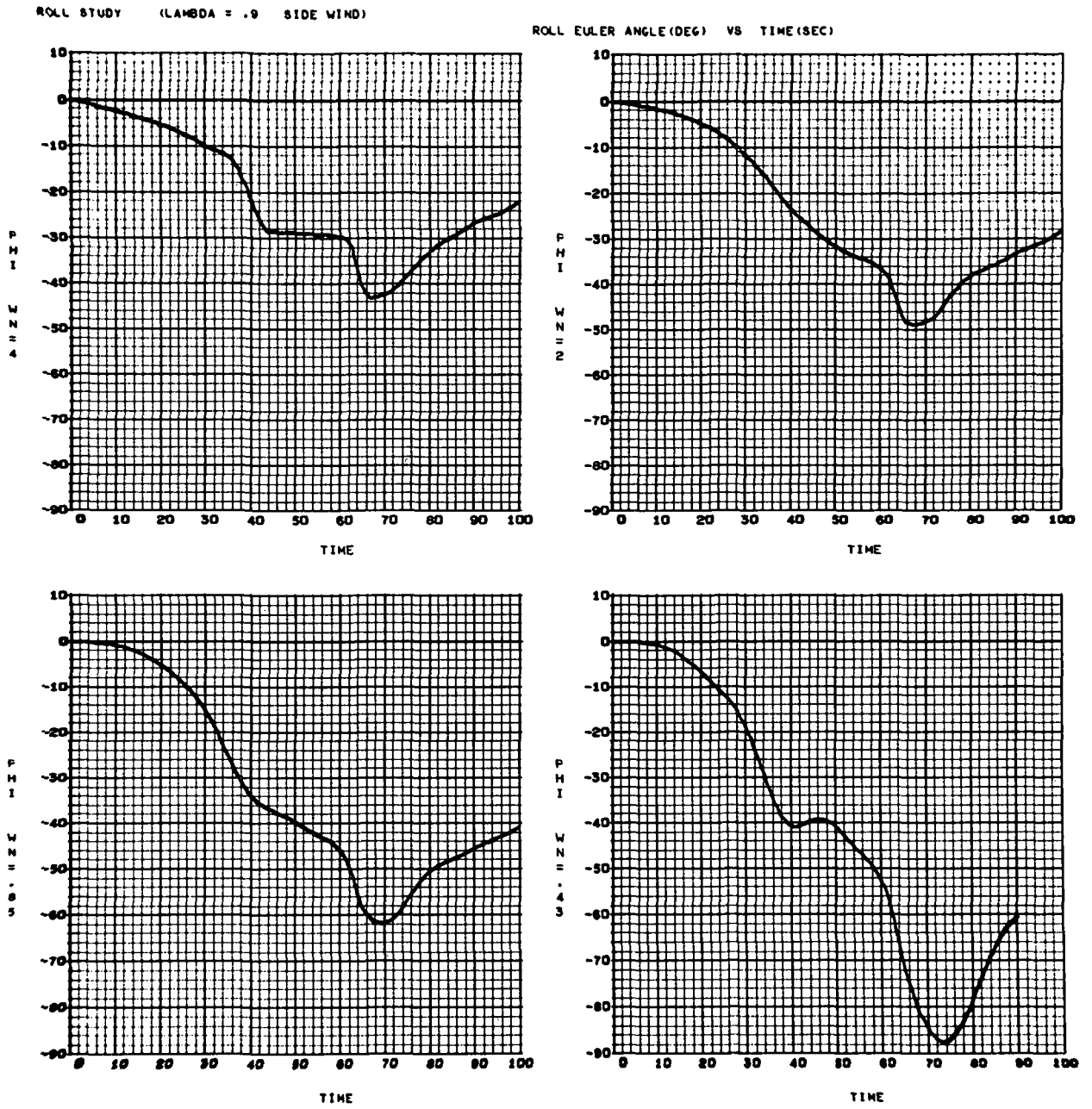
Out-of-plane winds can cause large roll torques, as illustrated below by using maximum q from the nominal trajectory and assuming a sidewind with no load or roll relief.

$$\text{Roll moment } = qSL\, C_{\ell_\beta}\, \beta = (630)\,(17000)\,(202)\,(.0008)\,(13) = 15.7 \times 10^6 \text{ ft-lb}$$

Roll Control. One of two basic methods of controlling roll angle during sidewinds is to counter the roll moment due to sideslip with combined aerodynamic and thrust vector control. The other is to allow the vehicle to roll into the wind and employ only thrust vector control.

The latter method was selected as the baseline on the basis of considerable study.

The following considerations were important in selecting this method:

- Guidance accuracy and path control is completely independent of roll orientation.

- The vehicle has greater aerodynamic stability in pitch than yaw.

- Excessive differential engine deflections would be required to counteract the wind induced roll torques shown above. This point is illustrated by the following equation for roll gimbal deflection:

$$\text{Sin } \delta = \frac{L}{NT\ell_R} = \frac{15.7 \times 10^6}{(8)\,(450,000)\,(13)} = 0.34 \text{ rad}$$

$$\therefore \underline{\delta > 18^\circ}$$

Figures 10-23, 10-24, and 10-25 show time-histories of $q\alpha$, δ_{pitch}, and roll angle for a sidewind with four values of roll autopilot natural frequency.

The straightforward roll control system shown below yields the desired independence of $q\alpha$ max from wind azimuth. This result, along with the associated roll control deflections, is shown by Table 10-12 and Fig. 10-26. Roll control deflection may appear on the pitch gimbals, yaw gimbals, or both as shown under Gimbal Plan.

$$\phi_\epsilon = -\bar{Z}_M \cdot \bar{n}$$

$\bar{n}$ — unit orbit normal

$\bar{Z}_M$ — unit vector along vehicle z-axis

$$\delta_{R_C} = \left\{ K_\phi \left[\phi_\epsilon\right]_{\phi_{\epsilon_{lim}}} - K_{PP} \right\}_{\delta_{R_{C_{lim}}}}$$

$\phi_{\epsilon_{lim}}$ = 0.115 deg

$\delta_{R_{C_{lim}}}$ = 3.44 deg

$\zeta = \dfrac{K_P L_\delta}{2\omega_n}$ = 0.8

$\omega_n = \sqrt{K_\phi L_\delta}$ (See Table 10-12)

L_δ = 1.37 rad · sec^{-2}

Table 10-12

ROLL CONTROL RESULTS OBTAINED FOR SIDEWIND, γ = .9

ω_n rad/sec	$\dot{P}_{max}$ (deg/sec^2)	P_{max} (deg/sec)	ϕ_{max} (deg)	ϕ_{peak}* (deg)	δ_{roll} (deg)	$\dot{\delta}_{roll}$ (deg/sec)	$q\alpha_{max}$ (psf-deg)
4	-4.5	-4.1	-31.6	-43.3	±3.44	±4.8	3036
2	-3.7	-3.7	-38.6	-48.7	±3.44	±2.4	2805
.85	-2.5	-3.7	-50.1	-61.7	±3.44	±2.4	2492
.43	-2.4	-4.9	-59.2	-87.4	±2.46	±1.2	2278

*ϕ_{peak} is the value of ϕ at maximum wind speed.
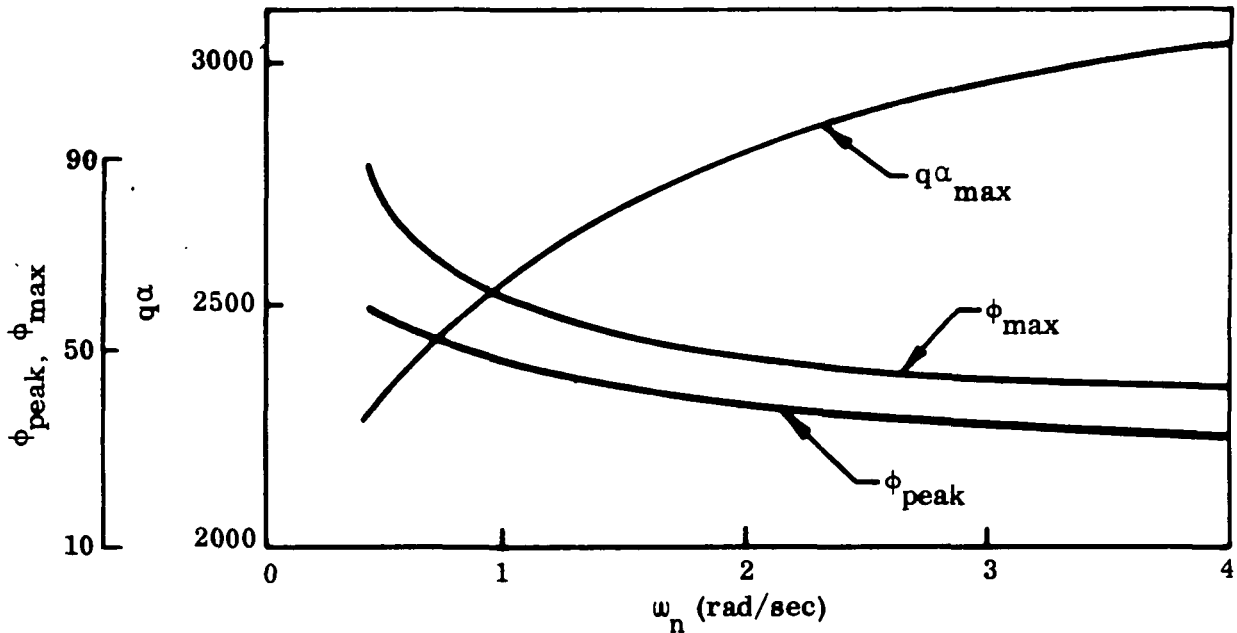
Fig. 10-26

Engine Out. Table 10-13 shows no significant variation in loads or propellant margins when one engine is out and the others are throttled up to maintain total thrust. For the particular engine failed (number 6 in the gimbal plan below) an additional 0.2 degree of pitch gimbal arc is required for cg trim just prior to staging. The effect of failing an engine used for roll control has not been determined.

Table 10-13

ENGINE OUT RESULTS, $\lambda = 0$

Wind	$q\alpha_{max}$	A_z	δ_{pitch}	$\dot{\delta}_{pitch}$	δ_{yaw}	$\dot{\delta}_{yaw}$	ΔW
H	3294	+0.8	+7.3	+6.5	+0.7	+0.4	-1018
		-22.5	-13.7	-5.8	-0.2	-0.2	
T	2740	+12.1	+2.3	+6.0	+0.3	+0.1	+770
		-22.7	-13.7	-8.5	-0.5	-0.4	

10-47

Gimbal Plan. The engine arrangement used is shown in Fig. 10-27. Control torques were developed as follows:

- Pitch — all engines deflected equally
- Yaw — all engines deflected equally
- Roll — eight outside engines deflected differentially in yaw (plan 1, Fig. 10-27)

Equivalent roll torques may be developed by deflecting the eight outside engines differentially in pitch and yaw (plan 2) or, with larger deflections, by differential pitch deflections only; that is, required roll deflections can be added to either or both gimbal directions as desired. Plan 1 was chosen to minimize pitch gimbal arc. For the roll control method adopted, yaw control torques required are relatively small, as shown by the yaw deflections of Table 10-11. Since the vehicle is aerodynamically stable, it may well be possible to substantially reduce even this torque requirement. Differential throttling of ±25 percent could be used to develop the same yaw torque as 1 degree of gimbal deflection, as shown in the next equation.
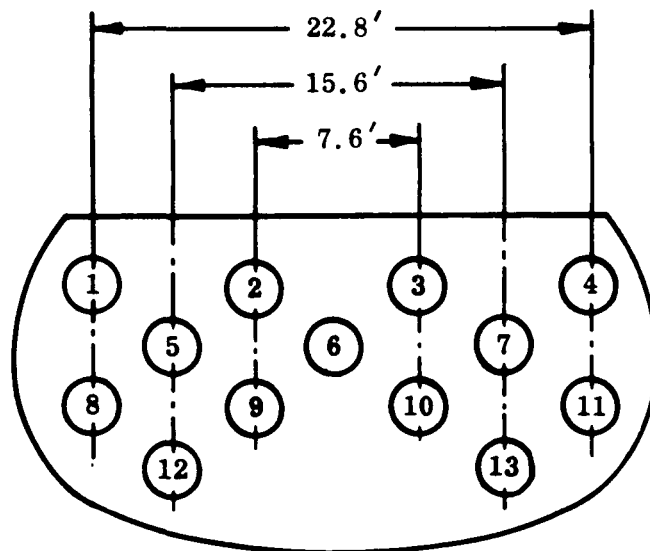
Fig. 10-27 Booster Engine Arrangement

Roll plan 1: (1, 2, 3, 4) and (8, 11, 12, 13) deflected differentially in yaw

Roll plan 2: (1, 8) and (4, 11) deflected differentially in pitch and (2, 3) and (12, 13) deflected differentially in yaw

Roll plan 3: (1, 2, 5, 8, 9, 12) and (3, 4, 7, 10, 11, 13) defelcted differentially in pitch

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Yaw torque by differential throttling:

$$\sum (\Delta T)\,(D) = N = N_\delta\,\delta = 10 \cdot 10^6 \text{ ft-lbs for } \delta = 1^0$$

$$\Delta T = 10^7/\Sigma D = 10^7/\left|(2)\,(22.8) + (2)\,(15.6) + (2)\,(7.6)\right|$$

$$\cong \underline{110,000} \text{ LB}$$

$$\frac{\Delta T}{T} \cong \frac{110,000}{440,000} = 25\%$$

At atmospheric entry, a simple dynamic pressure switchover from RCS to ACS, with an error signal level override to preclude excessive attitude errors when aerodynamic control is less effective than anticipated, is recommended (Fig. 10-28).
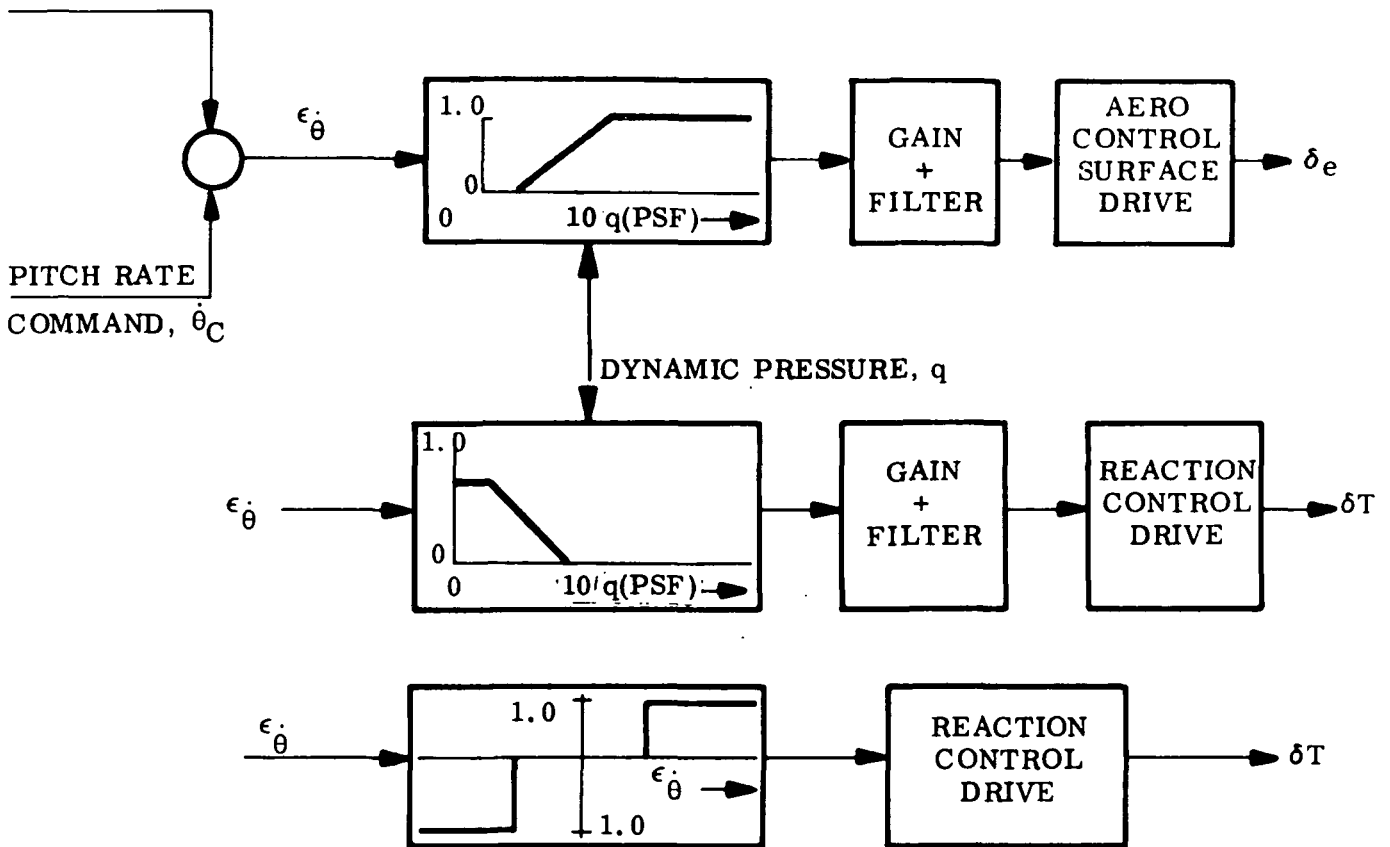
Fig. 10-28 Switching Logic

During reentry, it is recommended that flight control gains be adjusted or scheduled as functions of computed flight condition variables, such as Mach number, dynamic pressure, and angle of attack. There does not appear to be a need for self-adaptive control; neither the range nor rate of change of flight condition is anywhere near as severe as with the X-15, for example. Scheduled gain systems have a long history of use and require shorter development spans than a self-adaptive system.

10.2.3.4 Rendezvous and Docking Subsystem. Rendezvous and docking sensors must provide range and range rate and line-of-sight angle and rate to the target.

The logistics mission imposes the following requirements:

- Rendezvous
  Acquisition range: up to 70 nm
  Minimum range: 1000 ft
  Search angles: ±15 deg square pattern
  Acquisition range rate: up to 600 fps

- Docking
  Maximum range: 1000 ft
  Automatic stationkeeping within 10 feet of target
  Automatic hard docking to space station/base

Representative docking tolerances are as follows:

Relative position		Relative velocity
Radial	±1 ft	0.2 fps
Axial	±10 deg	0.2 fps
Angular	±5 deg	1.0 deg/sec

A cooperative laser system has sufficient range and accuracy to perform both the rendezvous and the docking functions if, for automatic docking, an additional laser receiver in the target is added for angle measurements.

The recommended laser system will be a derivative of the one under development by ITT for NASA-MSFC.

Several alternate radar and laser systems have been reviewed for their applicability to the rendezvous or docking functions. The characteristics of these systems are summarized in Table 10-14 (rendezvous) and Table 10-15 (docking).

Table 10-14

RENDEZVOUS SENSOR CHARACTERISTICS

	Laser (Coop)	Laser (Non-Coop)	Radar (Coop)	Radar (Non-Coop)	Radar (Non-Coop)
Search angles	$\pm15^{o}$ x $\pm15^{o}$	$\pm15^{o}$ x $\pm15^{o}$	$\pm90^{o}$ x $\pm90^{o}$	$\pm45^{o}$ x $\pm45^{o}$	$\pm185^{o}$ x $\pm165^{o}$
Range, max	75 nm	1-2 nm	400 nm	110 nm	41 nm
Range rate, max	10,000 m/sec	1000 m/sec	4900 ft/sec	11,000 ft/sec	5000 ft/sec
Accuracy					
Range	±0.05%	±0.25%	>50 nm ±0.1% <50 nm ±80 ft	±0.5%	±0.5%
Range rate	±1 m/sec	±5 m/sec	±1.0 ft/sec	±1.0 ft/sec	±1.0 ft/sec
LOS angles	±0.02 deg	±0.1 deg	±0.1 deg	±0.6 deg	±0.2 deg
LOS angle rates	±0.003 deg/sec	±0.015 deg/sec		±0.1 deg/sec	±0.2 deg/sec
Shuttle					
Size	7.5 in. dia x 17 in. (cyl) and electronics 6 in. x 12 in. x 12 in.	7.5 in. dia x 17 in. (cyl) and electronics 6 in. x 2 in. x 12 in.	28 in. x 8 in. x 5 in.	11.1 ft^3	72 in. x 15 in. dia
Weight	24 lb	24 lb	75 lb	234 lb	190 lb
Power consumption	20 watts	20 watts	250 watts	1100 watts	1200 watts
Antenna aperture	Xmtr 0.5 in. dia Rec 2-3 in. dia	Xmtr 0.5 in. dia Rec 2-3 in. dia	24 in. dia	6 ft dia	40 in. dia
Space Station					
Size	4 in. dia	NA	12 in. x 7.5 in. x 6 in.	NA	NA
Weight	2 lb	NA	14.5 lb	NA	NA
Power consumption	NA	NA	75 watts	NA	NA
Antenna aperture	NA	NA	4 in. dia	NA	NA
Source	ITT	ITT	RCA (Apollo)	Texas Instr.	Emerson Elec.

Table 10-15

AUTOMATIC DOCKING SENSOR CHARACTERISTICS

	Laser (Coop)	Laser (Non-Coop)	Radar (Coop)	Radar (Non-Coop)
Accuracy				
Range	±10 cm			
Range rate	± 3 cm/sec	Information not available for ranges required in docking (500 ft to 1 ft)		
LOS angles	± 0.02 deg			
LOS angle rate	± 0.003 deg/sec			
Shuttle				
Size	7.5 in. dia x 17 in. (cyl) and electronics 6 in. x 12 in. x 12 in.	7.5 in. dia x 17 in. (cyl) and electronics 6 in. x 12 in. x 12 in.	28 in. x 8 in. x 5 in.	11.1 ft^3
Weight	24 lb	24 lb	75 lb	234 lb
Power consumption	20 watts	20 watts	250 watts	1100 watts
Antenna aperture	Xmtr, 0.5 in. dia Rec, 2-3 in. dia	Xmtr, 0.5 in. dia Rec, 2-3 in. dia	24 in. dia	6 ft dia
Space Station				
Size	6 in. dia x 15 in. (cyl) 6 in. x 6 in. x 6 in.	NA	12 in. x 7.5 in. x 5 in.	NA
Weight	20 lb	NA	14.5 lb	NA
Power consumption	12 watts	NA	75 watts	NA
Antenna aperture	2-3 in. dia	NA	4 in. dia	NA
Source	ITT	ITT	RCA (Apollo)	Texas Instruments

10.2.3.5 <u>Approach and Landing Subsystem</u>. The approach and landing subsystem must guide the Space Shuttle from the termination of reentry, arbitrarily established at a 100,000-foot altitude and about 100 nm distance from runway touchdown, under zero visibility conditions. At a 25,000-foot altitude, a decision is made as to whether a normal power-on landing can be executed; if it is to be unpowered, a steep curved approach trajectory will be necessary rather than a low-elevation straight path. The system must be adaptable to either trajectory.

The recommended system is an adaptation of the Navy-Bell Aerosystems SPN-42, an operational carrier automatic landing system. Ground-based radar tracks the aircraft beacon and transmits its azimuth, elevation, and slant range via the UHF data link to the onboard computer, which establishes the trajectory. The computer compares the actual to the reference or desired trajectory, computes steering and throttling commands, and transmits them to the flight controls. A scanning beam ground reference system (such as AILS) holds promise for use as an alternate to the ground radar. Scanning beams are presently under evaluation by the FAA and may be operational by 1975.

Also considered were an augmented ILS-type system and an onboard mapping radar. The ILS system, augmented by a radar altimeter for glide slope extension, is not easily adaptable to the steeper emergency path; in addition, repeatability and reliability would require improvement. Contemplated use of the onboard radar concept is, at present, limited to providing an image of the runway to the pilot. Further development is necessary to achieve the required accuracy for automatic landing use.

10.2.3.6 <u>Navaid Subsystem</u>. A complement of aircraft navigation and landing aids will be required to permit test, ferry, and cruise-back operations. Navigation equipment contemplated includes ADF and VOR-DME. An ILS system, augmented by a radar altimeter for glide slope extension, will back up the automatic approach and landing system and allow low-visibility landings at any commercial or military airfield.

All-weather operations introduce the possible need for one or more additional radars for:

- Terrain and collision avoidance (up to 30 miles)
- Weather display (up to 30 miles)
- Ground terminal beacon location (up to 150 miles)

Radar functions can be provided individually by self-contained units or combined into one or more multipurpose units, which share elements such as RF generators, control electronics, and antennas.

The single-function radar approach minimizes performance compromises, eases testing, and facilitates isolation from interference. Multifunction systems, by sharing components, reduce the size, weight, and power penalties for redundancy.

Up to this point, the recommended guidance, navigation, and flight control system has two different multipurpose radar units — a combined altimeter and attitude sensor and a combined rendezvous and docking sensor. A third unit is recommended for the aforementioned all-weather operations.

The integrated system for terrain, collision, and weather avoidance would have several antenna types and locations. Current aircraft radar systems weigh from 200 to 450 pounds and required from 2 to 4.5 kw input power. They use both conventional dish-scan and phased-array antennas. The recommended multimode system will require a ±60 degree square pattern about an axis of 60 degree forward and below the vehicle longitudinal axis. Nose-cap thermal protection is metallic, precluding use of this location for an antenna. High-temperature antennas or rigid silica inserts elsewhere in the heat shield would provide RF windows.

10.2.3.7 <u>Computer Subsystem</u>. The computations can be performed in either a dedicated computer or as part of a federated computer system. Estimates of the computational equipment are given in Tables 10-16 and 10-17. Computer selection tradeoffs are discussed in Section 4, Volume III.

Table 10-16

REPRESENTATIVE COMPUTER CHARACTERISTICS FOR GN & C FUNCTIONS

Mission Phase	Storage (words)	% Utilization (1)
Prelaunch		
Mission Planning	7800	20 (2)
Strapdown algorithm	750	15.6
Ascent (atmospheric)		
Navigation	430	0.3
Guidance	1000	5.0
Attitude control	5260	30.5
Strapdown	750	15.6
Ascent (exo-atmospheric)		
Navigation	430	0.3
Guidance	3000	7.0
Attitude control	5260	30.5
Strapdown	750	15.6
Parking orbit and transfer		
Navigation	5400	13.9
Guidance	600	0.5
Attitude control	750	7.9
Mission planning	3000	all remaining time
Strapdown	750	15.6
Terminal rendezvous		
Navigation	430	0.3
Guidance	3000	7.5
Attitude control	730	7.9
Strapdown	750	15.6
Retro/deorbit initialization		
Mission planning	6000	20 (2)
Strapdown algorithm	750	15.6
Reentry/landing		
Guidance/navigation	1200	4.0
Attitude control	5260	30.5
Strapdown	750	15.6

(1) Based on IBM 4π EP with floating point arithmetic
(2) This represents alignment function only.

Table 10-17

AVERAGE INSTRUCTION PER SECOND BY PHASES

Prelaunch	50,120
Atmospheric ascent	72,400
Ascent (exo-atmospheric)	75,200
Parking orbit and transfer	53,340
Terminal rendezvous	44,120
Retro/deorbit initialization	50,120
Reentry/landing	70,530

10.3 COMMUNICATIONS

10.3.1 Recommended Configuration

The communications requirements for the Space Shuttle differ substantially for the five major operational phases of the vehicle. A preliminary indication of communication link requirements is shown in Table 10-18.

Already developed systems appear to provide satisfactory performance for the launch and orbit injection phase and for the landing phase. For the other three phases, many of the requirements are not yet fully determined. Numerous tradeoff studies and integration analysis are necessary before the preferred systems and techniques can be determined for the orbital rendezvous, docking, and reentry phases of the program. For this reason, it is considered premature to recommend a communication system configuration at the present stage of the investigation.

A tentative system configuration, minus precise mechanization details, is presented in Fig. 10-29. The long-range communication link employs a low-power solid-state transceiver. An additional power amplifier stage (TWT, Amplitron, etc.) is required when a communication satellite is in the link. Antennas are flush mounted on both top and bottom of the vehicle with omnidirectional capability to the ground and electronically or electromechanically controlled directivity for communications with a satellite. Short-range communication during ascent, rendezvous and docking, and landing will be by a VHF or UHF link. This will minimize the number of configurations and will still conform to FAA air traffic control requirements during the landing phase. The VHF/UHF antenna should not represent any major technological problems.

10.3.2 Alternative Approaches

During the orbital phase, communications will be necessary between the Space Shuttle and the ground and also between the Space Shuttle and the Space Station. Several alternatives are possible for these links, some of which are shown in Table 10-19.

Table 10-18

COMMUNICATION SUBSYSTEM – COMMUNICATION REQUIREMENTS VS MISSION PHASE

(Preliminary Link Requirements)

Function	Pre-launch	Launch	Ascent	Parking Orbit	Trans-fer	Rendez-vous	Docking	Orbital Operations	De-orbit	Entry	Landing
Voice											
Spacecraft to Ground	P	P	P	P	P	P	S	P	P	T	P
Spacecraft to Space Station					S	P	P	S			
Data											
Spacecraft to Ground	P	P	P	P	P	P	S	P	P	T	
Spacecraft to Space Station						S	P	S	S		
Command and Control											
From Ground		P	P	P	P	P	S	P	P	T	P
From Space Station						P	P				

P = Primary
S = Secondary
T = Tentative
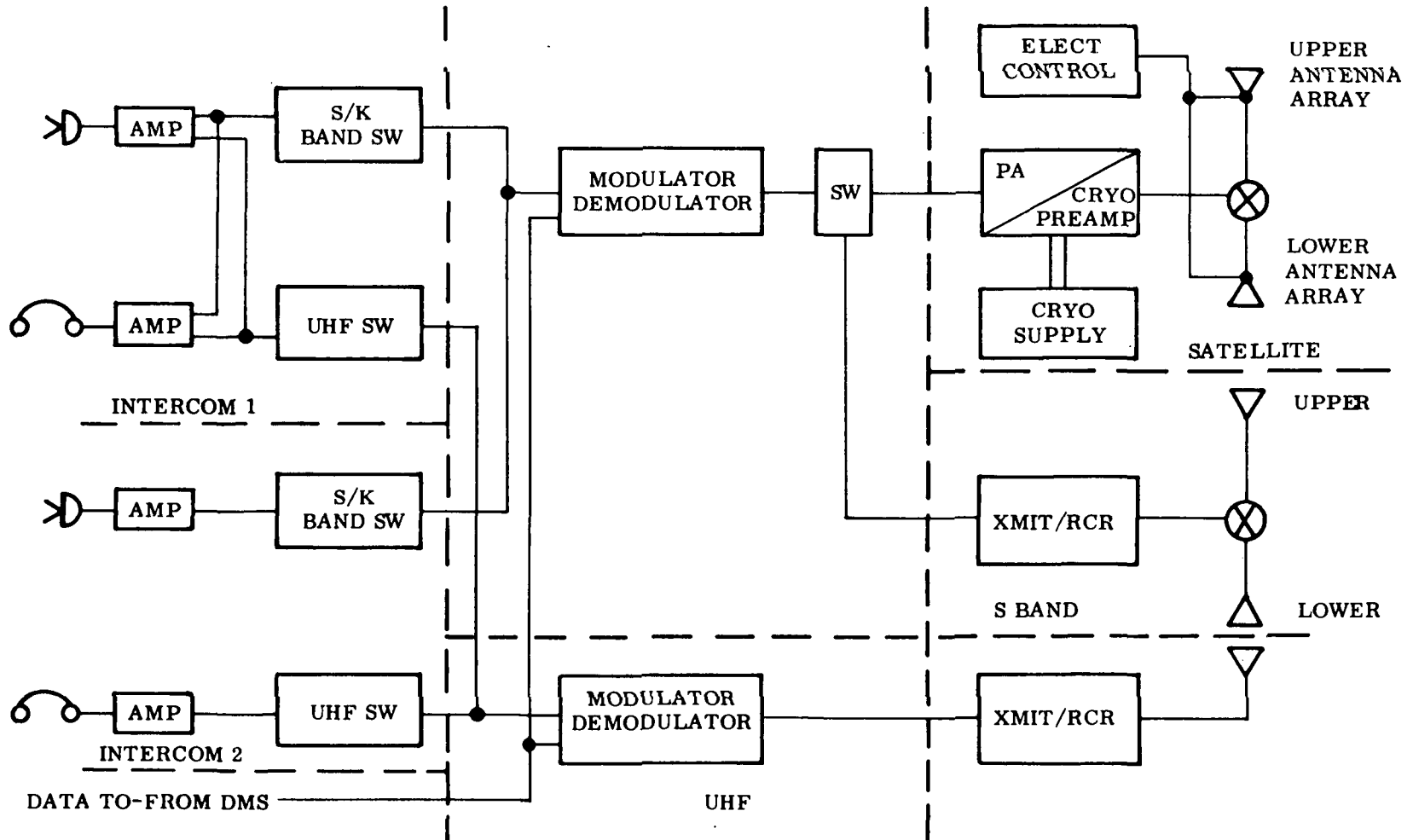
Fig. 10-29 Communication System

Table 10-19

ALTERNATIVE APPROACHES

Mode	Shuttle to Ground	Shuttle to Space Station
Direct – S-band	X	X
Direct – high frequency	X	X
Direct – (COMSAT freq)	X	X
Direct – laser link		X
COMSAT relay (1 hop)	X	X
COMSAT relay (2 hops)	X	X
Space Station relay	X	
Space Station to COMSAT	X	X
Rendezvous radar		X
Laser link to COMSAT	X	X

The communication configuration will depend not only on the requirements and characteristics of the Space Shuttle but also upon the characteristics of other systems, such as the communication satellite relay.

Other alternatives involve the technology to be employed. One of the important technological areas pertains to the use of phased arrays versus more conventional antennas. Phased arrays are very attractive because of their flexibility and aerodynamic properties; however, the complexity and expense of these systems require the full examination of more familiar alternatives as well. High-temperature antenna technology may well be a pacing item in determining portions of the communication system configuration. Many of the design choices will depend on reentry profile; heat shield configuration; and operational requirements for voice, telemetry, tracking, and command during reentry, etc. If reentry blackout is determined to be acceptable, the possibility of retracting the antennas for heat shielding purposes may become an attractive alternative.

10.3.3 Evaluation of Alternatives

As previously indicated, the major communication system problems are encountered during the orbital phase, in which the shuttle will generally be beyond the line of sight of any given ground station, and during reentry, when high temperatures and plasma effects will make communication difficult. The joint solution of both these appears to be possible by the use of a frequency band in the region of 15 to 35 GHz.

The use of these frequencies is already being considered for advanced communication relay satellites. These frequencies should also be sufficiently high to penetrate the reentry plasma; thus an integrated system might be used for both satellite relay and for reentry. If the phased array antenna is unable to function during reentry because of high temperatures, a simpler low-gain, downward-looking antenna could be employed during that phase.

Preliminary analysis indicates that only modest data bandwidths can be transferred via COMSAT relay. Laser links to the relay satellite hold substantial promise for expanded data bandwidth capability during on-orbit operations.

10.4 DATA MANAGEMENT

The onboard data management system functions include data assessment, computer processing, realtime subsystem control and display, and checkout of subsystems. This system will allow complete autonomy of the Space Shuttle from ground monitoring and control stations and relieve the crew from having to perform the detailed subsystem checks and continuous status monitoring of the systems.

10.4.1 Recommended Configuration

A minimum level data management system presented herein is of a relatively easy to attain design, which is within the evolutionary trend exhibited throughout the industry. The special emphasis study on integrated electronic systems reported in Volume III contains recommendations for a more extensive integration of data management functions.

The following design concepts, proposed in this report, are expected to survive the individual subsystem configurations that may be employed to attain desired levels of integrations:

- Multiple data buses
- Parallel data processors across common input/output buses
- Distributed computer element applications
- Structured processor applications (computers responsible for computers)

10.4.1.1 <u>Functional Description</u>. The onboard data management system provides common hardware to supply time-shared services in the following areas:

- Instrumentation
- Display
- Crew control
- Onboard checkout
- Fault isolation

- Data formatting
- Housekeeping
- Data analysis
- Data storage
- Remote control

A functional schematic is presented in Fig. 10-30.

10-63
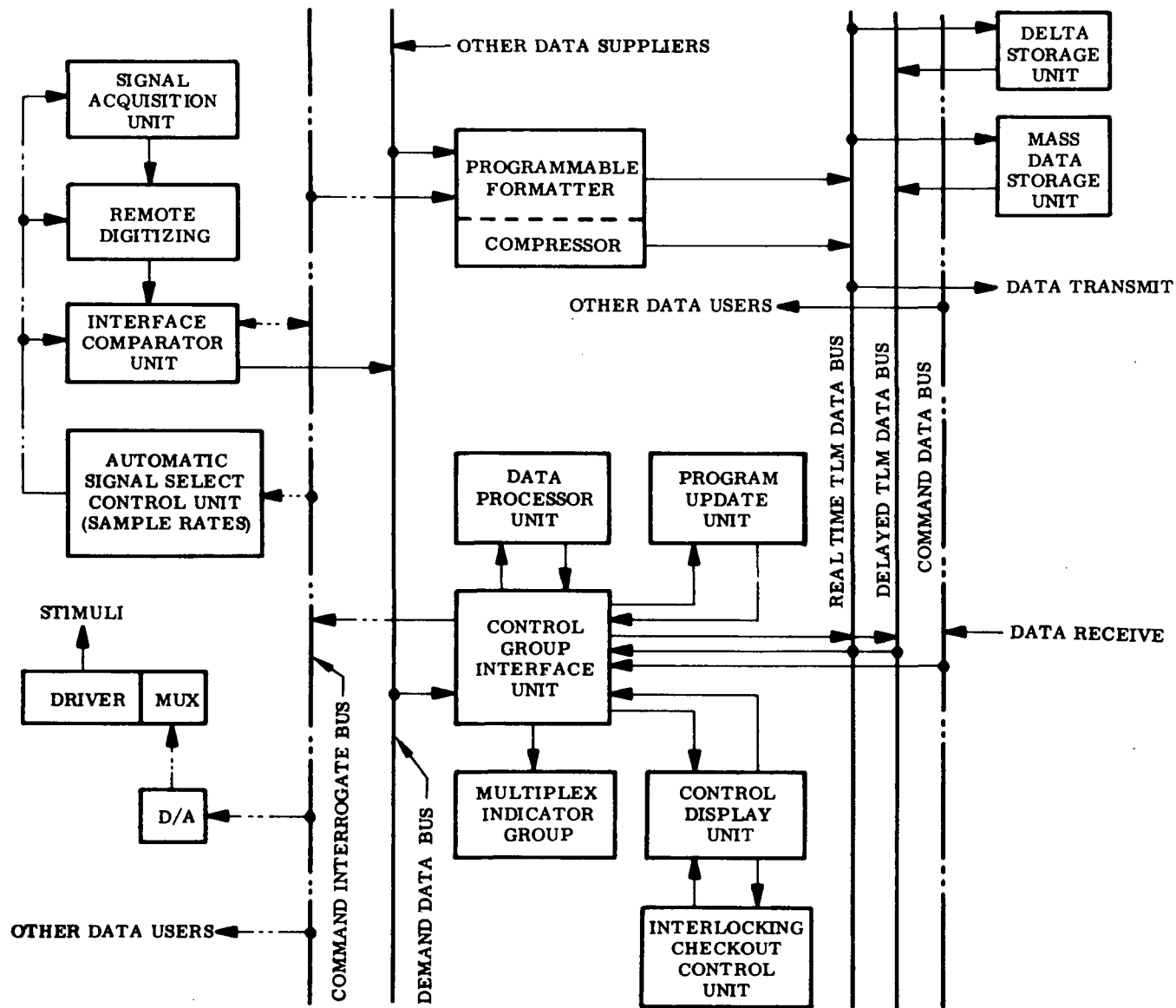
Fig. 10-30 Data Management Functional Schematic

Application of semiautomated operator instruction procedures permits logical trouble-shooting during flight to be extended beyond the level attainable with electronic signal processing. With this technique, the operator is employed as a stimulus generator/comparator. The technique yields a permanent record of the unit or units requiring maintenance action. Man/machine interface requirements and data-processing facilities are used for the normally required subsystem control and housekeeping functions on a time-shared basis.

The data management system is functionally designed as a distributed computer inter-connected with a high-speed communication network.

Coaxial cabling is used for high data rates, and twisted shielded pairs are used for less stringent requirements.

Major busing arrangements are command and interrogate, demand data, and telemetering.

Command and Interrogate Bus. This bus provides the instruction path for data management system control. Sample intervals, formats, and sequence are loaded into the automatic signal select control unit and altered as desired over this bus. This function is normally performed during final prelaunch preparations and at primary operating mode changes. A semimanual mode is available for use by skilled technicians in lieu of ground support equipment during normal hangar maintenance. Major program updates are accomplished by dumping prerecorded instructions (carried in plug-in cartridge form) through the program update unit.

Remote decision logic in the interface comparator units requests specific actions, which are implemented directly or provided by either preprogrammed response from the data processor/control group or manual response to the decoded request.

Demand Data Bus. The demand data bus is a group of time-shared multiplexed communication links. Major interfacing units of using subsystems have common switching logic interface modules, which maintain a present value of each signal. The present value, which is accessible in an asynchronous mode by any user on the same bus under

predefined priorities, is updated at a rate determined by either the natural frequency response of the signal or its sensitivity to elapsed time.

Telemetering Bus. A function similar to that supplied by the demand bus is supplied by interconnecting the basic control group with the formatter, data compressor, data storage, and external communications links. Realtime data are isolated for direct transmission to earth if desired for onboard storage and for compression if desired prior to either transmission or storage.

An uplink command data bus permits external coded command control through the control group interface unit. Uplink program changes are stored in the program update unit for subsequent readout or removal by cartridge for off-line retention. Access to the combined bus system by the central control group permits performance evaluation of the formatter, compressor, and data storage units.

10.4.1.2 Performance. The data management system satisfies the following performance requirements:

- Acquires data from interfacing elements
- Disperses data and commands to interfaced elements
- Provides a man/machine interface for both control and display
- Accepts remotely generated control commands
- Provides an accessible data bank for crew use
- Provides data storage with subsequent recall
- Measures, computes, and displays current performance of selected vehicle subsystems
- Performs failure diagnosis of subsystem malfunctions and isolation of subsystem faults to a packaged subsystem, component, or group of components packaged together and identified as a replaceable unit (in cooperation with builtin test circuitry, isolation is extended to replaceable modules).

10.4.1.3 Functional Characteristics. The system operates in a fully automatic mode (with manual reprogramming capability) for tasks that can be predefined in sufficient detail for mechanization. Specific tasks performed automatically include the following:

- Malfunction isolation to a replaceable unit
- Management, loading, and use of expendables
- Self-check and calibration
- Initiation of unsafe manual commands
- Formatting of telemetry data
- Data storage
- Performance of trending calculations on data with trend predictable characteristics
- Continual assessment of performance and display of a warning when measured values exceed the programmed limits
- Multiplexing functions for interfaced elements

A manual mode is provided to assist the crew in accomplishing complex tasks. Manual mode functions include:

- Presentation of time-line instructions with electrical interlocks for automatic switching and control assistance and to preclude operator error
- Step-by-step procedures for performing diagnostic troubleshooting with the instructions and controls available through the system and other available controls and indicators

10.4.1.4 <u>Performance Characteristics</u>. The system accepts compatible digital data through a common interface design or analog data, which are conditioned and digitized by data management hardware. The input hardware is modularized and located at strategic points to minimize wire weight.

The onboard checkout function is sufficient to validate the vehicle subsystem functional integrity established in hangar testing. Accomplishment of this function is independent of onboard personnel.

Time-line instructions are prepackaged by plug-in cartridge. A minimum of 5000 frames of instructions is available in each cartridge.

Any data accessed are presented at the operator's console on demand. Subsystem data required for continuous presentation are demultiplexed and presented by conventional instruments at the crew station.

10.4.1.5 <u>Physical Characteristics</u>. The elements of the system are discussed in detail in the following paragraphs.

The control display unit provides the primary man/machine interface for the system and includes the capability for displaying step-by-step instructions for the following:

- Prelaunch countdown
- Reentry status check
- Fault analysis
- Reactivation checkout
- Deactivation checkout
- Any loaded instruction set

Display is a combination of live signal display at a cathode ray tube and presentation of test and diagrams from film data contained in the interlocking checkout control unit. Manual controls include an alphanumeric keyboard and selector controls for the data management system. Display of data in engineering units and appropriate high-frequency waveform data are included.

The interlocking checkout control unit is a remotely located data source for the control display. Film-pack visual and digital data are accessed as required, visual data are displayed on the CRT of the control display, the digital data are utilized to activate or evaluate monitored subsystem status. This alerts the operator to improper responses and performs automatic data access to lessen operator requirements for attention to detailed functions.

The indicators within the multiplex indicator group provide a continuous visual output of multiplexed data, which are displayed for distribution or checkout purposes in lieu of providing parallel hard wire interconnects.

The program update unit provides temporary storage capacity for up-link instructions and crew initiated program changes. Cartridge loading permits a supply of prepared program changes or alternate program instructions to be carried onboard for use by the crew at appropriate points in the mission.

Interface control between the crew station complex and the data management elements is provided by the control group interface unit. Included are provisions for command override to permit remote control of the internal data subsystem.

The data processor unit is a general-purpose digital computer, used as the major data processing element of the data subsystem. Functions include the following:

- Onboard trend analysis
- Failure analysis
- Transfer function generation
- Logic evaluation
- Engineering unit conversion
- Operator calculation
- Limit determination
- Master clock

Commonality of building block units within the computer will be maintained for all data processing elements within the vehicles to minimize spares provisioning requirements.

The programmable formatter/compressor accesses data at a rate and sequence that is under program control. The interface comparator unit permits the formatting to be independent of individual channel sample rates. An optional data compressor function included may be used to lessen RF bandwidth requirements in realtime, lower storage capacity use as a function of time, or (in combination with the control complex) compress data for mass storage and reconstitute them for transmission.

A continuously updated record of the previous 15 minutes of data is provided by the delta storage unit, which:

- Permits recall by ground or crew of data surrounding the onset of a problem as an aid in identifying the source cause
- Provides a flight recorder function for "crash" investigation

The mass data storage unit provides data during periods in which downlink is not available or RF silence is desired. Tape recorders are used for these functions;

provisions are made for maintaining a positive nitrogen atmosphere internally to eliminate oxygen contaminant problems.

The interface comparator unit is functionally a part of the data processor. Two primary functions are to detect equal to or greater than and to provide accessible memory. Each incoming data sample (signal value) is assigned a memory position. Subsequent signal samples are compared to the stored value, and if different the memory position is updated. This continually updated memory value is accessed as required by any user connected to the demand data bus. Each signal has two limit values that may be set to initiate a request for a specific action when the signal value equals or exceeds the limit (high or low). One option permits limit penetration to be displayed as an alert to the crew, simultaneously updating the limit to its maximum design value. The next limit penetration is then interpreted as a failure.

An alternate option (precoded) initiates an action by direct response through a dedicated demultiplex/driver circuit combination to provide an appropriate stimulus or request a preloaded instruction to be executed by the data processor through the control group interface.

The signal acquisition units include signal conditioning for each signal, multiplex switching, output amplifiers, and calibration signal sources. Capacity for the 2000 signals that require conditioning and digitizing is provided by 16 units.

Multiplex digitizers (four each) are used to minimize the length of the analog circuits, thus reducing noise susceptibility and decreasing circuit settling time.

The sample rate of each signal is under the control of the automatic signal select control unit. The desired multiplexing format is designated by the control center via the command and interrogate bus. The designated format is executed until changes are requested. Request modes available are as follows:

- Preprogram cartridge input by the program update unit
- Up-link by the command data bus
- Operator input by the control display unit

10.4.2 Alternate Approaches

These are described in Volume III, Section 4.

10.4.3 Evaluation of Alternate Approaches

This is reported in Volume III, Section 4.

Section 11
CREW SYSTEMS

Major study areas have been focused on defining the role of the crew; determining crew size and composition; developing flight station concepts; and analyzing crew visibility requirements, habitability needs, control-display concepts, and crew safety requirements. Emphasis to date has been placed on the identification of crew factors that will have a major impact on vehicle system definition and configuration development.

11.1 CREW FUNCTIONS

A preliminary list of pilotage functions for the orbiter is contained in Space Shuttle Data, LMSC-A955317A, Vol. I, Section 5.2.1. Since the Space Shuttle is highly automated, the crew operates in essentially a manual backup mode, with override and takeover capability. Crew functions and associated time estimates were evolved through close coordination with subsystem design personnel, X-15 pilot interviews, LMSC experimental test pilot discussions, and extrapolation from related previous flight vehicle programs.

11.2 CREW SIZE AND COMPOSITION

Orbiter vehicle crew size and composition requirements were examined in relation to crew functions to accomplish mission objectives, mission duration, and work/rest cycle considerations. Current NASA guidelines call for a two-man crew with a flight system designed to be operable by one man. A 7-day self-sustaining mission duration is specified, with a 30-day mission duration capability, based on expendables provided from payload allocations.

The specified two-man crew suggests a pilot/copilot arrangement. Since the vehicle must be operable by one operator, both crew members must have overlapping skills in basic phases of system operation. Distribution of crew functions among two crew

members appears to be feasible for highly automated, short-duration missions. However, the full range of missions for the Space Shuttle has not been identified. As mission definitions are evolved, the magnitude of mission-peculiar crew functions may augment the flight crew workload. The two-man crew concept does not offer extensive flexibility for additional major crew tasks. This and other considerations indicate the need for continued analysis of crew workload to ensure that it can be effectively managed by a two-man crew.

11.3 CREW COMPARTMENT CONFIGURATIONS

Candidate crew compartment configurations have been developed for the Two-Stage and Triamese orbiter and booster. Maximum consideration has been given to the following NASA-desired system characteristics that influence compartment configurations:

- Two-man flight crew
- Design reference mission of 7 days
- Rapid and safe flight crew egress and abort capability
- Landing visibility comparable to that in high-performance aircraft
- Shirtsleeve environment with crew/passenger transfer conducted within pressurized compartments

Three major crew compartment areas have been developed for the candidate vehicles— the flight crew compartment, the tunnel (from the flight compartment to the cargo compartment), and the airlock. The payload compartment, in which the passengers or possible cargo payload management personnel will be located, is not included in the basic compartment analysis. Basic design criteria for compartment design are as follows:

- Design for 5th to 95th percentile crew anthropometrics.
- Provide normal and emergency ingress/egress traffic patterns.
- Design for external visibility requirements — rendezvous, docking, and landing.
- Provide capability for pressure suit don/doff.
- Consider comfortable living area envelope.
- Locate hatches for ease of access and design for simplicity of opening and closing.
- Integrate work and living spaces into a noninterference configuration to extent practical.

11-2

- Lay out flight instrument panel according to aerospace human engineering criteria.

- Provide for simplified and acceptable waste management, personal hygiene, and food/water management for missions to 7 days.

- Provide rapid and easy access to safety and emergency equipment.

- Reduce noise levels and provide external and internal illumination control within the cabin.

- Design for simplified internal compartment traffic flow and translation aids.

- Consider ejection-seat design constraints and provide eject-path geometry.

- Develop seat design for comfort and g-load protection.

- Consider isolation of sleeping crew member(s).

- Locate stowed items for ease of retrieval and stowage.

- Provide maximum modular design configuration for ease of installation and removal and integration.

A representative listing of crew provisions follows:

- Crew seats and eject system

- Flight instrument consoles

- Waste-management/hygiene station

- Food/water management station

- Hatches

- Emergency breathing system

- Payload display/control console (airlock)

- Survival kit

- First-aid and medical instrument kit

- Experiment support locker

- Tool kit

- Personal gear

- Emergency repair kit

- Exercise package

- Recreation package

Compartment volumes are presented in Table 11-1 for the flight crew, tunnel, and airlock areas. These volumes have been incorporated into the basic vehicle design and are practical to integrate within structural constraints.

Table 11-1

CREW COMPARTMENT VOLUME SUMMARY

Compartment Area	Volume

Orbiter Flight Station
    Basic flight station
    Recommended flight station
	338 ft^3
	568 ft^3

Orbiter Tunnel	552 ft^3
Orbiter Airlock	137 ft^3
Booster Flight Station	268 ft^3

Figure 11-1 presents the candidate flight compartment configuration for the booster (Two-Stage or Triamese); Figs. 11-2 through 11-5 illustrate various views of two candidate flight compartment layouts for the orbiter (Two-Stage or Triamese). The essential difference between the basic and recommended candidate compartments is volume. The basic flight compartment is specifically directed at the NASA-desired system characteristics of a two-man crew for only 7 days. The recommended flight compartment provides some growth and mission flexibility relative to additional crew inclusion and potential habitability extension beyond 7 days. Discussed in LMSC report A955317A, Vol. I, Section 5, is the possibility of additional crew members, habitability, and growth/flexibility potential if considered very early in the design process.

The tunnel from the flight crew compartment to the cargo payload area is sized for crew transfer, both in a shirtsleeve and suited mode. A transport track-sled and fireman's pole are candidates for simplified and rapid translation of the crewman. Figure 11-6 illustrates these concepts. An airlock, illustrated in Fig. 11-7, has been located directly adjacent to the cargo payload compartment and in a position to provide a hatch for extravehicular egress/ingress. It is sized to accommodate two crewmen, permits suit donning and doffing, provides for external viewing of the cargo payload during transfer, and provides for a small cargo payload unloading/loading control console.
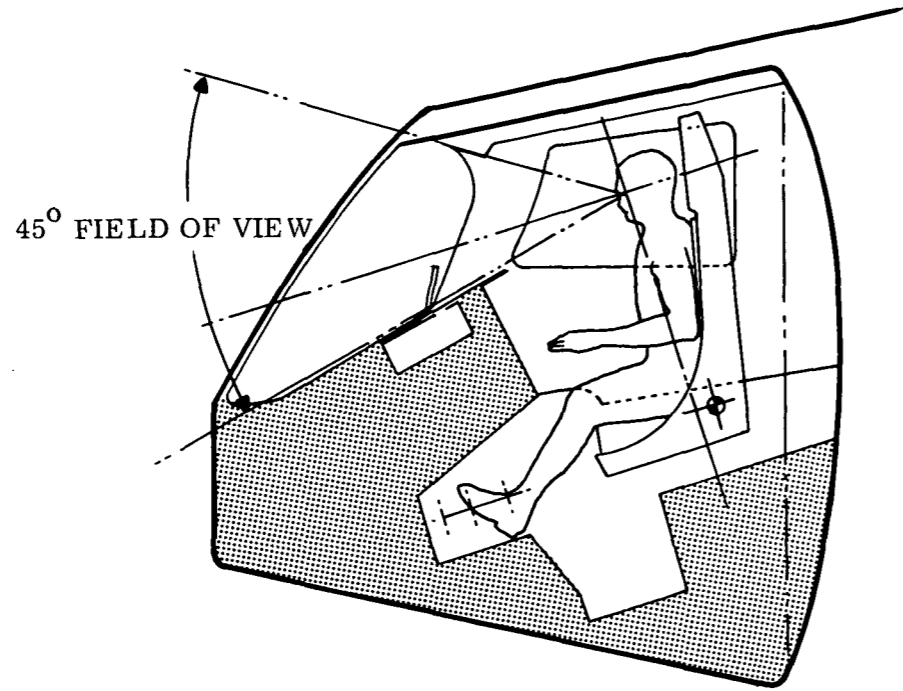
Fig. 11-1 Booster Flight Station
(Approx 80 in. Long by 76 in. High)
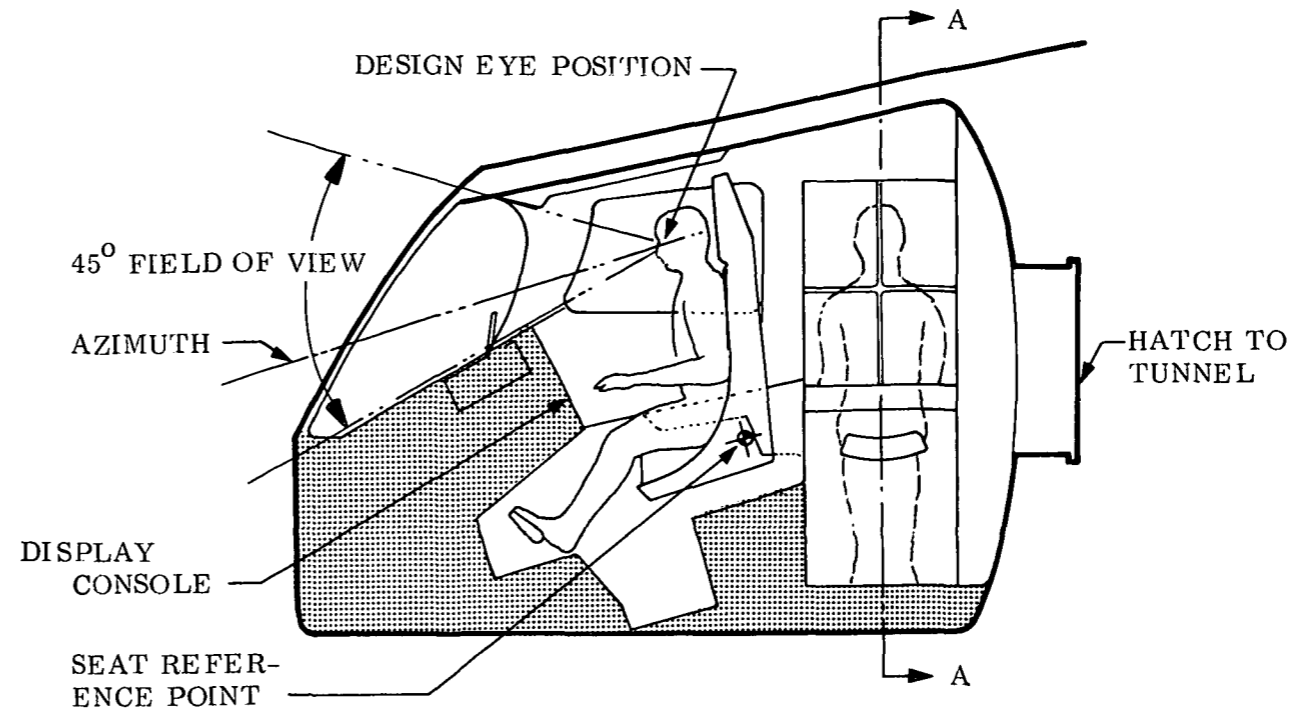
Fig. 11-2 Basic Orbiter Flight Station Compartment
(Approx 100 in. Long by 76 in. High)
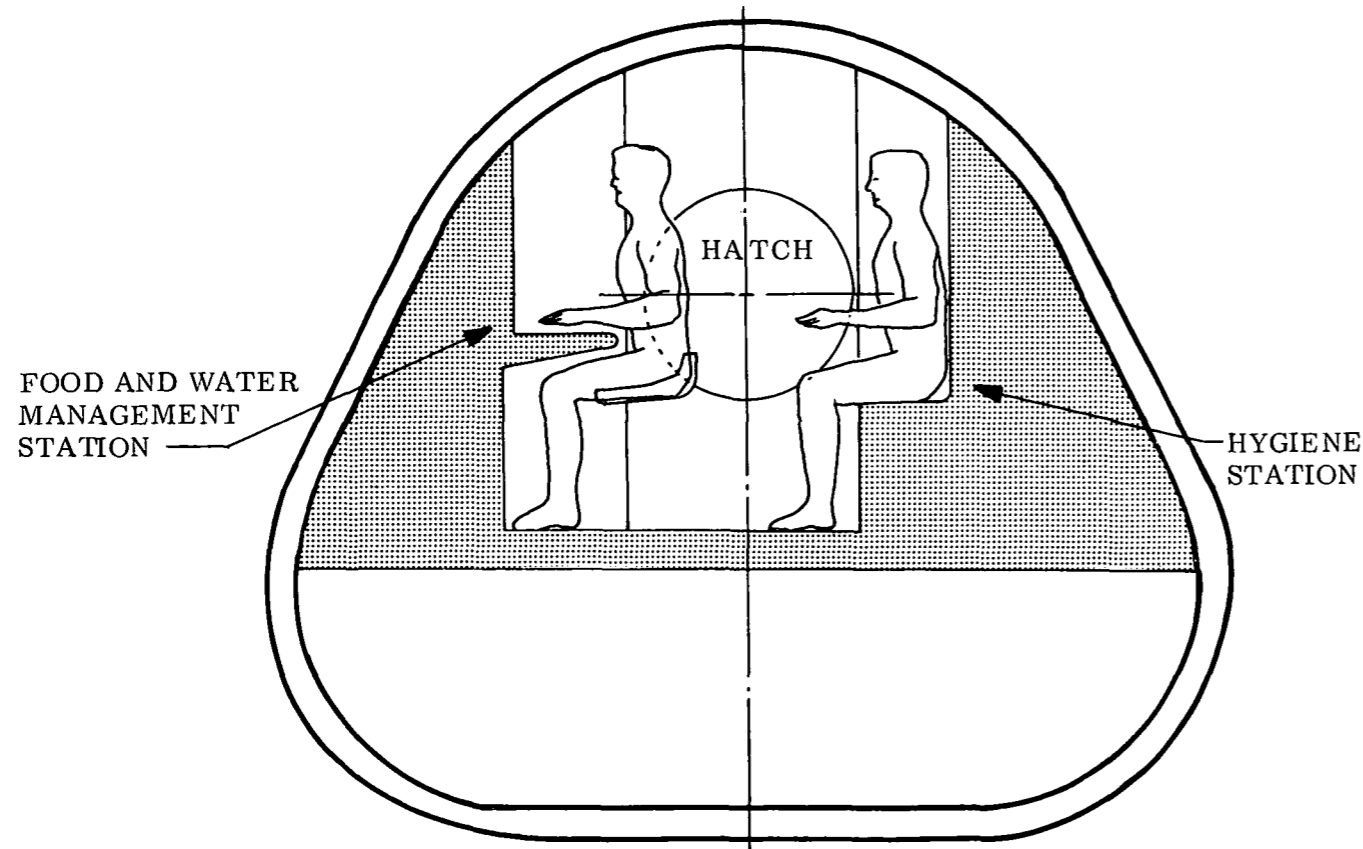
Fig. 11-3 Basic Orbiter Flight Station Crew
Support Area
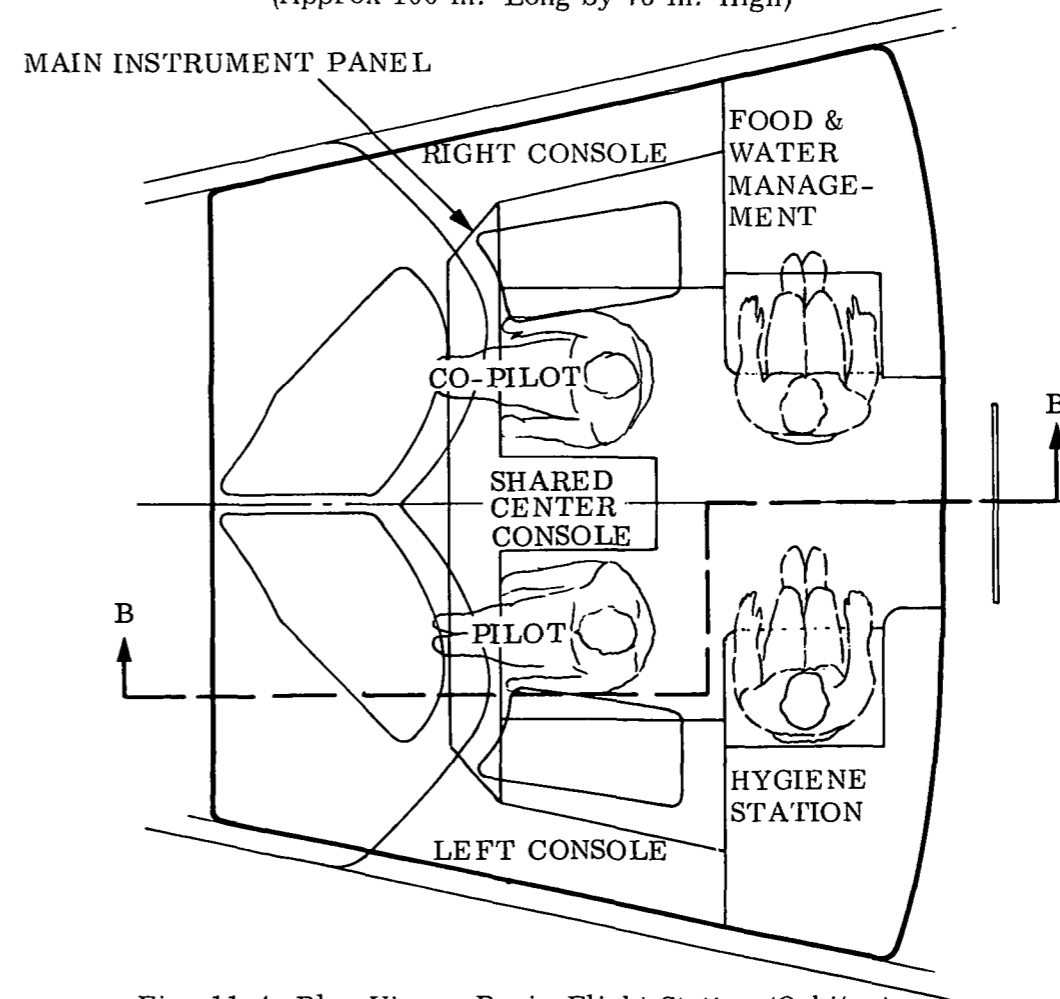
Fig. 11-4 Plan View – Basic Flight Station (Orbiter)

FOLDOUT FRAME 1

FOLDOUT FRAME 2

| BASIC FLIGHT STATION | EXTENDED SEGMENT GROWTH | CREW SUPPORT AREA |
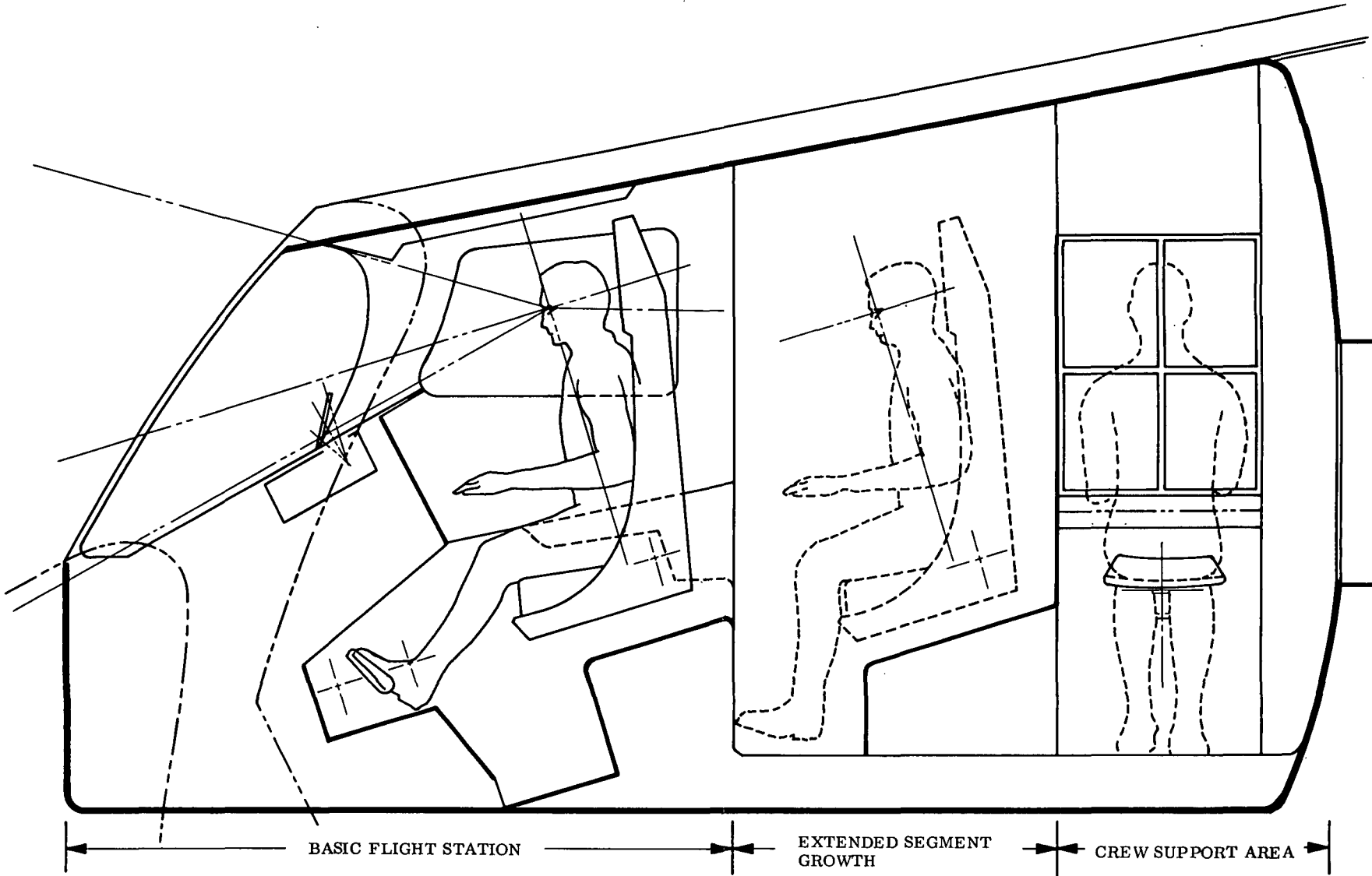
Fig. 11-5 Recommended Flight Station Compartment
(Approx 140 in. Long by 76 in. High)

BASIC FLIGHT
COMPARTMENT

HATCH

ALTERNATE
MOBIL SLED
CONCEPT

RAIL

TUNNEL TO
PAYLOAD AREA
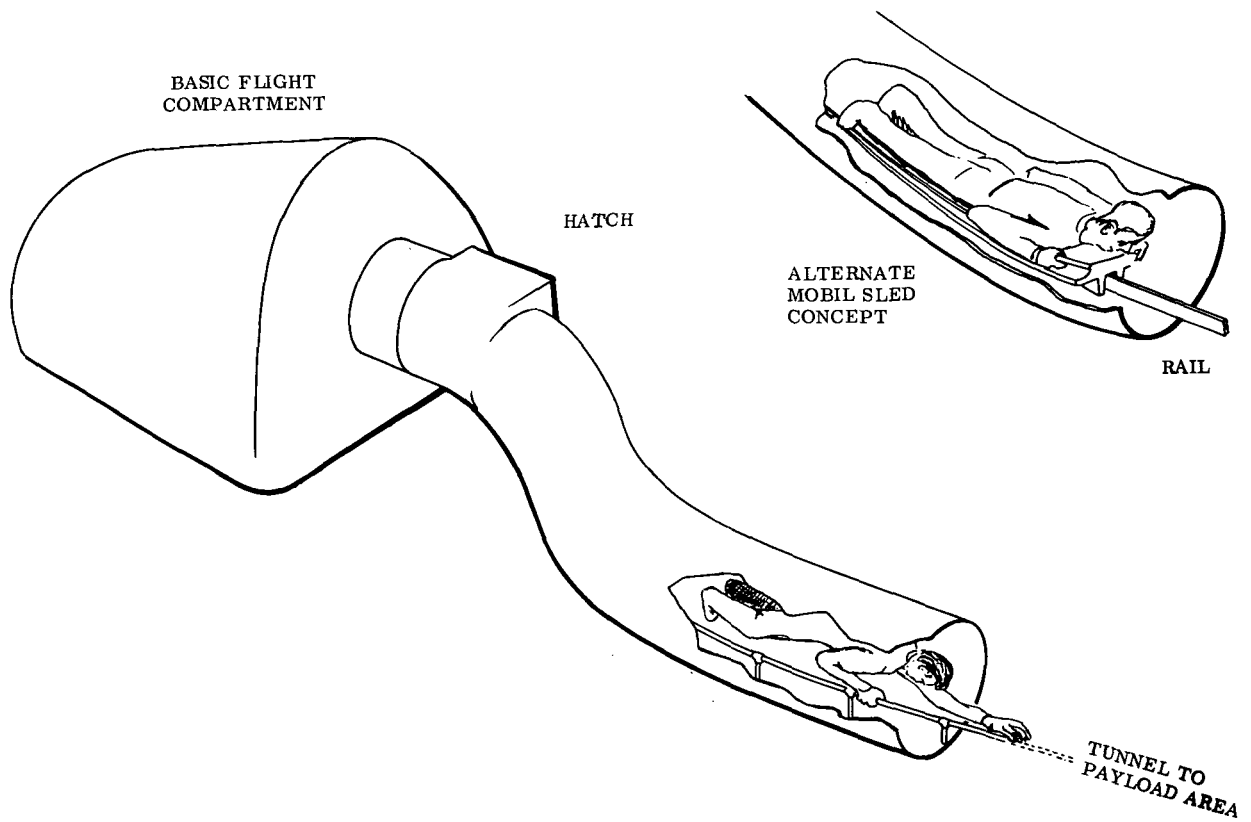
Fig. 11-6 Transfer Tunnel Hand Rail Locomotion Concept

HATCH WITH
GLASS CENTER

CARGO TRANSFER.
CONTROL
CONSOLE

VIEW OF
DOCKED
SPACE STATION,
ETC.

PAYLOAD
CARGO

HATCH

AIRLOCK

TUNNEL
TO PAYLOAD

HATCH

TRANSFER TUNNEL
TO CREW COMP.
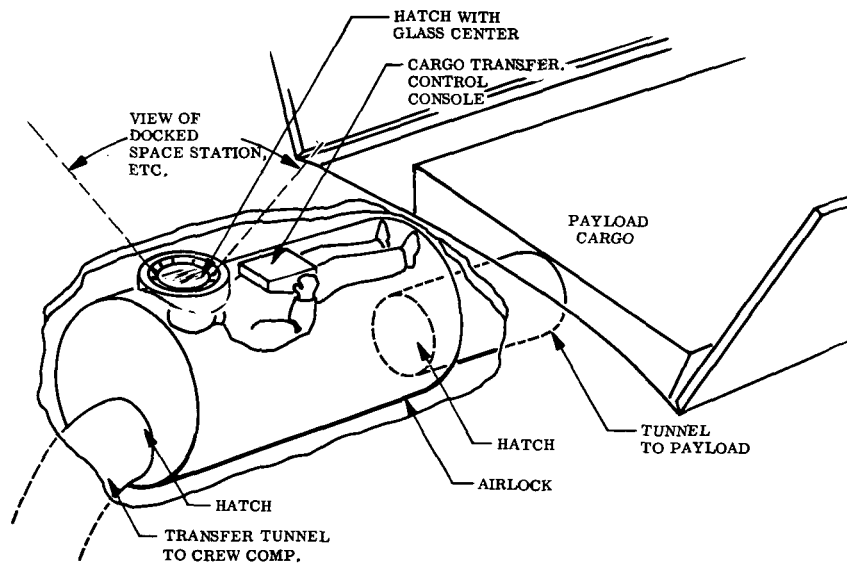
Fig. 11-7 Airlock Compartment

11-8

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11.4 CREW VISIBILITY CONSIDERATIONS

Crew visibility for landing and rendezvous/docking has become one of the key crew systems design issues. Two basic viewing candidates or a combination of the two appear to be applicable — out-the-window viewing and optical/sensor vision. Of the two, out-the-window viewing impacts on the vehicle design more significantly, particularly with respect to aerothermodynamics. Current conceptual schemes for crew visibility for the orbiter and the booster are as follows:

- Out-of-window
- Porthole
- Periscope and mirror
- Combinations
- CRT presentations
- Projection display presentations
- Fiber optics presentations

The NASA-derived system characteristics of landing visibility comparable to that with high-performance aircraft necessitates examination of MIL-STD-850A (Aircrew Station Vision Requirements for Military Aircraft), 8 June 1967, and Aerospace Standard 580A, revised 1 October 1968. The basic vision envelopes for the pilot designated in these documents are presented in Fig. 11-8. Out-the-window vision in the booster appears to be reasonably practical for either standard; and requirements can, in general, be met. These requirements are significantly more difficult to achieve for the orbiter. Therefore, both window and optical/sensor techniques should be examined. In inclement weather, both the orbiter and the booster would have to be provided with optical/sensor techniques for landing operations.

Out-the-window viewing has led to examination of the forward mold lines of the vehicle nose in relation to such visual considerations as landing field sighting, glide slope angle, vehicle attitude during glide, touchdown point, horizon, runway center line and touchdown point, touchdown angle of attack, go-around, collision avoidance, and space-terminal rendezvous. Figures 11-2 and 11-4 present representative layouts of out-the-window viewing for the orbiter. This generally meets MIL-STD-850A and AS 580A requirements. Figure 11-9 presents four candidate window-vehicle alternative configurations for meeting basic vision requirements; however, each alternative is being closely examined relative to its impact on design and aerothermodynamics.

CURRENT VISUAL
ENVELOPE

MIL-STD-850A

AS 850A
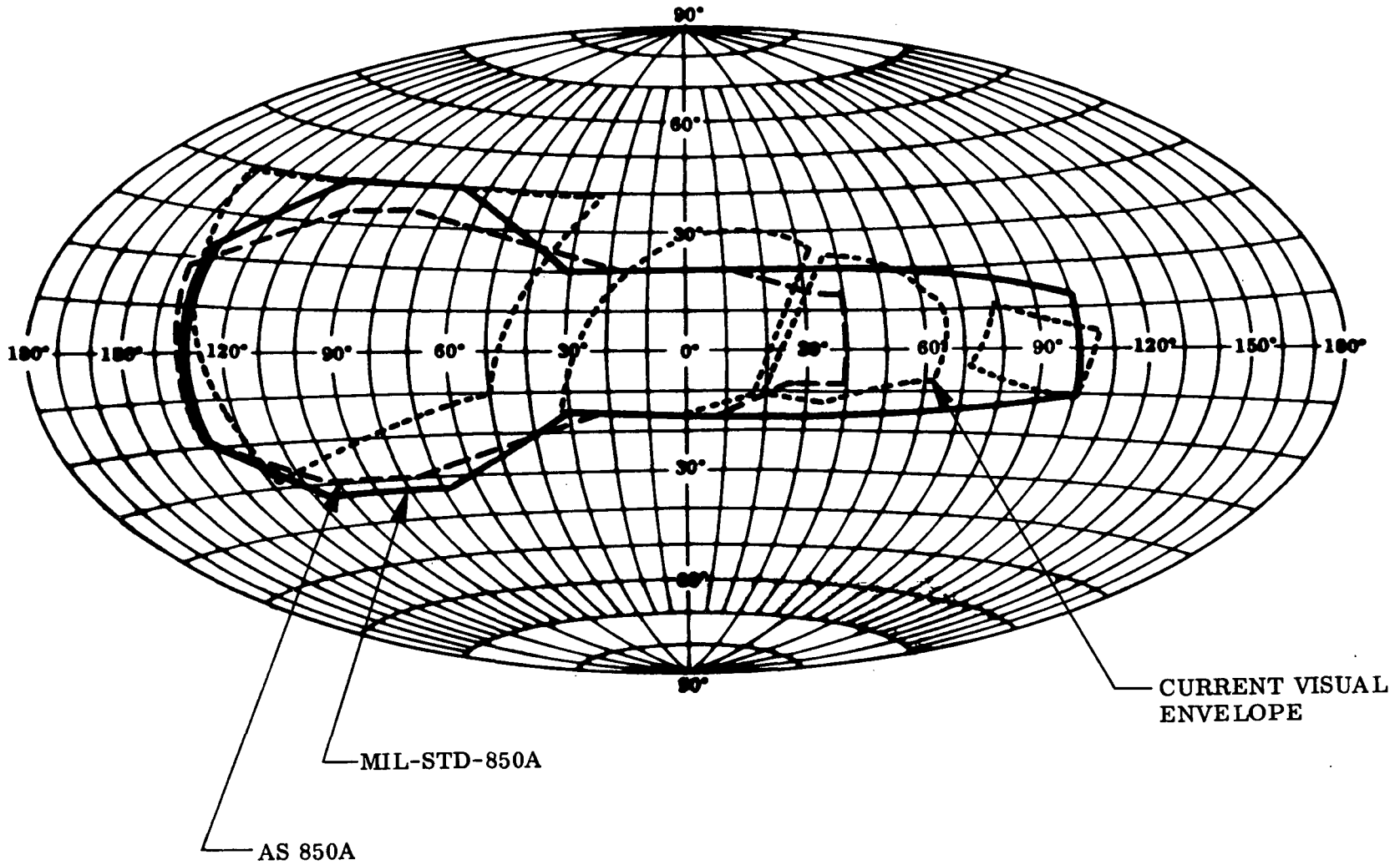
Fig. 11-8 Visual Envelope Comparison

BLOW-OFF CANOPY CONCEPT
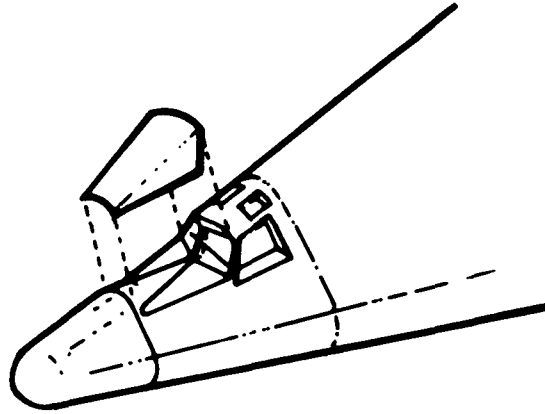
ELEVATED CREW COMPARTMENT CONCEPT
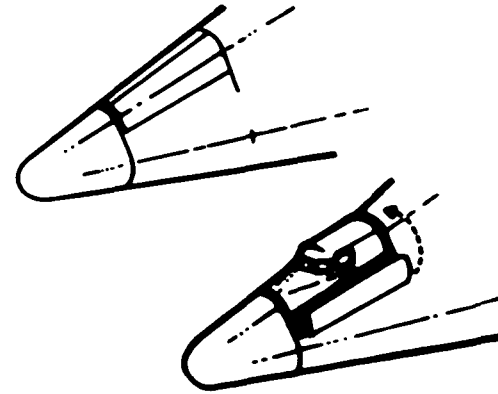
RAISED CANOPY CONCEPT
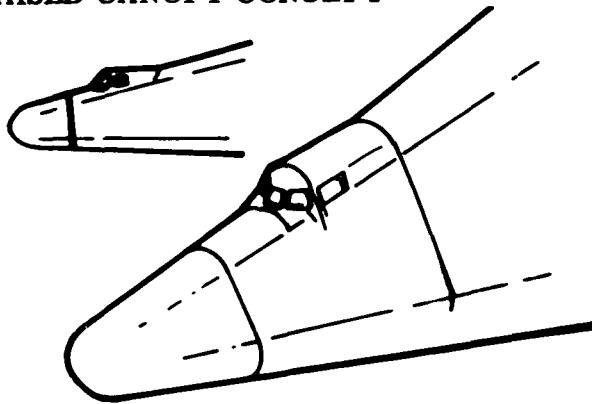
DROOP-NOSE CONCEPT
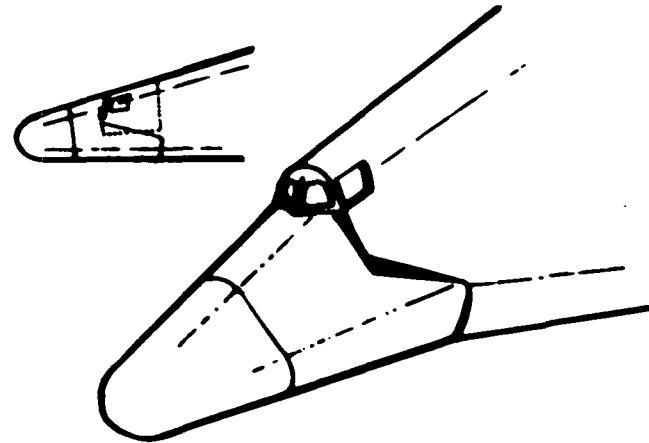
Fig. 11-9 Alternate Window Configurations for the Orbiter

Indirect viewing presents unique problems with respect to location of externally deployed equipment. These include equipment fidelity (e.g., three-dimensional presentation quality and image clarity), its proximity to the pilot's eye, glare, brightness/contrast, pilot acceptance, field of view, practicality, development index, and reliability. Typical of the instrument candidates are projection displays, cathode ray tubes, fiber optics, virtual image and real image presentations, or combinations. Table 11-2 indicates out-the-window versus optical/sensor viewing for each mission phase and the applicability of either concept.

Table 11-2

CREW VISION CONSIDERATIONS

Mission Phase / Vision	Launch and Ascent	Ren-dezvous	Docking	Cargo Trans-fer	Reentry	Landing* Powered	Landing* Unpowered	Landing* Go-around	Ferry* Flight
Out of window	Not reqd	Useful during terminal stage / Heads-up display useful	Reqd to assure proper alignment	Desirable to view station and cargo	Not reqd	Maximum forward viewing per MIL STD 850A / Heads-up display very useful	Maximum forward viewing ± from horizon / Heads-up display very useful	Forward and side viewing per conventional aircraft / Heads-up display very useful	Forward and side viewing per conventional aircraft
Optical/ sensor	Not reqd	Useful during terminal stage	1-to-1 scale reqd	Multi-camera reqd	Not reqd	Forward view 1-to-1 scale	Large field of view reqd	Side view presentation difficult	Not conventional technique Used in bad weather

*Includes landing area search, detection, approach and landing

In nearly every case except bad weather landing conditions, the desired viewing technique is to provide the pilot and copilot with out-the-window direct viewing comparable to or exceeding that from high-performance aircraft. This feature becomes particularly important in terminal rendezvous and docking operations, where critical eye/hand controller coordination may be required for a vehicle with the mass and potential velocity of the Space Shuttle. In the event that forward only docking is not practical, an additional window, as shown in Fig. 11-10, or supplemental docking pilot station will probably be required for vehicle translation in a vertical direction. If a supplemental station is required, it might be located in the airlock compartment and integrated with the payload cargo management control console (Fig. 11-7). Rendezvous and docking is compounded by the variability of external illumination interactions encountered. LMSC is using its extensive simulation capability to examine this problem, drawing on the experience gained in the recently completed NASA studies on this subject relative to Apollo flights 10 and 11.

11.5 CONTROL AND DISPLAY CONCEPTS

The basic flight program for the booster and the orbiter is planned for automatic control, with the crew providing dynamic monitoring of the flight display management system with takeover capability provided at any point in the mission profile. This requirement strongly suggests an integrated display management and control system in addition to minimal basic flight instruments. The basic underlying philosophy of this system is to provide the pilot/copilot with flight and vehicle display information necessary only for that specific flight phase. A candidate approach is to present this information to him, mission phase by mission phase, through a dynamic display system that is totally programmable and controllable through pilot input command to the computer complex. Thus, during any mission phase, the pilot/copilot can call up desired information and position it on various of his display devices. Information would be composed of two primary modes — static and dynamic. Static information would be generally categorized as reference data, while dynamic information would be categorized as status, trend, predictive, and tolerance data.
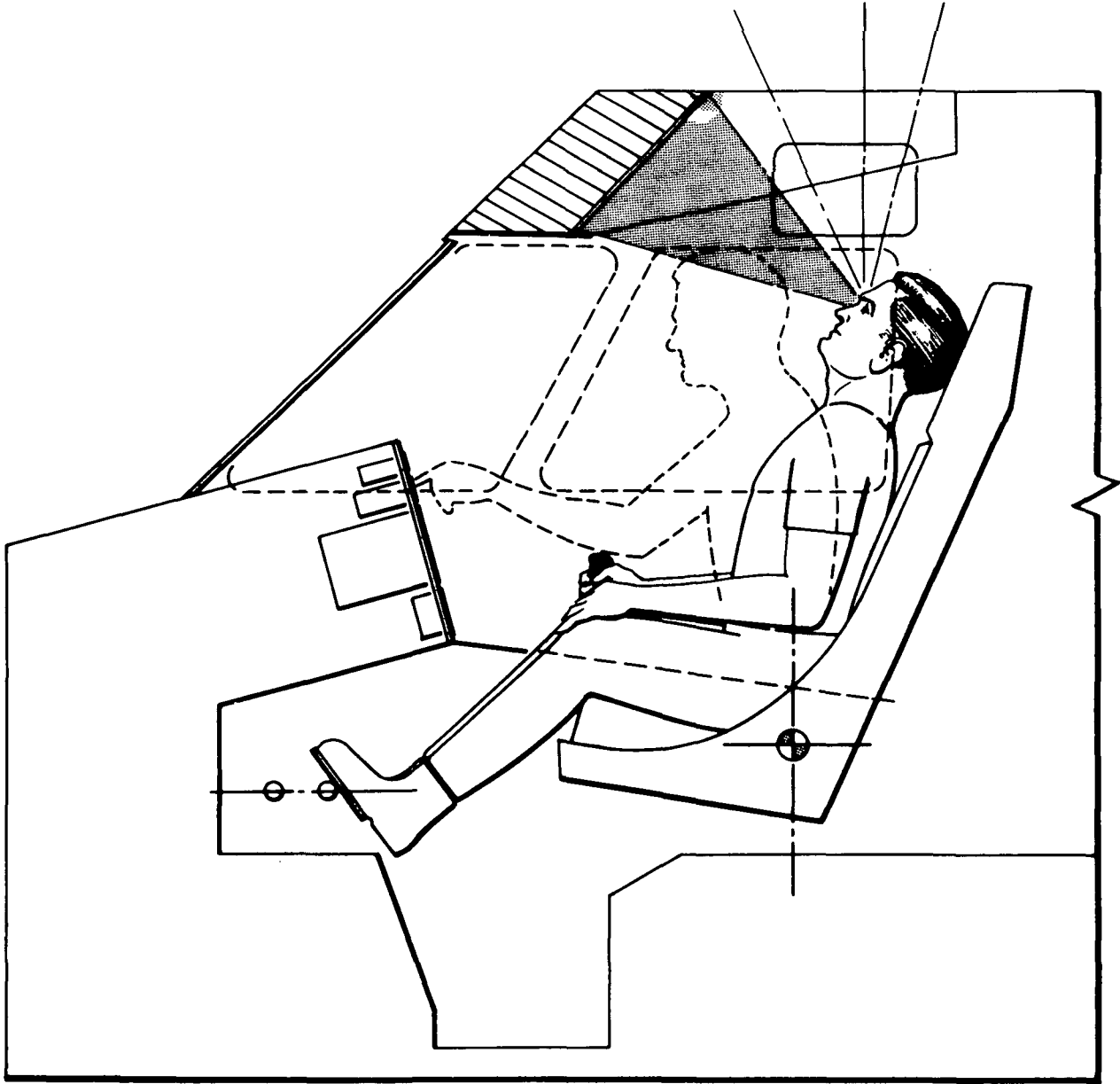
Fig. 11-10 Supplemental Docking Window Concept

The computer system would automatically configure each display element for the next sequence, thus reducing crew task time considerably. Inherent in this management display system is a built-in computer program, which commands the display system to return to the automatically programmed configuration as the flight progresses. This is necessary since the pilot can call up desired information, thereby changing his displays to a highly nonroutine configuration. Thus, the pilot cannot become "lost" in the display network and configuration. A simplified control command is provided to permit him to reset the display system to the automatic mode, thus reestablishing the display configuration to that particular mission mode and phase.

The pilot and copilot flight station was developed on the basis of the NASA-desired system characteristic of a two-man pilotage crew and operable by one crewman. This characteristic suggests conventional side-by-side, pilot/copilot arrangement with a shared center display/control panel. Both crewmen will have a similar set of display/control instruments directly in front of their operating positions for vehicle and flight management control. An important NASA-derived system characteristic is the goal to eliminate toggle switches and electro-mechanical gages and meters and replacing these with electronic displays. Thus, projection and CRT devices have been selected as the primary candidate presentation techniques for the basic display management system.

Dynamic flight data will be presented on the four to six display devices in front of the pilot and copilot and displayed for each mission phase. In addition, a minimal basic set of flight instruments will also be provided to augment the computer-driven flight management display/control system. This displayed information would aid him in making instantaneous decisions regarding takeover from the automatic system and enable him to take immediate action. Once he has taken command, the pilot must have all the information necessary for successful flight operations displayed directly ahead of him so that it is clearly recognizable while he is looking at the real world through his windshield (Fig. 11-2). In order to provide this capability, consideration must be given to a heads-up display system as a candidate method of displaying pertinent information. The display of critical flight management information is superimposed upon the external (real world) view as seen by the pilot through the windshield.

The heads-up display is focused at infinity with sufficient brightness to provide the contrast for rapid interpretation without detracting from the pilot's external view. Typical of the information presented would include attitude, altitude, heading, speed, localizer, glide-path, and flare-out. This information would be color coded and keyed to the pilot/copilot flight management panel.

In support of the basic flight management system would be a static navigation display, which is highly programmable according to pilot input command. This would be located in the center island console near the airbreathing engine controls. The center panel area shared by pilot and copilot would contain at least two projection or CRT displays, which would present system status information. The information would provide status, tolerance, trend, and predictive capability relative to each major onboard system. Data would be alarm displayed only if a system were out of tolerance to a degree that necessitated pilot/copilot attention during the flight phase. The display would also list specific or potential problems, alternative system paths, redundant networks, or recommended actions. Launch and orbital vehicle control capability should be provided to each crewman through input command to the computer. Standard input controls should be provided for normal functions such as internal illumination, environmental control, communication, timers, sensors, circuit breakers, landing gear, and a variety of other standard flight crew control tasks. The panels, consoles, and display/control devices are currently undergoing further definition and design refinement and will be integrated into a full-scale mockup.

11.6 HABITABILITY

Habitability refers to qualities of the operational environment that contribute to man's well-being in terms of sustained optimum performance. A primary factor in providing for crew habitability needs and a major determinant of crew compartment configuration is the volume required to conduct crew activities. Based on comprehensive analyses of volume allocations on current operational space vehicles, simulators, and underwater vehicles, the following free volume* requirements per man have been identified:

Mission Duration (Days)	Acceptable Range Cubic Ft/Man	Unacceptable Range Cubic Ft/Man
0.5	26-45	20-24
1	29-75	20-26
7	80-148	35-48
10	105-160	40-64
30	135-185	68-90

These volumetric guide lines were observed in developing a Space Shuttle crew compartment, sized to accommodate two crew members for a 7-day mission capability. In addition, the following contributors to adequate habitability were considered in evolving the crew compartment layouts depicted in Figs. 11-1 through 11-5 and Fig. 11-7:

- Ease of crew movement and traffic patterns
- Seat and hatch ingress/egress
- Exercise provisions
- Adequacy of sleeping provisions
- Eating provisions
- Waste management (personal hygiene techniques)
- Integration of living and work space envelopes
- Adequacy of storage provisions
- Privacy arrangements

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*Based on the assumption that free volume is 60 percent of total compartment volume.

- Noise
- Illumination
- Temperature/humidity/airflow/atmosphere
- Decor factors

11.7 CREW HAZARDS ANALYSIS

Crew safety considerations are of paramount importance in the development of a Space Shuttle concept. The approach followed to date includes an examination of each mission phase for both space and atmospheric flight regimes. The effort was initiated with a preliminary mission hazards analysis by mission phase, with a contiguous analysis of operational events also identified by mission phase. Figure 11-11 illustrates a simplified example of the hazards analysis related to mission phases. Included as a second step in this process is the examination of potential hazards pinpointed to each area in which the crew or passengers would or might perform functions. These analyses are currently underway and are being further expanded in depth and refinement. Key crew safety issues became immediately apparent for the following mission elements:

- Preliftoff crew/passenger abort
- Pad abort of crew/passengers with near or actual catastrophic event
- Launch ascent mission abort
- Reentry mission abort
- Atmospheric mission abort (landing or takeoff)

Current analysis suggests that both booster and orbiter crews be provided with at least seat eject (preferably capsule) abort capability for the R&D vehicles. It would also seem reasonable to recommend crew eject capability for the operational booster vehicle, in light of the minimum weight/volume penalty.

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CAUSE AND EFFECT	SYSTEMS AND DESIGN CONSIDERATIONS									MISSION PHASES				
HAZARD DESCRIPTION.	POTENTIAL FAILURES	DEGREE OF CRITICALITY	GENERAL CREW REACTION TIME	MISSION IMPLICATION	REMEDIAL ACTION	CREW HAZARD	HAZARD DETECTION	ESCAPE TECHNIQUE	RESCUE TECHNIQUE	PRELAUNCH	LAUNCH AND ASCENT	ORBIT	REENTRY	LANDING
PRESSURE LOSS														
ATMOSPHERE CONTAMINATION														
TEMP/HUMIDITY OUT OF LIMITS														
ELECTRICAL POWER LOSS														
FLIGHT CONTROL LOSS														
ORBIT DECAY														
RADIATION														
FOOD/WATER CONTAMINATION/LOSS														
BUILDUP OF DANGEROUS BACTERIA														
CREW INJURY														
FIRE														
● ELECTRICAL														
● CHEMICAL														
● OTHER														
EXPLOSION														
● LIQUID														
● GAS														

FLIGHT STATION COMPARTMENT

CREW ESCAPE CONDITIONS / ESCAPE TECHNIQUE CANDIDATES	ABANDON COMPART.	RELOCATE IN CARGO/ PAYLOAD AREA	RELOCATE IN TUNNEL	DON SPACE SUIT	DON SPACE SUIT AND TRANSLATE EVA	ENTER ESCAPE CAPSULE	ARM AND EJECT SEAT	EGRESS TO PAD ESCAPE DEVICE	ETC.	MISSION PHASES
FIRE										
● ELECTRICAL										
● CHEMICAL										

Fig. 11-11 Crew Hazards Analysis

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Section 12
ENVIRONMENTAL CONTROL AND LIFE SUPPORT

The environmental-control and life-support system (EC/LSS) provides the crew with an environment that ensures their safety and functional capability throughout the mission. The system also provides total internal thermal control, passive and active, for vehicle systems, compartments, and components. The orbiter EC/LSS concepts presented herein are compatible with the Space Station atmosphere even if it is increased from its present 10 psia to 14.7 psia. The primary impact will be to increase gas quantities. The booster EC/LSS is similar to that of the orbiter with exceptions resulting from shorter operations.

Figures 12-1 and 12-2 are simplified schematics for the orbiter and booster EC/LSS. Tables 12-1 through 12-5 show vehicle design parameters, metabolic and cabin data, power requirements, thermal loads, and weights. The EC/LSS will occupy approximately 24 cubic feet in the orbiter and 9 cubic feet in the booster. These volumes include expendables.

Two modes of operation are provided. The primary mode occurs with the crew in shirtsleeves with the gas circulating loop open to the cabin. An alternate, closed-loop suited mode is possible for use during ascent, descent, and a limited number of mission tasks.

12.1 ATMOSPHERE SUPPLY AND CONTROL

The system for the orbiter supplies a 10 psia nitrogen-oxygen atmosphere with an oxygen partial pressure of 2.7 psia. Each gas is stored in super-critical cryogenic tanks, and oxygen is shared with the electrical power subsystem. The gases are heated by the cryogenic heat exchangers prior to admittance to the cabin. A gaseous supply, stored in accumulators from tank venting is maintained for emergency or high-usage rates. Tank pressures are reduced by regulators. Oxygen partial pressure control is identical for either suited or shirtsleeve operation. Oxygen partial pressure
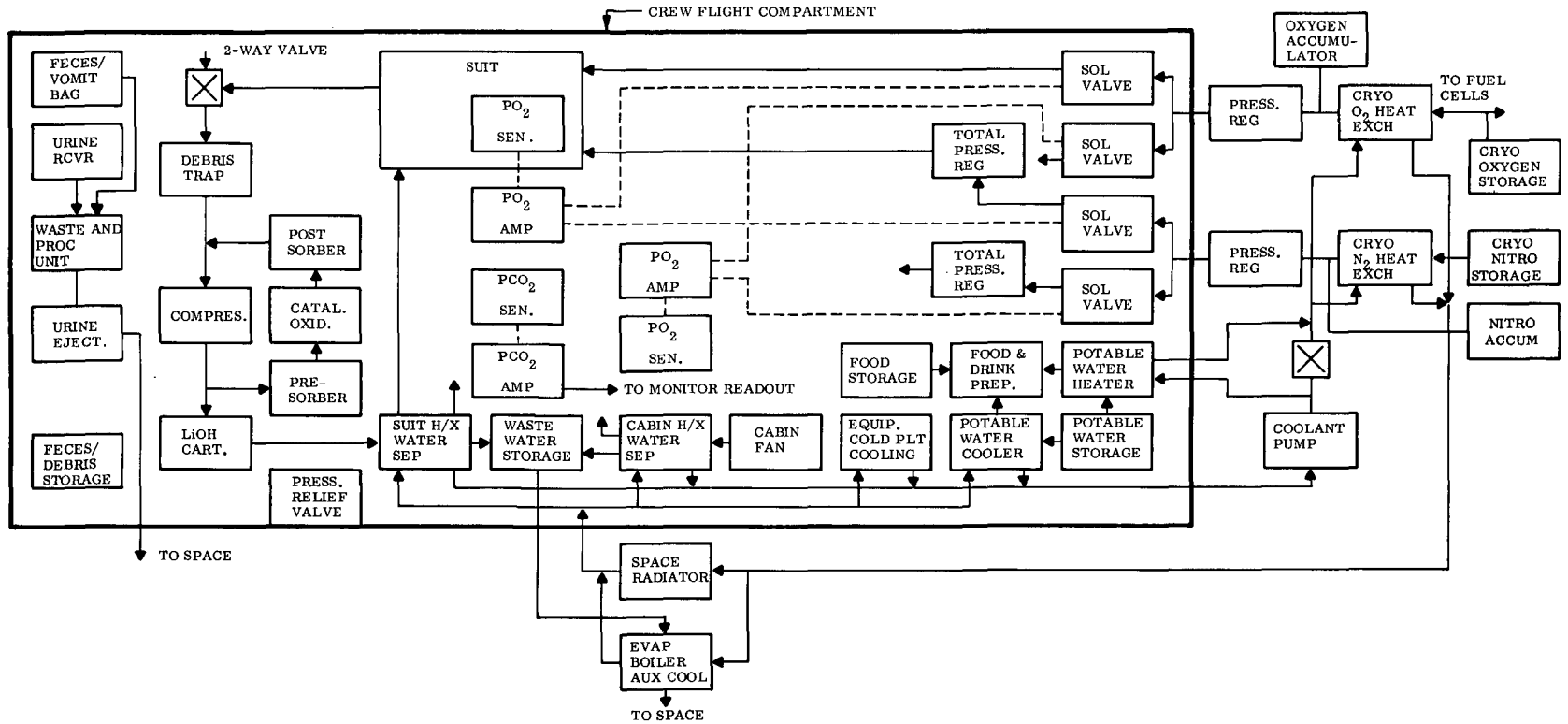
Fig. 12-1 Orbiter Environmental Control and Life Support System — Schematic

CREW FLIGHT COMPARTMENT

2-WAY VALVE

SUIT

VOMIT BAG

PO₂ SEN.

SOL VALVE

PRESS. REG

HI-PRESS. OXYGEN STORAGE

DEBRIS TRAP

PO₂ AMP

SOL VALVE

URINE RCVR

PO₂ AMP

TOTAL PRESS. REG

URINE PROC UNIT

COMPRESS.

PO₂ SEN.

SOL VALVE

PRESS. REG

HI-PRESS. NITROGEN STORAGE

PCO₂ SEN.

TOTAL PRESS. REG

URINE EJECT.

PCO₂ AMP

TO MONITOR READOUT

SOL VALVE

DRINK STATION

LiOH CART.

SUIT H/X WATER SEP

WASTE WATER STORAGE

CABIN H/X WATER SEP

CABIN FAN

EQUIP COLD PLT COOLING

POTABLE WATER STORAGE

POTABLE WATER STORAGE

COOLANT PUMP

VOMITUS DEBRIS STORAGE

PRESS. RELIEF VALVE

TO SPACE

FREON OR AMMONIA STORAGE

EVAP BOILER COOL

TO SPACE

Fig. 12-2 Booster Environmental-Control and Life Support System — Schematic

Table 12-1

BOOSTER AND ORBITER DESIGN PARAMETERS

	Orbiter	Booster
● Mission duration	7 days	3.5 hours
● Crew size	2	2
● Cabin volume (nominal)	400 ft^3	200 ft^3
● EVA required	None scheduled	None
● Number of repressurizations	2	None
● Operational modes	Suited and shirtsleeve	Suited and shirtsleeve
● Payload bay ECS	Integral with payload	—

Table 12-2

BOOSTER AND ORBITER
METABOLIC AND CABIN DATA

	Orbiter	Booster
● Cabin atmosphere constituents	O_2, N_2	O_2, N_2
● Cabin total pressure	10 psia	14.7 psia
● Oxygen partial pressure	2.7 psia	3.1 psia
● Nitrogen partial pressure	Remainder of atmosphere	Remainder of atmosphere
● Carbon dioxide partial pressure limit (nom.) (emergency)	5 mm Hg 10 mm Hg	5 mm Hg 10 mm Hg
● Cabin temperature	65 – 75°F	65 – 75°F
● Cabin relative humidity	35 – 65%	35 – 65%
● Cabin leakage rate	2 lb/day	2 lb/day
● Oxygen consumption (metabolic)	1.84 lb/man day	1.84 lb/man day
● Carbon dioxide generation	2.12 lb/man day	2.12 lb/man day
● Latent heat (metabolic)	3340 Btu/man day	3340 Btu/man day
● Sensible heat (metabolic)	7860 Btu/man day	7860 Btu/man day
● Dry food consumed	1.4 lb/man day	⚠1
● Water of oxidation	0.8 lb/man day	0.8 lb/man day
● Water consumed	5.8 lb/man day	5.8 lb/man day
● Evaporative water	3.3 lb/man day	3.3 lb/man day
● Urine output	3.2 lb/man day	3.2 lb/man day
● Fecal output	0.35 lb/man day	⚠1

⚠1 Because of short mission duration, no food or fecal management is provided.

Table 12-3

ECS/LSS POWER REQUIREMENTS (WATTS)

	Prelaunch	Launch Ascent	Orbit	Rendezvous and Docking	Deorbit	Reentry	Landing
Orbiter							
Suit compressors	100/200	100/200	100/200	100/200	100/200		100/200
Cabin blower	80	80	80	80	80	80	80
Cabin vent fans	80	80	80	80	80		80
Coolant pumps	125/250	125/250	125/250	125/250	125/250	125/250	125/250
Water and waste management	△1	–	△1	–	△1	–	△1
PO$_2$ monitor and control	50	50	50	50	50	50	–
Heaters	–	–	△2	△2	△2	–	–
Total avg/peak	435/860	435/660	435/1360	435/1160	435/1360	255/380	385/810
Booster							
Suit compressors	100/200	100/200	–	–	–	–	100/200
Cabin blower	80	80	–	–	–	80	80
Cabin vent fans	80	80	–	–	–	–	80
Coolant pumps	125/250	125/250	–	–	–	125/250	125/250
Total avg/peak	385/610	385/610	–	–	–	205/330	385/610

△1 Intermittent operation (50% duty cycle at 200 watts)

△2 Intermittent operation (50% duty cycle at 500 watts)

3 Conditioning requirements – none; power may be supplied at nominal voltages.

Table 12-4

BOOSTER AND ORBITER THERMAL LOADS (BTU/HR)

	Orbiter						Booster			
	Prelaunch	Ascent	Orbit (Max)	Orbit (Min)	Entry	Landing	Prelaunch	Ascent	Entry	Landing
External cabin heat load	400	600	-400	-1000	*	*	400	600	*	*
Metabolic heat load (2 man totals)										
● Sensible	500	500	550	440	*	*	500	500	*	*
● Latent	600	600	900	220	*	*	600	600	*	*
Cabin electronic heat loads										
● Air cooled	1700	1500	2200	1050	1200	2000	1200	1000	800	1500
● Cold plate	2800	2500	4100	1300	1500	3900	2100	1000	1000	2500
Fuel cell heat loads	7800	8100	9200	3240	8200	10100	–	–	–	–
Battery heat loads	–	–	–	–	–	–	1700	1400	1800	2100

*Transient condition

Table 12-5

BOOSTER AND ORBITER ENVIRONMENTAL CONTROL SYSTEM WEIGHTS

	Weights (lb)	
	Orbiter	Booster
Thermal/humidity control and suit circuit	100	80
Heat transport (including radiators on orbiter)	500	95
Water and food management and supply	40	20
Atmosphere supply	150	30
Waste management and personal hygiene	35	20
Instrumentation, miscellaneous hardware, fluids in system	100	100
Fixed weight	925	345
Lithium hydroxide/charcoal	45	5
Consumable gases	120	2
Food	20	–
Ammonia	–	50
Water	–	10
Miscellaneous	50	20
Total expendables and miscellaneous	235	87
Total system weight	1160	432

is sensed by a polarographic sensor. The sensor signal is interpreted by an amplifier, which actuates a solenoid valve to admit oxygen as needed. At the same time, the amplifier closes the nitrogen solenoid valve. Total pressure is maintained with nitrogen by an absolute pressure regulator. For the suited mode, the total pressure is maintained slightly positive with respect to cabin pressure.

Operation of the booster control system is identical to that for the orbiter except that total pressure will be 14.7 psia, with an oxygen partial pressure of 3.1 psia. Gases will be stored in high-pressure tanks and used exclusively for environmental control, since fuel cells are not used on the booster.

Use of an emergency breathing face mask system in lieu of the suit loop operation during hazardous conditions appears to be possible. This method, in addition to an inert gas cabin purge system to eliminate contaminants or to reduce cabin oxygen partial pressure to extinguish a fire, merits further investigation.

12.2 THERMAL AND HUMIDITY CONTROL

Heat generated by onboard systems, power sources, and crew members is rejected passively to atmosphere and actively by heat exchangers and cold plates to a circulating coolant. The coolant transports the heat to a radiator system, which rejects it to space. Supplemental cooling for peak loads is provided by boiling excess fuel cell water. Humidity control is maintained by condensing the latent heat load generated by the crew in a condensing heat exchanger/water separator. The condensed water is stored either in the potable-water tank or in the waste water tank to supply the boilers used for auxiliary cooling.

Thermal and humidity control in the booster is identical to that for the orbiter except that the radiator has been eliminated; heat is rejected from the coolant loop by evaporating ammonia.

Further investigation of the feasibility of utilizing the heat sink capacity of onboard cryogenics for cooling is recommended. This approach offers the most promise during launch, ascent, and reentry phases of the mission.

12.3 CARBON DIOXIDE AND TRACE CONTAMINANT REMOVAL

Carbon dioxide and trace contaminants are added to the atmosphere by the crew, equipment, paints, volatiles, and other sources. The atmosphere, circulated by compressors, is passed through a lithium hydroxide/charcoal bed for carbon dioxide and odor removal and through a catalytic oxidizer to limit minor contaminants as well as major contaminants, such as methane, carbon monoxide, and hydrogen.

The booster removal system is identical to that for the orbiter with the exception that the catalytic oxidizer has been eliminated.

The orbiter catalytic oxidizer may be eliminated if vehicle leakage rates are sufficient to prevent the trace contaminants from reaching undesirable concentration levels. In the event the basic vehicle mission duration changes to exceed 15 days, a more advanced concept, such as molecular sieve carbon dioxide removal, could be considered. Chemical systems that also produce oxygen should be investigated if the shuttle is to be deactivated on orbit for long periods.

12.4 FOOD AND WATER

Food will be freeze dried and wrapped in individual servings. Reconstitution from the water system will be provided at the preparation station. Water is supplied from fuel cell output, which produces a quantity in excess of metabolic requirements. Heating or cooling of the water is accomplished by the coolant fluid. An electrical heater supplements the hot coolant for water heating.

No food will be provided on the booster; only cold water is supplied.

A more nearly normal food supply on the orbiter is desirable. Certain selected types of food for this purpose could be identified and added to supplement the freeze dried supply.

12.5 WASTE MANAGEMENT, PERSONAL HYGIENE, MEDICAL SUPPLIES

Urine and feces are collected at a waste management station in the orbiter. Urine is phase separated from atmospheric gas and dumped overboard through a heated dump valve. Feces is dried and stored in a container. Personal hygiene items to accommodate bathing, nail clipping, hair trimming, and shaving will be stored at the management station, together with medical supplies.

Only urine collection and dumping waste management capability is provided in the booster. No personal hygiene items are supplied and medical supplies are limited to first-aid equipment.

Methods that eliminate all personal contact or manual handling by the crew of the waste material should be investigated. Present concepts, involving slingers and similar techniques, are too complex for use on short duration missions, but simplified systems employing similar techniques should be investigated.

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Section 13

RELIABILITY AND MAINTAINABILITY

13.1 MISSION SUCCESS DESIGN CRITERIA

The safety of the crew and passengers and the safety of the equipment are related, so all subsystems will be designed to be fail operational-fail safe. Electronic subsystems will be designed to be fail operational-fail operational-fail safe. There will be no mission-critical single point of failure. All noncritical systems will be fail safe.

The following terms relate to the mission success criteria:

- Failure – the inability of a system, subsystem, component, or part to perform its required function

- Critical component – a required functional element of a subsystem essential for mission success

- Critical failure – any hardware failure that results in loss of life or loss of mission (when no backup capability exists)

- Fail operational capability – no degradation of a mission critical function subsequent to a hardware failure

- Fail-safe capability – no jeopardy of human safety subsequent to a hardware failure and safe return of crew and payload, i.e., intact abort

- Abort – premature termination of a mission because of existing or imminent degradation of mission success, accompanied by the decision to make safe return of the crew and payload the primary objective

Fail-operational or fail-safe capability can be enhanced by using the following techniques:

- Redundancy, active or standby, with automatic or manual fault detection, fault isolation, and switchover capability

- Alternate modes of accomplishing the primary task

13.2 RELIABILITY MODELS

Traditional reliability techniques were used in an investigation of the reliability of the candidate systems. Reliability models were developed in an informal manner in order to be responsive to the constantly changing design concepts. One completed example is shown in Figs. 13-1 and 13-2. Failure rates were based on discussions with potential vendors and current LMSC experience with the Agena. The failure rates expected by 1972 were estimated by extrapolating at a 20 percent per year improvement rate. The use of the mission success design criteria in a number of typical numerical calculations indicates that a high order of mission reliability ($P_s \geq 0.995$) for the combined vehicle is obtainable; however, any relaxation of these criteria will have an adverse effect on the potential mission reliability.

13.3 MAINTAINABILITY (M) CONSIDERATIONS

The qualitative and quantitative maintainability design characteristics of the Space Shuttle will be such that the planned mission can be accomplished with a minimum expenditure of maintenance manhours, maintenance ground equipment, and technical data to refurbish the orbiter, booster, and launch support equipment within allowable ground turnaround time.

Equipment-level maintainability quantitative parameters will be established by allocating the required overall mean time to refurbish down to the assembly, or black-box level, which will provide the maintainability engineer with a replace or repair time constraint on which to base the requirements for design. A similar allocation of time will be made from specified-allowable hardware-repair-maintenance hours per flight or flight hour. Although initial requirements specify that no maintenance is to be performed on board, a trade study will be performed to determine the most effective restorative method for failed prime equipment of a redundant configuration in the orbiter. Cost of such features as self-test, automatic fault isolation, automatic fault detection, and triple redundancy versus maintenance-on-board are principal considerations.

Fig. 13-1 Reliability Block Diagram — Orbiter Electrical Power System
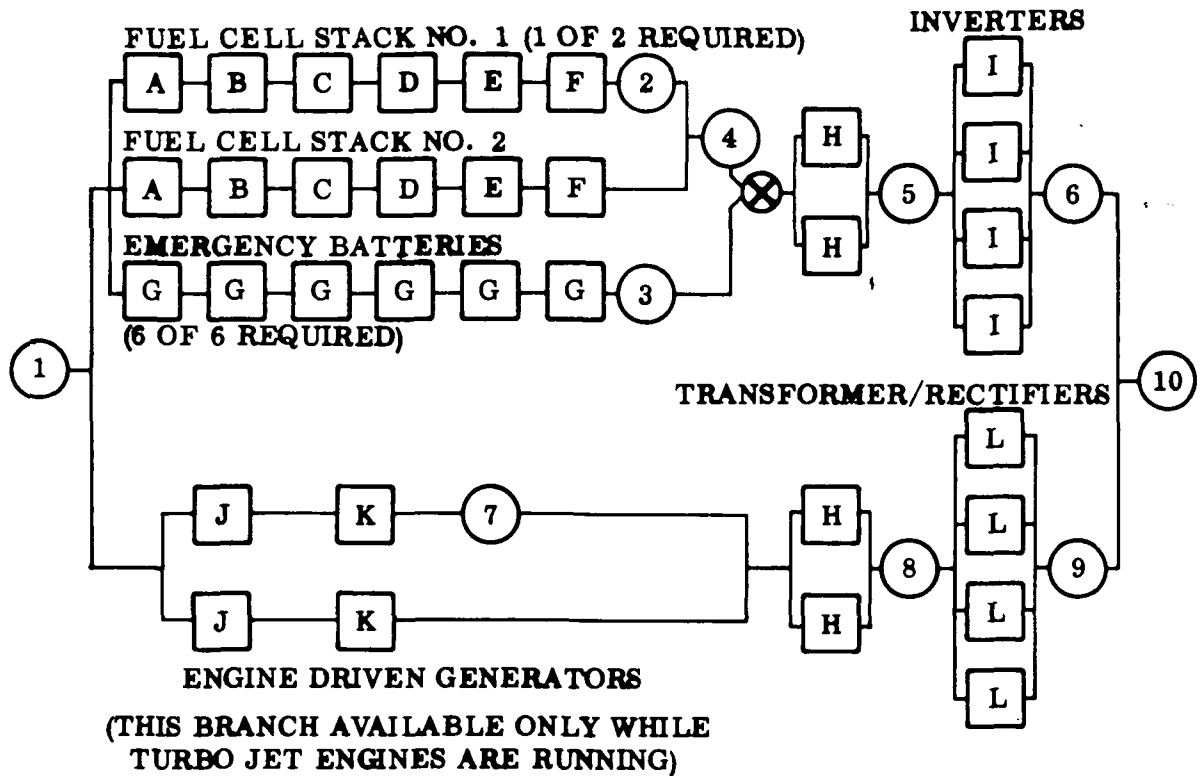
Block identification	Equipment	No. Required for Success	1972 Est/ Failure Rate ($\times 10^{-6}$)	Total Operating Time (hr)
A	Fuel cell module		28	
B	Reactant control assy		35	
C	Product water assy		5	
D	Thermal control assy		13	
E	System control assy		3	
F	Overload and reverse current protection		5	
	Fuel cell power system (ea)	1 of 2	89	168
G	Emergency batteries (ea)	6 of 6*	24	2.5
H	Buses and bus protection and switching Unit (ea)	1 of 2	25	168
I	400-cycle inverter (ea)	2 of 4	15	168
J	Engine-driven generator (ea)	1 of 2	18	0.7
K	Generator protection and control assy (ea)	1 of 2	5	0.7
L	Transformer/rectifier (ea)	2 of 4	9	0.7

*For emergency only — mission automatically aborted.

Reliability Equations — Main Power System	Estimated Reliability
$R_{(1-2)} = R_A \times R_B \times R_C \times R_D \times R_E \times R_F =$	0.985048
$R_{(1-4)} = 2R_{(1-2)} - \left[R_{(1-2)}\right]^2$	0.999777
$R_{(4-5)} = 2R_H - R_H^2$	0.999983
$R_{(5-6)} = R_I^4 + 4 R_I^3 (1-R_I) + 6 R_I^2 (1 - R_I)^2$	0.999999
$R_{(1-6)} = R_{(1-4)} \times R_{(4-5)} \times R_{(5-6)}$	0.999759
$*R_{(1-3)} = R_G^6$	0.998272
$R_{(1-10)} \cong R_{(1-6)}$	0.999759**

Reliability Equations — Auxiliary Power System (only available when turbo jet engines are operating)

	Estimated Reliability
$R_{(1-7)} = R_J \times R_K$	0.999984
$R_{(1-8)} = 2R_{(1-7)} - \left[R_{(1-7)}\right]^2 \times \left[2R_H - (R_H)\right]^2$	0.999999
$R_{(8-9)} = R_L^4 + 4 R_L^3 (1 - R_L) + 6 R_L^2 (1 - R_L)^2$	0.999999
$R_{(1-9)} = R_{(1-8)} \times R_{(8-9)}$	0.999998

*For emergency only — mission automatically aborted.
**The preliminary allocation of 0.999650 is exceeded.

Fig. 13-2 Reliability Mathematical Model — Orbiter Electrical Power System

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When maintainability quantitative values have been assigned to all functional groups of the system and the design features are identified to assure achievement of the allocated values, end-item maintenance sheets will be prepared to define the detail maintenance and support requirements. The data contained on these sheets, which will provide a basis for developing the required maintenance program, will identify all personnel skills, technical data, and maintenance ground equipment required to support the maintenance and refurbishment program at each launch site.

Corrective maintenance occurrences are a function of equipment failure rate; and the time to correct a malfunction is a function of design complexity. The apportioned electrical power system mean corrective maintenance time ($\overline{M}_{ct}$) at the functional group level is therefore a function of the combined effect of the failure rate of a group, the relative complexity of that group and the other 12 electrical power system functional groups.

The following equations will be applied to allocate the overall maintainability parameter down to functional groups of the system.

$$\overline{M}_{ct_i} = \overline{M}_{ct} \cdot k_i = \text{allocated mean corrective maintenace time} \qquad (1)$$
$$\text{for the ith functional group of EPS}$$

where

$\overline{M}_{ct}$ = the specified mean corrective maintenance time for EPS

and

$$k_i = \frac{X_i}{\dfrac{\Sigma X_i}{n}} = \frac{X_i(n)}{\Sigma X_i} \quad \text{Complexity factor for ith functional group} \qquad (2)$$

X_i = number of elements making up an ith functional group of the EPS i.e., assemblies, components, or parts. X_i for all groups of the EPS must be at the same level.

n = total number of functional groups making up the EPS, i.e., EPS, n = 13.

To illustrate the method, it is assumed that the mean corrective maintenance time for the reactant supply is 5.5 hr. This assumption is based on best maintainability engineering judgment.

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By substituting, $\overline{M}_{ct}$ for EPS can be calculated as follows:

$$\overline{M}_{ct} = \frac{\overline{M}_{ct_2}}{\frac{X_2(n)}{\Sigma X_i}} = \frac{5.5}{\frac{(24)\ 13}{138}} = 2.43\ hr$$

All $\overline{M}_{ct_i}$ can be allocated from the foregoing $\overline{M}_{ct}$ of 2.43 hr.

For functional group I of the electrical power system,

$$\overline{M}_{ct_i} = \overline{M}_{ct}\ \frac{X_1(n)}{\Sigma X_i} = 2.43\ \left(\frac{18\ (13)}{138}\right) = 4.1\ hr.$$

The following table provides an example of the allocated maintainability goals to all functional groups of the electrical power system of the orbiter. A similar set of goals will be allocated to items of other subsystems.

MAINTAINABILITY (M) GOALS

i Col. (1)	EPS Functional Groups Col. (2)	X_i Col. (3)	$\overline{M}^*_{ct_i}$ Col. (4)	λ Unreliability x 10^{-6} Col. (5)
n = 13	Electric power system	138	2.43 est	350**
1	Primary power source (2)	18	4.1	84
2	Reactant supply (pri)	24	5.5	23
3	Reactant supply (aux)	24	5.5	23
4	Overvoltage protection and control (pri)	4	0.9	5
5	Overvoltage protection and control (aux)	4	0.9	5
6	Inverters (2)	3	0.7	15
7	Auxiliary power supply	17	4.1	18

*Allocated mean corrective maintenance time
**The preliminary allocation of .999650 (350 x 10^{-6} unreliability) is used for this exercise. See Fig. 13-2

MAINTAINABILITY (M) GOALS (Cont.)

i Col.①	EPS Functional Groups Col.②	X_i Col. ③	$\overline{M}*_{ct_i}$ Col. ④	λ Unreliability x 10^{-6} Col. ⑤
8	Power switching int/ext (pri)	6	1.4	5
9	Power switching int/ext (aux)	6	1.4	5
10	Interlock system	4	0.9	5
11	Emergency battery power	9	2.1	144
12	Rectifiers (2)	4	0.9	9
13	Electrical power system control	4	2.3	10

*Allocated mean corrective maintenance time

The estimated $\overline{M}_{ct}$ of 2.43 hr established the allocations but must be replaced by the computed arithmetic mean in the equation

$$\overline{M}_{ct} = \frac{\Sigma M_{ct_i} \cdot f_{c_i}}{\Sigma f_c} \tag{3}$$

where

$\overline{M}_{ct_i}$ = value obtained from Column ④

f_{c_i} = λN per ith group, total failures x 10^{-6}

N = quantity of specific group indicated in Column ②

Hence

$$\overline{M}_{ct} = \frac{1366.4}{459} = 2.98 \text{ hr}$$

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Other maintainability goals for the orbiter electrical power system are as follows:

- A probability of 0.95 that corrective maintenance action will be completed within 9.4 hr m_{max}

  M_{max} is normally specified in all maintainability contracts and is the antilog $[\text{Log } \overline{M}_{ct} + 0.5]$

- That the mean refurbishment time for the electrical power system will be approximately equal to $\overline{M}_{ct}$ of 3.14 hr

A maintainability engineering program will be conducted in accordance with an approved program plan. Maintainability allocation and prediction techniques currently approved and in use will be applied to this program.

Used in the end-item maintenance data compilation, storage, and retrieval program currently undergoing validation at LMSC is a computer 80-column formatted end-item maintenance sheet. The use of this program for Space Shuttle would provide timely and effective maintenance and support planning prior to critical design review.

Section 14
SAFETY

14.1 FLIGHT SAFETY

Key concepts of the flight safety approach for the Space Shuttle are as follows:

- Operate a high-performance vehicle with proven techniques practiced in military and commercial aircraft test and operation.

- Utilize potential afforded by improved launch concepts, engines, and subsystems.

- Provide primary intact abort mode throughout development, initial operational, and operational phases.

- Provide crew escape backup during development test phase.

- Adopt best practices of Apollo-type critical component factory acceptance practices.

- Adopt military/commercial fail operational-fail operational fail safe concepts for vehicle subsystems, procedures, and operations.

Significant elements of a typical Space Shuttle system approach are outlined below.

    a. Engine-out intact abort from launch commit to injection

        (1) Direct return to launch site

        (2) Through orbit and return to launch site

    b. Advanced Engines

        (1) Benign idle thrust start mode similar to turbojet

        (2) Deep throttle mode with controlled runup to liftoff

        (3) Dependability based on extensive test and reuse

            − Typically more than 300,000 seconds of engine test stand operation

            − Multisuborbital and orbital demonstration multiengine flights prior to Initial Operation Phase

            − Concurrent engine qualification and vehicle flight test in the actual combined environments

c. Incremental Flight Development

    (1) Systematic penetration of normal and abort flight envelopes

    (2) All design requirements demonstrated in flight prior to IOC

Engine-out capability is provided with the following additional considerations:

- An additional 15-percent thrust increase is available as a one-time emergency rating.

- Benign aborts with thrust loss preventing through-orbit abort involve a loiter mode in throttled condition in near-vertical attitude to consume the propellants until an altitude of 50,000 feet is reached, followed by a return to base.

Phases of flight safety development associated with increasing confidence gained in the system reliability and performance are illustrated in Fig. 14-1. Throughout all three phases, intact abort is the primary abort mode for both the booster and the orbiter.
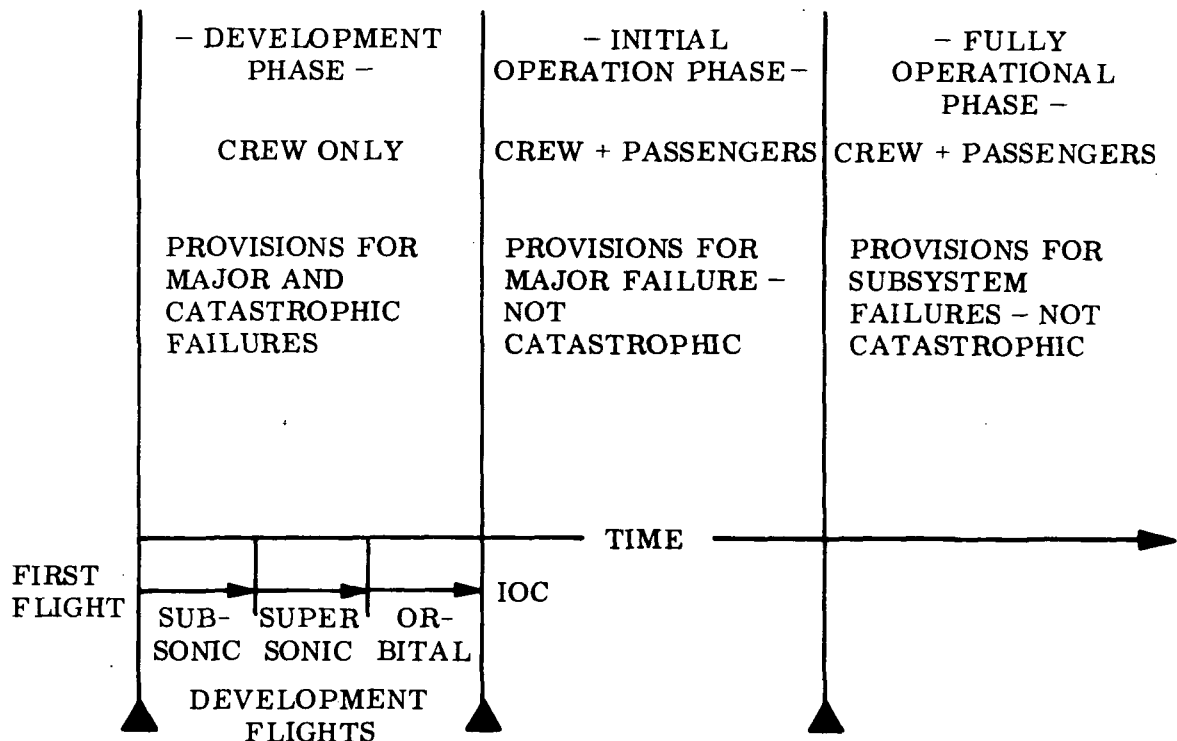
Fig. 14-1 Flight Safety Program Phases

14.1.1 Phase I — Development

During this phase, which includes all flight testing modes — subsonic, supersonic, and orbital — only the flight crew will be aboard. The basic escape provisions are designed to protect against all abort circumstances up to and including catastrophic system failure from liftoff to landing. A reaction time of 2.5 seconds and an initial acceleration of 10 g are design goals.

14.1.2 Phase II — Initial Operational

This phase covers initial orbital flights carrying operational payloads and extends to an as yet undefined time when the total vehicle system has demonstrated a safety level equivalent to standard commercial aircraft operation. The primary abort mode in this phase would be escape-to-orbit if possible. Worst-case abort mode would be return-to-earth with passenger and crew escape at low altitude if the orbiter is unable to make a safe landing.

14.1.3 Phase III — Fully Operational

This phase covers all flights after the spacecraft has demonstrated a safety level equivalent to standard commercial aircraft operation. The primary abort mode is an intact abort, however, escape to orbit or return to earth with emergency spacecraft landing or ditching is still possible. Survival equipment and personnel egress and procedures would be consistent with commercial aircraft practices.

14.2 CREW SAFETY

To assure that consideration is given to all aspects of system design and system operation, as related to system failure and the potential consequence of a hazardous condition, the Space Shuttle program will be constrained to follow an approach that accomplishes the following:

- Identifies critical functions, events, and equipment
- Provides an assessment of failure modes by failure frequency as related to abort or catastrophe
- Optimizes reliability through fail-safe methods of redundancy configuration and failure effect containment
- Implements investigations resulting in optimization of automatic fault detection/identification and correction
- Implements a hazard analysis aiding in development of a method for estimating the probability of crew survival for all phases of the mission and bases for design alternatives for equipment essential to a safe mission abort

The most significant feature of this study is the assessment of crew safety in terms of the total system. The absolute value of the numbers is useful only for measuring tradeoff alternatives.

An ingredient in the hazard analysis is a logic block diagram for each system under study, as shown in Fig. 14-2. This is a "tree" of dichotomous events of success (R = reliability) and failure (Q = unreliability) probabilities detailing the paths of operation from mission start to either mission completion or mission failure. This approach is further expanded to form a logic diagram of mission-abort capabilities that provide paths to distinguish between mission-fail-abort and mission-fail-catastrophe. An example is given in Fig. 14-3. To draw the tree, each event in each mission phase is related to the portions of the system in each phase that must not fail. However, if failure occurs, a determination is made as to whether the terminal point is "abort" or "catastrophe."

With the logic trees completed, a hazard probability apportionment is provided for guidance and predictions are subsequently made for comparison. The apportionment must include consideration of factors such as differences in system/phase operating times, equipment complexity, equipment failure criticalities, and effects of environment. Once the permissible failure probabilities have been more firmly

14-4

R_I = RELIABILITY OF IGNITION
R_L = RELIABILITY OF LAUNCH PHASE
R_1, R_2 ... R_N = RELIABILITY OF EACH PHASE
R_A = RELIABILITY OF SUCCESSFUL ABORT
Q = UNRELIABILITIES
C = CATASTROPHE
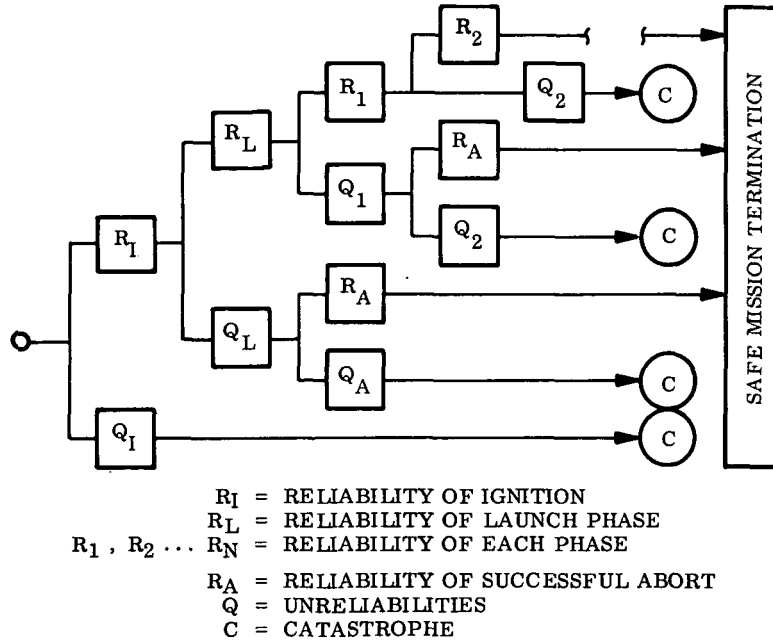
Fig. 14-2 Hazard Analysis Logic Block Diagram
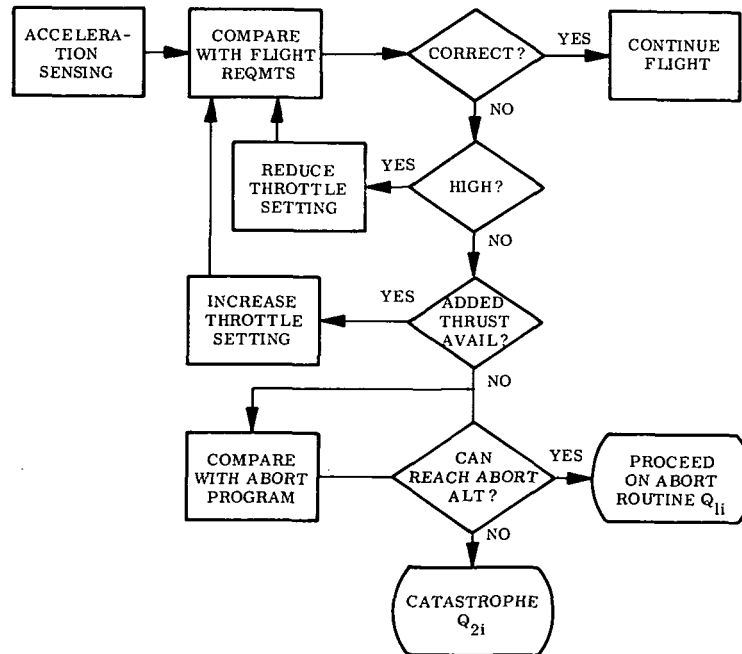
Fig. 14-3 Ascent Phase — Main Propulsion Logic Diagram (Typical)

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established and the major hazard events more completely identified, the techniques of logic diagrams and fault trees (with associated failure mode and effects analyses) can be iterated to aid in design optimization.

An apportionment is provided for the current concept of the Space Shuttle, and a probability model has been derived to define the probability of crew survival for each mission phase. The derivation of the logic sequence provides a basis for a successful abort sequence.

A decision logic/quantitative technique has been used to determine the probability of crew survival. Each phase of the mission, starting with preflight, has been considered from the standpoint of criticality of equipment operation and mission sequence mode.

The following definitions are pertinent:

- Crew survival/safe mission termination − the safe return of all personnel aboard the vehicle, either as a result of a successful completed mission or a required mission abort

- Abort − premature termination of a mission because of existing or imminent degradation of mission success, accompanied by the decision to make safe return of the crew the primary objective

- Probability models − probability of success + probability of failure = 1

$$P_i + Q_i = 1$$

$$Q_i = Q_{1i} + Q_{2i}$$

- Mission completion probability (M_i) − the unconditional probability that an abort-type failure will not occur through the completion of Phase (i)

- Probability of crew survival (P_{cs}) − the probability that the mission is not aborted or the mission is aborted at any time and the crew returns safely

$$P_{cs} = M_5 + \sum_{i=1}^{i=5} M_{i-1} \times Q_{1i} \times R_{Ai}$$

- Phase completion probability (P_i) − the probability that an abort-type failure does not occur during phase, given that the previous phase was completed without an abort-type failure

- Phase escape probability (R_{A_i}) — the probability that the vehicle escape equipment functions successfully during Phase (i), given that the failure is not one from which crew escape is impossible

- Abort-type failures — probability of failure (Q_i) consisting of the probability of failures from which a successful abort can be executed (Q_{1i}) and those that will result in an unsuccessful (not crew initiated) abort (Q_{2i})

The launch phase has been divided into three time sequences to identify different failure modes and escape probability. The probability of crew survival, P_{cs}, given below, is based on calculations from the preceeding equation. A reliability success ratio of $M_5 = 0.995$ has been assumed. Table 14-1 shows phase relationships.

Table 14-1

ALLOCATION OF PROBABILITY OF CREW SURVIVAL

Phase (i) Term	1 Pre-launch	2A Ascent 0 to 20 sec	2B Ascent 21 to 120 sec	2C Ascent to Inject	3 On-Orbit	4 Reentry	5 Approach and Landing
M_i	0.999950	0.999825	0.999700	0.999500	0.995659	0.95300	0.99500
$Q_{1i} \times 10^{-6}$	47	96	118	199	3823	297	274
$Q_{2i} \times 10^{-6}$	3	29	7	1	27	53	26
P_i	0.999950	0.999875	0.999875	0.999800	0.996150	0.999650	0.999700
R_{A_i}	0.99	0.90	0.96	0.995	0.995	0.98	0.96

Note: $Q_{2i} \times 10^{-6}$ values are from Table 14-2.

Table 14-2

PROBABILITY OF A CATASTROPHIC OCCURRENCE* (Q_{2i})

Phase	Pre-launch	0 to 20 sec	20 to 120 sec	To Inj	On-Orbit	Reentry	Approach and Landing
Hazard							
Structure breakup	–	0.4	3.0	0.5	10	4.6	4
Explosion or fire	3	0.1	0.25	0.25	1	–	1.6
Rupture of crew comp	–	–	–	0.10	10	–	–
Two or more engines out	–	11	–	–	–	–	–
Liftoff AGE malfunction	–	10	–	–	–	–	–
IMU and RGs	–	2.5	1.25	0.05	–	–	–
FCC	–	2.5	1.25	0.05	–	–	–
TVC	–	2.5	1.25	0.05	–	–	–
Electrical power	–	**	**	**	1	**	0.4
Loss ECS	–	–	–	–	–	–	–
Loss RCS	–	–	–	–	5	2.4	–
Loss hyd	–	–	–	–	–	–	16
Jet engine system	–	–	–	–	–	–	4
Total hazard	3	29	7	1	27	53	26

*x 10^{-6} occurrence/mission
**Negligible

Sample calculation for Table 14-1:

$$P_3 + Q_{13} + Q_{23} = 1$$

$$0.99615 + 3823 \times 10^{-6} + 27 \times 10^{-6} = 1$$

$$
\begin{aligned}
P_{cs} = 0.995 \ & + \ 1 \, (47 \times 10^{-6}) \ (0.99) \\
& + \ (0.999950) \ (96 \times 10^{-6}) \ (0.90) \\
& + \ (0.999825) \ (118 \times 10^{-6}) \ (0.90) \\
& + \ (0.999700) \ (199 \times 10^{-6}) \ (0.995) \\
& + \ (0.999500) \ (3823 \times 10^{-6}) \ (0.995) \\
& + \ (0.995659) \ (297 \times 10^{-6}) \ (0.98) \\
& + \ (0.995300) \ (274 \times 10^{-6}) \ (0.96)
\end{aligned}
$$

$$P_{cs} = 0.995 + 0.004798$$

$$P_{cs} = 0.999798$$

Note that only Q_{1i} is used in the equation for crew survival, P_{cs}.

14.3 ASCENT SAFETY PROPULSION CONSIDERATIONS

The effect of losing one or more engines on the ability either to complete the mission or to terminate the mission safely has been evaluated. The thrust/weight available at any time is given by the following relationship, which is valid up to the point in ascent where throttling is required.

$$\frac{T}{W} = \frac{1}{\dfrac{1}{(T/W)_o} - \dfrac{t}{I_{sp}}}$$

14-9

where

T = nominal thrust

W = weight

I_{sp} = specific impulse

t = burntime

o = the time of liftoff

Thrust requirements decrease as the vehicle weight decreases. With a given thrust/weight requirement, the preceding equation can be solved for the time in flight at which a certain number of engines may be lost.

Two regimes are of interest here:

- Thrust required for nominal mission completion to provide a T/W = 1.43, with 115 percent of normal thrust rating used (Use of this option would necessitate engine refurbishment.)
- Thrust required for zero deadband, here identified with a minimum T/W = 1.1 at 115 percent rating, which permits slow climbout, separation, and return to the landing site

After an acceleration of 3 g is reached, continuous throttling begins, which means that additional engines may be lost without compromising mission completion.

The number of engines (X) of a total of (N) engines that may be lost if a $(T/W)_{REQ}$ is required at an engine rating R is given by

$$X = N - N(T/W)_{REQ} \left[\frac{1}{R(T/W_o)_{NOM}} - \frac{t}{I_{sp}} \right]$$

An evaluation of this equation has been made for the Two-Stage configuration, as shown in Fig. 14-4.
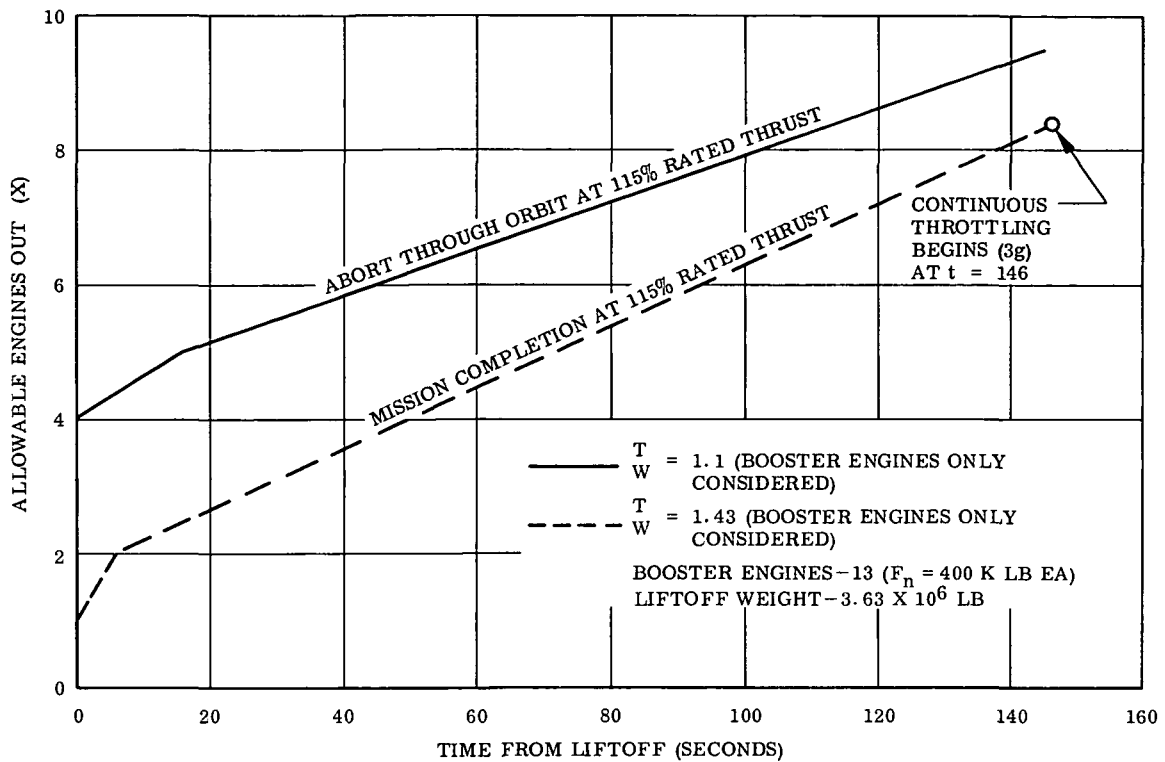
Fig. 14-4 Allowable Engines Out — Two-Stage Configuration

The normal mission velocity requirement is based on a liftoff weight of 3.63×10^6 pounds and 13 engines, each capable of 400,000 pounds thrust at 100 percent of rated. These engines can be run up to 115 percent emergency rating with the resulting penalty that the engines must be refurbished. The orbiter engines are not running at liftoff. However, they may be started and brought up to 100 percent of rated thrust in 5 seconds to provide additional safety margin in low thrust to weight situations.

The use of 115 percent percent takeoff rating permits loss of one engine from liftoff without affecting mission completion. The probability of losing one engine at any time prior to the time that continuous throttling begins (143 seconds) is approximately 1 in 1000 flights.

Increasing numbers of engines may be lost, as indicated, as the vehicle weight is reduced. Considerable redundancies exist, particularly as the mission proceeds. A deadband resulting from a major propulsion failure is discussed here, although such

14-11

an event is extremely unlikely. In such a case, a $T/W = 1.1$ is considered to be a minimum for climbout. The number of engines that can be safely lost if the 115 percent emergency rating is used is shown in Fig. 14-4. If an abort through orbit were attempted, the mission velocity available would be reduced by about 1500 fps because of the lower $T/W = 1.1$. This brackets the effect of all conceivable booster engine failures and demonstrates the forgiveness of the multiengine installation.

14.4 SAFETY TASKS

Since a study effort is iterative, the information developed under each task listed below, must be updated and refined as dictated by the results of the other tasks.

    a. Identify mission phases.

    b. Identify potential hazards per phase.

    c. Prepare a matrix of interaction of operational events and hazards.

    d. Perform operational/safety hazard analyses.

    e. Establish operational/safety guidelines.

    f. Assess impact of operational/safety considerations on vehicles.

    g. Validate guidelines against NASA-furnished Space Shuttle concepts.

    h. Prepare and submit reports.

The functional flow and interrelationships of these tasks are shown in Fig. 14-5.

NASA STUDY REVIEWS

(FINAL PRESENTATION)

TASK I
STUDY DIRECTION

TASK II
IDENTIFY STUDY PHASES

- PAD OPERATIONS
- LAUNCH OPERATIONS
- ON ORBIT
- DOCKING OPERATIONS
- RE-ENTRY OPERATIONS
- LANDING OPERATIONS

TASK III
IDENTIFY MISSION
PHASE HAZARDS

- EXPLOSION
- ENGINE OUTAGE
- FIRE
- GN&C LOSS
- SUBSYSTEM FAILURE
- STRUCTURAL FAILURE
- OTHER

TASK IV
DETERMINE MISSION
OPERATIONAL EVENTS

- TRANSPORT
- ERECT
- MATE
- CHECKOUT
- FUEL LOAD
- CREW INGRESS

TASK V
PERFORM
OPERATIONAL/
SAFETY
HAZARD ANALYSES

TASK VI
DEVELOP
OPERATIONAL/
SAFETY
GUIDELINES

- DESIGN
- PROCEDURES
- OPS PLANS
- MISSION PROFILE
- OTHER
- CREW FUNCTIONS, REQMTS,
  AND PROCEDURES
- GROUND EQUIPMENT
  OPERATIONS
- GROUND CREW, FUNCTIONS

TASK VIII
VALIDATE
GUIDELINES
AGAINST
CANDIDATE
VEHICLE
CONCEPTS

TASK IX
REPORT

TASK VII
DETERMINE IMPACT OF
OPERATIONS ON VEHICLE
CONSIDERATIONS ON VEHICLE

- VEHICLE DESIGN
- VEHICLE GSE INTERFACE
- MATERIALS
- SUPPORT EQUIPMENT
- LOGISTICS
- PAYLOAD
- OTHER

START
PHASE B

END
PHASE B

Fig. 14-5 Block Diagram – Space Shuttle Operations and Safety Considerations

Section 15

OPERATIONS

15.1 SYSTEM OPERATIONS PROFILE

Space Shuttle operations and facilities will be designed for economical routine support of reusable spacecraft as "common carriers" for space cargo and passengers in the late 1970s and the 1980s. This approach parallels that used in today's air cargo operations, particularly those specializing in charter service and special cargos. Differences are primarily in frequency of operation, integration of payload, and use of dedicated launch and primary landing facilities. The concept of routine operations is one of the significant factors in producing an economical space delivery system. Mission success is expected because of thorough flight testing and advanced designs and operations for high reliability similar to those used with today's high passenger density aircraft.

Figure 15-1 depicts an operational cycle for a Space Shuttle, starting with a payload to be transported into space. After installation or loading of the payload into the spacecraft in the Maintenance and Assembly Building (MAB) or hangar, the complete system is checked out. Optical jigs will be used for mechanical alignment, and the onboard computerized checkout system will check out and record the status of each subsystem. The RF systems, operating in a closed-loop mode, will tie into the ground management and data system. Upon completion of checkout, the vehicle would be held in readiness for the scheduled launch. A few hours before launch, the Space Shuttle is towed out of the MAB and positioned at the launch pad; commodities such as jet fuel, water, supercritical cryogens for the ECS and EPS, RCS propellants, and pressurant gases are loaded. The main propellants are then loaded, the fuel cells are started and checked out, and the crew boards. Then the vehicle is checked out by the crew, using the onboard systems. Upon clearance from the control center, the crew initiates the terminal countdown and launch. The ground computer in the operations management center will compare data from the spacecraft with launch tracking data during ascent and will

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ORBITAL OPERATIONS
- STATUS CHECKS
- PROPELLANT UTILIZATION
- GROUND UPDATES

⑦

DE-ORBIT
- PREPARATION FOR RETRO
- RETRO BURN
- DUMP RESIDUALS
- REPRESSURIZE TANKS

⑧

ASCENT
- PROPELLANT UTILIZATION
- PERFORMANCE MONITORING

⑥

FLIGHT OPERATIONS

COUNTDOWN & LAUNCH
- PROPELLANT LOADING
- PRESSURIZATION
- FINAL STATUS CHECKS
- IGNITION

⑤

LAND AND PROTECT
- COOL SPACECRAFT
- PURGE SYSTEMS
- PRESERVE
- UNLOAD PAYLOAD
- TRANSPORT

⑨

GROUND OPERATIONS

PREPARE FOR REUSE
- DETERMINE MAINTENANCE REQ'D
- REPAIR OR REPLACE
- SYSTEM TEST
- STORE

⑩

④ ESTABLISH FLIGHT READINESS
- SPACECRAFT MATE TO BOOSTER
- PAYLOAD INSTALLATION
- COMPLETE SYSTEMS TEST (CST)
- FINAL PREPARATIONS

① ② ③

ACCEPTANCE TESTS
- MFG/VENDOR

VEHICLE TESTS
- SUBSYSTEM
- ASSEMBLY

SYSTEM TESTS
- END-TO-END
- SIMULATED ENVIRONMENT
- DD-250

Fig. 15-1 Space Shuttle Reuse Cycle

process and record spacecraft data during the flight. All mission operations are performed.as required. Reentry is controlled by the spacecraft with authority and backup position updating from the ground control center. Landing aids on the ground supplement onboard systems to enable hands-off recovery under normal weather conditions. After landing, the spacecraft is purged of any residual propellants and sensitive systems are protected against corrosion or damage. Removal of payload (unless the same payload is to be reused for the next flight), inspection, and refurbishment operations then prepare the spacecraft for its next use.

In view of the "common carrier" mode of operation conceived for the Space Shuttle system, careful attention to downtime in both ground operations and orbital operations is required to maintain high utilization of the spacecraft. With the relatively small number of vehicles planned in the inventory, the economic effectiveness considerations are as important as performance optimization. These tradeoffs are analogous to the aircraft industry marketing analysis for transport aircraft cargo accommodation provisions, in relation to traffic and distance and terminal handling considerations. Factors influencing fast turnaround in ground operations as well as orbital operations are treated in the subsequent sections, along with the basic functions of vehicle recovery, maintenance, and launch operations.

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15.2 GROUND OPERATIONS

Ground operations segments of the reuse cycle are shown in Fig. 15-1. The first three entail the introduction of new vehicles or components into the operational system. Segments 4 and 5, commonly called "Launch Operations," are the prelaunch and pad activities that culminate at liftoff. Flight Operations, sometimes called "Mission Operations," include the next three segments. The Landing, or post-flight phase, commencing with touchdown, is Segment 9. In the Refurbishment phase, Segment 10, the boosters and orbiters are readied for reuse, commencing again with Segment 4. The 2-week turnaround goal for an operational Space Shuttle that has had a nominal flight includes the activities in Segments 9, 10, 4, and 5.

A substantial portion of the cost of space transportation will be for operations and maintenance. The ground operations are shown in Figs. 15-2 and 15-3, in which the heavy lines show the main operations required for every cycle.

Branch loops define items that support main-line operations or items that require taking the vehicle off line. The Two-Stage vehicle flow diagram shows separate refurbishment and repair for each of the two vehicles, while the Triamese chart shows a common refurbishment and repair for all stages. The major difference between the ground operations of the two vehicles is that the Two-Stage vehicle is mated horizontally prior to pad delivery, while the Triamese vehicle is mated vertically at the pad.

Figures 15-4 and 15-5 show preliminary estimates of the ground operation timelines for the Two-Stage and Triamese vehicles. Both require about the same time for ground operations. However, more jobs and men are required to support the Triamese refurbishment and on-pad buildup because of the higher number of flight elements. The time spans remain about the same because of the capability of parallel operations on each vehicle element. It is anticipated that the Two-Stage configuration could achieve a 10-shift turnaround (including 5-1/2 shifts for routine maintenance after an uneventful flight) within the first year after IOC.

```
                    ┌──────────────┐      ┌──────────────┐
                    │  COOL S/C    │◄─────│    LAND       │◄────────────────┐
                    │  AND UNLOAD  │      │  SPACECRAFT   │                 │
                    │    CREW      │      │  AND BOOSTER  │                 │
                    └──────────────┘      └──────────────┘                 │
```

| SPACECRAFT STRUCTURAL REPAIR | DECONT. ROUGH CLEAN AND INSPECT | BOOSTER STRUCTURAL REPAIR |

SPACECRAFT   BOOSTER

| INSPECT COMPONENTS & DETERMINE DISPOSITION | REFURBISH SPACECRAFT | REFURBISH BOOSTER | INSPECT COMPONENTS & DETERMINE DISPOSITION |

| REFURBISH COMPONENTS AT MAB | STORAGE | REFURBISH COMPONENTS AT MAB |

SPACECRAFT   BOOSTER

| REFURBISH COMPONENTS AT FACTORY | SHIP COMPONENTS TO MAB | SHIP COMPONENTS TO MAB | REFURBISH COMPONENTS AT FACTORY |

| ASSEMBLE AND CHECKOUT TOTAL PAYLOAD | INSTALL PAYLOAD IN SPACECRAFT | MATE BOOSTER & SPACECRAFT AND CST | INSTALL VEHICLE ONTO LAUNCHER AT PAD & ERECT | LOAD COMMODITIES, PROPELLANT, AND CREW | FINAL VEHICLE TEST AND LAUNCH |

| MANUFACTURE PAYLOAD AND CHECKOUT | PROVIDE SUPPORT FLUIDS | REFURBISH LAUNCH PAD |

MISSION

Fig. 15-2 Two-Stage Ground Operations Flow Chart

```
┌─────────────────┐        ┌─────────────┐        ┌──────────────┐
│ DECONTAMINATE   │◄───────│  COOL S/C   │◄───────│     LAND     │
│ ROUGH CLEAN     │        │ AND UNLOAD  │        │  SPACECRAFT  │◄──┐
│ AND INSPECT     │        │    CREW     │        │ AND BOOSTERS │   │
└─────────────────┘        └─────────────┘        └──────────────┘   │
        │                                                            │
        ▼                        ▼                  ┌──────────────┐ │
┌─────────────────┐        ┌─────────────┐          │   INSPECT    │ │
│  STRUCTURAL     │───────►│  REFURBISH  │─────────►│ COMPONENTS & │ │
│  REPAIR         │        │             │          │  DETERMINE   │ │
└─────────────────┘        └─────────────┘          │ DISPOSITION  │ │
                                                     └──────────────┘ │
                     ┌─────────────┐     ┌──────────────┐             │
                     │   STORAGE   │     │  REFURBISH   │          MISSION
                     │             │     │ COMPONENTS   │             │
                     └─────────────┘     │   AT MAB     │             │
                                         └──────────────┘             │
           SPACECRAFT                                                 │
                                 ┌──────────────┐ ┌──────────────┐    │
                                 │     SHIP     │ │  REFURBISH   │    │
                       BOOSTERS  │ COMPONENTS   │◄│ COMPONENTS   │    │
                                 │   TO MAB     │ │ AT FACTORY   │    │
                                 └──────────────┘ └──────────────┘    │
```

| ASSEMBLE AND CHECK OUT TOTAL PAYLOAD | MATE PAYLOAD TO SPACECRAFT | PREPARE FOR LAUNCH AND CST | INSTALL VEHICLES ONTO LAUNCH PAD & MATE | LOAD COMMODITIES, PROPELLANT, AND CREW | FINAL VEHICLE TEST AND LAUNCH |

| MANUFACTURE PAYLOAD AND CHECK OUT | | | | PROVIDE SUPPORT FLUIDS | REFURBISH LAUNCH PAD |

Fig. 15-3 Triamese Ground Operations Flow Chart
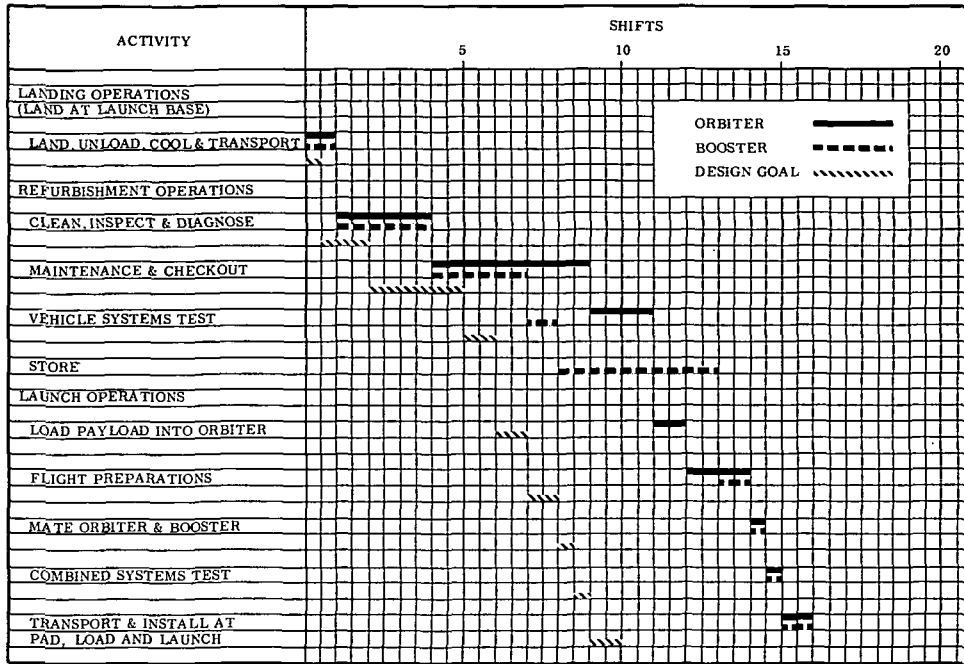
Fig. 15-4 Turnaround Estimates for Two-Stage Vehicle
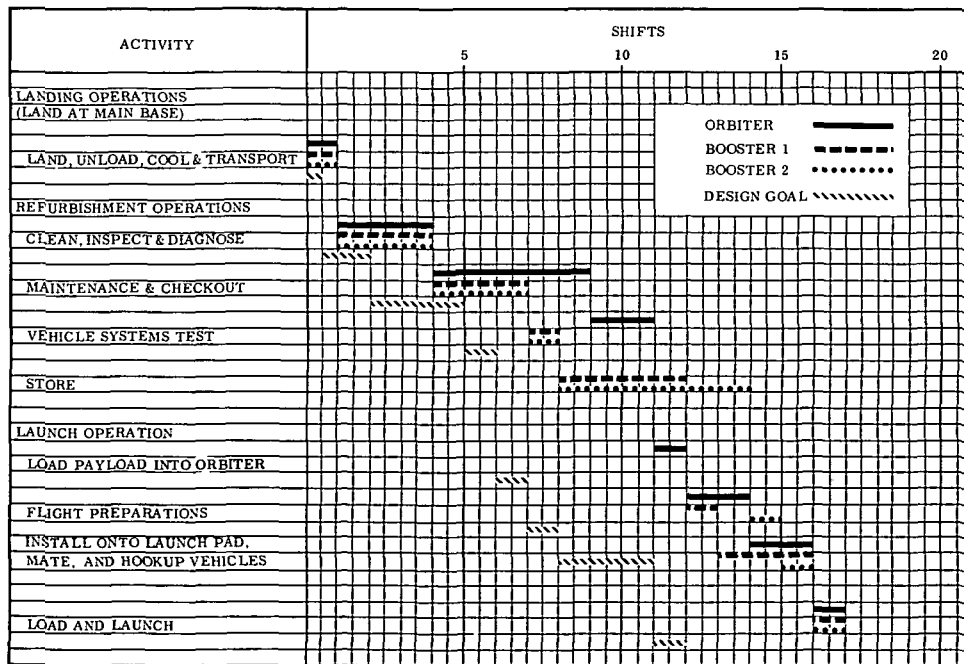
Fig. 15-5 Turnaround Estimate for Triamese Vehicle

15-7

Vital to the development of an economical space transportation system are the requirements to design for ease of maintainability, the use of a self-testing onboard checkout and status monitoring system, and the simple fast repair by replacement. Key development requirements are as follows:

- Design for automatic onboard checkout and status monitoring. Include appropriate transducers in mechanical/pneumatic/hydraulic systems for performance verification or malfunction detection. Self-test is essential.

- Provide hard points for handling large components, the spacecraft, and the complete launch vehicle. Allow for standard ground shock and vibration environment (in most cases not a limiting item).

- Provide field joints to enable breaking down large items to transportable size.

- Design for maintainability. Locate limited life components so that they are accessible for easy replacement. Use a modular approach wherever possible. Locate flight critical items for access in both horizontal and vertical vehicle positions.

- Incorporate startup provisions for the fuel cells in the electrical power system. Dummy loads may be required either as AGE or AAE.

- Provide permanent optical targets for rapid alignment checks during assembly and on the launch pad.

- Use an "inspect and repair as necessary" type of maintenance plan.

15.3 FACILITIES AND OPERATIONS

A horizontal takeoff flight test program is required to establish capability of the Space Shuttle for return flight, landing, and self-ferry. A logical place to conduct such a program is the NASA Flight Test Facility at Edwards Air Force Base, where ideal runways and much instrumentation already exist. Additional instrumentation and support buildings will be required to assemble, maintain, and test the vehicles. While vertical flights out of Edwards are not planned during the development phase, the existence of the support facilities and the all-azimuth launch capability in the operational phase of the program suggest that a vertical launch pad could be installed there for considerably less cost than a completely new launch base. However, the most flexibility and range safety would result from using MILA for easterly launches and the South Vandenberg Area of VAFB for polar launches.

Vertical flight testing leads directly into the operational phase at the IOC milestone. The operational facilities must therefore be provided early enough to support the earlier developmental flights. Figures 15-6 and 15-7 show suggested locations for integrated landing, maintenance, and launch facilities at ETR and at WTR. In Fig. 15-6, the circles drawn around Pads 1 and 2 are respectively 10,948 feet and 14,597 feet in diameter and show the 0.4-psi and 0.28-psi overpressure boundaries. These are based on 3 million pounds of LO_2/LH_2 at 60 percent TNT equivalency. For the Two-Stage configuration, the time on the pad is so short that a much smaller separation between the prime pad (1) and the backup pad (2) could be tolerated. For on-pad buildup of the Triamese configuration, the 0.4-psi spacing probably should be maintained to minimize safety interference with work schedules.

The Maintenance and Assembly Building (MAB) and the landing field or runway should be located far enough (0.2 psi or 18,246 feet) from the pads to avoid suspension of operations during launch. It appears that the sound pressure level will be about 120 db for this separation.
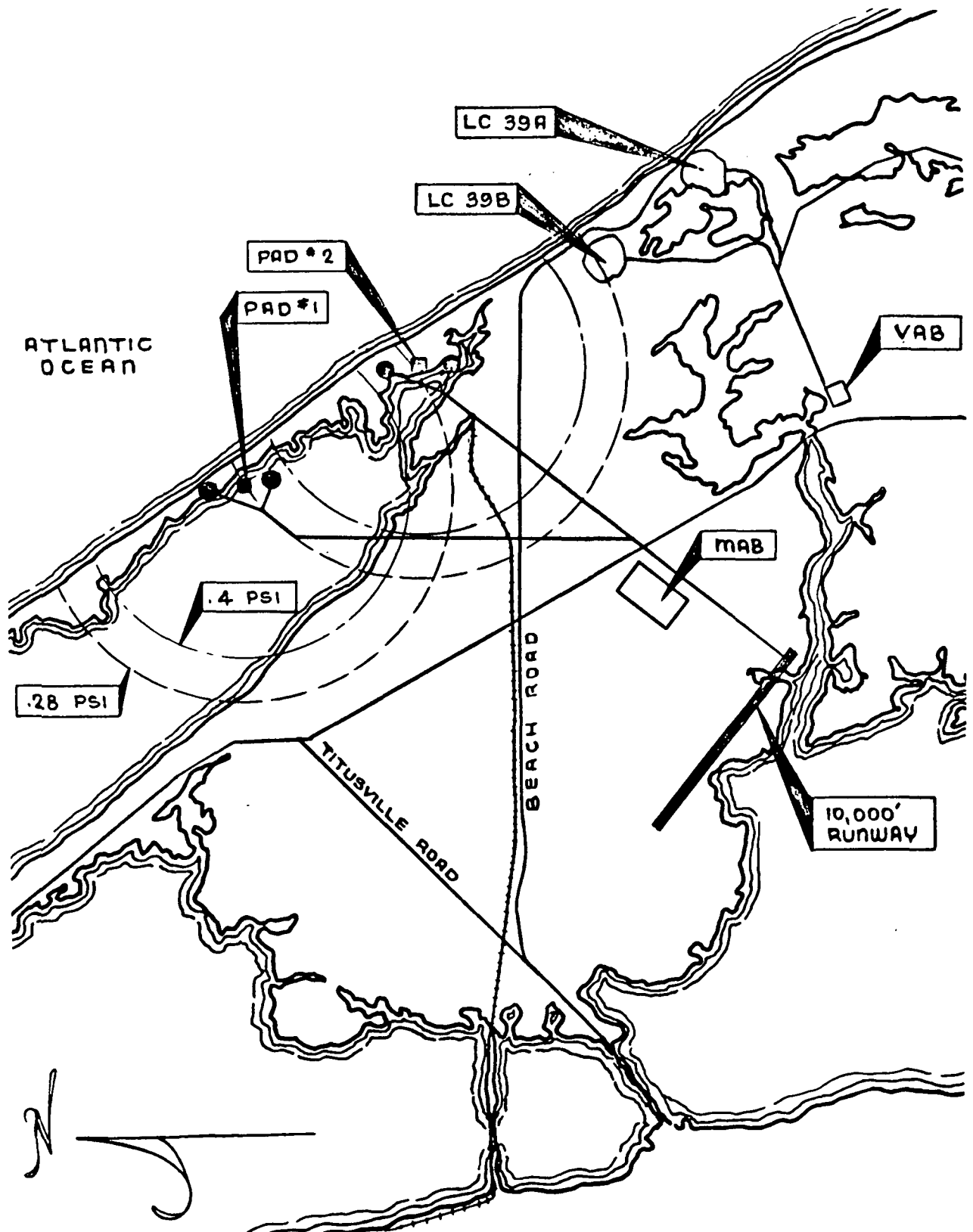
Fig. 15-6 ETR Space Shuttle Base

15-10
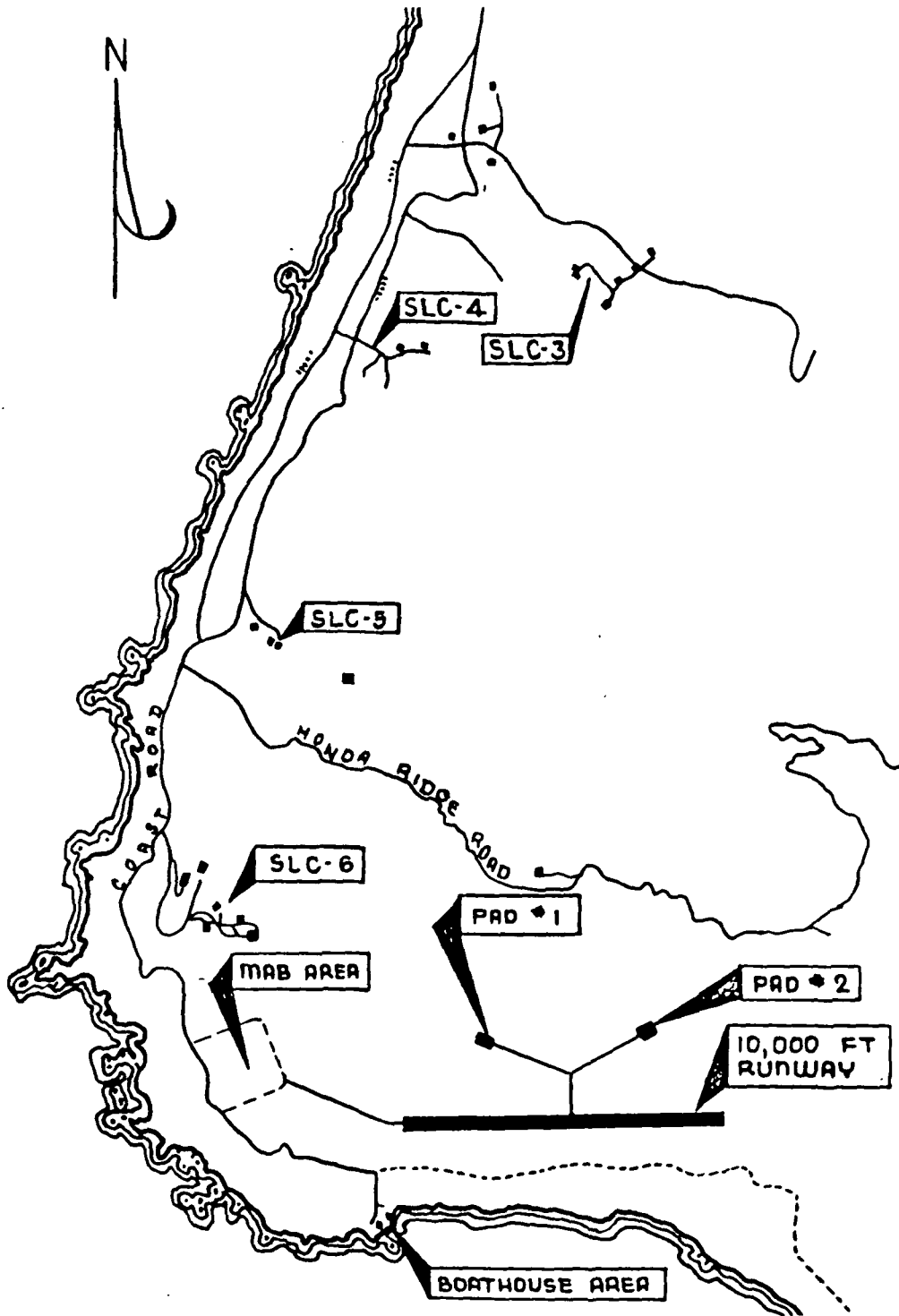
Fig. 15-7   Possible WTR Space Shuttle Base

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All of the Space Shuttle vehicles are being designed to land at moderate speeds (120 to 150 knots), and a runway length of 10,000 feet appears to be adequate. Use of longer runways might enable some savings in weight of landing systems, but this would drastically reduce the number of alternate fields available for emergency landings. (Landing systems are discussed in Vol. III, Section 5.) Landing weights of the four ILRV baseline configurations vary from 207,000 pounds to 375,000 pounds, all within the envelope of current jet transport aircraft. Therefore, though a heavy-duty runway is required, it does not present a unique problem. Actual runway thickness specifications will depend on the flotation provided by the landing gear, the bearing strength of the soil,and the type of construction materials chosen. (For comparison purposes, the maximum ramp weight of the L-1011 Tri-Star is 411,000 pounds and the maximum allowable weight on the main landing gear is 390,000 pounds.) A 300-foot width is suggested to provide more margin for cross winds, low visibility, and maybe different aerodynamic response from that found with commercial planes on 150-foot wide strips. On most military fields, runways are made 200 to 300 feet wide for these same reasons. In addition to the standard communications, fire protection, lighting, maintenance, and landing aid facilities and services required by active airports, the Space Shuttle system will require a few special items of support equipment because of the size and nature of the vehicle. These include the following:

- Special access stands for unloading crew and passengers
- Cooling carts to prevent overheating of the orbiter after reentry and landing
- Purge carts to eliminate or blanket residual propellant gases as a safety precaution
- Special payload unloading slings or fixtures for the yet undefined payloads, to permit rapid unloading if necessary (A long-boom mobile crane or a fixed derrick could be used at the landing field.)

The equivalent of these items, probably in air-transportable versions, may be desirable for use at an alternate landing site preselected before deorbit. Standard aircraft-type ground power carts, tugs, access platforms, wheel chocks, etc., will, of course, be used.

A major part of the turnaround cycle is the refurbishment or maintenance process. The maintenance facility must be set up to handle both scheduled maintenance and occasional unscheduled maintenance or repair activities. Figures 15-2 and 15-3 show both nominal on-line activities required for every turnaround and some of the special repair or major maintenance requirements that might be occasioned by a hot reentry, a hard landing, or excessive time on a vehicle or component.

The cleaning operations are particularly important for safety, corrosion protection, and (in the case of orbital vehicles) for personnel comfort on the next flight. Inspection includes all types of examinations from a visual check to taking samples of fluids and gases for contamination checks and any mechanical, x-ray, dye check, and radioisotope inspections required to detect failures or incipient failures of critical internal or external mechanical systems, such as heat shields, landing gears, engine nozzles, controls and control surfaces, and vehicle skins. Also, a complete vehicle self-check is performed by using the onboard checkout system.

The diagnostic analysis, decisions, and subsequently the actual refurbishment work plan for an individual vehicle is based on the following:

- Time oriented and normal maintenance requirements for each individual vehicle as provided by the computerized ground management system
- Onboard records of vehicle state and areas of concern or failures as reported by the onboard checkout system for the flight just completed
- Pilot squawk sheets or crew ratings of vehicle status
- Results of the various ground inspections and checkouts just performed

The actual maintenance of the vehicle follows the work plan, as determined and scheduled from the diagnostic analysis previously described. The maintenance approach is to replace components only as necessary and to eliminate all onboard repair. Individual subsystems are checked out to determine acceptable status as soon as their respective maintenance action items have been accomplished. Complete vehicle checkout is performed by using the onboard checkout system, supplemented as necessary by some ground-based stimuli generators and calibration reference equipment, in addition, of course, to ground power supplies.

After completion of all maintenance items, a complete vehicle checkout is performed by using the onboard checkout system. This constitutes a system acceptance test and revalidates the vehicle as operational and ready for flight. The vehicle can then be delivered for its next flight assignment or placed in controlled storage. Protection requirements have not been fully determined; but some isolation from sand, dust, rain, and salt air seems desirable to minimize degradation of critical thermal surfaces and instrumentation.

The size of the facility required for maintenance or refurbishment is a function of the launch rate, number of vehicles in the fleet, size of the vehicles, and the minimum allowable turnaround time. One concept for a MAB, patterned after aircraft operations, is shown in Fig. 15-8. The dimensions shown are approximate, and there is no requirement that all operations be under a single roof. Movement of vehicles from one position to another is by removal from the bay and external transfer to the next station, either on the integral landing gear (first few operations) or on transporters or bogies.

The vehicle work areas consist of various bays designed to provide a specific vehicle operation in the most effective manner. The manufacturing bay is a work area, which provides the assembly fixtures, work stands, special optical and support tooling, and the variety of fabrication and inspection equipment necessary to assembly and check out a complete spacecraft. This bay also provides the capability for structural repair and major maintenance and rework of an operational vehicle. Most operations in this area involve cutting, fastening, and other contaminating procedures. All work done in this bay is off the main-line cycle and would generally consist of operations such as repair or replacement of heat shields, tankage, aerodynamic control surfaces, and landing gear.

The refurbishment bay is a work area that provides the necessary access and support equipment to allow least time turnaround of a spacecraft. The basic operational concept for this bay is to remove and replace specified spacecraft items only as necessary. A combination of onboard checkout, realtime flight recorder information, and ground data/logistics management system inputs provides a total health picture of the vehicle to be refurbished. Based on these inputs, pilot squawk sheets, and the results of a thorough visual inspection, the refurbishment schedule plan for a vehicle is determined and accomplished in this bay.

2180

CRANE RAILS

EXTERNAL
COVERED
AREA

275

195	195	195	195	280	280	280	280	280
DECONT. CELL	MFG CELL	REFURB CELL	REFURB CELL	VEHICLE MATE	P/L LOADING	REFURB CELL	MFG CELL	DECONT. CELL

240

320

6 FLOORS OF SUPPORT AREAS

100'

SHIPPING DOCK

90' HIGH
BUILDING

PAYLOAD AREAS

150' CRANE RAILS
250 TON CRANE

120' HIGH BAY
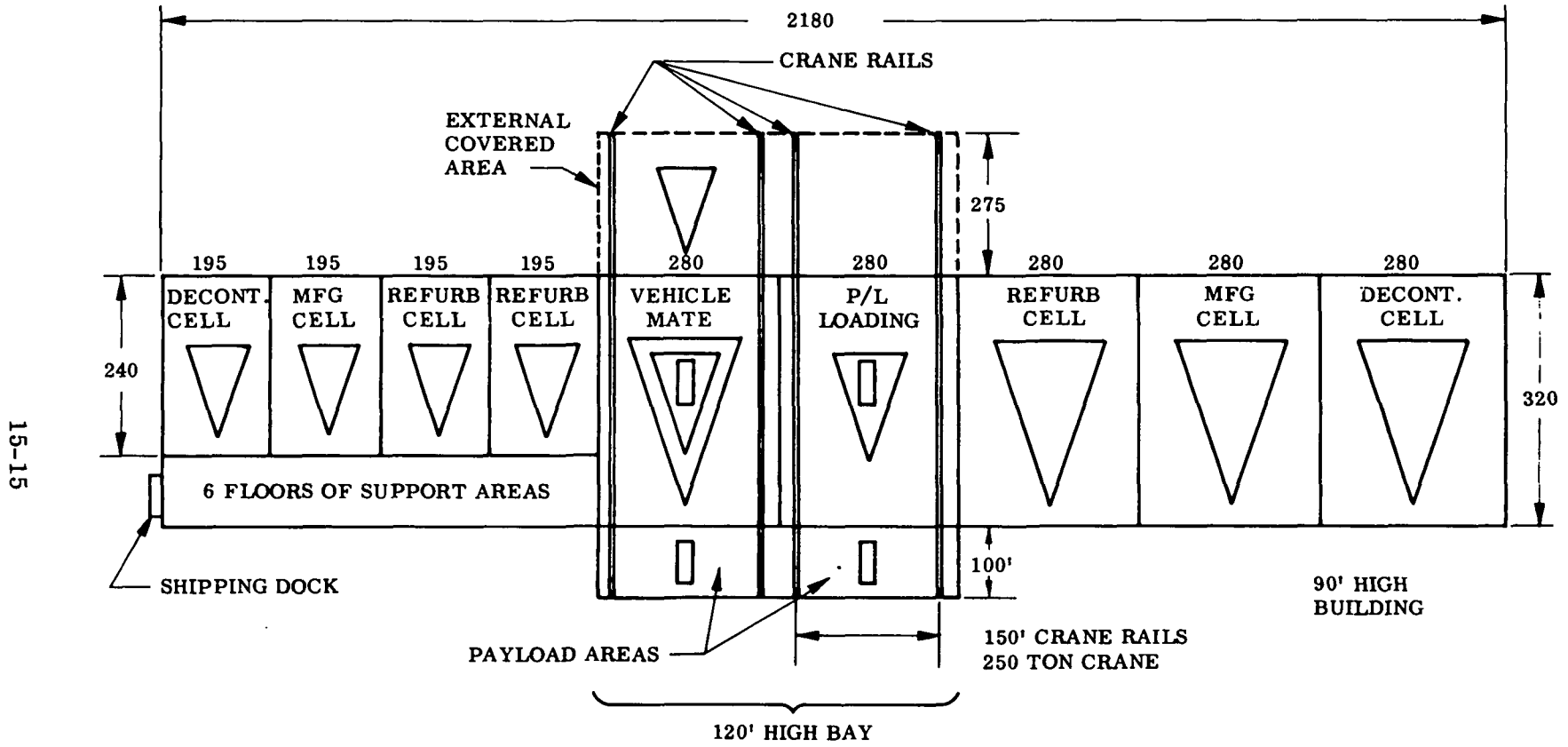
692K FT^2 EXCLUDING
SUPPORT AREAS

Fig. 15-8 Plan View of Two-Stage Maintenance and Assembly Building (MAB), Sized for 5-Vehicle Fleet

An item that cannot be remedied by the remove and replace concept will require the vehicle to be taken off the main line and delivered to the manufacturing bay for rework. Standard main-line replacement items consist of jobs such as recharging life support and environment systems, replacing failed instruments or transducers, installing batteries, changing tires, and replacing black box type components.

The hazardous bay provides blast walls and other safety precautions that allow operations such as tank purge, clean and passivate, tank proof tests, fluid delivery system clean and proof tests, and battery charging. This bay provides all the access platforms; handling equipment; purge, pressurization, flushing, and passivation facilities; and toxic gas disposal systems to support the types of hazardous operations described above.

The loading and unloading areas are serviced by high-bay overhead bridge cranes, which extend into the payload areas. The high-bay provides space and equipment for installing the payload or cargo modules into the spacecraft and checkout of the total combination. For the Two-Stage vehicle, these areas and their overhead cranes also provide the means for vehicle mate; and one high bay cell incorporates mating and access equipment to support the mating operation. To make this payload area more flexible, the concept of providing an outside covered payload handling capability by extending the crane rails outside the building is shown. This outside area could be used for unloading a just-returned spacecraft.

The payload area provides a controlled area for test, package, integration, etc., of any payload or cargo component, either in preparation for launch or of return payload.

The support function areas consist of various shops, laboratories, storage facilities, and checkout and test rooms to support the Space Shuttle subsystems. Examples of these are machine shops, electronic shops, valve shops, antenna laboratories, tool cribs, part storage, component storage, instrument calibration and repair shop, hydraulic flow facility, clean rooms, computer rooms, x-ray and ultrasonic checkout facility, and subsystem test facilities. These could be located on various floors adjacent to the vehicle operation bays. The tool cribs and part storage areas should be located for most efficient access to the work bays they are supporting.

The administrative areas consist of facilities to provide engineering support, records and files, and clerical and administration functions for the total refurbishment and launch operation. These facilities should be located on the various floors adjacent to their applicable work support areas. While it is not absolutely necessary to have the support and administrative areas in the MAB, the efficiency of the operation would be much higher than with remote locations.

Launch operations commence in the MAB, with the preparation of the vehicles for flight, the installation of payload or payload tanks, and flight readiness verification checks. For the Two-Stage configurations, mating would take place horizontally in the MAB; in the Triamese configurations, the mating would be accomplished vertically on the launch pad. Alternate methods of mating, transportation, and erection have been studied; some of the concepts are illustrated in Fig. 15-9. In all of these systems, a launch mount adapter is employed to provide the interface between the flight vehicle and the launch facility. This adapter is basically a thrust structure, incorporating holddown provisions. It is mated to the vehicles while they are still in the controlled environment of the MAB. This provides an easy mate procedure for such precision-located items as fill umbilicals and launch holddowns. Use of these adapters allows a greater tolerance for locating the large vehicle on the launcher and minimizes pad operations.

The method anticipated for locating the vehicle on the launcher is use of airlift pads under the vehicle wheels. A rack and pinion drive is used for X and Y translation, and a hydraulic jack is used for Z adjust.

The rationale for the operating methods described may be expressed as follows:

- Horizontal mate of the two vehicles in the MAB involves use of a standard bridge crane, simpler access platforms, precision alignment for mate, weather protection, and payload change or access after mate.

- Use of the first-stage vehicle landing gear would eliminate the need for a boggie wheel delivery system, but it may impose an appreciable weight penalty for heavier gear.

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- Use of launch mount adapters provides for mate of the vehicle-to-ground umbilicals and flight restraint mechanism by adding the adapters to the vehicle. This operation, performed in the refurbishment cell, allows an easy, precise mate and hookup, which would be much more difficult if it were done by installing the vehicle onto pad-mounted adapters.

- The positioning plates provide for three-dimensional adjustment of the vehicle for the mate to the launcher/erector.

- The balance type launcher/erector provides both the launch pad for vehicle liftoff and the method of erecting the vehicle. It also incorporates a personnel ingress/egress and rapid escape system and a stabilization support (wind brace).

- The existing Saturn V crawler could be used to deliver the launcher-vehicle combination if off-pad erection is desired. A simple and easily refurbished launcher would be required, and it would be possible to use one of the existing LC-39 launch pads. An expanded-capability parallel propellant loading system would be required, and a new flame deflector may be necessary.

- Access to the crew compartments and cargo/passenger areas is provided by a personnel access tower and swing arm ramps. This system also provides emergency vehicle egress up to liftoff.

The favored launch operating concept is for a minimum vehicle-on-pad time, which requires a fast launch pad delivery method, rapid and easy vehicle-to-pad installation operations, and rapid vehicle chilldown and propellant loading. Figure 15-10 provides a preliminary list of pad operations and their associated times for the Two-Stage configurations. Five convenient hold points are shown for the Two-Stage vehicle. The first hold point is just prior to roll out from the MAB; and the duration is limited only by a few components, such as primary batteries. The next hold, at T-280 minutes, just before commodity loading, is limited primarily by the lack of environmental protection. At T-3 hours, just prior to erection, the ability of the vehicles to hold the supercritical cryogens governs. After erection, the vehicle may be held at T-2 hours by delaying the start of main propellant loading.
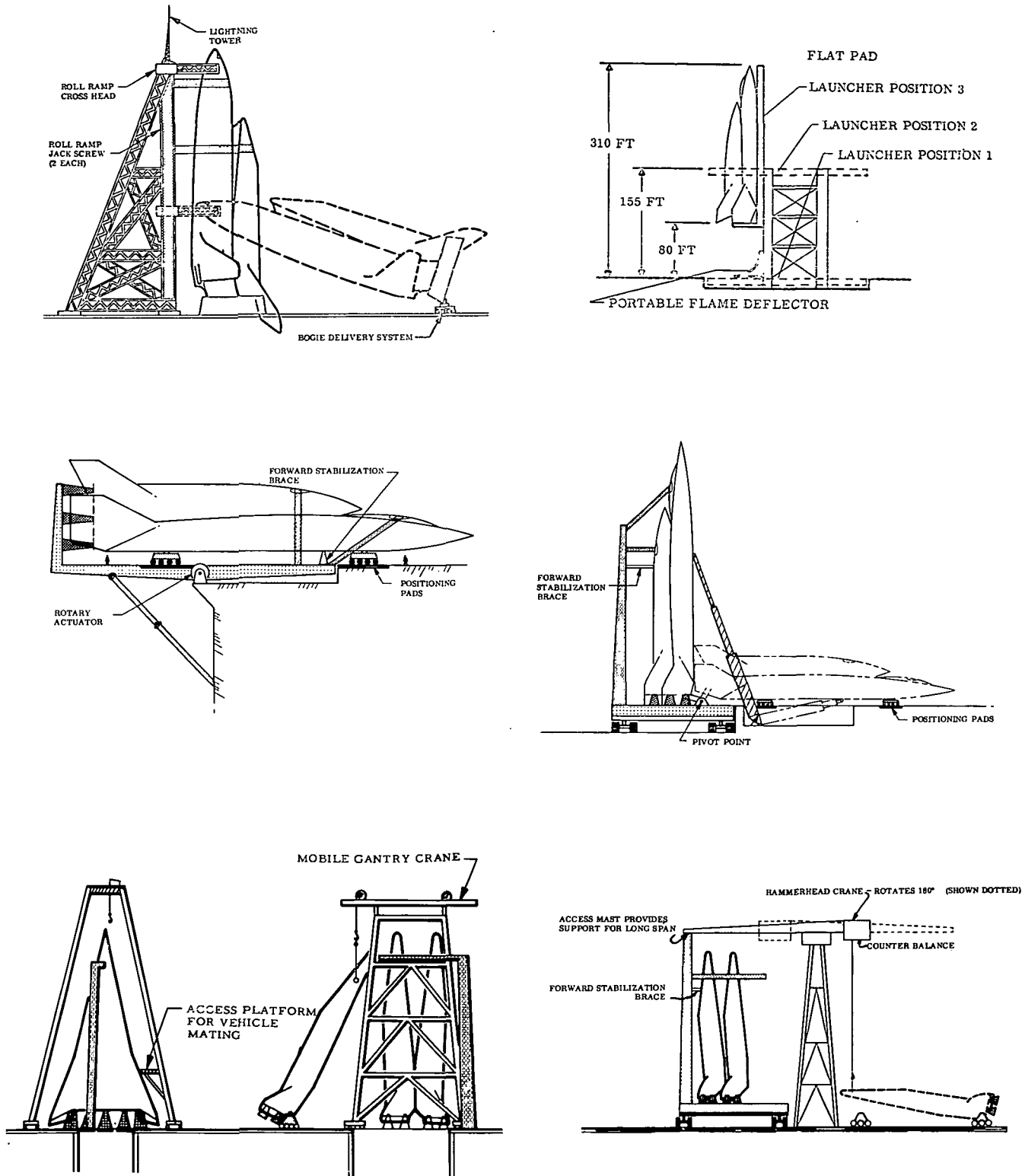
Fig. 15-9 Vehicle Erection Concepts

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	HOURS	T-6	T-5	T-4	T-3	T-2	T-1	
	MINUTES	-360	-300	-240	-180	-120	-60	0
HORIZONTAL TRANSPORT TO PAD								
POSITION & PREPARE FOR ERECTION								
PURGE S/C TANK INSULATION WITH GHe								
LOAD JP-4 FUEL								
LOAD SUPERCRITICAL LOX								
LOAD SUPERCRITICAL LH$_2$								
LOAD HIGH PRESSURE GHe								
LOAD HIGH PRESSURE GN$_2$								
ERECT VEHICLE								
COUPLE UMBILICALS TO PAD FACILITY								
PURGE VEHICLE BASE AREAS W/GN$_2$								
CLEAR PAD AREA								
CHILLDOWN AND SLOW FILL LH$_2$								
BOOSTER 3K GPM								
SPACECRAFT 1K GPM								
FAST FILL LH$_2$								
BOOSTER 20K GPM								
SPACECRAFT 10K GPM								
REPLENISH LH$_2$								
CHILLDOWN AND SLOW FILL LOX								
BOOSTER 2K GPM								
SPACECRAFT 1K GPM								
FAST FILL LOX								
BOOSTER 10K GPM								
SPACECRAFT 3K GPM								
REPLENISH LOX								
ACTIVATE FUEL CELLS								
REPLACE GROUND CREWS WITH FLIGHT CREWS								
COMPLETE LAUNCH READINESS CHECKS								
RANGE SAFETY CHECKS AND CLEARANCE								
TERMINAL COUNTDOWN								
LAUNCH								
"HOLD" POINTS		0		0	0	0	0	

Fig. 15-10 Launch Time Line – Two-Stage Space Shuttle

Finally, upon completion of propellant loading, at T-1 hour, the hold capability is limited by the vehicle tank insulation systems; this hold capability actually extends to about T-2 minutes, when the terminal countdown sequence is initiated in the booster cockpit. At any time prior to launch commit (determined by thrust buildup), the launch may be scrubbed and the engines shut down. In this event, the main propellants would be drained back into the storage dewars by use of the fill/vent system, which remains connected to the vehicle until liftoff. It is anticipated that this dump operation may take 2 to 3 hours, and it could be remotely controlled if the crew desired to leave the vehicle via the personnel access tower. Upon completion of the drain operations, the vehicle may be lowered to the horizontal position. Depending upon the reason for the scrub, the vehicle may be recycled at the pad or may be returned to the MAB for corrective maintenance.

Figure 15-11 shows a more detailed analysis of the operations and times associated with propellant loading. Typical propellant quantities required for each launch are shown in Table 15-1.

A preliminary breakdown of the manning levels required for the Space Shuttle base facility is shown in Table 15-2.

MINUTES	0	10	20	30	40	50	60	70	80

START LOADING PROPELLANTS
CHILLDOWN AND SLOW FILL
    BOOSTER LH_2 3K GPM
    BOOSTER LOX 2K GPM
    SPACECRAFT LH_2 1K GPM
    SPACECRAFT LOX 1K GPM
FAST FILL
    BOOSTER LH_2 20 GPM
    BOOSTER LOX 10K GPM
    SPACECRAFT LH_2 10K GPM
    SPACECRAFT LOX 3K GPM
REPLENISH LH_2
        LOX
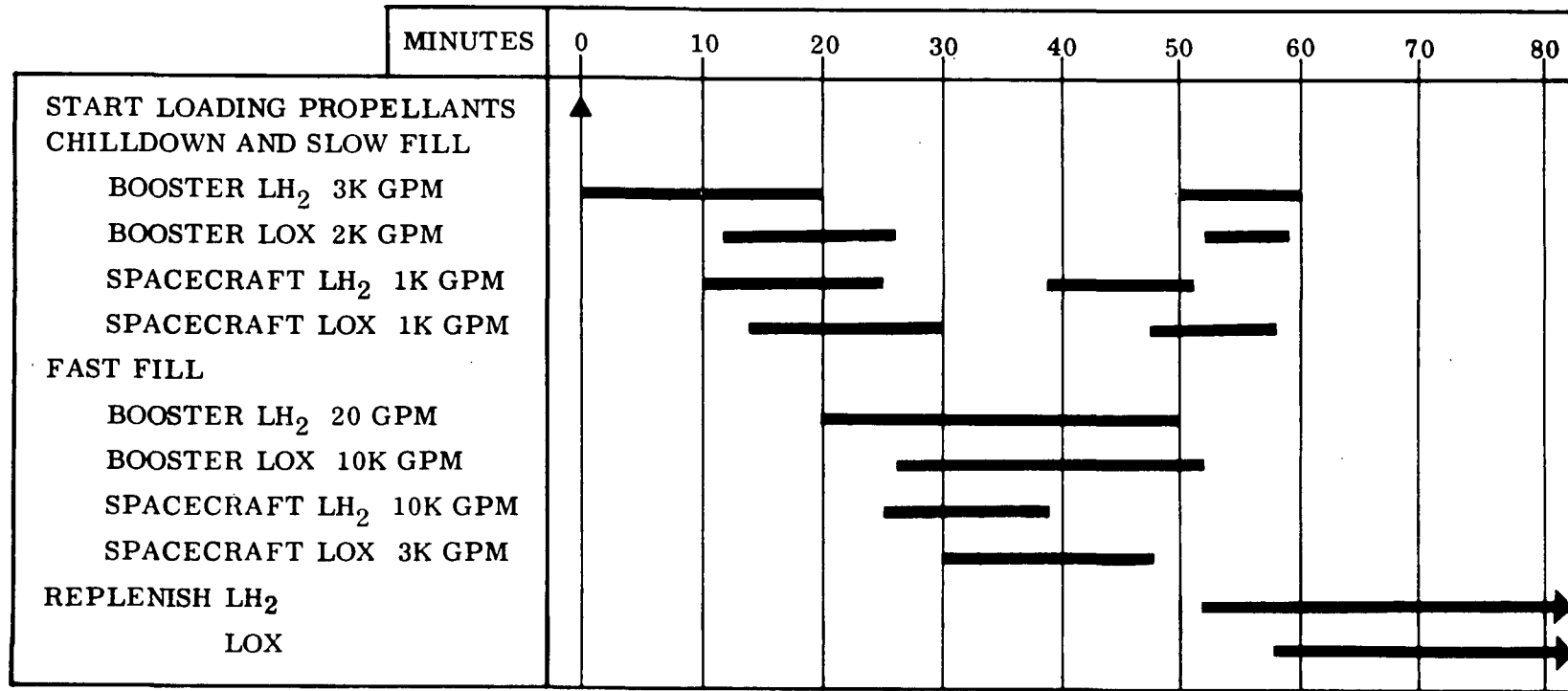
Fig. 15-11 Main Propellant Loading Plan — Two-Stage Space Shuttle

Table 15-1

PRIMARY PROPELLANT REQUIREMENTS COMPARISON

	Units	Vehicle Configuration							
		Two-Stage				Triamese			
Payload		25K		50K		25K		50K	
Gross Launch Weight	10^3 lb	3,058		3,735		3,609		4,340	
Propellant Type		LO_2	LH_2	LO_2	LH_2	LO_2	LH_2	LO_2	LH_2
Quality									
Booster 1	10^3 lb	1,684	241	2,051	293	1,012	145	1,213	173
Booster 2	10^3 lb	–	–	–	–	1,012	145	1,213	173
Orbiter	10^3 lb	460	66	586	84	460	66	589	84
Subtotal	10^3 lb	2,144	307	2,637	377	2,484	356	3,015	430
Facility and Chill-down Losses	10^3 lb	715	102	837	126	828	119	1,005	143
Total	10^3 lb	2,859	409	3,516	503	3,312	475	4,020	573
	10^3 cu ft	40.1	93.4	49.3	114.8	46.4	108.4	56.1	130.8
	10^3 gal	230	699	369	859	348	811	422	979

Table 15-2

PRELIMINARY CREW REQUIREMENTS FOR
A TWO-STAGE SPACE SHUTTLE

50 FLIGHTS/YEAR

BASIC CREWS –

Vehicle Mate Area		24
Mechanics	15	
Technicians	6	
Inspectors	1	
Foremen	2	
Hazard Cell		12
Mechanics	7	
Technicians	3	
Inspectors	1	
Foreman	1	
Refurbishment Cell (2)		48
Mechanics	25	
Technicians	15	
Inspectors	4	
Foremen	4	
Manufacturing Cell		12
Mechanics	6	
Technicians	3	
Inspectors	2	
Foreman	1	
Launch Pad (2)		48
Mechanics	32	
Technicians	10	
Inspectors	2	
Foremen	4	
Payload Area		24
Mechanics	10	
Technicians	10	
Inspectors	2	
Foremen	2	

Table 15-2 (Cont.)

SPECIAL CREWS –

Environmental Life Support   16

Engineers	4
Mechanics	5
Technicians	5
Inspectors	1
Foreman	1

Controls and Displays/Instruments   24

Engineers	8
Mechanics	3
Technicians	10
Inspectors	1
Foremen	2

Onboard Comp./Communications   34

Engineers	10
Mechanics	4
Technicians	16
Inspectors	2
Foremen	2

Electrical Power   16

Engineers	4
Mechanics	3
Technicians	7
Inspectors	1
Foreman	1

Guidance Navigation/Stabilization Control   26

Engineers	10
Mechanics	4
Technicians	8
Inspectors	2
Foremen	2

Mechanical Systems   28

Engineers	4
Mechanics	12
Technicians	8
Inspectors	2
Foremen	2

Table 15-2 (Cont.)

SPECIAL CREWS – (Cont.)

Rocket Engines		16
Engineers	4	
Mechanics	6	
Technicians	4	
Inspectors	1	
Foreman	1	
Turbojet Engines		16
Engineers	4	
Mechanics	6	
Technicians	4	
Inspectors	1	
Foreman	1	
Propellant Systems		32
Engineers	8	
Mechanics	12	
Technicians	8	
Inspectors	2	
Foremen	2	

SUPPORT AREAS –

Receiving and Shipping	12
Inspection, X-ray, etc.	36
Stores	12
Tool Rooms	12
Valve Shop	8
Electronic Shop	16
Leakcheck and Proof Test	12
Installation, Calibration and Repair	12
Fuel Cell and Mechanical System	16
Hydraulic Flow	8

Table 15-2 (Cont.)

SUPPORT AREAS — (Cont.)

Clean Rooms	8
Cleaning Facility	8
Machine Shop	16
Emergency/Fire, etc.	12
Equipment and Facility Maintenance	16
Design and Softwear	125
Clerical and Records	85
Management	20
Total	800

15.4 OPERATIONS MANAGEMENT AND SUPPORT

An operations management function is required to provide direction and support to the Space Shuttle system. It combines the management, sales, maintenance, personnel, and flight operations activities typical of every airline operation. While it is not mandatory that all these functions be co-located with the launch site, many advantages accrue if this is done. In concept the operations management center calls for facilities and personnel necessary to carry out the functions of ground-based flight support, launch/land operations, crew selection, training and assignment, passenger orientation, shuttle refurbishment operations and associated logistic support, and master flight scheduling.

Since the shuttle is designed for autonomous flight, with ground backup control capability, the support tasks would typically be limited to the following types of activity:

Ground-based flight support —

- Select specific communications mode for specific shuttle flights.
- Notify appropriate space communications network of specific flight support requirements.

- Arrange for specific frequency and channel assignments for active flight periods.

- Provide ground monitor and abort assistance during launch and ascent phases.

- Establish return airport and contingent landing site support arrangements for each flight.

- Prepare a return activity checklist for both flight and maintenance crews for use immediately subsequent to landing.

- Prepare the flight plan, flight profile, communications plan, time line of activities, and a safety and contingency plan.

Launch/land operations −

- Provide technical support and direction for the actual launch and landing phases of the Space Shuttle operation. This includes monitoring and providing backup support of the onboard system for operations such as propellant loading, final launch countdown, vehicle targeting, descent and return flight planning, final landing approach, and ferry-mode flights. This operation combines the basic function of a launch control center and an FAA control tower, but in a backup or advisory position only, because the primary direction and control would rest with the onboard system and crew. All communications and verifications of onboard systems is through an RF link, and no hard lines between this center and the flight vehicle are planned.

Crew/personnel operations −

- Evaluate flight profile, communications plan, etc., and define crew training requirements.

- Assess crew training requirements for determination of simulators and simulator programs needed.

- Record, reduce, and assess simulator data.

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● Assess flight data and pilot debriefing for improved simulator design and control.

● Orient passengers as to abort, space transfer, safety, and reentry/ landing routines.

● Train flight, operations, and refurbishment crews by use of simulators.

Refurbishment operations —

● Evaluate shuttle maintenance handbooks, crew member logs, onboard checkout records, and other appropriate data for the purpose of developing and upgrading the logistics support capability and the maintenance and repair guides for use by refurbishment personnel.

● Schedule maintenance, repair, and refurbishment for specific shuttles in consonance with the master schedule.

● Direct maintenance crew training.

● Operate and maintain facilities and manage the resources to provide adequate capability to maintain a flight-ready shuttle fleet.

Master flight scheduling —

● Receive the detailed request for shipment notices and translate them into specific flight plans and directions. Typical of data covered on a shipping request are the following:

Product name
Mass characteristics
Weight and volume
Center of gravity
CG drift with attitude
Shuttle power required
Monitoring necessary
Environment requested
Storage data
Radiation character
Cryogens
    Tanks required
      Loading requirements

Experiments
Maneuvers peculiar to stability,
  pointing accuracy, and
  ephemeris
Destination data
Interfaces
    Personnel
    Cryogen
    Access
Communications
Umbilical definition
Shipment need dates
Flight profile plan

Based on this information, provide specific flight-related data, typically including such items as:

Specific flight	Crew Requirements
Ephemeris	Simulation requirements
Ascent profile	Passenger list
Flight profile	Cargo manifest
Time line activity	Passenger orientation checklist
Booster selection	Maintenance planning scheduling
Orbiter selection	Flight guidance and control programs
Flight scheduling	Simulation control program requirements
	Onboard checkout program

15.5 GROUND SUPPORT EQUIPMENT

Adequate support equipment is essential to the achievement of the rapid turnaround. However, the design for autonomous operation and similary to aircraft eliminates the necessity for extensive special checkout equipment and allows use of support equipment similar to that used for the jumbo jets and large military transport aircraft. Special transporters and erector systems, as well as jigs, fixtures, and other vehicle-peculiar handling equipment will be required. Propellant servicing equipment may be patterned after existing hardware, with parallel systems to meet the required time lines. Table 15-3 is a generic list of support equipment.

Table 15-3

GROUND HANDLING AND SUPPORT EQUIPMENT

MAIN PROPULSION SUBSYSTEM

Rocket Engines

Access and inspection stands
Engine installation and removal fixture
Miscellaneous engine components removal equipment
Engine maintenance stand
Engine storage racks
Engine shipping container
Engine handling fixture − slings, dolly
Engine skirt removal fixture
Engine skirt shipping container

Table 15-3 (Cont.)

Rocket Engines (Cont.)

      Engine servicing equipment
      Engine inspection equipment
      Engine component shipping containers
      Engine control system checkout equipment – electrical

Airbreathing Engines

      Access and inspection stands
      Engine installation and removal fixture
      Miscellaneous engine components removal equipment
      Engine maintenance stand
      Engine storage racks
      Engine shipping containers
      Engine handling fixture – slings, dolly, etc.
      Engine servicing equipment
      Engine component shipping containers
      Engine control system checkout equipment – electrical

Engine Outriggers

      Installation and removal fixture
      Storage racks
      Handling fixtures – slings
      Shipping container

Engine Outrigger Control System

      Access and inspection stands
      Special inspection tools
      Hydraulic cart adaptors
      Outrigger control system checkout equipment – electrical

JP-4 Tank

      Installation and removal fixture
      Handling fixture – slings, dolly
      Storage rack
      Shipping container
      Propellant sampling equipment

JP-4 Delivery System

      Access and inspection equipment
      Flushing system adaptors

Main Propellant Tanks

      Access and inspection equipment
      Installation and removal fixture
      Handling fixtures – slings, dollies, etc.

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Table 15-3 (Cont.)

Main Propellant Tanks (Cont.)

Storage racks
Maintenance and cleaning stands
Flushing and purging adaptors
Shipping containers
Insulation installation equipment

Main Propellant System

Access and inspection stands
Installation and removal fixtures for large manifolds,
valves, and flex joints
Special handling equipment – slings, dollies
Storage racks
Flushing and cleaning racks
Maintenance stands
Cleaning and purging adaptors
Shipping containers
Special valve servicing equipment
Valve, electrical activation checkout equipment
Propellant control system checkout equipment – electrical

General Propulsion Subsystem

Hydraulic flushing unit
Hydraulic operating unit
Flush and place equipment – fuel and LO_2
Leak check equipment
Bored scope and optical inspection equipment
Valve flow bench and activation equipment
Clean room equipment
Cleanliness sampling and analysis equipment
Degreaser and ultrasonic cleaning equipment
Pressure test equipment – pumps, etc.
Special hand tools
Inspection jigs, fixtures, and gages
General work stands
Part delivery carts
Prime mover for handling
Ordinance installation and checkout equipment
Fluorine passivation system
LN_2 coolant supply system
GN_2 supply and control unit
Helium purge system
Lubrication equipment
Jet engine starting units
Power drives

Table 15-3 (Cont.)

REACTION CONTROL SUBSYSTEM

Thruster Assemblies

Access and inspection equipment
Removal and installation equipment
Shipping containers
Thruster checkout probe system
Thruster optical alignment system

Tanks, Lines, and Valves

Removal and installation equipment
Flush, purge, and pressurize adapters

Control System

Control system checkout equipment — electrical

LIFE SUPPORT SUBSYSTEM

Air sampling and analysis unit
Waste products removal and cleaning system
Life support items loading system
Oxygen system sampling and analysis unit

GUIDANCE AND NAVIGATION SUBSYSTEM

Two-axis rate table — portable
Auxiliary power supply — gyros
Optical alignment system — refurbishment area
Optical alignment system — launch pad
Sensor calibration equipment
Radar calibration equipment
Special handling and installation fixtures

ELECTRICAL AND ELECTRONIC SUBSYSTEMS

Ground data management system
Tape fill unit
Computer monitor unit — ground terminal
Ground computer complex
RF checkout system
Fuel cell run and monitor unit
Battery chargers
Battery installation and handling equipment

Table 15-3 (Cont.)

ELECTRICAL AND ELECTRONIC SUBSYSTEMS (Cont.)

Fuel cell installation and handling equipment
Electronic special checkout equipment

AIR FRAME AND STRUCTURE

Aircraft jacks
Total spacecraft lifting equipment
Spacecraft delivery system
Boggie wheel sets
Prime mover
Launch adapter
Spacecraft support fixture
Spacecraft assembly alignment fixtures
Spacecraft components lifting sling sets
Spacecraft components storage racks
Access and inspection stands
Heat shield installation equipment
Heat shield inspection equipment
Spacecraft surface and structure inspection equipment
Payload installation – fixtures, slings

LANDING AND LAUNCH

Spacecraft cooling cart
Environment protection kit
Air conditioning unit
Ground power unit
Water loading cart
JP-4 fueling cart
LH_2 fill cart
LN_2 fill cart
ECS/EPS OX fill cart
ECS/EPS LH_2 fill cart
GH_e fill cart
GN_2 fill cart
Crew ingress/egress cart – portable
Spacecraft access equipment – portable
Vehicle erection system
Crew access and escape system – vertical
Air transportable flyback support equipment
Food loading unit

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15.6 LOGISTIC MISSION OPERATIONS

The three segments of flight operations indicated in Fig. 15-1 are ascent, orbital operations, and deorbit. Ascent flight operations, generally covered in Section 4 and Section 5, include the main flight performance and functional aspects such as trajectory constraints, throttling profiles, and staging through injection to 45-nm transfer orbit. The baseline mission for the ILRV study is Space Station/Base logistics.

A typical rendezvous profile for the advanced logistics system is shown in Fig. 15-12. A nominal orbit for station/base is 270 nm, a 55-degree inclination, and circular.

The Space Station is tracked to gain and maintain an accurate ephemeris. This information is processed to the Space Shuttle onboard computer, which then determines precisely the launch time that would allow rendezvous with the station/base. The method used accepts the station/base ephemeris and generates a flight plan. The information of use to the launch director consists of the earliest liftoff time, the liftoff time for minimum total velocity, the latest liftoff time, the liftoff time for a given azimuth, and the liftoff time for minimum total flight time.

Ideally, station/base ephemeris would be precisely known and it would be favorably positioned (phased) in the orbit that passes exactly through the launch site. For this ideal situation, rendezvous with the Space Station could be accomplished in less than 2 hours by using only 660 ft/sec incremental velocity (transferring from a 45/100-nm injection orbit). However, the effects of launch delays, launch guidance errors, station/base ephemeris determination inaccuracies, and unfavorable Space Station phasing would cause either increased propellant consumption or extended mission durations.

Rendezvous
Burn No. 2

Injection Into
Initial Transfer Orbit
(H = 45 nm)

Transfer Orbit
Injection
(Nth Crossing)

Transfer
Orbit

Parking Orbit
Injection
(H = 100 nm)

Rendezvous
Burn No. 1

Drift Orbit Injection
(H = 260 nm)

Parking Orbit

Drift Orbit

Target Orbit

Parking Orbit/Target
Orbit Line of Nodes
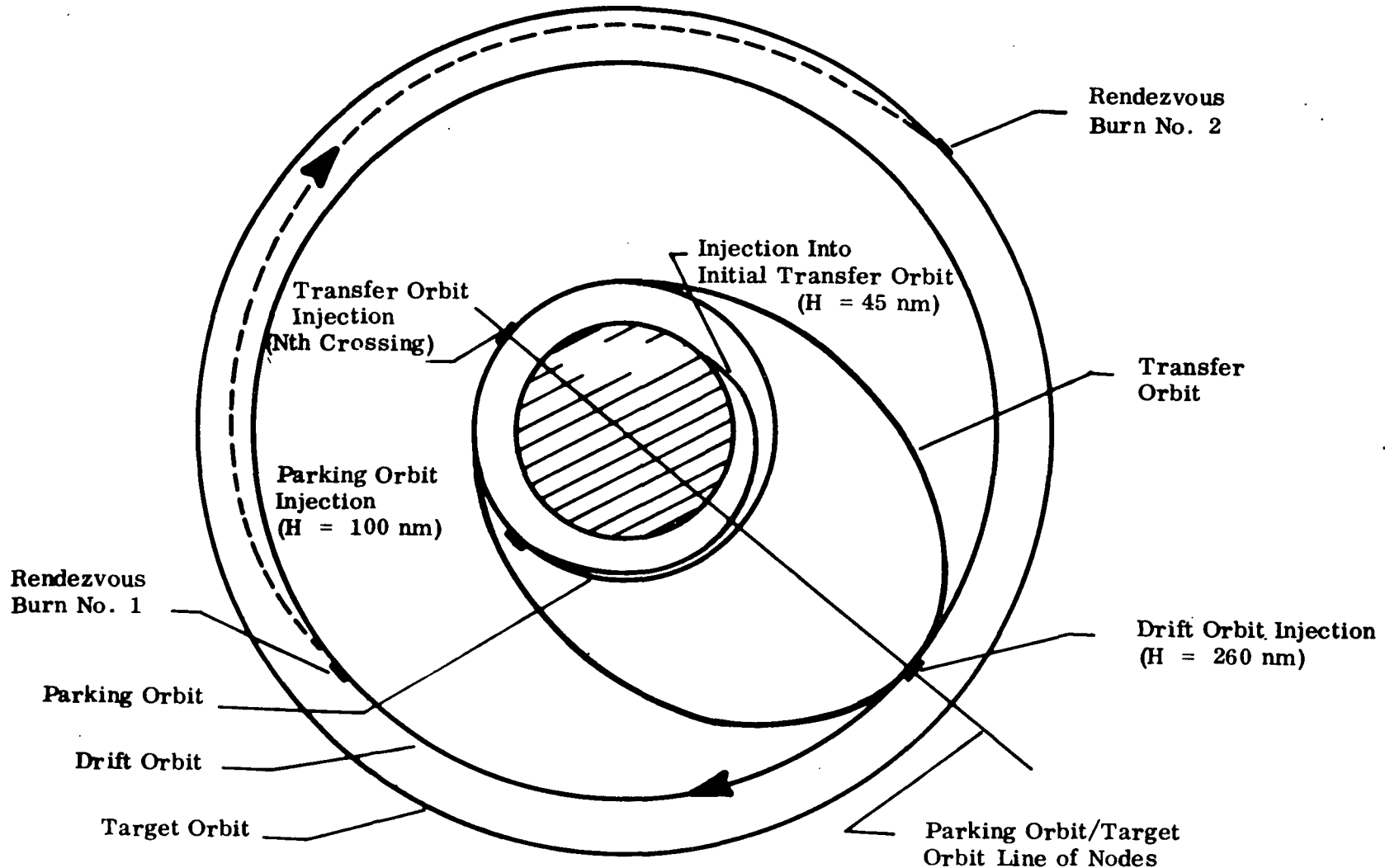
Fig. 15-12 Typical Rendezvous Technique – Logistics Mission

The rendezvous mission sequence of events begins with powered flight from the launch site at an initial launch azimuth of 141 degrees. Orbit injection occurs 370 seconds* later at perigee of an elliptical orbit having a perigee of 45 nm and an apogee of 100 nm. At this point, the shuttle is between 800 and 1000 nm downrange of the launch site.

The ΔV allotment for the transfer orbit is about 860 ft/sec, with 200 ft/sec allotted to account for launch dispersions and plane-change penalties.

The terminal rendezvous phase begins with a radar search pattern and with target lockon. This is generally considered to occur from a position below and back of the target, with a range closing rate of perhaps several hundred feet per second to a distance of about 1000 feet from the station/base. From this position to about 100 feet, the range closing rate is decreased to 10 ft/sec or less. At 100 feet, the docking phase begins.

A gross time line for a typical mission profile appears in Table 15-4.

The on-orbit stay times can vary between less than 1 day to up to 7 days, or even more, depending on the cargo, checkout, and return window phasing. A typical on-orbit event allowance for the crew and cargo transfer to and from the station/base and checkout for preparation to return events in slightly greater than 8 hours. Orbital standby for return phasing is dependent on the orbital position and velocity, landing site availability, and the Space Shuttle entry operating characteristics.

15.7 DOCKING OPERATIONS

Principal factors considered in docking operations are that maneuvers must not require special orientation or maneuverability of the station/base and that provision for both manual and automatic modes are provided at the option of the flight crew. Both hard docking and stationkeeping modes are also involved in the operations. Especially in the initial phases of the station/base program, the shuttle is substantially larger than the station and potential physical interference with antennas and other devices deployed from the station upon close approach by the shuttle is a hazard to this equipment.

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*Based on 3-g limited ascent trajectory.

Table 15-4

LOGISTICS MISSION EVENT TIMES

Time (hr)	Event	Time (hr)	Event
T −2	Begin fuel loading	T_O +22.0	Dock with Space Station
T −1.5	Activate fuel cells	T_O +144.0	Prepare for return
T −1.0	Crew ingress	T_O +166	Separate from Space Station
T 0	Lift off	T_O +166.7	Deorbit
T +0.05	Perform staging	T_O +167.3	Enter atmosphere
T_O +0.1	Perform orbit injection	T_O +167.8	Approach landing
T_O +0.8	Transfer to 100-nm circular orbit	T_O +168	Touch down
		T_B +0.15	Deploy cruise engines
T_O +5.0	Correct for boost and launch errors	T_B +0.17	Begin cruise to base
T_O +18.4	Transfer to 100/260 nm	T_B +1.50	Approach landing
T_O +19.2	Transfer to 260 nm	T_B +1.70	Touch down
T_O +20.0	Begin terminal rendezvous		

In general, shuttle maneuvers required to dock or to attach external modules to the station/base require very careful control; and collision in any respect involves potentially hazardous damage to the shuttle heat shield, engines, and control surfaces. Fully automatic docking systems have not yet been accomplished; they require precision laser or radar systems for sensing and complex vehicle onboard guidance and control and dynamics computations. Manual systems based on pilotage skills have been fully demonstrated on Gemini and Apollo vehicles, which are, of course, much smaller than the shuttle. With much larger masses, the inertial and attitude sensing skills of flight crew members may not suffice; and the larger offsets between eyeball, docking port, and vehicle center of gravity is expected to generate complex motions during fine maneuvers. Also, approach and docking are limited by visibility and illumination constraints and sun angles.

Use of manual control with large logistics vehicles is likely to result in apparent control axes cross couplings during docking pilotage. The consequent control disharmonies can produce greater propellant usage, time requirements, and error potential for the docking maneuver than flight experience to date would seem to indicate.

These effects will be accentuated whenever the docking capture mechanism is not within the field of view, and the pilot must use a docking target positioned at a location different from the docking contact point.

Figure 15-13 shows a control geometry reference system with orthogonal rotational axes intersecting at a common center of rotation, which coincides with the vehicle center of gravity, and parallel motion translation along these same axes with thrust aligned through the center of gravity. The characteristics of current manned space vehicles and logistics vehicles as currently conceived are approximated. In Gemini and Apollo, the docking mechanism is centered and the eyepoint would nearly coincide with the rotational center and with the docking line of sight along a control axis, normally roll.

As vehicles become large, system design influences dictate wide separations between these points; and this can lead to manual control disharmonies. For instance, when the eyepoint is located on the roll axis but separated from the center of rotation, pitch or yaw rotation appears to have heave or sway components of a magnitude increasing with eyepoint distance from center of rotation. In another example, separation of the docking mechanism center from the roll axis results in apparent pitch and yaw rotational changes at the docking contact point when the vehicle is rolled, with the magnitude increasing as a function of docking-mechanism center distance from the roll axis.

Other considerations, such as thrust misalignments and control sensitivities (for which considerable data and experience on manual control effectiveness exist), will interact and further affect these new problems.

Visibility for docking depends on three primary factors. First, the geometric relationships among target vehicle, docking vehicle, sun, and earth determine ambient illumination of objects within the field of view. Second, the shapes and surfacings of the two vehicles determine contrast within the field of view, particularly glare and shadow. Third, window position and field-of-view size and shape interacting with the first two factors determine occurrence of sun shafting through the viewport and scene veiling effects caused by scatter within the window media. Nonoptimum visibility increases
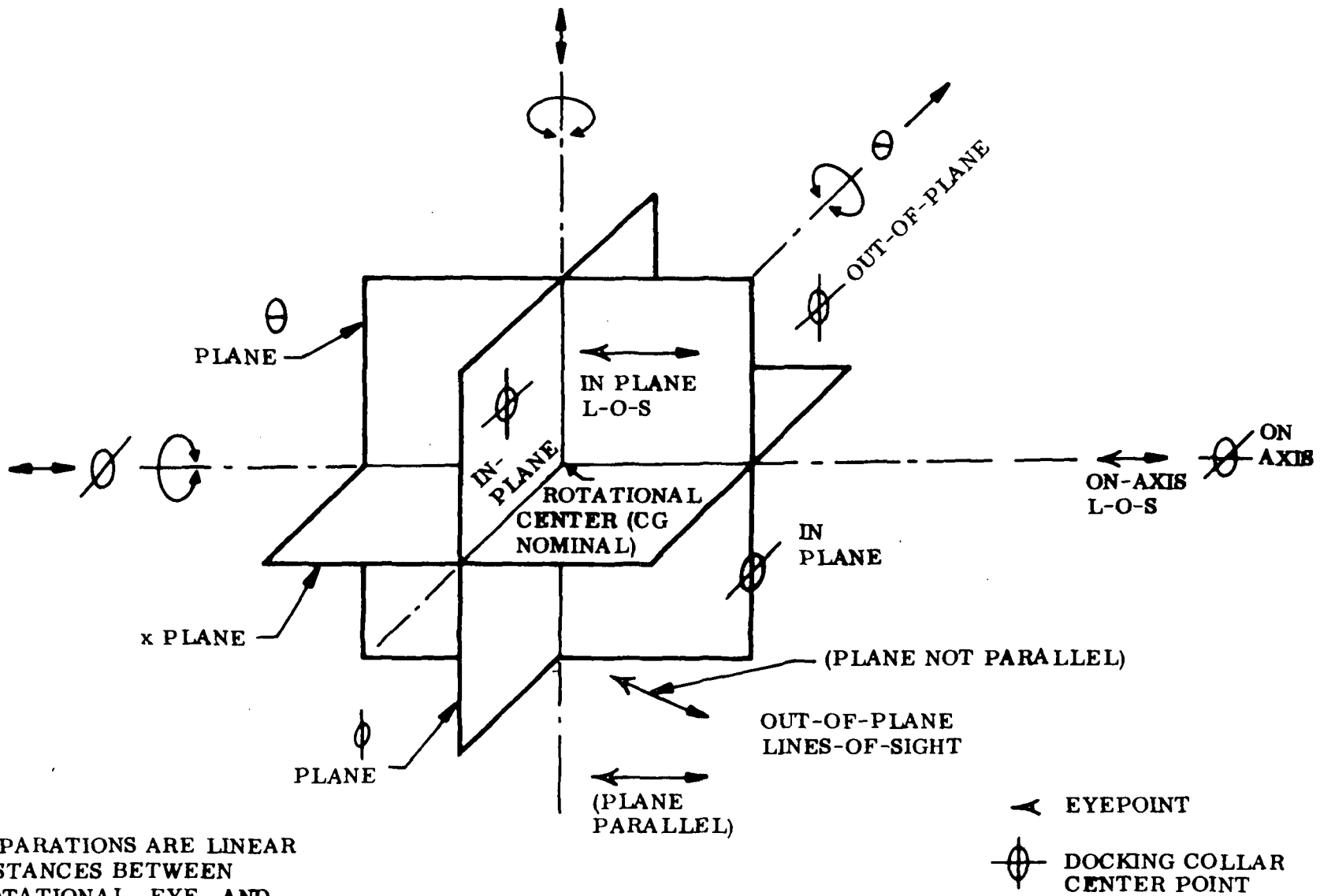
SEPARATIONS ARE LINEAR
DISTANCES BETWEEN
ROTATIONAL, EYE, AND
DOCKING MECHANISM
CENTER POINTS

Fig. 15-13 Docking Control Geometry

propellant consumption, error possibility, and safety hazards for the docking operation. Visibility considerations constrain docking port location and orientation, the capture mechanism design, and the target design at each docking location.

An analysis was made for a station/base oriented with solar arrays perpendicular to sun's rays and roll axis parallel to sun's rays. In this flight mode, a constant attitude is maintained with the Space Station roll axis parallel to the sun line and with the yaw axis maintained in the orbital plane.

The Space Station was modeled as a cylinder with solar arrays, as shown in Fig. 15-14. Cylinder side docking ports are assumed to be oriented with the docking maneuver centerline perpendicular to the cylinder centerline. The docking port centers are located in either the pitch or yaw planes or clocked to the midpoint (45 degrees) between the two planes. Cylinder end docking ports are assumed to be oriented with the docking centerline parallel to the Space Station centerline. The docking port locations and orientations include conditions in both Space Station configurations at their present level of definition.

The model used in the analysis for the logistics vehicle is a Space Shuttle docking in the normal nose-in mode and with current thermal control surfacings. Sun angles (sunlight incidence angle) between 60 and 140 degrees are assumed to be acceptable, where the sun angle is defined as the angle between the docking maneuver centerline (active docking vehicle roll axis) and the sun line and where a zero-degree sun angle has the +X direction toward the sun (Fig. 15-15). It is also assumed that sun-angle constraints are the same for all vehicle angles, where vehicle angle is determined by roll position with respect to the sun line. For sun angles of less than 60 degrees, sun shafting through the docking viewport or sun incidence causing veiling (light scatter in the viewport optics) are possibilities. For sun angles greater than 140 degrees, obscuration of the Space Station docking target and port by logistics vehicle shadow is a possibility. The simplifying assumption of a constant sun angle constraint for all vehicle angles would probably be inappropriate for the more complex shapes and docking concepts envisioned for the Space Shuttle.
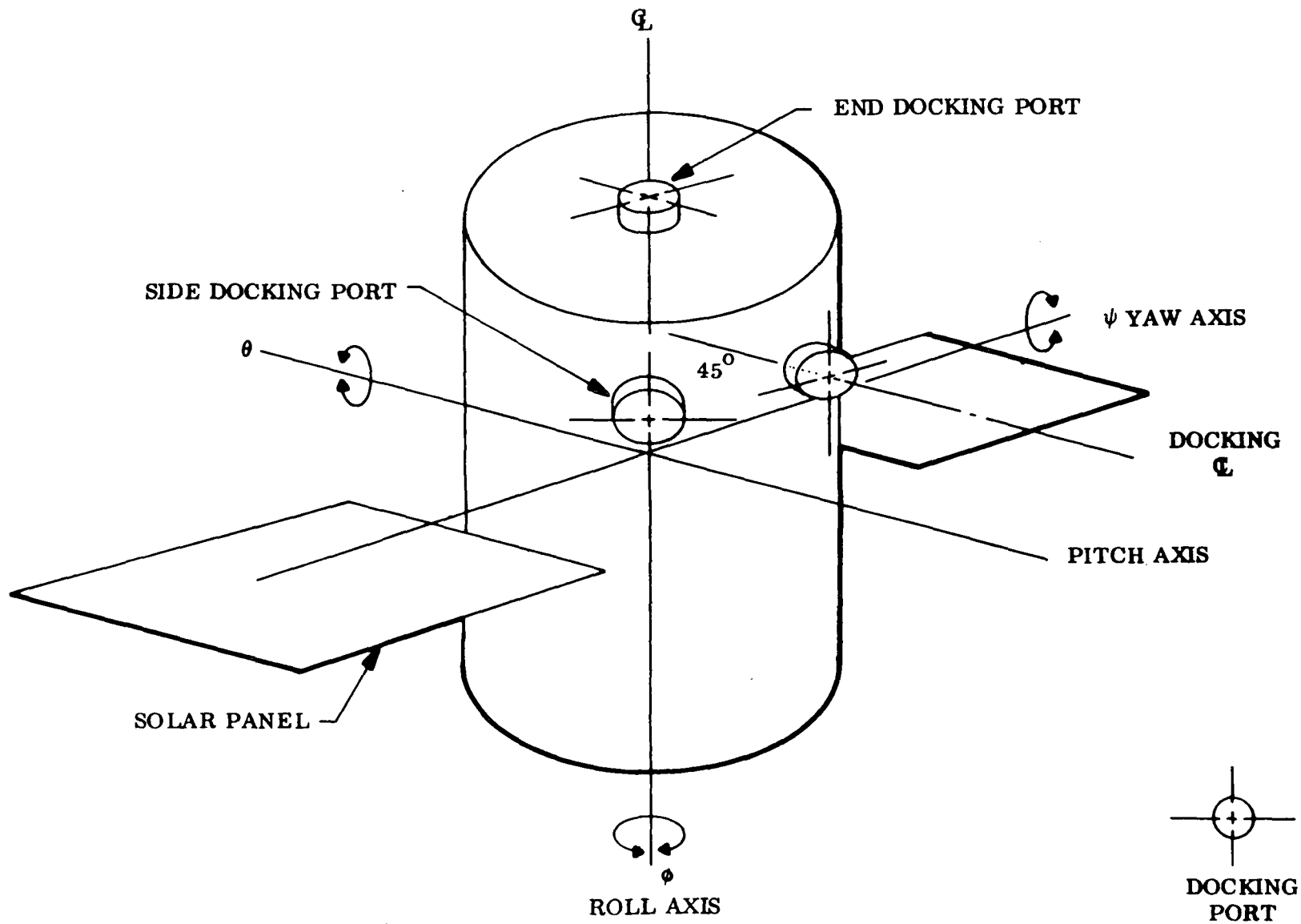
Fig. 15-14 Space Station Model

DOCKING C̶L −X

ACTIVE VEHICLE
(SHUTTLE)

÷X

DOCKING
PORT C̶L
STATION/BASE

60°

SUN ANGLE

80°

SUN LINE
LIMIT

ACCEPTABLE
ENVELOPE

SUN LINE
LIMIT

SUN
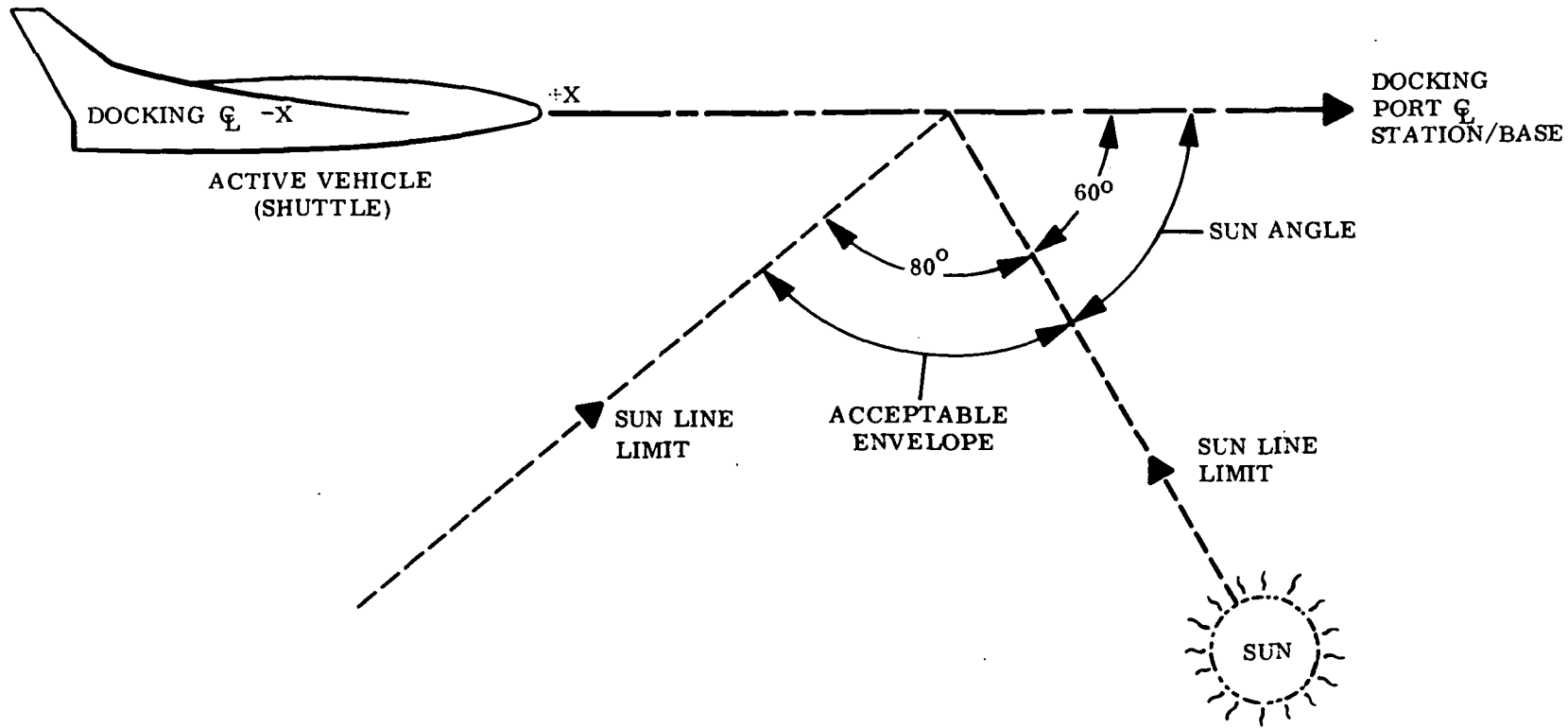
Fig. 15-15 Docking Sun Angle

The following assumptions were made:

- A 270-nm orbit at 55-degree inclination

- All docking in daylight

- All docking in zero-gravity flight modes

- No reorientation of the Space Station to accommodate the docking maneuver

- All external Space Station surfacings highly reflective (greater than 50 percent) and lambertian diffuse, except for the solar arrays with their solar cells 20 percent reflective and highly specular in mounts of matte-finished aluminum with cell-to-mount packing factors of .80 to .85

- Sky or dark earth background during the docking maneuver, with the docking port illuminated, in order of preference, by sun and earth reflection, sun only, or earth reflection only

- Flat Space Station surfaces preferable to curved surfaces in the active vehicle field of view during docking maneuvers a condition probably difficult to attain in many cases

This mode presents the simplest case for analysis because a constant attitude is maintained with respect to the sun line at all times. All cylinder-side docking ports are acceptable from a sun-line standpoint, because the sun angle to the docking vehicle in a normal docking maneuver is always 90 degrees.

All dark-end docking ports are therefore unacceptable on the daylight side of the orbit. The sun-end port with the docking approach parallel to the Space Station centerline is unacceptable, because the sun angle is 180 degrees, which is more than the maximum acceptable position of 140 degrees. Additionally, this docking approach is close to the glare envelope resulting from the highly specular solar array cells, because the plane of solar array is nominally perpendicular to the sun line as is the docking approach. However, sun-end approaches offset 45 degrees to the Space Station centerline result in a sun angle of 135 degrees, which is acceptable because it is less than the maximum acceptable angle of 140 degrees. This alternate approach, in which glare from the solar arrays is likely, is also well outside the envelope.

Both direct sun and earth albedo (sun reflection by earth) illumination of the Space Station docking target and port are desirable. The earth subtends ±68 degrees in a 270-nm orbit. A full earth would illuminate 316 degrees of the cylinder circumference at any point along its length. Earth albedo considerations provide no obvious criteria for docking port and target location. Earth-side docking maneuver approaches provide both the desirable dark-sky background and the optimum illumination conditions.

15.8 LOGISTICS CARGO TRANSFER

The common-carrier mode of operation conceived for the Space Shuttle system re-quires efficient use of a minimum fleet of vehicles in logistics support of Space Station/Base, as well as a broad spectrum of alternate missions. Fast turnaround in the recovery, maintenance, and re-launch cycle of the shuttle itself appears to be feasible from the operations standpoint discussed in preceding sections. Fast turn-around for payload transfer is equally important to economic effectiveness, both on the ground and in orbital operations. This leads directly to functional requirements for standardized container modules and pallets that decouple payload preparation cycles from the logistics flight scheduling on the ground and in orbit. Additional functional requirements for deployment of external stores and modules in the vicinity of the station/base, or externally attached in some instances, must be recognized.

The shuttle cargo bay, typically 15 feet in diameter, 60 feet long, and 10,000 cubic feet in volume, is similar to that of cargo aircraft in that it has no provisions for cargo accommodation other than hard points for primary structural support loads. Payload accommodation weights come out of the nominal 50,000-pound capability of the vehicle; and commercial and military air cargo experience in the Lockheed C-141, C-130, C-5A, and L-500 indicates a minimum of 1.25 pounds per cubic foot of total packaging wieght penalty for all types of cargo. This includes the weight of con-tainers for separation and structures to withstand basic flight and landing loads and to be retained within the airframe. Extrapolation for the shuttle indicates payload ac-commodation weights on the order 20 percent of the nominal 50,000-pound capability. Analysis based on the following considerations of payload accommodation and cargo transfer tend to confirm this.

Two principal modes of cargo accommodation and transfer to station/base have been considered. One is the unpressurized accommodation of bulk containerized cargo with a pallet or space frame accepting the structural loads; the other is a pressurized cylindrical module with bulkheads and internal racks. For the unpressurized pallet case, it is assumed that the station/base is fitted with a large airlock or hanger that can accept the entire pallet from the shuttle for subsequent unloading by the station/

base crew. The pallet would translate into the air lock in a rigid docked configuration. This case involves cycling of the air lock with each logistic operation.

The pressurized compartment of the alternate mode would include the necessary passenger accommodations on some flights in a mix with pressurized cargo. In a rigid docked configuration, the pressurized compartment would translate out of the payload bay under mechanical constraint and register with a hatch interface to accommodate pressurized transfer of personnel and cargo. An alternate stationkeeping mode would provide for the compartment to fly a short distance out of the shuttle bay and dock to the station/base by means of the reaction control system.

With the weight penalty for cycling the air lock included, typical weights associated with delivery of an unpressurized pallet range between 8,000 and 11,000 pounds, depending on whether a pump down cycle is used or makeup atmosphere is provided as part of the logistics payload. Delivery of a full-size pressurized module costs about 10,000 pounds for either the hard docked or stationkeeping modes. Thus, noncargo weight penalties seem to be on the order of 20 percent, making this a key aspect of future analysis to achieve effective use of the shuttle.

If the pressurized mode with the compartment docked to the station/base is assumed, a number of cargo handling considerations must be evaluated. Manual handling is limited by crew capabilities and time to small articles. Very limited NASA and DOD effort in zero-g cargo handling and transfer has been accomplished in the KC-135 aircraft simulation and in some underwater simulation. Guidelines indicated something less than a 24-inch cube for one man and a 20 x 30 x 40-inch object weighing no more than 250 pounds for two men.

Based on these assumptions, it would take approximately 200 trips over an average round-trip distance of 60 feet to empty the compartment loaded with 200-pound containers, or about four 8-hour shifts for two crewmen to handle the cargo. One conclusion is that the shuttle should not remain on orbit awaiting cargo transfer both up and down; another is that mechanical aids are required. Furthermore, one concept suggested for future study is to use the 15 x 60-foot pressurized cargo/passenger

module deployed from the shuttle as a semipermanent warehouse attached to the station/base, thereby eliminating a requirement to off-load cargo before the shuttle can deorbit and return. In this mode of operation, the cargo is off-loaded as needed and the cargo/passenger module filled with return cargo as needed in preparation for a return flight. The operations profile would then amount to an exchange with the shuttle of standardized up and down cargo modules on each logistic flight, allowing the shuttle to return immediately to earth.

A cargo handling system envisaged as an integral part of the pressurized module is illustrated in Figs 15-16 and 15-17. It provides restraint of the cargo containers at all times during the unloading operations; any shape of container can be handled. The cargo module has a center access tunnel; and the containers are mounted around the center tunnel, attached to the outer shell of the cargo module. The cargo handling system consists of a pallet to which the containers are secured for unloading. The pallet slides along on two sets of parallel rails. Each of the parallel end tracks, mounted in a circular end ring, can rotate around within the end ring; therefore, the rails and the pallet can be indexed under any container within the module. The rails and pallet are raised and lowered by an electrically driven, closed-loop cable system at each of the end tracks. An electric motor-driven roller system at one end ring provides the power to rotate the platform within the module.

Containers will vary in shape and size, but all containers will have to pass through a common hatch (5 feet in diameter at present). The containers may be 3 feet deep and may vary from 5 feet to 10 feet or more in length, depending on storage room volume and shape at the station/base. Their weights may range up to 2000 pounds. The container structure will have to take the load of the launch booster, typically 39 vertical, plus 2 g transverse. The cross-sectional shapes of containers may be circular, rectangular, or trapezoidal; and several shapes may be used within one module.

CARGO CONTAINER ATTACHMENT
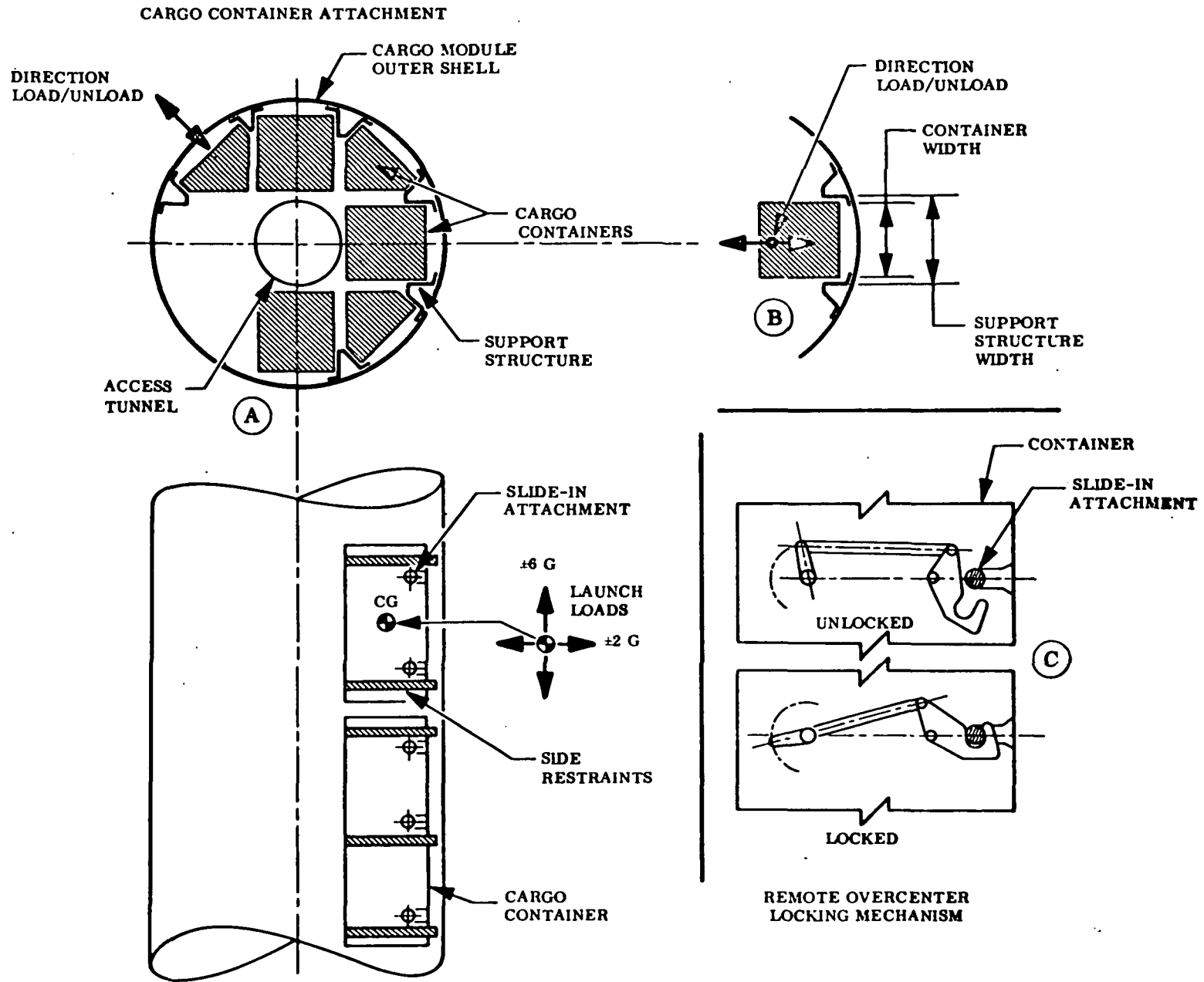
Fig. 15-16   Typical Methods of Cargo Container Attachment

REMOVAL OF CARGO MOUNTED ABOVE RAILS

RING

CARGO CONTAINER

END RING
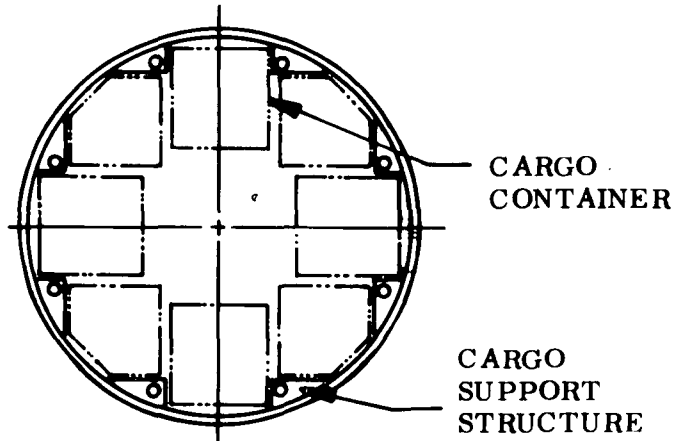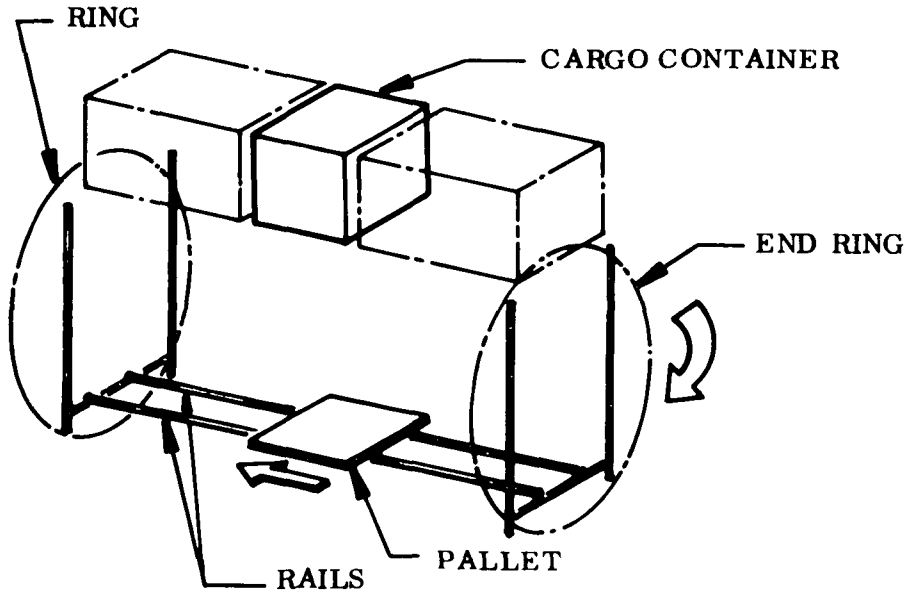
PALLET & RAILS RAISED.
CARGO CONTAINER ATTACHED TO
PALLET.   CONTAINER DISENGAGED
FROM LAUNCH SUPPORT STRUCTURE.

PALLET

RAILS

CARGO
CONTAINER

CARGO
SUPPORT
STRUCTURE
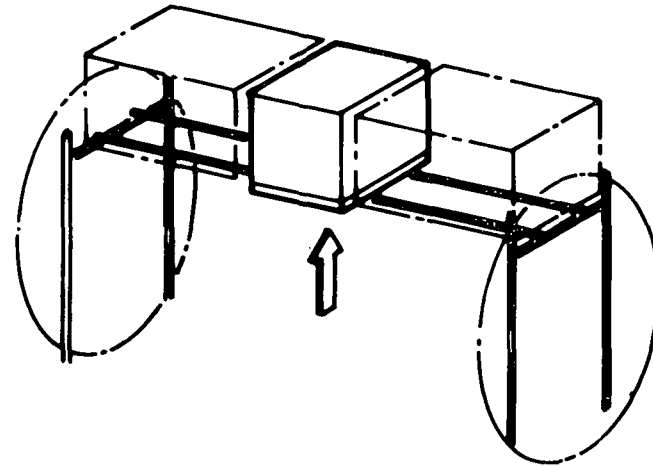
Fig. 15-17 Cargo Unloading Sequence

Section 16
TEST AND PRODUCTION

16.1 ENGINEERING TEST PROGRAM

Analyses indicate that application of aircraft buildup flight test concepts will permit schedule compression and reduce emphasis on ground testing while maintaining acceptable test risk.

The interaction between ground and flight testing is illustrated in Fig. 16-1, in which typical growth or learning curves are presented to relate design maturity to operational experience. The flight buildup test concept permits flight operational experience prior to completion of ground testing.

Major design, manufacturing, and development test elements are identified and interrelated to provide a preliminary assessment of scope for the test program illustrated in Fig. 16-2. The various elements and relationships are established on a basis of
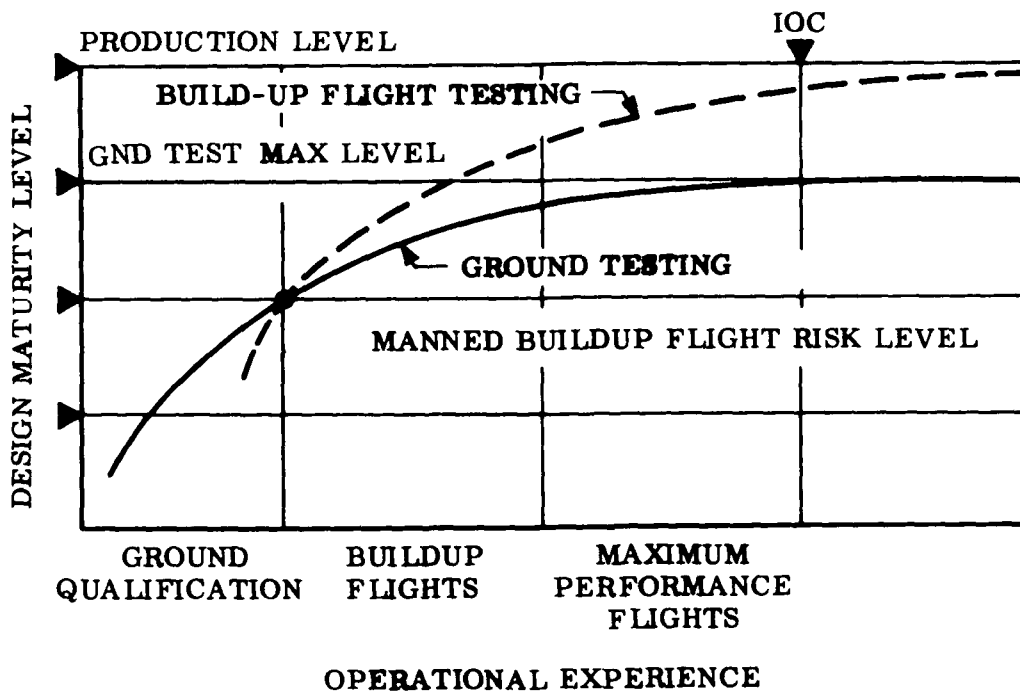
Fig. 16-1 Design Maturity vs Operational Experience
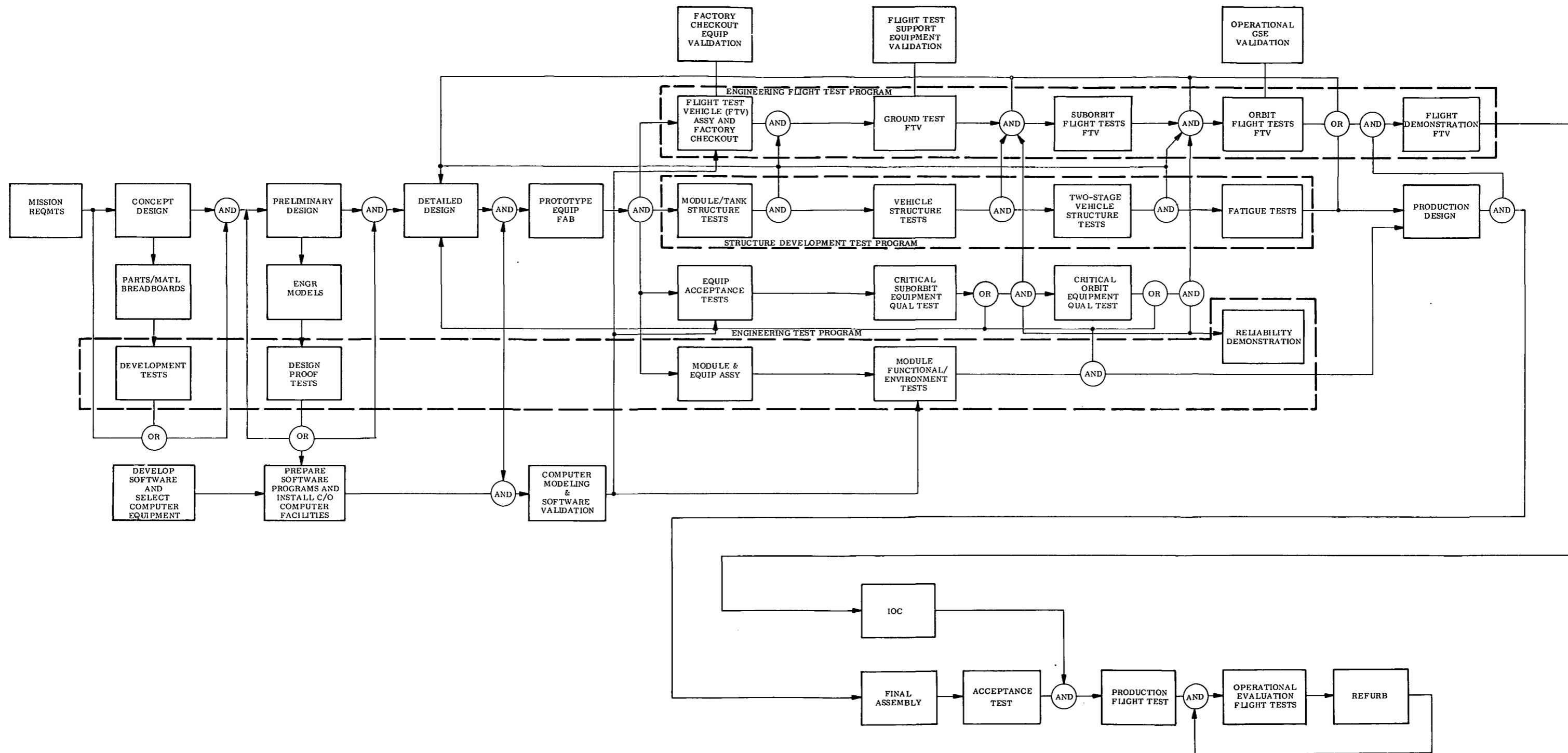
Fig. 16-2 Overall Test Program

attaining required levels of design maturity to support engineering flight testing and flight demonstrations. "And" and "or" gates are provided to indicate alternatives and interactions that may arise during the test program.

16.1.1 Engineering Flight Test Program

The test concept for the Space Shuttle, summarized in Fig. 16-3, is based on aircraft-type buildup flight testing to minimize test risk, to obtain design maturity through operational experience, and to permit schedule compression by flight setback in respect to availability of selected subsystems. Major program elements are indicated in Fig. 16-4.

In essence, the buildup testing concept calls for development of flyable airframe and propulsion subsystem based on low loads and minimum support subsystems. After initial familiarization to provide the pilot and test monitoring personnel with a feel for the vehicle, maneuvers and environmental loads are increased. Actual performance is monitored in respect to predicted performance, and inflight performance is restricted to safe stress levels. When data return indicates safe conditions, stresses are increased; if marginal conditions are disclosed, the vehicle is returned for retrofit. Thus, over-stress conditions are minimized and critical weakness is detected early and strengthened prior to catastrophic failure. All flights are conducted under conditions involving low flight risk. Proposed testing includes a 28-month flight test program, including performance demonstration, performed in parallel with ground qualification of noncritical items. Ground qualification of critical items is conducted prior to flight on an expedited basis.
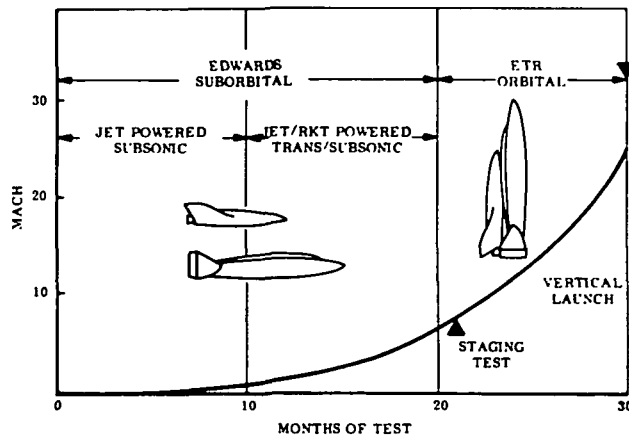
Fig. 16-3 Flight Test Concept

16-5
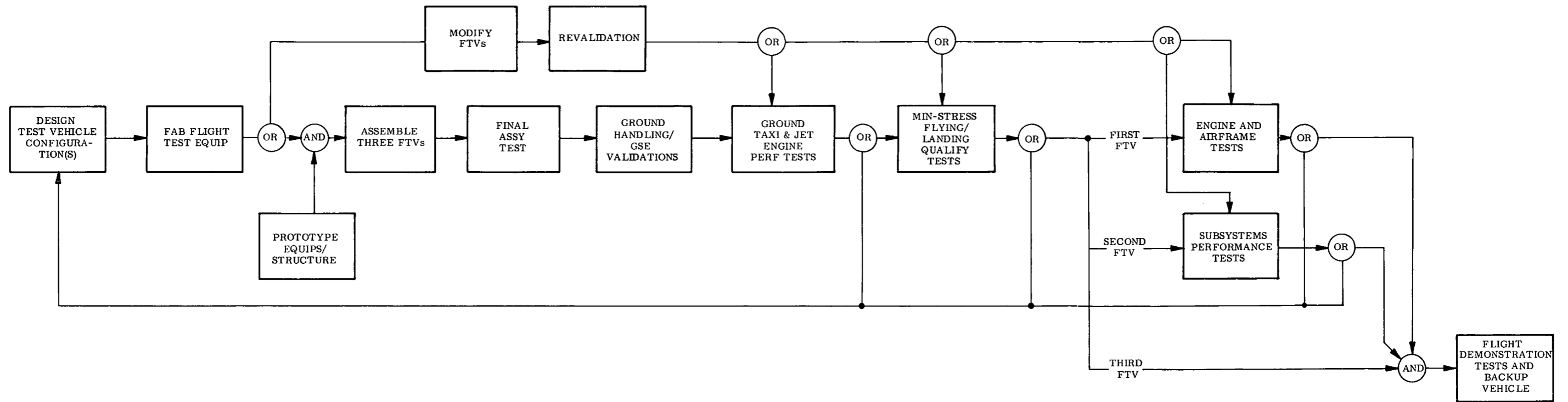
Fig. 16-4 Engineering Flight Test Program

16.1.1.1 <u>Suborbital Flight Test.</u> During suborbital flight tests, performed in the subsonic and low supersonic flight regimes, the go-around and self-ferry requirement will provide inherent capability of horizontal takeoff with jet engines and subsonic flight testing according to current aircraft practices. Performing these tests at Edwards Air Force Base would minimize test risk; but at the present time, there is uncertainty concerning achievable Mach levels under horizontal takeoff conditions.

Transonic and supersonic flight test requires rocket engine operation or equivalent thrust capability. While propellant weight can be substituted for payload weight, there are not enough data to predict propellant loading limitations because of permissible bending loads during horizontal vehicle operations; additionally, propellant flow problems and in-flight rocket engine start may preclude achieving higher Mach levels with horizontal takeoff and main-engine thrust.

Vertical takeoff of individual stages could be performed at Edwards AFB or the operational launch site. While Edwards presents minimum tesk risk, ETR has greater launch support capabilities. From either location, it is desirable to provide velocity buildup in incremental stages because of uncertainty in aerothermal load prediction techniques. State-of-the-art accuracy of thermal prediction and the supporting theory of turbulent heating entail an uncertainty in peak reentry temperature of approximately $200^{\circ}F$. Ground testing could reduce this uncertainty, but flight testing will be required to solve the problem of heat shield optimization.

A flexible heat shield design is proposed to permit use of alternate materials, such as ablators, LI-1500, or metallics. Initial hypersonic flights at lower heating levels may require radiative (nonablative) systems to provide the required thermal instrumentation accuracy. Initial trajectories will be sized to limit temperature; subsequent trajectories will be sized to provide temperature buildup from the lower levels. Vehicle return approaches will require restricted bank angles (low cross range) and angle-of-attack modulation to control aerothermal heating.

16.1.1.2 <u>Orbital Flight Testing.</u> Orbital flight testing, which is a continuation of suborbital testing, demonstrates abort modes, as well as operational mission capabilities. Major considerations are mated vehicle ascent; staging; booster return; orbit injection; rendezvous and docking; cargo transfer; communication link operations; deboost and reentry; and descent, flair, and landing. This should be coordinated with ground operational support and facilities capability demonstrations.

16.1.1.3 <u>Flight Test Instrumentation.</u> Since the flight test vehicles will serve in lieu of the ground test laboratory, they will require extensive modification from the IOC vehicle configuration to provide the required data return. This additional instrumentation cannot be provided by the production and data management systems planned for operational use.

Basic vehicle tests will require thousands of strain gages, themocouples, and pressure indicators, as well as numerous wideband accelerometers, acoustic devices, and sensitive calorimeters. Both in-flight and ground data recorders and displays are required for readout and data storage. Data storage must be identified and time correlated with vehicle flight profiles.

It is anticipated that new-generated avionics, associated electromechanical devices, and the various prime movers and slaves will require detailed data return, as well as normal input, output, and power consumption data from black-box and subsystem terminals.

Data return from both the basic vehicle and from critical components should be capable of indicating variables, relationships, degradation trends, and capability of the basic elements within the black boxes.

Early recognition of these instrumentation requirements would permit incorporation of development test instrumentation design at the time of first article manufacture, thereby providing higher quality data return at lower cost through the need for extensive engineering test modifications.

16.1.2 Engineering Ground Test Program

Ground testing includes all testing in support of hardware design.

16.1.2.1 Critical Equipment Qualification. Qualification testing, at the critical equipment and logistics supply level, covers electrical, mechanical, hydraulic, pneumatic, and pyro devices, according to detail specification requirements. Functional operation is verified under environmental stresses that exceed those of expected flight environments.

16.1.2.2 Structural Testing. The structural testing concept, illustrated in Fig. 16-5, calls for use of one complete vehicle, minus installation hardware, as a test specimen. At the lower assembly levels, structural components and mechanical features are subject to mechanical functional tests and static and dynamic loads tests. These items are repaired as necessary and assembled into modules or segments with hard interfaces suitable for industry-standard static and dynamic loads testing. In this test series, landing gear would be subjected to special deployment, retraction, and landing shock loads testing.

Following this series, modules are assembled to comprise the structural test vehicle (less landing gear).

The entire vehicle structure would be static-loads tested in a horizontal position under standard temperature. Then a longitudinal modal survey would be made with the vehicle in the vertical position. Localized heating and variable tankage simulation would be provided. Normal aircraft fatigue life tests would be performed on the vehicle in the horizontal position. Vertical and horizontal tests will be required for the booster, the orbiter, and the launch vehicle configuration.

16.1.2.3 Subsystem Engineering Testing. This testing is required to demonstrate equipment and subsystem functional capability and subsystem compatibility under environmental stress conditions. Test elements and interrelations are shown in Fig. 16-6.

Ground demonstration of equipment installations on booster and the orbiter impose severe requirements for environmental facilities and equipment because of the vehicle size.
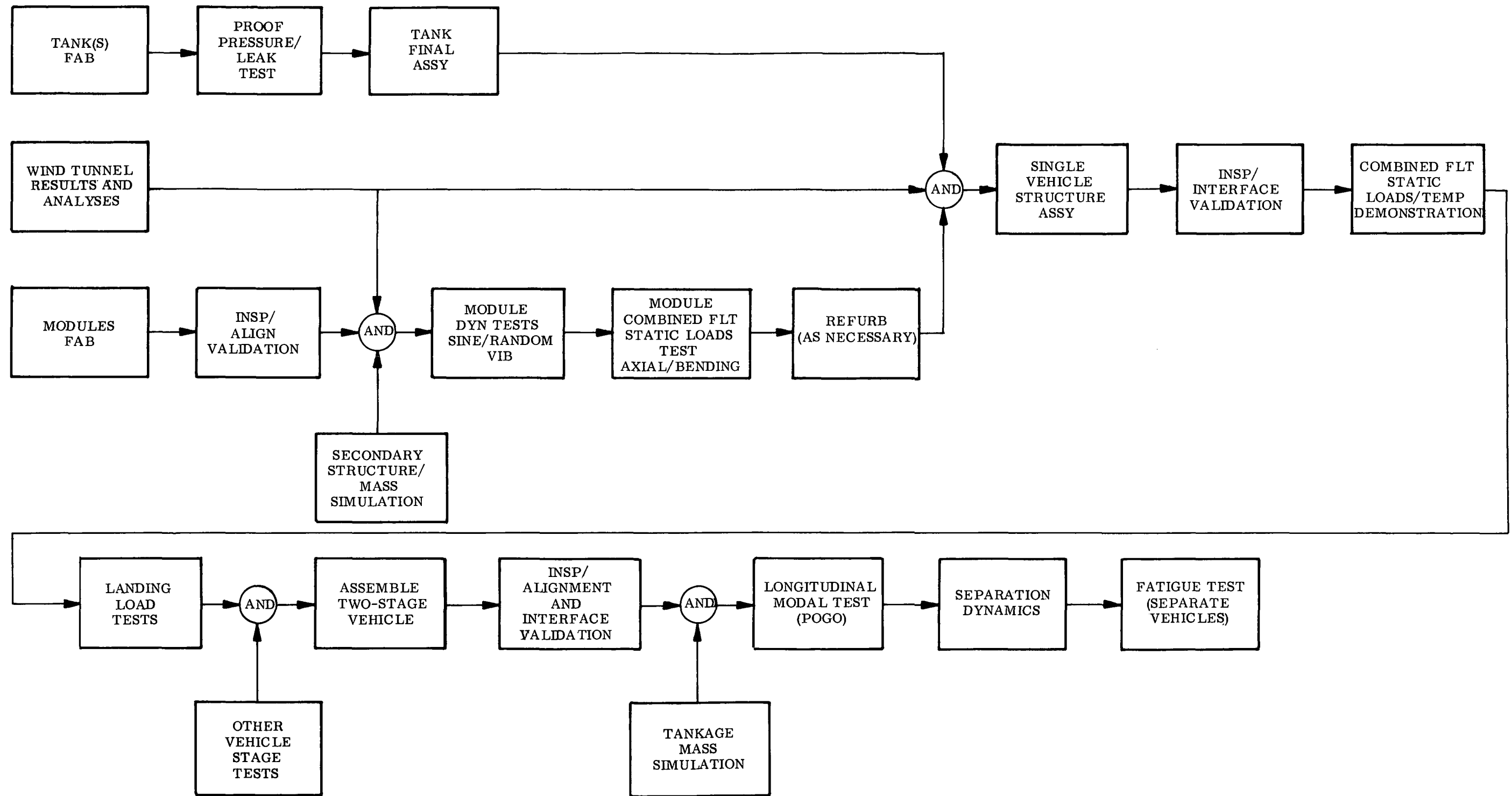
Fig. 16-5 Structural Testing Concept
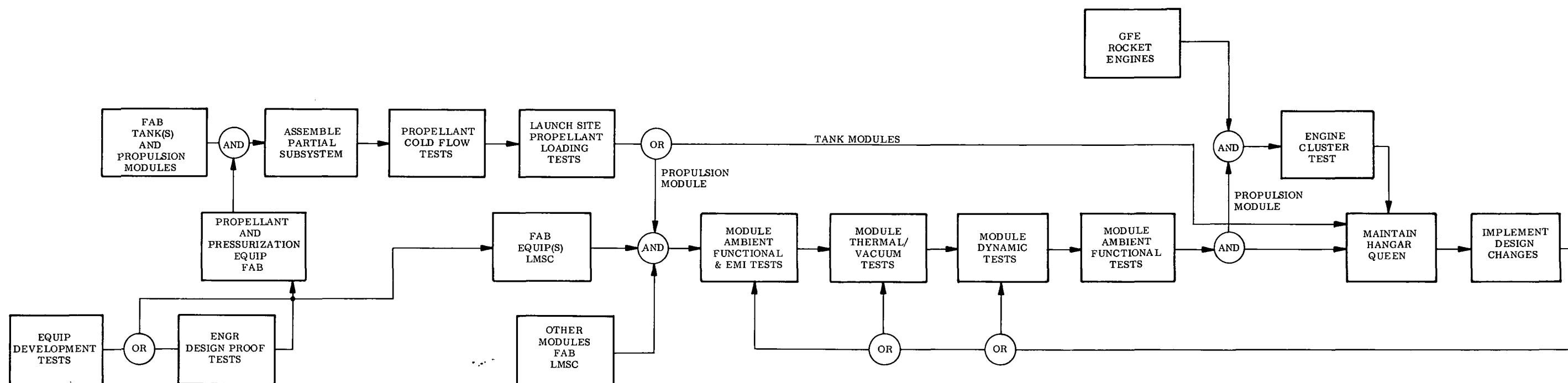
Fig. 16-6 Ground Environmental Test Program

Development of thermal/vacuum cycling and dynamic vibration capability to accommodate the entire vehicle would represent a major design effort. Therefore, it appears that a practical approach would be to establish installation qualifications at a major subassembly or module level, where the maximum module size is constrained by existing environmental stress capability.

In a strict sense, implementation of this concept would require deviation from standard ground test practice. Equipment installations in a given module will not in all instances comprise a functional subsystem; in general, systems compatibility tests cannot be performed. However, current second-generation and anticipated third-generation computer capability can contribute to acceptable test risk. A functional vehicle simulator (or functional computer software model) can be established; and actual hardware under test can be substituted during test to permit simulated subsystem and compatibility testing.

16.1.3 Factory Checkout and Flight Readiness

The factory test concept for prototype hardware, summarized in Table 16-1, is based on the pyramid concept, wherein, begining at the lowest material or part level, verification of required properties is made as manufacturing or assembly operations are performed and as additional functional capability is accumulated. This additional capability is then verified before this hardware is integrated into the next higher level assembly. This process is repeated until the finished product is obtained. Thus, after completion of the end item, functional tests would have been performed in the overall procurement/manufacturing process to verify every possible function. Typical test flow is shown in Fig. 16-7.

Table 16-1

FACTORY TEST CONCEPT

Vehicle Assembly Level

    Self-checking after onboard checkout (OBC) installation

    Prior to OBC installation, ground computer equivalent used

    No environmental tests at final assembly level

Module Assembly Level (including equipment installations)

    Module interconnected into centralized computer model of orbiter through a remote terminal and all functions exercised for systems compatibility and individual black box performance capability

Installation-Level Components

    Procured items:) interconnected with central computer (environmental
    Manufactured items:) stress for critical items provided)

    Environmental stress applied:

        temperature/altitude profiles

        dynamic vibration

The pyramid concept would ensure vehicle flight readiness if it were not for reliability and human error factors. To detect such error, critical parameters must be selected at all functional levels and made subject to test at the vehicle level. Thus, parts, materials, and assemblies are designated as critical according to mission criticality;

and provision is made to verify critical functions of these items at the end-item or vehicle level of assembly. Unfortunately, the use of breakout boxes and cables, interconnection of ground test equipment, etc., introduces additional operations and equipment into the system, which greatly enhances the probability of test-system error. Such error may be caused by erroneous readings of properly operating vehicle hardware or, more seriously, human mistakes in the test operation; or test equipment malfunction may cause vehicle hardware failure. Thus, in actual practice, tradeoffs must be made between testing error and depth of functional verification.

Experience has shown that testing system error is reduced by minimizing vehicle/ground equipment interconnections and access to the vehicle and standardizing the test methodology and instrumentation setup to obtain consistent data return throughout the testing system.

Based on these criteria, onboard checkout is recommended. This involves installation of onboard command and control capability for exercising vehicle functions at all desired levels (normally to include all critical functions and to include input/output verification at the logistics replacement level) and simulation for flight conditions such as gyro displacement and earth reference. Detailed features are tabulated as follows:

- Ground-to-vehicle connections — limited to power sources for electricity, hydraulic and pneumatic connections, air conditioning, and telemetry transmission capabilities
- Onboard design capable of commanding at the logistics replacement level
- Onboard interpretation of command response on a qualitative basis
- Variable data-return to ground via RF link for selected critical functions
- Vehicle-to-space station to ground RF link for checkout of station to earth flights
- Computer control of onboard test command capability
- Computer analysis of data return, comparison with fail-or-pass criteria, and fail-or-pass decision

16.1.3.1 <u>Vehicle Systems Test</u>. Vehicle final checkout with the OBC computer is proposed to establish end-to-end performance capability of each vehicle subsystem, to establish subsystem compatibility, and to establish vehicle-to-launch pad compatibility.

COMPONENT/BENCH LEVEL TESTING

1. ELECTRONIC BLACK BOXES
2. JUNCTION BOXES ELECTRICAL/ELECTRONIC
3. WIRE HARNESSES
4. TRANSMISSION LINES
5. ANTENNAS
6. SERVO CONTROL ASSEMBLYS
7. MECHANICAL COMPONENTS
8. ACTUATORS
9. COMPONENT PRESSURE & LEAK TESTS
10. TANK/PRESSURE VESSEL TESTING
11. STRESSED AIRFRAME & STRUCTURAL COMPONENTS
12. VALVES, COUPLINGS, REGULATORS, ETC.

TEST REQUIREMENTS

FUNCTIONAL TEST
  STATIC, QUASI-STATIC, DYNAMIC

ENVIRONMENTAL TEST
  TEMPERATURE, ALTITUDE, VIBRATION

PRESSURE TEST
  LEAK CHECK

P/L BAY SECTIONS
(SEG 430, 420, 410)

PROOF PRESSURE & LEAK TESTS
  TANKS
  LO_2
  LH_2
  LONG LIFE LO_2
  LONG LIFE LH_2
  RETRO

TEST EQUIPMENT
  TANK TEST COMPLEX
  OPTICAL ALIGNMENT EQUIP.

VERIFICATION CHECKS
  L/G INTERCONNECTS
  P/L INTERCONNECTS
  STRUCTURAL INTERFACE
    ALIGNMENTS
  TANK INSTALLATION
  TRANSFER VALVES &
    PLUMBING
  P/L ATTACH POINTS
    VERIF

P/L BAY SECTION
(SEG 400)

FUNCTIONAL TESTS
  TANKS, L/P & LEAK TESTS
  PAYLOAD SECTION FIT & CLEARANCE CHECK
  TRANSFER SYSTEMS FUNCTIONAL TEST
  PAYLOAD SECTION INTERFACE TESTS
  LANDING GEAR ACTUATION & CONTROL CHECKS

TEST EQUIPMENT REQUIRED
  OPTICAL ALIGNMENT EQUIPMENT
  PRESS & LEAK CHECK TEST FACILITY
  ACTUATOR INITIATION EQUIPMENT
  AUTOMATIC CIRCUIT ANALYZER
  COMPUTER SUBSTATION

FWD SECT.

FUNCTIONAL TESTS
  FWD SECTION ALIGNMENT CHK
  CREW COMPARTMENT FIT & CLR.
  CREW COMPARTMENT INTERFACE
  ECS FUNCTIONAL TEST
  REACTION CONTROL SUBSYS CHK
  NAV-AID, AVIONICS INSTALLATION &
    FUNCTIONAL CHECK
  FUEL CELL SYSTEMS CHK.
  NOSE CONE FUNCTIONAL TEST
  POWER SYSTEMS CHECK
  CREW COMFORT SYSTEMS TEST
  APU TESTS

TEST EQUIPMENT
  COMPUTER SUB-STATION
  AUTO-CIRCUIT ANALYZER
  INTERFACE & ADAPTER EQUIP.
  ECS TEST COMPLEX
  PNEUMATIC TEST CONSOLE
  MECHANICAL COMPONENT
    INSTRUMENTATION
  POWER SUPPLY TEST CONSOLE
  OPTICAL ALIGNMENT EQUIPMENT

PROP BAY ASSY

FUNCTIONAL TESTS
  ROCKET ENGINE MTG BRKT ALIGN.
  TURBO-JET ENGINE ALIGN.
  ENGINE CONTROLS & INSTR LINES
    & COMPONENTS
  THRUSTER ALIGNMENTS AND
    FUNCTIONAL TEST
  CONTROL SURFACE DRIVE ELECT.
  RATE GYRO OPERATIONAL TEST
  THRUST VECTOR CONTROL ELECT.
  SERVO-SYS TEST & CHECK-OUT

TEST EQUIPMENT
  SPECIAL ALIGNMENT FIXTURES
  COMPUTER SUB-STATION
  STIMULI & CONTROL CONSOLE
  SERVO-SYSTEMS TEST CONSOLE
  INTERFACE AND ADAPTER EQUIP.

ELEVON **VERTICAL FIN ASSY**

FUNCTIONAL TESTS
  ATTACH POINT VERIFICATION
  WIRE HARNESS CHECKOUT
  CONTROL SURFACE RIGGING CHK
  SENSOR INSTALLATIONS & CHECKOUT
  ACTUATOR TESTS

TEST EQUIPMENT
  AUTOMATIC CIRCUIT ANALYZER
  SENSOR STIMULUS & READOUT EQUIP
  RIGGING TEST FIXTURE
  MATCHMATE VERIFICATION FIXTURE

AFT FWD SECT.
(SEG 340)

FUNCTIONAL TESTS

ELECTRONIC SUBSYSTEMS
  NAVIGATION
  FLIGHT CONTROLS
  ENVIRONMENTAL CONTROL SYST.

VERIFICATION OF INTERFACE POINTS
  PRESSURE/LEAK CHECKS
  ELECTRICAL INTERFACE CHECKS
  STRUCTURE ALIGNMENT VERIF.

TEST EQUIPMENT
  CENTRAL COMPUTER COMPLEX
  COMPUTER SUBSTATION
  HYD/PNEU PRESS, SOURCES AND
    MONITOR EQUIPMENT

NOSE ASSY
(SEC 310)

FUNCTIONAL TESTS
  NOSE CONE TRANSLATION CHECK
  SENSOR LOCATION & VERIFICATION CHECKS
  NOSECONE INTERFACE TESTS
  NOSECONE STRUCT ALIGNMENT TESTS
  WIRE HARNESS INSTALLATION TEST

TEST EQUIPMENT
  INTERFACE FIXTURE
  EXTENSION TEST CONSOLE
  CIRCUIT ANALYZER
  ACTUATOR CONTROL CONSOLE
  SENSOR SIGNAL STIMULUS & SIG. COND.
  COMPUTER SUB-STATION

ENVIRONMENTAL TEST COMPLEX

CENTER SECT.

FUNCTIONAL TESTS
  CENTER SECT. INTERFACE VERIF
  CENTER SECT. ALIGNMENT VERIF
  TRANSFER SYS TEST
  GUID/NAVIG/F/L COMPUTER
    INSTALLATION TEST
  SENSOR TESTS & VERIFICATION
  NAVIG AIDS OPERATIONAL TEST
  FLIGHT INSTRUMENT ELECT.
  HG ACTUATION & CONTROL CHK.

TEST EQUIPMENT
  COMPUTER SUB-STATION
  SIGNAL SIMULATION CONSOLE
  SIGNAL CONDITIONING EQUIP.
  AUTO-CIRCUIT ANALYZER
  OPTICAL ALIGNMENT EQUIP.
  INTERFACE & ADAPTER EQUIP.
  PRESSURIZATION & LEAK CHECK
    EQUIP.

CENTER – P/L SECT. MATE

FUNCTIONAL TESTS
  INTERCONNECT & INTERFACE VERIF
  VERIFICATION OF INTERCONNECTED
    SYSTEMS
  MATCHMATE OF P/L & CENTER
    SECTION JOINT
  OPTICAL ALIGNMENT OF ASSY.

TEST EQUIPMENT
  INTERFACE & ADAPTER EQUIP.
  COMPUTER SUBSTATION
  OPTICAL ALIGNMENT EQUIP.

FWD/AFT SECT. MATE

FUNCTIONAL TESTS
  FUEL MANAGEMENT SYSTEMS TEST & CHECKOUT
  INTERFACE CHECKS & ALIGNMENTS
  INTEGRATED SYS TEST, ELECTRONIC/AVIONIC
  ELECTRICAL VERIFICATION TESTS

TEST EQUIPMENT
  INTERFACE & ALIGNMENT TEST FIXTURE
  COMPUTER SUB-STATIONS
  OPTICAL ALIGNMENT EQUIPMENT
  PRESSURE/LEAK CHECK EQUIPMENT
  SIGNAL SIMULATORS & STIMULI CONSOLE
  AUTOMATIC CIRCUIT ANALYZER

ELEVON/VERTICAL FIN MATE

FUNCTIONAL TESTS
  LIFE SUPPORT SYSTEMS
  HYDRAULIC SYSTEMS CHECKOUT
  FUEL SYSTEMS

FINAL ASSEMBLY

FUNCTIONAL TESTS
  GUIDANCE SYSTEMS
  FLIGHT CONTROL SYSTEMS
  FAILURE DETECTION SYSTEMS
  ELECTRICAL POWER SYSTEMS
  MECHANICAL/STRUCTURAL ALIGNMENTS
  POWER PLANTS
  CONTROL SURFACES

TEST EQUIPMENT
  CENTRAL COMPUTER COMPLEX
  HYDRAULIC SYSTEMS TEST CONSOLE
  HYDRAULIC FILL & BLEED CART
  PNEUMATIC SYSTEMS TEST CONSOLE
  PRESSURE SOURCE
  ELECTRICAL SYSTEMS GROUND POWER
    UNIT
  OPTICAL ALIGNMENT FACILITY
  RF SYSTEMS TEST CONSOLE
  DUMMY LOADS
  SIMULATORS
  INTERFACE AND ADAPTER EQUIPMENT

Fig. 16-7 Typical Spacecraft Factory Test Plan

The onboard computer is supported by a ground centralized computer, which operates in parallel with the vehicle unit to verify OBC capability.

16.1.3.2 Module (Segment) Level Tests. The manufacturing assembly sequence provides for producing a group of modules prior to final assembly. Where possible, equipment and critical component installation should be performed at the module level. A standard computer software model should be constructed for the entire vehicle for central computer programming. As each module is subjected to electrical/electronic test, the electrical interface sould be substituted into the computer model and all installed equipment should be exercised to establish functional capability after installation and to establish subsystem compatibility. Where practical, pneumatic and hydraulic leak checks should be performed.

16.1.3.3 Equipment Tests. Functional performance should be verified before equipment is assembled into modules. Because it is anticipated that the logistics replacement will be at the equipment level, complete verification of performance and interchangeability should be performed. It is anticipated that a large number of contractors will participate in the testing, and this will compound control of functional interfaces and use of standard methods.

Use of the centralized computer with remote terminals located at the equipment supply point to perform equipment interface testing and subsystem compatibility testing by substitution into the vehicle model appears to offer maximum test control over quality and interface compatibility.

16.1.4 Test Support Requirements

Major test specimen requirements are summarized in Table 16-2.

Table 16-2

TEST SPECIMEN REQUIREMENTS

Test Specimens	Quantity	
	Triamese	Two-Stage
Suborbital flight test vehicle		
Boosters	3* ea	3* ea
Orbiters	3* ea	3* ea
Orbital flight test vehicle (FTV)		
Boosters	4 ea	2 ea
Orbiters	2 ea	2 ea
Engineering ground test vehicle (ETV)		
Booster	1 ea	1 ea
Orbiters	1 ea	1 ea
Structural test vehicle		
Booster	1 ea	1 ea
Orbiters	1 ea	1 ea
Prototype subsystem hardware		
Booster, FTV, and ETV	6 sets	4 sets
Orbiter, FTV, and ETV	4 sets	4 sets
Ground qualification	1 set	1 set

*Two vehicles transfer to orbital test phase.

16.1.4.1 <u>Test Facility Support</u>. Major facility support requirements are summarized in Table 16-3 for ground testing and in Table 16-4 for flight testing. Factory test will be accomplished on an in-line basis.

These requirements are based on the assumption that the ground test facility is located near the final assembly building and that there will be no problem in delivering the flight test vehicles to the engineering flight test site.

16.1.4.2 <u>Development Test Equipment Support</u>. The following points are significant to test equipment support:

- There will be a relatively few vehicles; therefore, extensive factory production equipment is undesirable.

- Test equipment ought to be available at the assembly facility to confirm suspected failures and integrety of new assemblies at the time of installation.

- System test equipment must be capable of assessing performance of onboard checkout equipment and testing the vehicle with an RF data link with no hardwired lines except for power, hydraulics, and penumatics.

- Use of similar (preferably the same type) equipment at all test facilities should be considered as highly advantageous.

- Test equipment must support engineering development test by providing and storing internal fault-isolation and degradation data from critical components.

Test equipment support is required at supplier, fabrication and assembly, and engineering flight test facilities. During development activities, correlation of data return from these locations is highly desirable to aid in developing design maturity.

Three basic test console concepts considered for the various in-process locations are manual test consoles, automatic consoles of custom design with a dedicated application, and a universal test station with capacity to support the maximum scope of test. Trades for these approaches are given in Tables 16-5, 16-6, and 16-7. It is concluded that the universal test station approach would offer the best technical approach, as summarized in Fig. 16-8.

This approach represents a solution for meeting test objectives by implementing the universal test station configuration as compatible versions at launch, factory, and vendor installations. The launch and system configuration represents a complete

Table 16-3

FACILITY REQUIREMENTS – GROUND TEST

Function	Facility Requirements	Need Date	Equipment Installation Completion Date
Vehicle Tests			
Structural tests: Static/dynamic load horizontal mated vehicles vertical model determination horizontal separation	Test building – 200 wide, times 300 long, times 300 high, Seismic block with tiedown and 250-ton bridge crane; thermal heating capability	Mar 73	Oct 73
Combined engine cluster (aft end of booster)	Test stand, cryogenic feed, flame bucker for down thrust	Nov 73	Apr 74
– and –			
Propellant cold flow (without tanks)		Sept 72	Jan 73
Module Tests	Environment chambers, static and dynamic load sources	(Exist) Oct 72	(Exist) Jan 73

Table 16-4

FACILITY REQUIREMENTS – FLIGHT TEST

Function	Facility Requirements	Need Date	Equipment Installation Completion Date
Suborbit Test Booster test support and hangar space (2 vehicles)	120,000 sq ft	Dec 73	Feb 74
Spacecraft test support and hangar space (2 vehicles) flight test runways and operational test range hydraulic, pneumatic, vacuum, and instrumentation labs white room and life support lab data acquisition and reduction	97,000 sq ft	Dec 73	Feb 74
Orbit Test Two-Stage vehicle	operational launch site capability to provide adequate orbit test support	Dec 74	Aug 75

Table 16-5

VENDOR TEST OPERATION TRADEOFFS

Remote Access to Central Software Vehicle Simulation Model	
Advantage	Disadvantage
Uniformity of test data management program	High reprogramming costs to standardize computer network for software compatibility
Vendor-Supplied Development Test Equipment	
Advantage	Disadvantage
Lower equipment investment	Difficult test data reconciliation
Factory Derivative/Standard Automatic Test Station	
Advantage	Disadvantage
Same design (basic architechure) as factory and launch systems test equipment	Manufacturing cost to replicate test station at vendor facilities

Table 16-6

FACTORY TEST OPERATION TRADEOFFS

Manually Operated Test Stations	
Advantage	Disadvantage
Low implementation cost	High human contamination of test data
Automatic Dedicated Test Station(s)	
Advantage	Disadvantage
Highly effective for high- production rate programs	Inefficient design/equipment utilization in low production rate programs
Automatic Universal Test Station	
Advantage	Disadvantage
Single test station accommodates multiple test applications	Requires development of complex time- shared software programs

Table 16-7

LAUNCH DEVELOPMENT TEST OPERATION TRADEOFFS

Dual Program Execution — Flight/Ground Model

Advantage	Disadvantage
Proven concept, minimal development costs	Requires complex comparative hardware/software mechanization

Factory–Compatible Dynamic System Stimulation

Advantage	Disadvantage
Very closely resembles operational flight environment	Requires more study to efficiently exercise vehicle test instrumentation transducers

FACTORY

LAUNCH

VENDOR

SYSTEMS CHECKOUT

INTEGRATED SYSTEM TEST

● RF ENVIRONMENT

OBC EVALUATION/CERTIFICATION

● VEHICLE SYSTEMS SIMULATION

SUBSYSTEM SUBSTITUTION

● SIM. MISSING EQUIPMENT

MALFUNCTION ISOLATION

● SUBSYSTEM LEVEL DIAGNOSIS

COMPONENT DIAGNOSIS

SUBSYSTEMS/TEST

SUBSYSTEM CHECKOUT

● UNIVERSAL TEST SYSTEM

VEHICLE SIMULATION

● SUBSYSTEM COMPATIBILITY

● OBC COMPATIBILITY

MALFUNCTION ISOLATION

● LRU LEVEL DIAGNOSIS

COMPONENT DIAGNOSIS

VENDOR TEST

SUBSYSTEM DEVELOPMENT

● WORST-CASE PARAMETERS

VEHICLE SIMULATION

● MINIMUM MODEL

MALFUNCTION ANALYSIS

● ERROR PROPAGATION

COMPONENT DIAGNOSIS

Fig. 16-8 Ground Test Operational Plan

design, including provision for integrated system checkout as well as subsystem checkout and substitution. By using a lesser configured production version, just the universal system checkout capability would be provided for factory area; and a minimal configured economy version would be used at all applicable vendor facilities. Thus, one basic design calling for identical operational techniques and applicable test procedures becomes attractive.

The test station could be configured in a modular fashion with standard building-block hardware, a patchboard, a standard unit under test interface, programmed computer control, and data processing hardware (Fig. 16-9). Since there is a definite commonality between package/subsystem and total integrated system stimuli and measurement hardware, a growth version of the same equipment becomes ideal for applications from vendor through launch facilities.

16. 1. 4. 3 Dynamic System Stimulation. An automatic test system for an assembled Space Shuttle should accomplish the following basic tasks:

- Establish end-to-end performance capability of each vehicle subsystem
- Establish subsystem compatibility
- Establish subsystem hardware status
- Establish internal faults for engineering test

To perform these tasks, the checkout equipment should have some means of testing all critical avionic systems operating under simulated flight conditions. The scope of testing can be broadened to include the following tasks:

- Substitute missing subsystem components during avionics assembly, development, and system integration
- Aid in troubleshooting by simulating dynamic inputs to units under investigations and isolating faulty components
- Aid in crew training by simulation of malfunction and system activation

This system could allow tests to be performed with the vehicle grounded as well as in flight. By using the world-wide tracking facilities, continuous monitoring of critical avionics systems could be performed, and all required checkout and simulation operations could be started from ground.

MODULAR
BUILDING BLOCKS

PATCHBOARD

PROGRAM
CONTROL

DATA PROCESSING
AND
MAN/MACHINE INTERFACE

STD UUT
INTERFACE

UUT

CRT

HARD
COPY

MAG TAPE
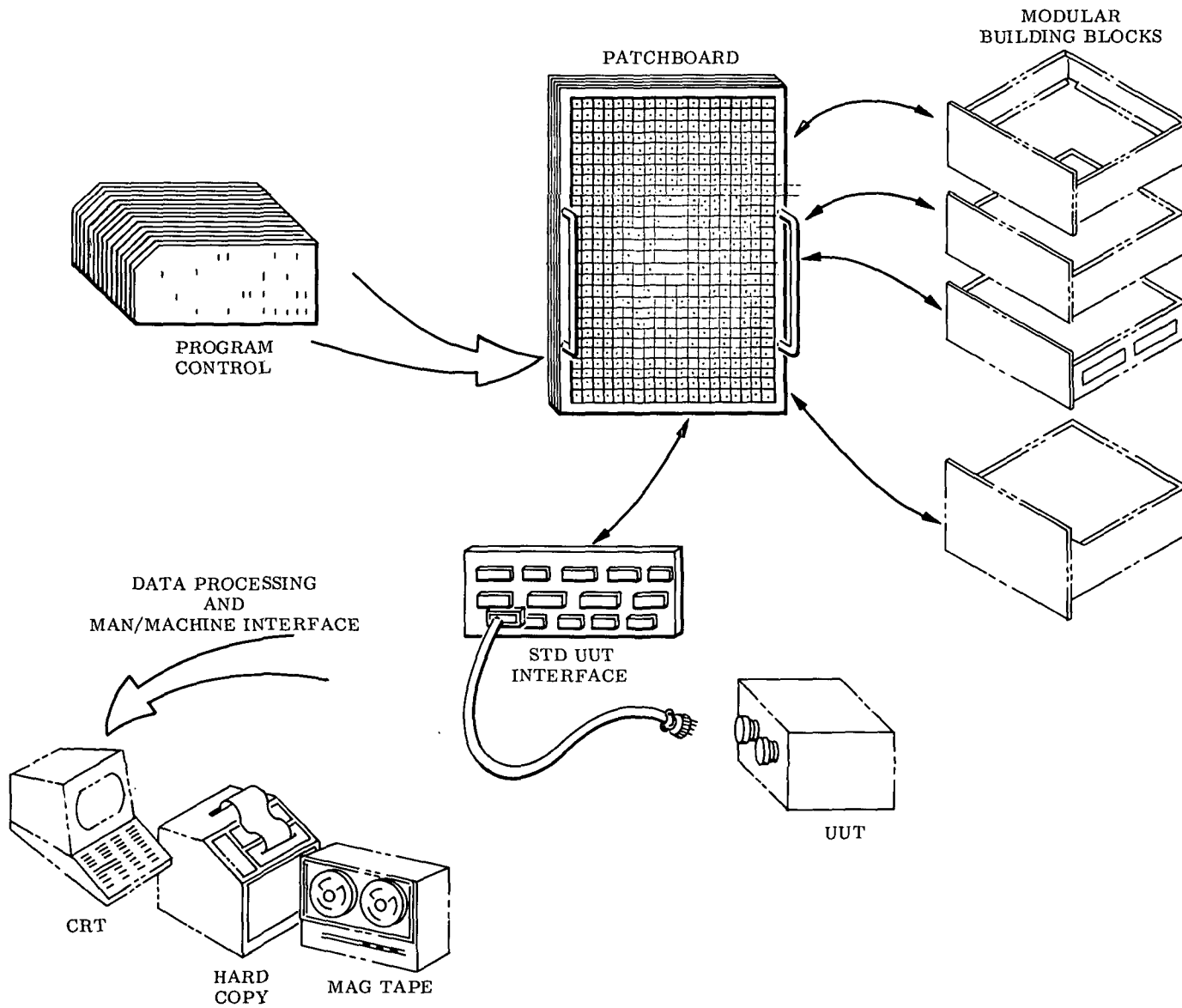
Fig. 16-9 Test Station

The basic building blocks of the test system are as follows:

- An RF source unit having capabilities of producing a signal or signals in the frequency range from zero to approximately 60 MHz

- A monitoring unit having receiver-down conversion and demodulation capabilities of up converting or down converting signal frequencies, attenuating or amplifying, mixing, filtering signals, and adding noise to signals to reach signal-to-noise ratios

- A switching matrix to send required signals between aforementioned units and the avionics

- A computer for command and control of test sequences and for mathematical computations

A larger degree of flexibility in the location of these building blocks should be provided. It is foreseen, however, that the signal conditioners would be installed close to the avionics to be tested whenever RF design considerations dictate. Studies should be conducted to assess the degree of integration between the signal conditioner and the avionics to be tested. A very detailed analysis of the hardware characteristic and circuitry should be performed for each avionics subsystem component. Critical parameters could be monitored and the outputs sent to the computer for further processing and decision. This process should be an integral part of system so that a component malfunction can generate a diagnostic test sequence. Such decision can also be made by an operator.

Onboard checkout facilities should be suitable for testing all the vehicle avionics except the antennas. Proper coupling of the stimulation signals at the antenna feedline will substitute outside inputs. The antenna itself could be tested by using reflectometer methods. Areas of investigations should include the location of the basic system building block and the use of an onboard special computer or of the existing vehicle computer. To enhance flight diagnosis capabilities and safety, use of tracking station telemetry links should be considered. As a general design criteria, the maximum degree of flexibility should be provided so that any of the onboard checkout building blocks could be substituted with a similar ground unit.

A realistic dynamic simulation of the integrated avionics system can be achieved in a ground facilitated test whereas the actual simulations are done through an open loop.

That is, the stimuli are emitted by quasi-radiating elements, such as horn antennas and phased arrays in a free space environment of an anechoic facility. The target array (antenna) associated with each unique avionics item should be located strategically in the ground facility to establish a known and constant spatial coordinate with respect to the axis of the vehicle under simulation.

Mathematical models would simulate such factors as sea state, terrain, and atmosphere. Similarly, flight profiles would be introduced into the inertial navigation loop to simulate the 6 degree-of-freedom of aerodynamics. This can also benefit training of flight crews, mission planning, and research and engineering.

Consideration should be given to relating the onboard and ground function through RF links, as illustrated in Fig. 16-10 to facilitate testing, monitoring, and flight statusing of the entire vehicle avionics system. Certain advantages can be expected from this scheme; for example, ground-based flight test control can direct and independently interrogate any onboard system for a realtime assessment of the system relevant to the test mission success and alleviate certain monitoring tasks from the engineering test flight crews during critical maneuvers.

Throughout the engineering flight test program, up and down links must be provided for realtime or near realtime analysis. Flight status of airframe structures, thermodynamics, aerodynamics, and other pertinent data sensed by their appropriate transducers can be combined in a common-carrier scheme with appropriate encoder/decoder elements and sharing a segment of the baseband combiner in the telemetry system. These data can be PCM formatted in the downlink and processed by the ground PCM decommutation system. Additional capability, such as predetection recording, can be employed by taking the raw PCM at the output of the down converter in the RF signal conditioning unit and routing directly to a broadband video recorder.

Communication satellites can be employed for remote programming of the test functions with minimal additional frequency translation of the up and down links. Existing global tracking stations can be updated to provide this remote testing capability to support the Space Shuttle program.

16-35

SENSOR

TM

XMTR 1

XMTR 2

MPX

RCVR
SYS

DECODER

DAC
(TYP)

AIRBORNE SYSTEM

L-BAND CMD, RANGING, VOICE

COHERENT DOPPLER, TM, RANGING, VOICE

HIGH PCM DATA

PARABOLIC
DISH

ANGLE
DATA

ANT
CONT

GROUND

DI-
PLEXER

$\dot{R}$

RANGE
RATE

RCVG
SYS

TM

VOICE

PCM

FM/FM

PAM/FM

XMTR

RANGE

PCM

CMD

VOICE

R
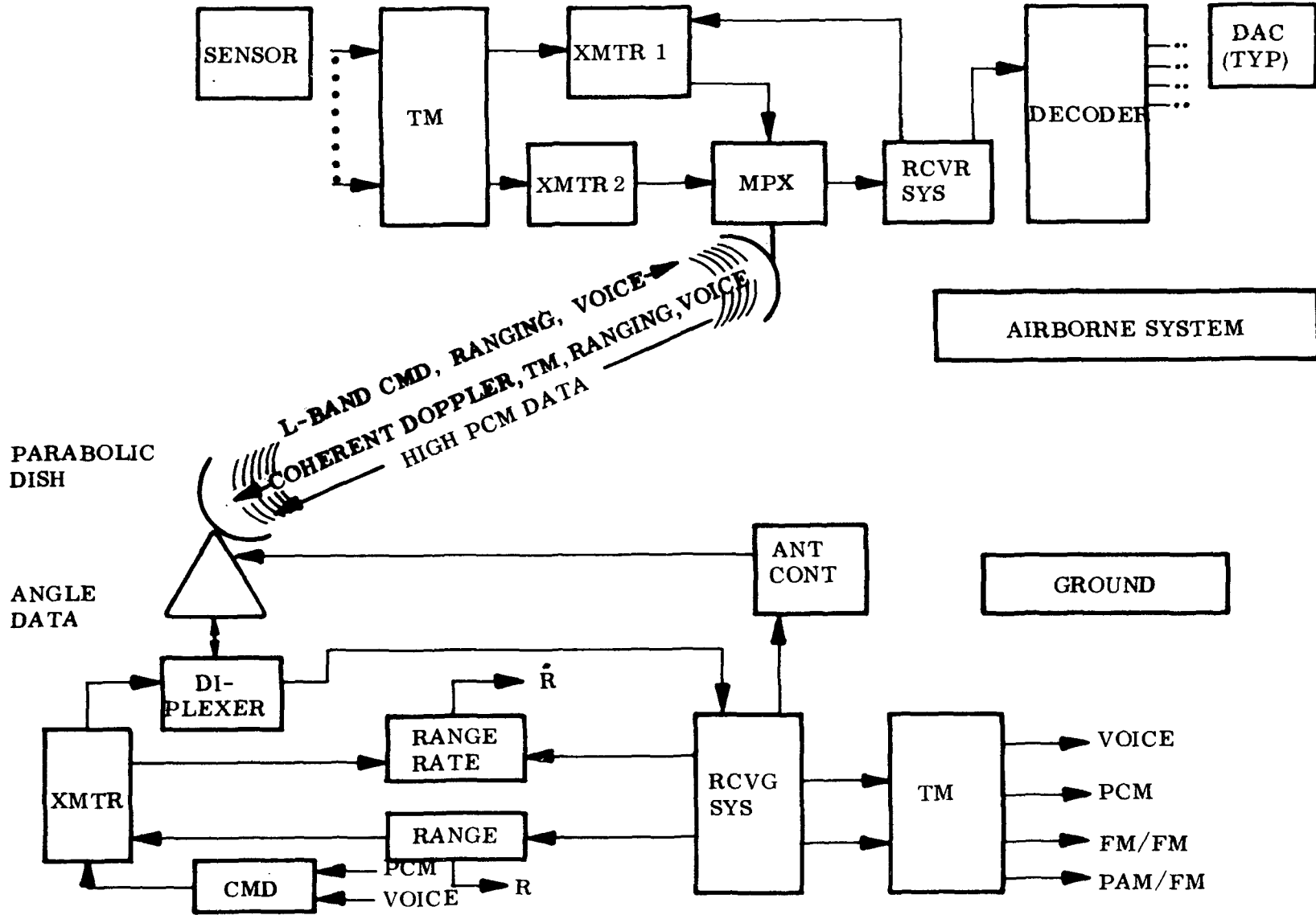
Fig. 16-10 RF Link System — Simplified Functional Block Diagram

This same remote programming of the test functions by means of the command and PCM telemetering links of the various link system can be employed effectively to the fully integrated system simulation in the ground facilitated complex illustrated in Fig. 16-11.

LOCKHEED MISSILES & SPACE COMPANY

AIRBORNE

DECODER

DAC

OB
SIM
TEST
UNIT

RF
DIST
(MPX)

SGLS

CMD
L-BAND

ENCODER

OB
COMPUTER

TM DATA
S-BAND

AVIONIC

UNIQUE AVIONICS

AVIONIC

GROUND

ANECHOIC TEST FACILITY

7000
ATE
(UNIQUES)

GND
SGLS
SYS

RAW
PCM

PRE-D
RECORD

PCM
DECOMM
SYS

7000
COMPUTER
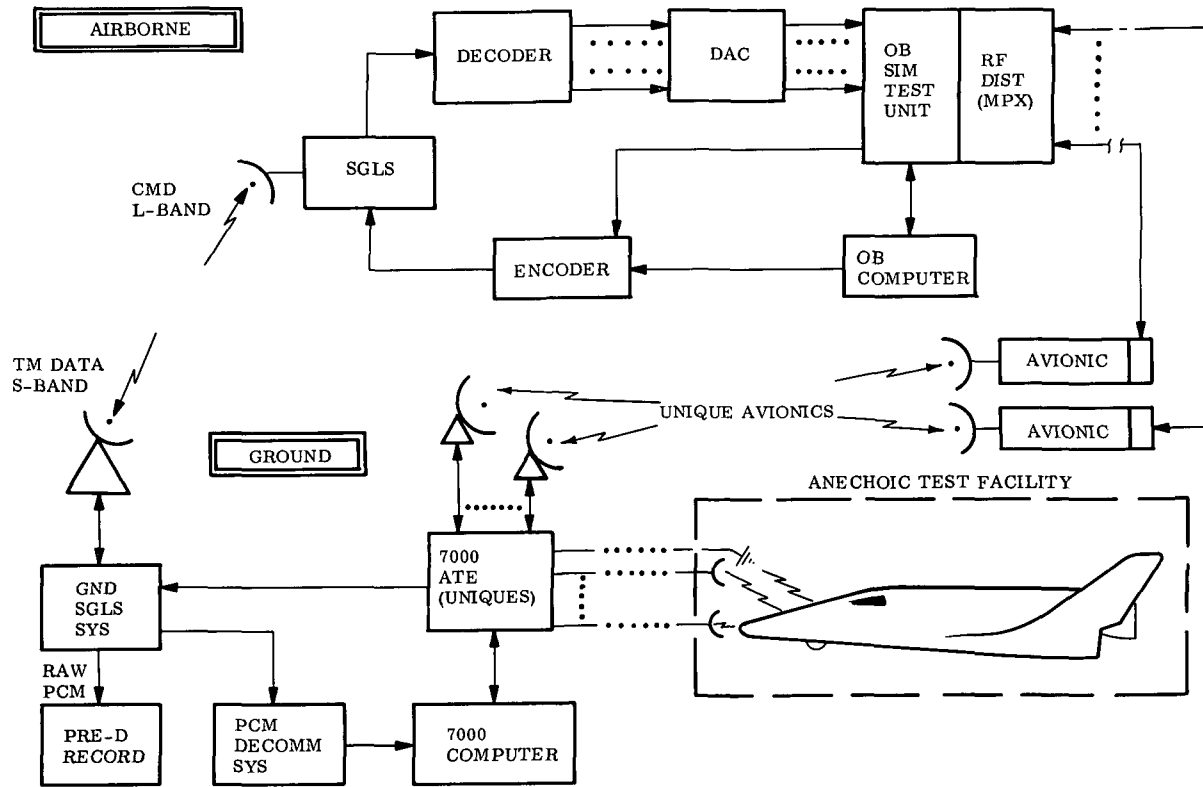
Fig. 16-11 Typical Air Link Test Loop

16.2 MANUFACTURING

Analyses and studies have been conducted in conjunction with the preliminary design effort to establish the basic manufacturing approach and its relation to the supporting functions. Attention has been directed to a manufacturing development plan; an assembly breakdown; a tooling policy; and packaging, handling, and transportation problems.

16.2.1 Fabrication

Many of the fabrication problems will be identical to those encountered in fabricating large aircraft; however, a manufacturing development program must cope with new materials and processes, as well as new technologies.

16.2.2 Major Assembly

Major assembly span charts presented in Figs. 16-12 through 16-14 are the result of preliminary analysis of the current design for each major assembly or function. Major segment spans were determined for the orbiter (applicable to both concepts), the Triamese booster, and the Two-Stage booster.

Typical of the analysis required to isolate problems associated with manufacturing development is the study made on the main cryogenic propellant storage tanks.

The current concept, calling for 2219-T87 aluminum, can be fabricated by using fusion butt-welding techniques. Compound curvatures, encountered on several tank components, may present access problems during welding operations. However, several approaches, including the use of laser techniques, are being considered. Polyurethane insulation on interior as well as the exterior tank surfaces may require spray-on application, as opposed to blanket-type methods.

MANUFACTURING MONTHS

| 30 | 29 | 28 | 27 | 26 | 25 | 24 | 23 | 22 | 21 | 20 | 19 | 18 | 17 | 16 | 15 | 14 | 13 | 12 | 11 | 10 | 9 | 8 | 7 | 6 | 5 | 4 | 3 | 2 | 1 |

NOSE ASSEMBLY

LWR FWD SECTION

UPR FWD SECTION

AFT FWD STRUCTURE

CREW COMPARTMENT

LIFE SUPPORT SYSTEM

AFT FWD SECT

FWD SECTION

TURBO-FAN ENGINES

LWR CENTER SECT

UPR CENTER SECT

CENTER SECTION

FINAL ASSEMBLY

SYS TEST

FWD P/L BAY ASSEM

CTR P/L BAY ASSEM

AFT P/L BAY ASSEM

PAYLOAD BAY SECT

PROPULSION BAY ASSEMBLY

ELEVON ASSEMBLIES

AFT SECTION

ROCKET ENGINES (SUBCONTRACT)

VERTICAL FIN ASSEMBLIES

PAYLOAD PALLET ASSEM

FUEL TANKS

LANDING GEARS

Fig. 16-12 Major Assembly Spans for Orbiter

MANUFACTURING MONTHS

| 33 | 32 | 31 | 30 | 29 | 28 | 27 | 26 | 25 | 24 | 23 | 22 | 21 | 20 | 19 | 18 | 17 | 16 | 15 | 14 | 13 | 12 | 11 | 10 | 9 | 8 | 7 | 6 | 5 | 4 | 3 | 2 | 1 |

FINAL ASSY 1200

FAN JET ENGINES

ELEVONS

ROCKET ENGINES

MATE FWD SECT TO B.S. 2000

FWD SECTION

VERTICAL FINS

SUPT STRUCTURE – GENERAL

MATE JET SUPT STR F.C. SWING 3000

FWD B.S.

CENTER B.S.

AFT B.S.

LDG GR FAIRING

LDG GEAR

FUEL SECTION – JET

SUPT STR – FAN JETS

WINGS

BASIC STRUCT ASSY

MATE TANK TO BASIC ST

LO$_2$ TANK

LH$_2$ TANK

Fig. 16-13 Major Assembly Spans for Triamese Booster

FINAL ASSY

NOSE ASSEMBLY

ROCKET ENGINE

JET ENGINES

VERTICAL FIN L.H.

VERTICAL FIN R.H.

FWD PLANT SUR

JET FUEL TANKS

LH$_2$ TANK ASSY

LO$_2$ TANK ASSY

KEEL ASSEMBLY

MATE UPPER FWD WINGS

LWR FWD SECT L&R.

LWR FWD MID-BODY

LWR CTR MID-BODY

LWR AFT MID-BODY

LWR AFT FWD SECT

KEEL BOX BEAM

NOSE LOG GR

MAIN LOG GR

MATE TANK TO KEEL ASSY

WING ASSY L.H.

WING ASSY R.H.

ELEVON L.H.

ELEVON R.H.

UPPER FWD BUS SECT

UPPER FWD MID-BODY

UPPER CENTER MID-BODY

UPPER AFT MID-BODY

UPPER AFT FWD SECT

Fig. 16-14 Major Assembly Spans for Two-Stage Booster

16.2.3 Tooling

One of the major costs in the Space Shuttle program will be that for the special (project) tooling – jigs, fixtures, etc. Some preliminary tooling approaches for minimizing this expense have been defined, as follows:

- Minimum tooling– Because of the low production rate of two to four units per year, minimum duplication of tooling will be required.

- Valve engineering techniques – Applied during the design will be techniques such as tradeoff studies and analyses of costs versus tolerance specifications.

- Minimum tool inspection documentation– Tool designs will define critical specifications only, and tool inspection will document only the critical tooling. Low-cost shop practices will prevail wherever critical specifications are not imposed.

- Envelope-style tooling– Assembly fixtures will be designed to locate exterior contours and critical mating surfaces with only a minimum of interior structure locators. Basic tooling structures will be fabricated from square tubing, which allows maximum alteration at a minimum cost.

- Laser and optical techniques – Extensive use of laser and optical techniques in establishing alignments, perpendicularity, and distance measurements is anticipated. Large sections will be mated optically rather than through the use of physical masters.

- Numerical control – Numerical control techniques will be used for contour machining, multiple details, hole patterns, etc.

- New techniques and materials – This is the key to minimizing costs in a program of this magnitude. A large-vehicle tooling technique group should be established to explore the use of such materials as foamed aluminum, honeycomb, plastics, and composites.

Section 17

COSTS AND SCHEDULES

Costs and schedules have been developed for both the Two-Stage and the Triamese
baseline configurations. The cost estimates were made from parametric cost models
as well as from a preliminary bottom-up analysis. All output data for both costs and
schedules were designed to conform to the format detailed in NASA "Specification for
the Reporting of Costs and Schedule Plans for New Space Projects, Phase A," dated
January 30, 1969.

17.1 WORK BREAKDOWN STRUCTURE

The Work Breakdown Structure (WBS) used for both the parametric and bottom-up cost estimates was established in accordance with the NASA specification cited above. In general, this specification requires the reporting of costs and schedules to the third level for launch and flight operations; to the fourth level for ground support equipment and program managment; and to the fifth level for flight hardware, test hardware, facilities, and systems support. To these levels, the same basic WBS applies to both the Two-Stage and the Triamese configurations. In the cost and schedule data following, Two-Stage WBS elements are identified by the 102-XX-XX family of numbers and Triamese by the 103-XX-XX family. In both cases, the last two pairs of WBS digits have been omitted, since reporting is not required below the fifth level.

17.2 PARAMETRIC COSTING

Parametric cost estimates were made for the Two-Stage and Triamese configurations, both sized to meet the 50,000-pound ascent and return payload capability. These estimates were made by means of two independent methods: (1) a parametric cost model specifically derived for comparative costing as a part of Task 1 of the ILRV study; and (2) a cost model developed by the Air Force for use in the Space Transportation System study.

17.2.1 Cost Model Descriptions

The cost model developed by LMSC for the ILRV study is based on a model originally developed by the Institute for Defense Analysis (IDA) for reusable space launch vehicles. The IDA model was selected as a point of departure because of its good overall coverage and the compatibility of its required vehicle parametric input with the output of the LMSC vehicle sizing program designated MAGIC. However, the original IDA cost model tended to lump many cost blocks of the WBS into a few elements and did not provide the identification of costs to the depth required by the NASA specification. Therefore, the original IDA cost model was substantially modified to provide an output that is more closely aligned to the ILRV WBS.

17-2

In general, these modifications consisted mainly of establishing new cost estimating relationships (CERs) for elements not covered in the original model and adjusting the original CERs to account for these subtracted costs. In some cases, new CERs were developed from more current data and substituted directly for corresponding CERs of the original model.

The Air Force cost model used for parametric costing is described in detail in Annex 6 of the Air Force Statement of Work for the Space Transportation Systems Study, dated August 7, 1969. This model does not align with the NASA WBS format as well as the ILRV cost model, but it serves as a good common yardstick for assessing system costs from various sources. Therefore, costs by the Air Force model are summarized and presented for comparative purposes.

17.2.2 Costing Ground Rules and Assumptions

For both the Two-Stage and Triamese configurations, certain common ground rules and assumptions were postulated on the basis of mission requirements. These are summarized in Table 17-1. Test hardware is based on the assumption that the Two-Stage configuration will require the equivalent of five orbiters and five boosters and that the Triamese will require five orbiters and seven boosters. It is also assumed that two orbiters and two boosters of the Two-Stage configuration will survive the development phase and be carried over into the operational fleet. For the Triamese configuration, two orbiters and three boosters will be carried over into the operational fleet. Required production quantities to meet the flight rate of 100 flights/year are tabulated below.

	Number required		Number available	
	Flight	Spares	Development	Production
Orbiter (Both configurations	6	1	2	5
Booster Two-Stage Triamese	2 4	2 3	2 3	2 4

Table 17-1

COMMON COSTING ASSUMPTIONS

Parameter	Assumption
Number of development flights	
Horizontal	175 per stage
Vertical	25 complete vehicles
Number of operational launch facilities	1
Number of operational flights	1000
Operational period	10 years
Spares factor, development	20 percent
Spares factor, production	10 percent
Learning rates	
Engines	95 percent
Other	90 percent
On-orbit time, orbiter	7 days
Ground turnaround time (design goal)	
Orbiter	14 days
Booster (two shifts per day refurbishment)	7 days
Maximum flights/year	
Orbiter	17
Booster	51
Boost vehicle-orbital vehicle commonality	
Structures	0
Heat shield	20 percent
RCS	80 percent
EPS	50 percent
Communications	80 percent
ECS	80 percent
Rocket propulsion	97 percent
Airbreather propulsion	Modify existing engines
Average number of reuses (including use of spares)	
Orbiter	143
Booster (two-stage)	250
Booster (triamese)	286

Other common assumptions:

Subsystem costs do not include weight contingencies. Cost of production vehicles includes only the number of vehicles required for 100 flights per year for 10 years and does not reflect procurement of vehicles necessary for continuation of the program.

17-4

17.2.3 Parametric Cost Estimates for Two-Stage Configuration

Parametric estimates of RDT&E and recurring costs for a 1000-flight operational program for the 50,000-pound payload Two-Stage configuration are shown in Table 17-2. First unit costs for the vehicle are shown in Table 17-3. These costs are based on the LMSC ILRV cost model. The total RDT&E cost is $5512 million and the total recurring cost for the 10-year operational span is $1255 million.

For the purpose of comparison, equivalent costs (in millions of dollars) by the ILRV cost model and by the Air Force model are summarized below.

	ILRV Model	AF Model
Total RDT&E	$5512.4	$6219.6
Vehicle first unit	186.4	232.7
Production vehicles	501.7	1013.3
Ten-year operating cost	753.5	2830.0
Total recurring cost/flight	1.26	3.84
Operations cost/flight	0.754	2.83

The Technical Characteristics List for this configuration is contained in Table 17-4.

Table 17-2

TWO-STAGE PARAMETRIC COSTS
ILRV COST MODEL
(Million of Dollars)

		RDT&E		Recurring (100 flt/yr)
102-01-01	Structures		719.0	131.1
	Airframe	373.8		
	Heat shield	345.2		
-02	Subsystem Installation		7.2	24.0
-03	Propulsion		482.3	46.7
	Rocket	452.3		
	Airbreather	30.0	34.2	
-04	RCS		50.0	12.5	4.0

Table 17-2 (Cont.)

		RDT&E		Recurring (100 flt/yr)	
102-01-05	EPS		56.0		12.1
-06	ECS		30.0		6.4
-07	Communications		40.0		12.1
-08	Stabilization and Control		0.0		0.0
-09	Launch Escape		0.0		0.0
-10	Guidance and Navigation		110.0		14.1
-13	Instrumentation		0.0		0.0
-15	Landing and Recovery		8.7		11.8
-16	Crew Systems		10.0		2.0
-92	Systems Support		499.7		24.0
-95	Facilities		175.0		–
-97	Test Hardware		310.6		–
102-01-00			2498.5		288.3
102-04-01	Structures		769.9		68.2
	Airframe	509.7			
	Heat shield	260.2			
-02	Subsystem Installation		7.7		15.4
-03	Propulsion		43.5		67.6
	Rocket	13.5		59.3	
	Airbreather	30.0		8.3	
-04	RCS		10.0		1.8
-05	EPS		28.0		0.8
-06	ECS		6.0		1.7
-07	Communications		8.0		1.7
-08	Stabilization and Control		0.0		0.0
-09	Launch Escape		0.0		0.0
-10	Guidance and Navigation		22.0		5.8
-13	Instrumentation		0.0		0.0
-15	Landing and Recovery		11.5		6.4
-16	Crew Systems		1.0		0.2

Table 17-2 (Cont.)

		RDT&E	Recurring (100 flt/yr)
102-04-92	Systems Support	398.1	15.4
-95	Facilities	191.4	—
-97	Test Hardware	493.2	—
102-04-00		1990.3	185.0
102-18-18	GSE	290.4	0.0
-92	Systems Support	72.6	0.0
102-18-00	Total GSE	363.0	0.0
102-93-00	Program Management	309.7	28.4
102-00-00	Total Vehicle System	5161.5	501.7
502-01-01	Vehicle Launch Operation	7.7	84.0
-03	Propellants and Gases	7.5	312.6
-92	Systems Support	48.8	16.8
-95	Launch Facilities	179.8	—
502-01-00	Vehicle Operations	243.8	413.4
502-93-00	Program Management	15.6	24.8
502-00-00	Total Launch Operations	259.4	438.2
602-01-00	Ground Command and Tracking	2.5	16.0
602-02-00	Ground Crew Training	2.0	0.0
602-03-00	Flight Crew Training	10.0	0.0
602-05-00	Refurbishment	10.5	232.7
602-06-00	Horizontal Flight Operations	43.8	—
602-92-00	Systems Support	17.2	49.7
602-93-00	Program Management	5.5	16.9
602-00-00	Total Flight Operations	91.5	315.3
	Total	5512.4	1255.2

$$\text{Recurring cost per flight} = \frac{1255.2}{1000} = 1.26$$

$$\text{Operations cost per flight} = \frac{753.5}{1000} = 0.754$$

Average operation cost per flight is distributed as follows:

Operation	Cost ($ x 10^3)
Launch	84.0
Providing propellants and gasses	312.0
Ground control and tracking	16.0
Refurbishment	232.7
Subsystem support	66.5
Program management	41.7
Total	753.5

Table 17-3

TWO-STAGE FIRST UNIT PARAMETRIC COSTS
(ILRV COST MODEL)
(Millions of Dollars)

102-01-01	Structures		33.5
	Airframe and heat shield	30.0	
	Crew compartment	3.5	
-02	Subsystem Installation		6.1
-03	Propulsion		12.3
	Rocket: 3 at 2.7	8.1	
	Airbreather: 4 at 1.05	4.2	
-04	RCS		1.0
-05	EPS		3.0
-06	ECS		1.6
-07	Communications		3.0
-08	Stabilization and Control[1]		0.0
-10	Guidance and Navigation		3.5
-13	Instrumentation[2]		0.0
-14	Ordnance		0.0
-15	Landing and Recovery		2.9
-16	Crew Systems		0.5
-92	Systems Support		6.1
-96	Information Management[2]		0.0
102-01-00	Total Orbiter		73.5

Table 17-3 (Cont.)

102-04-01	Structures		42.0	
	Airframe and heat shield	39.3		
	Crew compartment	2.7		
-02	Subsystem Installation		9.4	
-03	Propulsion			
	Rocket: 13 at 2.7	35.1	41.1	
	Airbreather: 4 at 1.5	6.0		
-04	RCS		1.0	
-05	EPS		0.5	
-06	ECS		1.0	
102-04-07	Communications		1.0	
-08	Stabilization and Control[1]		0.0	
-10	Guidance and Navigation		3.5	
-13	Instrumentation[2]		0.0	
-14	Ordnance		0.0	
-15	Landing and Recovery		3.9	
-16	Crew Systems		0.1	
-92	Systems Support		9.4	
-96	Information Management[2]		0.0	
102-04-00	Total Booster			112.9
102-00-00	Total Vehicle			186.4

[1]Included in RCS

[2]Included in Communications

Table 17-4

50K TWO-STAGE VEHICLE TECHNICAL
CHARACTERISTICS DATA FORM

Identification Number	WBS Identification	WBS Level	Quantity or Value		Units of Measure	Characteristics	Notes
			Current	New			
102-00-00	Vehicle	3		3,734,906	Pounds	Weight – liftoff	
				50,000	Pounds	Weight – payload	
				3,045,868	Pounds	Weight – propellant	
102-01-00	Orbiter	4		202,217	Pounds	Weight – dry	
				50,000	Pounds	Weight – payload	
				164	Feet	Length	
102-01-01	Structures	5		129,326	Pounds	Weight	
102-01-03	Propulsion	5		12,493	Pounds	Weight – rocket engine	
				3	Each	No. of rocket engines	
				460,315	Pounds	Thrust/rocket engine	
				20,806	Pounds	Weight – aero engine	
			4		Each	No. of aero engines	
			19,000		Pounds	Thrust/aero engine	
102-01-04	Reaction control	5	2,059		Pounds	Weight – dry	
102-01-05	Electrical power	5		5,108 Fuel cells	Pounds –	Weight Type of source	
102-01-06	Environ. control	5		1,231	Pounds	Weight	
102-01-07	Communications	5		141	Pounds	Weight	
102-01-10	Guidance and navigation	5		842	Pounds	Weight	

Table 17-4 (Cont.)

Identification Number	WBS Identification	WBS Level	Quantity or Value		Units of Measure	Characteristics	Notes
			Current	New			
102-01-13	Instrumentation	5		200	Pounds	Weight	
102-01-15	Landing and recovery	5		10,380	Pounds	Weight	
102-01-16	Crew systems	5		1,248	Pounds	Weight	
102-04-00	Booster	4		354,808	Pounds	Weight — dry	
				237	Feet	Length	
102-04-01	Structures	5		199,129	Pounds	Weight	
102-04-03	Propulsion	5		58,720	Pounds	Weight — rocket engine	
				13	Each	No. of rocket engines	
				460,315	Pounds	Thrust/rocket engine	
			32,341		Pounds	Weight — aero engine	
			4		Each	No. of aero engines	
			29,000		Pounds	Thrust/aero engine	
102-04-04	Reaction control	5		4,387	Pounds	Weight — dry	
102-04-05	Electrical power	5		9,010 Fuel cells	Pounds	Weight — Type of source	
102-04-06	Environmental control	5		1,181	Pounds	Weight	
102-04-10	Guidance and navigation	5		842	Pounds	Weight	
102-04-13	Instrumentation	5		200	Pounds	Weight	
102-04-15	Landing and recovery	5		14,994	Pounds	Weight	
102-04-16	Crew systems	5		1,248	Pounds	Weight	

17.2.4 Triamese Parametric Costs

Cost estimates (in millions of dollars) by the ILRV cost model for the 50,000-pound payload Triamese configuration are shown in Tables 17-5 and 17-6. Total RDT&E cost is $5525 million, and total recurring costs for 1000 flights is $1493 million. Comparisons of the key costs with the equivalent costs by the Air Force cost model are summarized below.

	ILRV Model	AF Model
Total RDT&E	$5525.0	$6131.0
First unit	254.3	301.4
Production vehicle	615.0	1209.0
Ten-year operating	877.6	3418.0
Total recurring cost/flight	1.49	4.63
Operations cost/flight	0.878	3.42

The Triamese configuration Technical Characteristics List is contained in Table 17-7.

Table 17-5

TRIAMESE PARAMETRIC COSTS
ILRV COST MODEL
(Millions of Dollars)

			RDT&E	Recurring (100 flights/yr)
103-01-01	Structures		744.0	140.8
	Airframe	398.8		
	Heat shield	345.2		
-02	Subsystem Installation		7.4	24.6
-03	Propulsion		465.4	42.9
	Rocket	435.5	33.4	
	Jet	30.0	9.5	
-04	RCS		50.0	4.0
-05	EPS		56.0	12.1
-06	ECS		30.0	6.4

Table 17-5 (Cont.)

			RDT&E	Recurring (100 flights/yr)
103-01-07	Communications		40.0	12.1
-08	Stabilization and Control		0.0	0.0
-09	Launch Escape		0.0	0.0
-10	Guidance and Navigation		110.0	14.1
-13	Instrumentation		0.0	0.0
-15	Landing and Recovery		8.7	11.8
-16	Crew Systems		10.0	2.0
-92	System Support		505.2	24.6
-95	Facilities		180.0	–
-97	Test Hardware		319.3	–
103-01-00			2526.1	295.4
103-04-01	Structures		705.9	120.7
	Airframe	472.4		
	Heat shield	233.5		
-02	Subsystem Installation		7.1	23.8
-03	Propulsion		43.1	85.2
	Rocket	13.1		71.3
	Jet	30.0		13.9
-04	RCS		10.0	3.2
-05	EPS		28.0	1.6
-06	ECS		6.0	3.1
-07	Communications		8.0	3.1
-08	Stabilization and Control		0.0	0.0
-09	Launch Escape		0.0	0.0
-10	Guidance and Navigation		22.0	11.0
-13	Instrumentation		0.0	0.0
-15	Landing and Recovery		9.0	9.3
-16	Crew Systems		1.0	0.3
-92	System Support		386.6	23.8

Table 17-5 (Cont.)

			RDT&E	Recurring (100 flights/yr)
103-04-95	Facilities		161.3	–
-97	Test Hardware		545.0	–
103-04-00			1933.0	285.1
103-18-18	Vehicle GSE		306.9	0.0
-92	System Support		76.7	0.0
103-18-00	GSE		383.6	0.0
103-93-00	Program Management		309.1	34.8
103-00-00	Total Vehicle System		5151.8	615.3
503-01-01	Vehicle Launch Operations		8.2	99.9
-03	Propellants and Gases		8.6	358.1
-92	System Support		51.7	20.0
-95	Launch Facilities		190.0	–
503-01-00	Total Vehicle Operations		258.5	478.0
503-93-00	Program Management		16.5	28.7
503-00-00	Total Launch Operations		275.0	506.7
603-01-00	Ground Command Control and Tracking		2.5	16.0
603-02-00	Ground Crew Training		2.0	0.0
603-03-00	Flight Crew Training		15.0	0.0
603-05-00	Refurbishment		10.5	75.6
603-06-00	Horizonal Flight Operations		43.8	–
603-92-00	System Support		18.5	58.3
603-93-00	Program Management		5.9	21.0
603-00-00	Flight Operations		98.2	370.9
	Total		5525.0	1492.9

$$\text{Recurring Cost/Flight} = \frac{1492.9}{1000} = 1.49$$

$$\text{Operations Cost/Flight} = \frac{877.6}{1000} = 0.878$$

Table 17-6

TRIAMESE FIRST UNIT PARAMETRIC COSTS
(ILRV COST MODEL)

103-01-01	Structures		35.2	
	Airframe and heat shield	31.7		
	Crew compartment	3.5		
-02	Subsystem Installation		6.2	
-03	Propulsion		11.0	
	Rocket: 3 at 2.6	7.8		
	Airbreather: 4 at 0.8	3.2		
-04	RCS		1.0	
-05	EPS		3.0	
-06	ECS		1.6	
-07	Communications		3.0	
-08	Flight Controls and Stabilization[1]		0.0	
-10	Guidance and Navigation		3.5	
-13	Instrumentation		0.0	
-14	Ordnance		0.0	
-15	Landing and Recovery		2.9	
-16	Crew Systems		0.5	
-92	System Support		6.2	
-96	Information Management[2]		0.0	
103-01-00	Total Orbiter			74.1
103-04-01	Structures		40.0	
	Airframe and heat shield	36.2		
	Crew compartment	3.8		
-02	Subsystem Installation		7.5	
-03	Propulsion		25.0	
	Rocket: 8 at 2.6	20.8		
	Airbreather: 4 at 1.05	4.2		
-04	RCS		1.0	
-05	EPS		0.5	

Table 17-6 (Cont.)

103-04-06	ECS		1.0	
-07	Communications		1.0	
-08	Flight Controls and Stabilization[1]		0.0	
-10	Guidance and Navigation		3.5	
-13	Instrumentation[2]		0.0	
-14	Ordnance		0.0	
-15	Landing and Recovery		3.0	
-16	Crew Systems		0.1	
-92	System Support		7.5	
-96	Information Management[2]		0.0	
103-04-00	Total Boost Vehicle			90.1
103-00-00	Total Vehicle			254.3

[1]Included in RCS

[2]Included in Communications

Table 17-7

50K TRIAMESE VEHICLE TECHNICAL
CHARACTERISTICS DATA FORM

Identification Number	WBS Identification	WBS Level	Quantity or Value		Units of Measure	Characteristics	Notes
			Current	New			
103-00-00	Vehicle	3		4,340,230	Pounds	Weight — liftoff	
				50,000	Pounds	Weight — payload	
				3,481,921	Pounds	Weight — propellant	
103-01-00	Orbiter	4		202,719	Pounds	Weight — dry	
				50,000	Pounds	Weight — payload	
				164	Feet	Length	
103-01-01	Structures	5		129,350	Pounds	Weight	
103-01-03	Propulsion	5		12,544	Pounds	Weight — rocket engine	
				3	Each	No. of rocket engines	
				434,624	Pounds	Thrust/rocket engine	
			30,806		Pounds	Weight — aero engine	
			4		Each	No. of aero — engines	
			19,000		Pounds	Thrust/aero-engine	
103-01-04	Reaction control	5		2,059	Pounds	Weight — dry	
103-01-05	Electrical power	5		5,108	Pounds	Weight	
103-01-06	Environ. control	5		1,231	Pounds	Weight	
103-01-07	Communi-cations	5		141	Pounds	Weight	

Table 17-7 (Cont.)

Identification Number	WBS Identification	WBS Level	Quantity or Value		Units of Measure	Characteristics	Notes
			Current	New			
103-01-10	Guidance and navigation	5		842	Pounds	Weight	
103-01-13	Instrumen-tation	5		200	Pounds	Weight	
103-01-15	Landing and recovery	5		10,380	Pounds	Weight	
103-01-16	Crew systems	5		1,248	Pounds	Weight	
103-04-00	Booster	4		251,965	Pounds	Weight – dry	
				180	Feet	Length	
103-04-01	Structures	5		117,688	Pounds	Weight	
103-04-03	Propulsion	5		34,045	Pounds	Weight – rocket engine	
				8	Each	No. of rocket engines	
				434,624	Pounds	Thrust/rocket engine	
			22,818		Pounds	Weight – aero engine	
			4		Each	No. of aero engines	
			21,000		Pounds	Thrust/aero engine	
103-04-04	Reaction control	5		2,618	Pounds	Weight – dry	
103-04-05	Electrical power	5		6,190 Fuel Cells	Pounds	Weight Type of source	
103-04-06	Environ. control	5		962	Pounds	Weight	
103-09-10	Guidance and navigation	5		842	Pounds	Weight	

Table 17-7 (Cont.)

Identification Number	WBS Identification	WBS Level	Quantity or Value		Units of Measure	Characteristics	Notes
			Current	New			
103-04-13	Instrumen-tation	5		200	Pounds	Weight	
103-04-15	Landing and recovery	5		10,603	Pounds	Weight	
103-04-16	Crew systems	5		1,248	Pounds	Weight	

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17.3 BOTTOM-UP COST ESTIMATE

The bottom-up cost estimate for the Two-Stage Space Shuttle is an engineering estimate. It was arrived at by breaking the program into a work structure at the subsystem level, then estimating the amount of effort required by function. These estimates were then cross checked for reasonableness with statistical profile data obtained from programs like Apollo Service Module, Apollo Command Module, Supersonic Transport, Lunar Excursion Module, and C-5A. In addition to the profile cross check, other cross checks were made on the basis of hours-per-pound data (fabrication only). The hours-per-pound data were based on statistical information from Lockheed programs in which large aircraft structures or structures that are similar in size to those of the Space Shuttle were built. A complexity factor was added because of the more expensive materials and larger size of the Space Shuttle.

The Triamese estimate was arrived at by an engineering analysis of the differences between the Two-Stage and the Triamese system. These differences were reduced to complexity factors, which were applied to the Two-Stage cost to derive Triamese cost.

The bottom-up cost data are displayed in the Cost Estimate Data Forms prescribed by the NASA "Specification for the Reporting of Costs and Schedule Plans for New Space Projects, Phase A," dated January 30, 1969. The highest cost and lowest cost columns of these forms are not filled in at this time because of the low degree of accuracy based on a confidence rating of 1 for the entire estimate.

Assumption and conditions:

- Estimates are in 1969 dollars unadjusted for inflation.
- Estimates are contractor costs only (no fee included)
- Hardware quantities

RDT&E (Two Stage)	Booster	Orbiter
Engineering test vehicle	1	1
Static vehicle	1	1
Flight test vehicles	3	3

	Booster	Orbiter
RDT&E Triamese		
Engineering test vehicle	1	1
Static vehicle	1	1
Flight test vehicle	5	3
Production vehicles		
Two Stage	2	5
Triamese	4	5

● One set of ground support equipment is included in RDT&E cost.

● One set of training simulators is included in the RDT&E cost.

● Training of the flight and launch crew is included in the RDT&E cost.

● RDT&E flights are to be 175 horizontal and 25 vertical.

● RTD&E development span is IOC 80 months after go-ahead.

● Operational phase is 10 years after IOC, with 1000 flights.

● The estimate does not include:

  NASA/DOD in-house cost

  Special-purpose manufacturing and test facilities

  Command, control, and tracking facilities

  Transportation costs for deliverable hardware

  Operational payloads or payload integration

● Operational phase includes:

  1000 flights over 10 years

  All gases and propellants

  Maintenance and refurbishment costs

17.3.1 Bottom-Up Estimates

Nonrecurring and recurring cost estimates for the 50,000-pound payload Two-Stage configuration are given in Tables 17-8 and 17-9. Total RDT&E cost is $5468 million, and total recurring cost for 1000 flights is $1626 million. The estimated funding schedule (including 7 percent fee) is shown in Table 17-10.

Corresponding data for the 50K Triamese configuration are given in Tables 17-11, 17-12, and 7-13. Total RDT&E cost for this configuration is $5172 million, and total recurring cost for 1000 flights is $1945 million.

17.3.2 Summary Comparison

A comparison of the key cost sums for both configurations, as estimated by the ILRV parametric cost model and the bottom-up analyses, is summarized in Table 17-14.

Table 17-8

TWO-STAGE COST ESTIMATE DATA FORM

_X_ Design and Development (Nonrecurring)
_____ Production and Operations (Recurring)

($ in Thousands)

Identification Number a	WBS Identification b	WBS Level c	No. Units d	Expect. Cost e	Highest Cost f	Lowest Cost g	Confid Rating h	T_d (mo) i	T_s (mo) j	Spread Funct k	Learn Index l
102-00-00	Space Shuttle	3	–	5,186,285			1	80	80	Summary Block	–
502-00-00	Launch Operations	3	–	94,669			1	9	9	2	–
602-00-00	Flight Operations	3	–	186,633			1	27	27	2	–
102-92-00	Systems Support	4	–	42,329			1	80	80	2	–
102-93-00	Program Management	4	–	32,219			1	80	80	2	–
102-18-00	GSE	4	–	51,956			1	59	68	1	–
102-01-00	Orbiter	4	–	2,752,866			1	80	80	2	–
102-04-00	Booster	4	–	2,306,915			1	80	80	2	–
102-01-01	Structures – Orbiter	5	–	206,933			1	42	68	1	–
102-04-01	Structures – Booster	5	–	139,632			1	42	68	1	–
102-01-02	Subsystem Installation (O)	5	–	138,263			1	20	40	3	–
102-04-02	Subsystem Installation (B)	5	–	168,789			1	20	40	3	–
102-01-03	Propulsion System (O)	5	–	603,969			1	74	80	2	–
102-04-03	Propulsion System (B)	5	–	68,333			1	74	80	2	–
102-01-04	Reaction Control Sys (O)	5	–	60,781			1	53	80	2	–
102-04-04	Reaction Control Sys (B)	5	–	48,759			1	53	80	2	–
102-01-05	Electrical Pwr (O)	5	–	54,740			1	53	80	2	–

O = Orbiter   B = Booster

Table 17-8 (Cont.)

_X_ Design and Development (Nonrecurring)
___ Production and Operations (Recurring)

($ in Thousands)

Identification Number a	WBS Identification b	WBS Level c	No. Units d	Expect. Cost e	Highest Cost f	Lowest Cost g	Confid Rating h	T_d (mo) i	T_s (mo) j	Spread Funct k	Learn Index l
102-04-05	Electrical Pwr (B)	5	-	40,274			1	53	80	2	-
102-01-06	Envir Cont Sys (O)	5	-	44,450			1	55	80	2	-
102-04-06	Envir Cont Sys (B)	5	-	34,101			1	55	80	2	-
102-01-07	Communications (O)	5	-	131,782			1	53	80	2	-
102-04-07	Communications (B)	5	-	86,497			1	53	80	2	-
102-01-08	Stab. and Control (O)	5	-	Included with structures			1	53	80	2	-
102-04-08	Stab. and Control (B)	5	-	Included with structures			1	53	80	2	-
102-01-09	Launch Escape (O)	5	-	9,227			1	55	80	2	-
102-04-09	Launch Escape (B)	5	-	6,468			1	55	80	2	-
102-01-10	Guidance & Navig (O)	5	-	30,966			1	55	80	2	-
102-04-10	Guidance & Navig (B)	5	-	26,012			1	55	80	2	-
102-01-13	Instrumentation (O)	5	-	19,579			1	55	80	2	-
102-04-13	Instrumentation (B)	5	-	13,239			1	55	80	2	-
102-01-14	Ordnance (O)	5	-	-			1	43	68	2	-
102-04-14	Ordnance (B)	5	-	-			1	43	68	2	-
102-01-15	Landing System (O)	5	-	22,290			1	54	80	2	-
102-04-15	Landing System (B)	5	-	17,525			1	54	80	2	-
102-01-16	Crew System (O)	5	-	25,126			1	54	80	2	-
102-04-16	Crew System (B)	5	-	16,442			1	54	80	2	-

O = Orbiter   B = Booster

Table 17-8 (Cont.)

<u>X</u> Design and Development (Nonrecurring)
___ Production and Operations (Recurring)

($ in Thousands)

Identification Number a	WBS Identification b	WBS Level c	No. Units d	Expect. Cost e	Highest Cost f	Lowest Cost g	Confid Rating h	T_d (mo) i	T_s (mo) j	Spread Funct k	Learn Index l
102-01-92	System Support (O)	5	–	985,628			1	80	80	3	–
102-04-92	System Support (B)	5	–	1,089,358			1	80	80	3	–
102-01-95	Facilities (O)	5	–	22,125			1	64	80	2	–
102-04-95	Facilities (B)	5	–	16,820			1	64	80	2	–
102-01-96	Information Mgmt	5	–	Included with communications			1	55	80	2	–
102-04-96	Information Mgmt	5	–	Included with communications			1	55	80	2	–
102-01-97	Test Hardware (O)	5	–	397,007			1	62	80	2	–
102-04-97	Test Hardware (B)	5	–	534,666			1	62	80	2	–

O = Orbiter
B = Booster

Total nonrecurring cost = $5,467,587,000

Table 17-9

TWO-STAGE COST ESTIMATE DATA FORM

—— Design and Development (Nonrecurring)
_X_ Production and Operations (Recurring)

($ In Thousands)

Identification Number a	WBS Identification b	WBS Level c	No. Units d	Expect. Cost e	Highest Cost f	Lowest Cost g	Confid Rating h	T_d (mo) i	T_s (mo) j	Spred Funct k	Learn Index l
102-00-00	Space Shuttle	3	—	688,458			1.	150	150	Summary Block	—
502-00-00	Launch Opers	3	—	937,364				120	120	*	—
602-00-00	Flight Opers	3	—	Included with launch operations				120	120	*	—
102-92-00	Systems Support	4	—	2,885				120	120	*	—
102-93-00	Program Mgmt	4	— ·	3,923				120	120	*	—
102-18-00	GSE	4	—	1,154				138	138	1	—
102-01-00	Orbiter	4	5	78,202				38	150	2	83%
102-04-00	Booster	4	2	144,743				38	150	2	
102-01-01	Structures (O)	5	5	35,402				22	150	3	
102-04-01	Structures (B)	5	2	58,706				22	150	3	
102-01-02	Subsys Install. (O)	5	5	17,186				12	128	3	
102-04-02	Subsys Install. (B)	5	2	28,987				12	128	3	
102-01-03	Propulsion Sys (O)	5	5	8,733			1	30	150	2	83%

NOTE: Expected cost is an average unit cost for the quantities indicated.
*Spread 1/120 of expected cost per month over 120 months
O = Orbiter   B = Booster

Table 17-9 (Cont.)

___ Design and Development (Nonrecurring)
X Production and Operations (Recurring)

($ In Thousands)

Identification Number a	WBS Identification b	WBS Level c	No. Units d	Expect. Cost e	Highest Cost f	Lowest Cost g	Confid Rating h	T_d (mo) i	T_s (mo) j	Spred Funct k	Learn Index l
102-04-03	Propulsion Sys (B)	5	2	28,961			1	30	150	2	83%
102-01-04	RCS (O)	5	5	850				30	150	2	
102-04-04	RCS (B)	5	2	1,428				30	150	2	
102-01-05	Elec Pwr (O)	5	5	3,400				30	150	2	
102-04-05	Elec Pwr (B)	5	2	5,626				30	150	2	
102-04-06	Envir Cont Sys (O)	5	6	1,873				30	150	2	
102-04-06	Envir Cont Sys (B)	5	2	3,094				30	150	2	
102-01-07	Communications (O)	5	5	3,400				30	150	2	
102-04-07	Communications (O)	5	2	5,627				30	150	2	
102-01-08	Stab & Cont (O)	5	5	Included with structures				30	150	2	
102-04-08	Stab & Cont (B)	5	2	Included with structures				30	150	2	
102-01-09	Launch Escape (O)	5	5	522				30	150	2	
102-04-09	Launch Escape (O)	5	2	894				30	150	2	
102-01-10	Guid & Nav (O)	5	5	3,767				30	150	2	
102-04-10	Guid & Nav (B)	5	2	6,242			1	30	150	2	83%

Table 17-9 (Cont.)

_____ Design and Development (Nonrecurring)

__X__ Production and Operations (Recurring)

($ In Thousands)

Identification Number a	WBS Identification b	WBS Level c	No. Units d	Expect. Cost e	Highest Cost f	Lowest Cost g	Confid Rating h	T_d (mo) i	T_s (mo) j	Spred Funct k	Learn Index l
102-01-13	Instr (O)	5	2	284			1	30	150	2	83%
102-04-13	Instr (B)	5	2	479			↑	30	150	2	↑
102-01-15	Landing Sys (O)	5	5	2,299				30	150	2	
102-04-15	Landing Sys (B)	5	2	3,882				30	150	2	
102-01-16	Crew System (O)	5	5	486			↓	30	150	2	↓
102-04-16	Crew System (B)	5	2	817			1	30	150	2	83%
102-01-92	Sys Support (O)	5	5	Included in the subsystem cost				—	—	—	
102-04-92	Sys Support (B)	5	2	Included in the subsystem cost				—	—	—	
102-01-95	Facilities (O)	5	5	—				—	—	—	
102-04-95	Facilities (B)	5	2	—				—	—	—	
102-01-96	Info Mgmt (O)	5	5	Included with communications				—	—	—	
102-04-96	Info Mgmt (B)	5	2	Included with communications				—	—	—	
102-01-97	Test Hardware (O)	5	—	—				—	—	—	
102-04-97	Test Hardware (O)	5	—	—				—	—	—	

Total recurring cost = $1,625,822,000

Table 17-10

TWO-STAGE FUNDING SCHEDULE

(Dollars in Millions)

Fiscal Year	RDT&E			Production and Operations			Total Program		
	Cost	Fee	Total	Cost	Fee	Total	Cost	Fee	Total
71	$ 122	$ 9	$ 131	$	$	$	$ 122	$ 9	$ 131
72	687	48	735				687	48	735
73	1,192	83	1,275				1,192	83	1,275
74	1,319	92	1,411				1,319	92	1,411
75	1,089	76	1,165	22	1	23	1,111	77	1,188
76	739	52	791	269	19	288	1,008	71	1,079
77	301	21	322	307	21	328	608	42	650
78	19	1	20	173	12	185	192	13	205
79				95	7	102	95	7	102
80				95	7	102	95	7	102
81				95	7	102	95	7	102
82				95	7	102	95	7	102
83				95	7	102	95	7	102
84				95	7	102	95	7	102
85				95	7	102	95	7	102
86				95	7	102	95	7	102
87				95	7	102	95	7	102
Total	$5,468	$382	$5,850	$1,626	$116	$1,742	$7,094	$498	$7,592

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LOCKHEED MISSILES & SPACE COMPANY

Table 17-11

TRIAMESE COST ESTIMATE DATA FORM

__X__ Design and Development (Nonrecurring)
_____ Production and Operations (Recurring)

($ In Thousands)

Identification Number a	WBS Identification b	WBS Level c	No. Units d	Expect. Cost e	Highest Cost f	Lowest Cost g	Confid Rating h	T_d (mo) i	T_s (mos) j	Spred Funct k	Learn Index l
103-00-00	Space Shuttle	3	—	4,890,348			1	80	80	Summary Block	—
503-00-00	Launch Opers	3	—	94,669				9	9	2	—
603-00-00	Flight Opers	3	—	186,633				27	27	2	—
103-92-00	Systems Support	3	—	42,329				80	80	2	—
103-93-00	Program Mgmt	4	—	32,219				80	80	2	—
103-18-00	GSE	4	—	51,956				59	68	1	—
103-01-00	Orbiter	4	—	2,752,866				80	80	2	—
103-04-00	Booster	4	—	2,010,978				80	80	2	—
103-01-01	Structures (O)	5	—	206,933				42	68	1	—
103-04-01	Structures (B)	5	—	111,706				42	68	1	—
103-01-02	Subsys Instl (O)	5	—	138,263				20	40	3	—
103-04-02	Subsys Instl (B)	5	—	135,031				20	40	3	—
103-01-03	Propulsion Sys (O)	5	—	603,969				74	80	2	—
103-04-03	Propulsion Sys (B)	5	—	54,666			1	74	80	2	—

O = Orbiter
B = Booster

Table 7-11 (Cont.)

__x__ Design and Development (Nonrecurring)
____ Production and Operations (Recurring)

($ In Thousands)

Identification Number a	WBS Identification b	WBS Level c	No. Units d	Expect. Cost e	Highest Cost f	Lowest Cost g	Confid Rating h	T_d (mo) i	T_s (mo) j	Spred Funct k	Learn Index l
103-01-04	Reaction Ctl Sys (O)	5	—	60,781			1	53	80	2	—
103-04-04	Reaction Ctl Sys (B)	5	—	39,007			↑	53	80	2	—
103-01-05	Elec Power (O)	5	—	54,740				53	80	2	—
103-04-05	Elec Power (B)	5	—	32,219				53	80	2	—
103-01-06	Envir Cont Sys (O)	5	—	44,451				55	80	2	—
103-04-06	Envir Cont Sys (B)	5	—	27,281				55	80	2	—
103-01-07	Communications (O)	5	—	131,782				53	80	2	—
103-04-07	Communications (B)	5	—	69,198				53	80	2	—
103-01-08	Stab & Control (O)	5	—	Included with structures				53	80	2	—
103-04-08	Stab & Control (B)	5	—	Included with structures				53	80	2	—
103-01-09	Launch Escape (O)	5	—	9,227				55	80	2	—
103-04-09	Launch Escape (B)	5	—	5,174				55	80	2	—
103-01-10	Guidance & Nav (O)	5	—	30,966				55	80	2	—
103-04-10	Guidance & Nav (B)	5	—	20,810				55	80	2	—
103-01-13	Instrumentation (O)	5	—	19,579			↓	55	80	2	—
103-04-13	Instrumentation (B)	6	—	10,591			1	55	80	2	—

Table 7-11 (Cont.)

<u>X</u> Design and Development (Nonrecurring)
___ Production and Operations (Recurring)

($ In Thousands)

Identification Number a	WBS Identification b	WBS Level c	No. Units d	Expect. Cost e	Highest Cost f	Lowest Cost f	Confid Rating h	Td (mo) i	Ts (mo) j	Spred Funct k	Learn Index l
103-01-14	Ordnance (O)	5	–	–			–	43	68	2	–
103-04-14	Ordnance (B)	5	–	–			–	43	68	2	–
103-01-15	Landing Sys (O)	5	–	22,290			1	54	80	2	–
103-04-15	Landing Sys (B)	5	–	14,020			↑	54	80	2	–
103-01-16	Crew System (O)	5	–	16,442				54	80	2	–
103-04-16	Crew System (B)	5	–	13,154				54	80	2	–
103-01-92	Sys Support	5	–	985,628				80	80	3	–
103-04-92	Sys Support	5	–	871,486				80	80	3	–
103-01-95	Facilities (O)	5	–	22,125				64	80	2	–
103-04-95	Facilities (B)	5	–	18,502				64	80	2	–
103-01-96	Info Mgmt (O)	5	–	Included w/communications				55	80	2	–
103-04-96	Info Mgmt (B)	5	–	Included w/communications				55	80	2	–
103-01-97	Test Hardware (O)	5	–	392,007			↓	62	80	2	–
103-04-97	Test Hardware (B)	5	–	588,133			1	62	80	2	–

Total nonrecurring cost = $5,171,650,000

TRIAMESE COST ESTIMATE DATA FORM

_____ Design and Development (Nonrecurring)
__x__ Production and Operations (Recurring)

($ In Thousands)

Identification Number a	WBS Identification b	WBS Level c	No. Units d	Expect. Cost e	Highest Cost f	Lowest Cost g	Confid Rating h	T_d (mo) i	T_s (mo) j	Spred Funct k	Learn Index l
103-00-00	Space Shuttle	3	—	773,739			1	150	150	Summary Block	—
503-00-00	Launch Opers	3	—	1,171,705			↑	120	120	· *	—
603-00-00	Flight Opers	3	—	Included with launch opers				120	120	· *	—
103-92-00	Systems Support	4	—	2,885				120	120	· *	—
103-93-00	Prog Mgmt	4	—	3,923				120	120	*	—
103-18-00	GSE	4	—	1,154				138	138	1	—
103-01-00	Orbiter	4	5	78,202				38	150	2	—
103-04-00	Booster	4	—	93,692				38	150	2	—
103-01-01	Structures (O)	5	5	35,402				22	150	3	83%
103-04-01	Structures (B)	5	4	38,637				22	150	3	↑
103-01-02	Subsys Instl (O)	5	5	17,186				12	128	3	
103-04-02	Subsys Instl (B)	5	4	18,446				12	128	3	
103-01-03	Propulsion Sys (O)	5	5	8,733			↓	30	150	2	
103-04-03	Propulsion (B)	5	4	18,228			1	30	150	2	83%

Note: The expected cost is an average unit cost for the quantities indicated.
O = Orbiter   B = Booster

*Spread 1/120 of expected cost per month for 120 months

Table 7-12 (Cont.)

_____ Design and Development (Nonrecurring)
__X__ Production and Operations (Recurring)

($ In Thousands)

Identification Number a	WBS Identification b	WBS Level c	No. Units d	Expect. Cost e	Highest Cost f	Lowest Cost g	Confid Rating h	T_d (mo) i	T_s (mo) j	Spred Funct k	Learn Index l
103-01-04	RCS (O)	5	5	850			1	30	150	2	83%
103-04-04	RCS (B)	5	4	917			↑	30	150	2	↑
103-01-05	Elec Pwr (O)	5	5	3,400				30	150	2	
103-04-05	Elec Pwr (B)	5	4	3,719				30	150	2	
103-01-06	Envir Cont Sys (O)	5	5	1,873				30	150	2	
103-04-06	Envir Cont Sys (B)	5	4	2,054				30	150	2	
103-01-07	Communications O)	5	5	3,400				30	150	2	
103-04-07	Communications (B)	5	4	3,719				30	150	2	
103-01-08	Stab & Cont (O)	5	–	Included with structure				30	150	2	
103-04-08	Stab & Cont (B)	5	–	Included with structure				330	150	2	
103-01-09	Launch Escape (O)	5	5	522				30	150	2	
103-04-09	Launch Escape (B)	5	4	563				30	150	2	
103-01-10	Guid & Nav (O)	5	5	3,767			↓	30	150	2	↓
103-04-10	Guid & Nav (B)	5	4	4,105			1	30	150	2	83%

Table 7-12 (Cont.)

_____ Design and Development (Nonrecurring)
__X__ Production and Operations (Recurring)

($ In Thousands)

Identification Number a	WBS Identification b	WBS Level c	No. Units d	Expect. Cost e	Highest Cost f	Lowest Cost g	Confid Rating h	T_d (mo) i	T_s (mo) j	Spred Funct k	Learn Index l
103-01-13	Instr (O)	5	5	284			1	30	150	2	83%
103-04-13	Instr (B)	5	4	305			↑	30	150	2	↑
103-01-15	Landing Sys (O)	5	5	2,299				30	150	2	
103-04-15	Landing Sys (B)	5	4	2,467				30	150	2	
103-01-16	Crew System (O)	5	5	486			↓	30	150	2	↓
103-04-16	Crew System (B)	5	4	532			1	30	150	2	83%
103-01-92	Sys Support (O)	55	–	–	–	–	–				–
103-04-92	Sys Support (B)	5	–	–	–	–	–				–
103-01-95	Facilities (O)	5	–	–	–	–	–				–
103-04-95	Facilities (B)	5	–	–	–	–	–				–
103-01-96	Info Mgmt (O)	5	–	–	–	–	–				–
103-04-96	Info Mgmt (B)	5	–	–	–	–	–				–
103-01-97	Test Hardware (O)	5	–	–	–	–	–				–
103-04-97	Test Hardware (B)	5	–	–	–	–	–				–

Total recurring cost = $1,945,444,000

Table 17-13

TRIAMESE FUNDING SCHEDULE

(Dollars in Millions)

Fiscal Year	RDT&E			Production and Operations			Total Program		
	Cost	Fee	Total	Cost	Fee	Total	Cost	Fee	Total
71	$ 115	$ 8	$ 123	$	$	$	$ 115	$ 8	$ 123
72	649	45	694				649	45	694
73	1,127	79	1,206				1,127	79	1,206
74	1,247	87	1,334				1,247	87	1,334
75	1,029	72	1,101	25	2	27	1,054	74	1,128
76	700	49	749	303	21	324	1,003	70	1,073
77	286	20	306	345	24	369	631	44	675
78	19	1	20	211	15	226	230	16	246
79				118	8	126	118	8	126
80				118	8	126	118	8	126
81				118	8	126	118	8	126
82				118	8	126	118	8	126
83				118	8	126	118	8	126
84				118	8	126	118	8	126
85				118	8	126	118	8	126
86				118	8	126	118	8	126
87				118	8	126	118	8	126
Total	$5,172	$361	$5,533	$1,946	$134	$2,080	$7,118	$495	$7,613

Table 17-14

SUMMARY COST COMPARISON

(Costs in Millions)

	Two-Stage		Triamese	
	Parametric	Bottom-Up	Parametric	Bottom-Up
RDT&E	5512	5468	5525	5172
Production Vehicles	502	689	615	773
Ten-Year Operating	754	937	878	1172
Total Recurring	1255	1626	1493	1945
Total Recurring/Flight	1.26	1.63	1.49	1.95
Operationa Cost/Flight	0.754	0.937	0.878	1.172
Total Program Cost (RDT&E + Recurring)	6767	7094	7018	7117

17.4 SCHEDULES

A detailed analysis of total development time required for each major development item, leading to IOC in 1977, has been made. Incremental development was the underlying approach.

To form the basis for the summary schedule, requirements and objectives were established for flight testing, development testing, and manufacturing.

Schedules were developed for each of these areas and measured against the time for design and detail specifications, vehicle layout, subsystem design, hardware assembly, installation drawings, critical design reviews, and design release points.

A series of key technical issues were also identified as follows and evaluated for impact on the development program schedules:

- Wind tunnel testing of candidate configuration concepts
- Reusable dependable thermal protection system
- Predictable recovery methods
- Flight control techniques
- Integrated avionics system
- Hot flow testing of full-scale integrated structural components
- Reusable nose cap
- Integrated onboard checkout system that includes capability of the flight crew to verify flight readiness

These issues were then evaluated for their effect on the concept feasibility, the criticality of their function and the time they required, and the impact on the total schedule if an alternate method had to be developed.

The effects of varying development timing were examined in terms of total system cost, maximum average annual funding, and technology risk level. Technological risk is either the degree to which the first stage, at the start of the development, is oversized in order to account for uncertainties in the reusable orbiter or the effect in both time and cost resulting from the failure to carry to development one of the initially

selected subsystems or concepts. Technology risk as it applies to the schedules deals with the latter description. While the first stage may be initially oversized for the specified payload, it may present the opportunity to increase the payload capability and has been previously discussed in this report.

These detailed schedules and comparisons, key technical issues, and analysis of technology risks fit into what is considered to be a realistic and achievable development schedule.

17.4.1 Development Schedule (Figure 17-1)

This schedule provides a summary of the major Phase C/D development spans. Since the vehicle does not require breakthroughs or new basic technologies, an orderly progression from Phase B to Phase C and Phase D is possible. Advanced development programs are already underway for the high-pressure LO_2/LH_2 rocket engines, which are an integral part of the Space Shuttle. The basic structural technology envisioned in the concept already exists.

Total time for development is 68 months, leading to IOC in 1977, an additional 11 months being required for Phase C. Alternate schedules leading to an IOC in 1976 have been developed, but this IOC date would involve increased costs especially in the Phase C period. Major milestones from Phase C are the following:

- Engineering design − 37 months, with CDR at month 24, 90 percent drawing release at month 25, and 100 percent drawing release at month 32

- Manufacturing − 28 months, including spans for systems checkout and acceptance test (Rollouts occur at 4-month intervals, with a total of three flight test vehicles for both the orbiter and the booster.)

- Testing − 28-month flight test program, including ground tests prior to the first flight and consisting of 15 months of suborbital test flights and 11 months of orbital flights (Qualified engines will be furnished 1 month prior to the first orbital flight.)

17.4.2 Development and Production Plan (Figure 17-2)

The work breakdown structure (WBS) schedule depicts the significant work spans and key events associated with specific level 3, 4, and 5 WBS blocks.

These schedule spans, developed from Lockheed and NASA historical and planning data, take into account technological risks that may arise from varying development times.

Spans shown on this schedule correlate with those shown in Fig. 17-1.

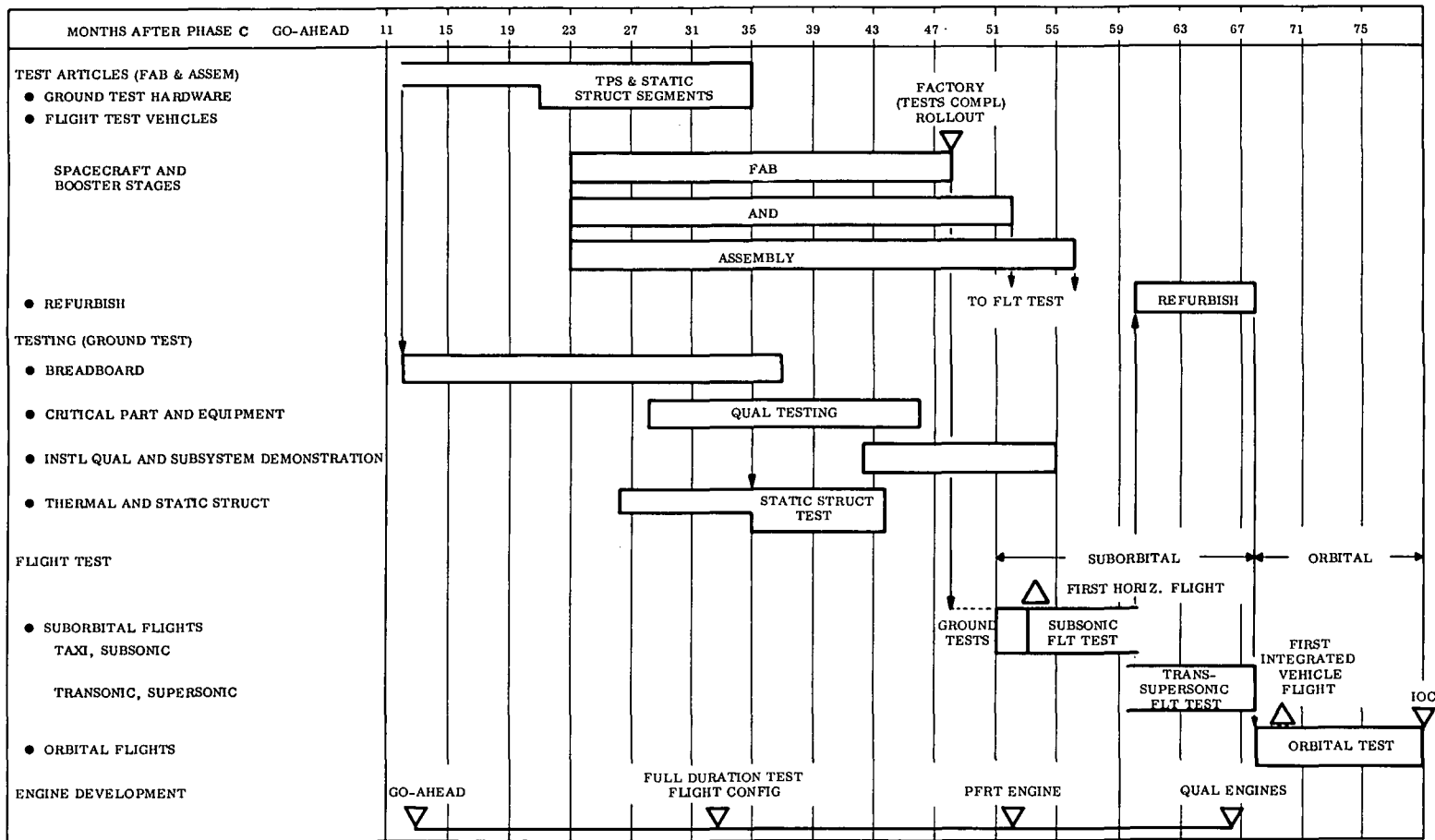17.4.3 Summary Manufacturing and Test Schedule (Figure 17-3)

This schedule depicts the manufacturing and test activities that are summarized in Fig. 17-1. The interrelationships between hardware and the tests are shown.

PHASES C & D SPACE SHUTTLE DEVELOPMENT SCHEDULE

LOCKHEED LMSC

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Fig. 17-1

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IDENTIFICATION NUMBER	WBS IDENTIFICATION	71	72	73	74	75	76	77
1XX-00-00	SPACE SHUTTLE	φC START / φD START		90% ENG / 100% ENG		1ST FLIGHT SUBSONIC	1ST ORBITAL FLIGHT	IOC
5XX-00-00	LAUNCH OPERATIONS ORBITER AND BOOSTER					GROUND TAXI TESTS	LAUNCH OPS	
6XX-00-00	FLIGHT OPERATIONS ORBITER & BOOSTER					TRANS & SUPERSONIC FLT. TEST		
1XX-92-00	SYSTEMS SUPPORT SPACE SHUTTLE	ANALYSES – EVALUATION – REVIEWS – INTEGRATION						
1XX-93-00	PROGRAM MANAGEMENT SPACE SHUTTLE	PROGRAM MANAGEMENT						
1XX-18-00	GROUND SUPPORT EQUIP SPACE SHUTTLE			FAB GSE RFU	TEST GSE RFU / SIMULATORS / FLT TEST RFU			
1XX-01-00	ORBITER VEHICLE	10% ENG / FTU #1 FAB / 90% ENG / 100% ENG			COMP FINAL / FTV #1 / 1ST FLIGHT HORIZ.		1ST VERT FLT	
1XX-04-00	BOOSTER VEHICLE	SAME MILESTONES AS ORBITER VEHICLE						
1XX-01-01 O / 1XX-04-01 B	STRUCTURES		STRUCTURES FAB & ASSEM	TESTING / START STATIC LOAD TESTS				
1XX-01-02 O / 1XX-04-02 B	SUBSYSTEM INSTALL.				FTV#1 / #2 / #3		1ST QUALIF. ENGINE	
1XX-01-03 O / 1XX-04-03 B	PROPULSION SYSTEM	DESIGN DEVEL. FAB & ASSY & TEST		HOT FIRE / ENGINE	PFRT / FLT / TEST ENGINE			
1XX-01-04 O / 1XX-04-04 B	REACTION CONTROL SYSTEM				TEST / FLT / UNIT #1 / #2 / #3		1ST OPNL ENGINE AVAIL.	
1XX-01-05 O / 1XX-04-05 B	ELECTRIC POWER	DESIGN ENGR / FAB		START ASSY / START TEST / FU #1	#2 / #3			
1XX-01-06 O / 1XX-04-06 B	ENVIR. CONTROL SYS.	DESIGN & DEVELOPMENT	FAB & FINAL ASSY / TEST	FU #1 / #2 / #3				
1XX-01-07 O / 1XX-04-07 B	COMMUNICATIONS	DESIGN & DEVELOPMENT	FAB & FINAL ASSY / TEST	FU #1 / COMP #2 / #3				
1XX-01-08 O / 1XX-04-08 B	STABILIZATION & CONTROL	DESIGN & DEVELOPMENT	FAB & FINAL ASSY / TEST	FU #1 / COMP #2 / #3				
1XX-01-09 O / 1XX-04-09 B	LAUNCH ESCAPE	DESIGN/DEVEL.	FAB & ASSY	TEST	FU #1 / #2 / #3			
1XX-01-10 O / 1XX-04-10 B	GUIDANCE & NAVIGATION	DESIGN/DEVEL.	FAB & ASSY	TEST	FU #1 / #2 / #3			
1XX-01-13 O / 1XX-04-13 B	INSTRUMENTATION	DESIGN/DEVEL.	FAB & ASSY	TEST	FU #1 / #2 / #3			
1XX-01-14 O / 1XX-04-14 B	ORDNANCE	DESIGN & DEVEL.	FAB & ASSY	TEST	FU #1 / #2 / #3			
1XX-01-15 O / 1XX-04-15 B	LANDING SYSTEM	DESIGN & DEVEL.	FAB & ASSY	TEST	FU #1 / #2 / #3			
1XX-01-16 O / 1XX-04-16 B	CREW SYSTEM	DESIGN & DEVEL	FAB & ASSY	TEST	FU #1 / #2 / #3			
1XX-01-92 O / 1XX-04-92 B	SYSTEMS SUPPORT	STAGE – SEI ANALYSES – EVALUATION – DESIGN REVIEWS – INTEGRATION						
1XX-01-95 O / 1XX-04-95 B	FACILITIES	RFU MFG FACILITIES	BEGIN FAB FLT FACIL		SUBORBITAL FACIL RFU	ORBITAL FACIL RFU		
1XX-01-96 O / 1XX-04-96 B	INFO. MGMT.	DESIGN/DEVEL.	FAB & ASSY	TEST	FU #1 / #2 / #3			
1XX-01-97 O / 1XX-04-97 B	TEST HARDWARE	BUILD MOCKUPS / STR. DEVEL.	FAB & ASSY / FTV #1	ACCEPT #1 / #2 / #3 / TEST				

Fig. 17-2 Development and Production Plan – Schedules and Milestones

LOCKHEED MISSILES & SPACE COMPANY

17-45

| MONTHS AFTER PHASE C GO-AHEAD | 11 | 15 | 19 | 23 | 27 | 31 | 35 | 39 | 43 | 47 | 51 | 55 | 59 | 63 | 67 | 71 | 75 |

TEST ARTICLES (FAB & ASSEM)
- GROUND TEST HARDWARE
- FLIGHT TEST VEHICLES

TPS & STATIC STRUCT SEGMENTS

FACTORY (TESTS COMPL) ROLLOUT

SPACECRAFT AND BOOSTER STAGES

FAB

AND

ASSEMBLY

TO FLT TEST

REFURBISH

- REFURBISH

TESTING (GROUND TEST)
- BREADBOARD
- CRITICAL PART AND EQUIPMENT

QUAL TESTING

- INSTL QUAL AND SUBSYSTEM DEMONSTRATION
- THERMAL AND STATIC STRUCT

STATIC STRUCT TEST

FLIGHT TEST

SUBORBITAL — ORBITAL

FIRST HORIZ. FLIGHT

- SUBORBITAL FLIGHTS
  TAXI, SUBSONIC

GROUND TESTS

SUBSONIC FLT TEST

FIRST INTEGRATED VEHICLE FLIGHT

  TRANSONIC, SUPERSONIC

TRANS-SUPERSONIC FLT TEST

IOC

- ORBITAL FLIGHTS

ORBITAL TEST

ENGINE DEVELOPMENT

GO-AHEAD

FULL DURATION TEST FLIGHT CONFIG

PFRT ENGINE

QUAL ENGINES

Fig. 17-3 Summary Manufacturing and Test Schedule

17.5 UNCITED REFERENCES

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