ATS-46876 N73-11861

SHUTTLE VEHICLE AND MISSION

SIMULATION REQUIREMENTS REPORT

VOLUME II

10/20/72

CASE FILE
COPY

SINGER
SIMULATION PRODUCTS

A DIVISION OF THE SINGER COMPANY • DEVELOPER AND MANUFACTURER OF THE **LINK** TRAINER SINCE 1929

CR-128622

SHUTTLE VEHICLE AND MISSION

SIMULATION REQUIREMENTS REPORT

VOLUME II

10/20/72

J. F. Burke
Principal Investigator
SMS Definition Study

This document is submitted in compliance with Line Item No. 2 of the Data Requirements List as Type I Data, Contract NAS 9-12836

SINGER COMPANY
SIMULATION PRODUCTS DIVISION

SHUTTLE VEHICLE AND MISSION SIMULATION REQUIREMENTS REPORT

VOLUME II

Prepared by:

J. F. Burke
Principal Investigator
SMS Definition Study

Approved by:

C. Olasky
NASA Technical Manager

Contracting Officer

This document is submitted in compliance with Line Item No. 2 of the Data Requirements List as Type I Data, Contract NAS 9-12836

SINGER COMPANY
SIMULATION PRODUCTS DIVISION

PREFACE

This document is submitted in compliance with Line Item No. 2 of the Data Requirements List as Type I Data, Contract NAS9-12836. The document is divided into four volumes for ease of handling. The contents of each volume is defined as:

Volume I: Includes sections entitled Introduction, Mission Envelope and Flight Dynamics which correspond to Sections 1.0, 2.0 and 3.0 of the Table of Contents.

Volume II: Includes sections entitled Introduction and Shuttle Vehicle Systems which correspond to sections 1.0 and 4.0 to 4.18 of the Table of Contents.

Volume III: Includes sections entitled Introduction and Shuttle Vehicle Systems which correspond to sections 1.0 and 4.19 to 4.22 of the Table of Contents.

Volume IV: Includes sections entitled External Interfaces, Crew Prodedures, Crew Station, Visual Cues and Aural Cues which correspond to sections 5.0, 6.0, 7.0, 8.0 and 9.0 of the Table of Contents.

TABLE OF CONTENTS

1.0 Introduction

2.0 Mission Envelope

2.1 Space Missions

2.1.1 Launch Phase

2.1.2 Orbit and Deorbit Phases

2.1.3 Entry and Landing Phases

2.1.4 Payloads, Deploy and Retrieval

2.1.5 Reference Missions

2.1.5.1 Vertical Tests

2.1.5.2 Easterly Launch

2.1.5.3 Resupply

2.1.5.4 South Polar

2.1.6 Timelines

2.1.7 Aborts

2.1.7.1 Pad and Low Altitude

2.1.7.2 Return to Site (unpowered)

2.1.7.3 Return to Site (powered)

2.1.7.4 Once-Around Orbit

2.1.7.5 To Orbit Degraded Mission

2.1.7.6 Booster Powered Glide Return

2.1.8 Mission Operations

2.1.8.1 Staging

2.1.8.2 Drop Tank Maneuver

2.1.8.3 IMU Alignment

2.1.8.4 Rendezvous

2.1.8.5 Docking

2.1.8.6 Payload Deployment

2.1.8.7 Payload Retrieval

2.1.8.8 Deorbit

2.1.8.9 Entry

2.1.8.10 Hypersonic-Supersonic Transition

2.1.8.11 Energy Management

2.1.8.12 Cruise

2.1.8.13 Landing and Rollout

2.2 Atmospheric Flights

2.2.1 Horizontal Flight Test

2.2.2 Ferry Flights

3.0 Flight Dynamics

3.1 Vehicle Configurations

3.1.1 Operational Space Mission Configuration

3.1.1.1 Orbiter Vehicle

3.1.1.2 Payload

3.1.1.3 Solid Rocket Motors

3.1.1.3.1 156" Booster SRM

3.1.1.3.2 156" Rocket Separation SRM's

3.1.1.3.3 Abort SRM's

3.1.1.3.4 External Tank Deorbit SRM

3.1.1.4 External Tank

3.1.1.5 Air Breathing Engines

3.1.1.6 OMS Engines

3.1.1.7 RCS

3.1.1.8 Docked Configurations

3.1.2 Operational Ferry Mission Configuration

3.1.3 Horizontal Test Configuration

3.1.4 Vertical Test Configuration

3.2 Equations of Motion

3.2.1 Coordinate Systems

3.2.2 Translational Equations of Motion

3.2.2.1 Forces

3.2.2.1.1 Earth Gravitational

3.2.2.1.2 Other Celestial Body Gravitational

3.2.2.1.3 Dynamic Body Forces

3.2.2.1.4 Payload Forces

3.2.2.1.5 Docking Effects

3.2.2.1.6 Staging Effects

3.2.2.1.7 Venting and Dumping

3.2.2.2 Trajectory Calculation Requirements

3.2.2.2.1 Orbiter

3.2.2.2.2 Target Vehicles

3.2.2.3 Relative Translational States

3.2.2.4 Accuracy

3.2.3 Rotational Equations of Motion

3.2.3.1 Moments

3.2.3.1.1 Dynamic Body Moments

3.2.3.1.2 Payload Moments

3.2.3.1.3 Docking Effects

3.2.3.1.4 Staging Effects

3.2.3.1.5 Venting and Dumping

3.2.3.2 Rotational State Calculation Requirements

3.2.3.4 Accuracy

3.3 Mass Properties

3.3.1 Orbiter

3.3.2 Solid Rocket Motors

3.3.3 External Tanks

3.3.4 Payload

3.3.5 ABES

3.3.6 Ferry ABES

3.3.7 Total Vehicles

3.3.7.1 Operational Space Vehicle

3.3.7.2 Operational Ferry

3.3.7.3 HFTS

3.3.7.4 VFT

3.3.8 Retrieval Satellites

3.3.9 Deployed Satellites

3.3.10 Space Station

3.4 Aerodynamic Characteristics

3.4.1 General Requirements

3.4.2 Aerodynamic Flight Regimes

3.4.2.1 Launch and Abort

3.4.2.2 Orbital Flight

3.4.2.3 Orbiter Only

3.5 Ephemeris

3.5.1 Celestial Bodies (stars, moon, sun, planets)

3.5.2 Coordinate Transformations (IM to IEO

4.0 Shuttle Vehicle System

4.1 Electrical Power

4.1.1 Electrical Power Distribution and Control (EPDC)

4.1.2 Power Distribution Equipment Description

4.1.2.1 Generator Control Units (GCU)

4.1.2.2 Power Contactors

4.1.2.3 Inverters

4.1.2.4 Transformer - Rectifiers (TR)

4.1.2.5 Battery Charger

4.1.2.6 Remote Power Controllers (RPC)

4.1.2.7 Sequencers

4.1.2.8 Interior Lighting

4.1.2.9 Exterior Lighting

4.1.2.10 Fuel Cell System

4.1.3 Electrical Power Operational Characteristics

4.1.4 Functional Interfaces and Support Requirements

4.1.4.1 Data Control and Management (DCMO

4.1.4.2 Displays and Controls (D&C)

4.1.4.3 Electrical Power Generation (EPG)

4.1.4.4 Electrical System Power Losses

4.1.4.5 Environmental Control Life Support (ECLS) Interface

4.1.4.6 Power Utilization Subsystems

4.1.4.7 Support Equipment GSE

4.1.4.8 Payload

4.1.4.9 Space Station Interface

4.2 Mechanical Power

4.2.1 Auxilliary Power Units (APU)

4.2.2 Hydraulic Power System

4.3 Main Propulsion Subsystem (MPS)

4.3.1 Engine

4.3.2 Control/Monitor System

4.3.2.1 Engine Thrust/Mixture Ratio

4.3.2.2 Engine Monitoring

4.3.2.3 SSME Controller Data Flow

4.3.3 SSME Operation Details

4.3.3.1 SSME Sequence Schedule

~~4.3.3.1.1 Engine Start~~

4.3.3.1.2 Engine Shutdown

4.3.3.2 SSME Flight Operations Monitoring and Continuous In-Flight Test

4.3.3.3 Data Transmission

4.3.3.4 Controller Build-In Test

F-398-8-A

4.3.3.4.1 Computer Checkout

4.3.3.4.2 Computer Interface Electronics Checkout

4.3.3.4.3 Input Electronics Checkout

4.3.3.4.4 Output Electronics Checkout

4.3.3.4.5 Power Supply Electronics Checkout

4.3.3.5 Sensor Build-In Test

4.3.3.5.1 Sensor Ground Checkout

4.3.3.5.2 Sensor In-Flight Monitoring

4.3.3.6 Actuator/Valve Build-In Test

4.3.3.7 Spark Igniter Built-In Test

4.3.4 Instrumentation Parameter List

4.3.4.1 Alternate Performance Control Parameters

4.3.5 Engine Actuator

4.3.5.1 Operating Characteristics

4.4 Reaction Control Subsystem

4.4.1 Configuration

4.4.2 Thruster Description

4.4.3 Propellant Tankage

4.4.4 Pressurization Subsystem

4.4.5 RCS Operation

4.5 Orbital Maneuvering Subsystem (OMS)

4.5.1 OMS Configuration

4.5.2 Engine

4.5.3 Tankage

4.5.4 Pressurization

4.5.5 Installation

4.5.6 Environmental Control

4.5.7 Maintenance

4.5.8 Operations

4.6 Airbreathing Propulsion Subsystem (ABPS)

4.6.1 Configuration

4.6.2 Fuel Tank

4.6.3 Ferry Configuration

4.6.4 Operation

4.7 Solid Rocket Motion (SRM)

4.7.1 Main SRM

4.7.2 Abort Solid Rocket Motor

4.7.3 Deorbit SRM for External Tank (ET)

4.8 External Tank Subsystem (ET)

4.8.1 Structure

4.8.2 Thermal Protection System (TPS)

4.8.3 Deorbit Motor

4.8.4 Avionics

4.8.4.1 Instrumentation

4.8.4.2 Separation

4.8.4.2.1 Electrical Power

4.8.4.2.2 Interface

4.9 Guidance, Navigation and Control (Less Computer)

4.10 Communications and Tracking
4.10.1 S-Band System
4.10.1.1 S-Band Voice
4.10.1.2 S-Band T/M
4.10.1.3 Command Data
4.10.1.4 Video Link
4.10.1.5 Wide Bank Data Link
4.10.1.6 Range Measurement
4.10.2 VHF System
4.10.3 UHF System
4.10.4 Audio Control Center
4.10.5 TACAN
4.10.6 Radar Altimeter
4.10.7 ATC Transponder (ATCRBS)
4.10.8 Instrument Landing System (ILS)
4.10.9 GCA Radar
4.10.10 Air Route Surveillance Radar
4.10.11 Precision Ranging System (PRS)
4.10.12 Microwave Landing System (MLS)

4.11 Operational Instrumentation

4.11.1 Recorders

4.11.2 Sensors and Signal Conditioning

4.11.3 Ground Support Equipment PCM Links

4.11.4 Caution and Warning System

4.12 Environmental Control and Life Support Subsystem (ECLS)

4.12.1 Atmospheric Revitalization

4.12.1.1 Pressure Control

4.12.1.2 Cabin Humidity, CO_2, Odor, and Temperature Control

4.12.2 Life Support

4.12.2.1 Food Management

4.12.2.2 Waste Management

4.12.2.3 Personal Hygiene

4.12.2.4 Fire Detection and Extinguishing

4.12.2.5 Extravehicular/Intravehicular Service and Recharge Station and Airlock

4.12.3 Thermal Control

4.12.3.1 Coolant Loops

4.12.3.2 Heat Sinks

4.12.3.3 Water Management

4.12.4 Fault Detection Management

4.12.5 Ground and Launch Operations

4.12.6 Cabin Noise Level

4.12.7 Corrosion and Contamination Control

4.13 Payload Accommodation System

F-398-8-A

4.13.1 Structural/Mechanical Interfaces

4.13.1.1 General Structural Attachment

4.13.1.2 Payload Deployment and Retrieval Mechanism

4.13.1.3 Payload Control/Display Panels

4.13.2 Propulsive System Fluid Interfaces

4.13.3 Electrical/Instrumentation Interfaces

4.13.4 Payload Avionics Signal Interface

4.13.5 Payload Environment Control

4.13.6 Payload Doors

4.13.7 Rendezvous and Docking Sensors

4.13.8 Payload Dedicated Recorder

4.13.9 Payload Handling Station

4.13.10 Payload Bay Lighting

4.14 Crew Station Instrumentation System

4.15 On-Board Computers

4.15.1 GN&C Computer

4.15.2 Payload Monitor System Computer

4.15.3 Operational Instrumentation Computer

4.15.4 Main Engine Computer

4.16 Miscellaneous Systems

4.16.1 Purge and Vent System

4.16.2 Landing/Braking System

4.16.3 Glide Brake System

4.16.4 Ejection Seat Mechanism

4.16.5 Docking Mechanism

4.17 Onboard Computer System

4.18 GN&C Onboard Computer and Interface Systems Requirements
4.18.1 GN&C General Functional Description
4.18.2 GN&C System Elements & Interfaces
4.18.2.1 Air Data Equipment
4.18.2.2 Rate and Accelerometer Sensors
4.18.2.3 IMU
4.18.2.4 Star Tracker System
4.18.2.5 Horizon Sensor
4.18.2.6 Analog S.A.S.
4.18.2.7 TVC Electronics
4.18.2.8 APS Logic Drive Unit
4.18.2.9 GN&C Computer Interface
4.18.2.9.1 Computer Data Acquisition & Data Conversion
4.18.2.9.2 Remote Multiplexing Unit
4.18.2.9.3 Input Buffer
4.18.2.9.4 Output Decoder
4.18.3 GN&C Computer
4.18.4 Orbiter GN&C Operational Flight Program
4.18.5 Rational
4.18.6 Assumptions
4.18.7 Data References

4.19 MDE

4.19.1 General Function

4.19.2 System Operation and Interfaces

4.19.2.1 Display Unit

4.19.2.2 Keyboard

4.19.2.3 IOB Adapter

4.19.2.4 Keyboard Adapter

4.19.2.5 Symbol Generator

4.19.2.6 PCM Adapter

4.19.2.7 Processor CPU-I/O

4.19.2.8 Lamp Buffer Adapter

4.19.2.9 System Operation

4 19.3 MDE Computer

4.19.4 MDE Software & Systems Applications

4.19.4.1 MDE Computer Software

4.19.4.2 GN&C MDE System Application

4.19.4.2.1 GN&C Display Systems Software

4.19.4.2.2 GN&C Performing Monitoring Software

4.19.4.3 Performance Monitor System

4.19.4.4 Payload System

4.19.5 Rationale

4.19.6 Assumptions

4.19.7 Data References

4.20 Main Engine Controller
4.20.1 Controller Design Requirements
4.20.2 Controller Design Description
4.20.3 Controller Interfaces
4.20.4 Controller Hardware
4.20.5 Controller Software
4.21 Thermal Protection System
4.22 Thermal Control System
5.0 External Interfaces
5.1 Display & Controls (D&C) Consoles
5.2 Universal Control System (UCS)
5.3 Command Acquisition Unit
5.4 Remote Interface Unit
5.5 Shuttle Vehicle Systems Interfaces

6.0 Crew Procedures
6.1 Mission Objectives
6.1.1 Ascent
6.1.2 Rendezvous
6.1.3 Orbit
6.1.4 Payload Operations
6.1.5 Return
6.1.6 Ferry Operations
6.1.7 Abort Operations
6.2 Requirements for Crew Participation
6.3 Task Requirements
6.3.1 Commander and Pilot
6.3.2 Mission Specialist
6.3.3 Payload Specialist
6.4 Work Station Intra-Relationship

8.0 Visual Cues
8.1 The Human Visual System - Some Observations
8.1.1 Introduction
8.1.2 Visual Simulation System Parameters
8.1.2.1 Field of View
8.1.2.2 Brightness
8.1.2.3 Contrast
8.1.2.4 Resolution
8.1.2.5 Color
8.1.2.6 References
8.2 Windows
8.2.1 Description
8.2.2 Field of View
8.2.2.1 Forward Windows
8.2.2.2 Payload Handling Station
8.2.3 Assumptions
8.3 Ascent Phase (Vertical Launch to Orbit Insertion)
8.3.1 Scene Content
8.3.1.1 Horizon
8.3.1.2 Terrain
8.3.1.3 Celestial Bodies
8.3.1.4 Own Vehicle
8.3.1.5 Atmospheric Effects
8.3.2 Color

8.3.3 Illuminators/Non-Illuminators

8.3.4 Displacement

8.3.4.1 Translation

8.3.4.2 Rotation

8.3.5 Velocity

8.3.5.1 Translation

8.3.5.2 Rotation

8.3.6 Acceleration

8.3.6.1 Translation

8.3.6.2 Rotation

8.3.7 Assumptions

8.4 Abort Phase (Vertical Launch)

8.4.1 Scene Content

8.4.1.1 Horizon

8.4.1.2 Terrain

8.4.1.3 Celestial Bodies

8.4.1.4 Orbiting Vehicles

8.4.1.5 Own Vehicle

8.4.1.6 Atmospheric Effects

8.4.1.7 Other Aircraft

8.4.2 Color

8.4.3 Illuminators/Non-Illuminators

8.4.4 Displacements

8.4.4.1 Translation

8.4.4.2 Rotation

8.4.5 Velocity

8.4.5.1 Translation

8.4.5.2 Rotation

8.4.6 Acceleration

8.4.6.1 Translation

8.4.6.2 Rotation

8.4.7 Assumptions

8.5 Orbital Operations

8.5.1 Scene Content

8.5.1.1 Horizon

8.5.1.2 Terrain

8.5.1.3 Celestial Bodies

8.5.1.4 Orbiting Vehicles

8.5.1.5 Own Vehicle

8.5.1.6 Atmospheric Effects

8.5.2 Color

8.5.3 Illuminators/Non-Illuminators

8.5.4 Displacement

8.5.4.1 Translation

8.5.4.2 Rotation

8.5.5 Velocity

8.5.5.1 Translation

8.5.5.2 Rotation

8.5.6 Acceleration
8.5.6.1 Translation
8.5.6.2 Rotation
8.5.7 Assumptions
8.6 High and Low Altitude Rendezvous Phase
8.6.1 Scene Content
8.6.1.1 Horizon
8.6.1.2 Terrain
8.6.1.3 Celestial Bodies
8.6.1.4 Orbiting Vehicles
8.6.1.5 Atmospheric Effects
8.6.2 Color
8.6.3 Illuminators/Non-Illuminators
8.6.4 Displacement
8.6.4.1 Translation
8.6.4.2 Rotation
8.6.5 Velocity
8.6.5.1 Translation
8.6.5.2 Rotation
8.6.6 Acceleration
8.6.6.1 Translation
8.6.6.2 Rotation
8.7 Docking and Undocking Phase
8.7.1 Scene Content

8.7.1.1 Horizon

8.7.1.2 Terrain

8.7.1.3 Celestial Bodies

8.7.1.4 Orbiting Vehicles

8.7.1.5 Atmospheric Effects

8.7.2 Color

8.7.3 Illuminators/Non-Illuminators

8.7.4 Displacement

8.7.4.1 Translation

8.7.4.2 Rotation

8.7.5 Velocity

8.7.5.1 Translation

8.7.5.2 Rotation

8.7.6 Acceleration

8.7.6.1 Translation

8.7.6.2 Rotation

8.8 Payload Operations Phase

8.8.1 Scene Content

8.8.1.1 Horizon

8.8.1.2 Terrain

8.8.1.3 Celestial Bodies

8.8.1.4 Orbiting Vehicles

8.8.1.5 Own Vehicle

8.8.1.5.1 Payload Bay Doors

8.8.1.5.2 Remote Manipulator System

8.8.1.5.3 TV Cameras and Monitors

8.8.1.6 Atmospheric Effects

8.8.2 Color

8.8.3 Illuminators/Non-Illuminators

8.8.4 Displacement

8.8.4.1 Translation

8.8.4.2 Rotation

8.8.5 Velocity

8.8.5.1 Translation

8.8.5.2 Rotation

8.8.6 Acceleration

8.8.6.1 Translation

8.8.6.2 Rotation

8.8.7 Assumptions

8.9 De-Orbit Phase

8.9.1 Scene Content

8.9.1.1 Horizon

8.9.1.2 Terrain

8.9.1.3 Celestial Bodies

8.9.1.4 Orbiting Vehicles

8.9.1.5 Atmospheric Effects

8.9.2 Color

8 9.3 Illuminators/Non-Illuminators

8.9.4 Displacement
8.9.4.1 Translation
8.9.4.2 Rotation
8.9.5 Velocity
8.9.5.1 Translation
8.9.5.2 Rotation
8 9.6 Acceleration
8.9.6.1 Translation
8.9.6.2 Rotation
8.10 Entry
8.10.1 Scene Content
8.10.1.1 Horizon
8.10.1.2 Terrain
8.10.1.3 Celestial Bodies
8.10.1.4 Atmospheric Effects
8.10.2 Color
8.10.3 Illuminators/Non-Illuminators
8.10.4 Displacement
8.10.4.1 Translation
8.10.4.2 Rotation
8.10.5 Velocity
8.10.5.1 Translation
8.10.5.2 Rotation
8.10.6 Acceleration

8.10.6.1 Translation

8.10.6.2 Rotation

8.10.7 Assumptions

8.11 Approach and Landing Phase

8.11.1 Scene Content

8.11.1.1 Horizon

8.11.1.2 Terrain

8.11.1.3 Celestial Bodies

8.11.1.4 Atmospheric Effects

8.11.1.5 Other Aircraft

8.11.2 Color

8.11.3 Illuminators/Non-Illuminators

8.11.4 Displacements

8.11.4.1 Translation

8.11.4.2 Rotation

8.11.5 Velocity

8.11.5.1 Translation

8.11.5.2 Rotation

8.11.6 Acceleration

8.11.6.1 Translation

8.11.6.2 Rotation

8.11.7 Assumptions

8.12 Ferry Flight Phase

8.12.1 Scene Content

8.12.1.1 Horizon

8.12.1.2 Terrain

8.12.1.3 Celestial Bodies

8.12.1.4 Other Aircraft

8.12.1.5 Own Aircraft

8.12.2 Color

8.12.3 Illuminators/Non-Illuminators

8.12.4 Displacement

8.12.4.1 Translation

8.12.4.2 Rotation

8.12.5 Velocity

8.12.5.1 Translation

8.12.5.2 Rotation

8.12.6 Acceleration

8.12.6.1 Translation

8.12.6.2 Rotation

8.12.7 Assumptions

9.0 Cue Requirements
9.1 Propulsion Cues
9.1.1 Main Rocket Engines
9.1.2 Solid Rocket Motors
9.1.3 Airbreathing Engines
9.1.4 Abort Solid Rocket Motors
9.2 System Equipment Cues
9.3 Aerodynamic Cues
9.4 Caution and Warning Cues
9.5 Landing Gear Cues
9.6 Malfunction Cues

1.0 Introduction

The objective of the Shuttle Vehicle and Mission Simulation Requirements report is to provide to NASA/MSC documentation of the requirements for faithful simulation of the Shuttle Vehicle, its systems, mission, operations and interfaces. To accomplish this objective the report was divided into eight topics which comprehensively cover the simulation requirements of the Shuttle mission and vehicle. The topics and their main objectives are summarized below.

Mission Envelope - This topic covers the space and atmospheric missions that are envisioned for the Shuttle program. The characteristics of each mission are described by an analysis of the mission phases, trajectory information, timelines and operations for nominal and abort conditions to the extent data was available.

Orbiter Flight Dynamics - This topic covers the flight regimes which the Shuttle vehicle will encounter in the accomplishment of its missions. The requirements were established in the following manner.

The vehicle configurations that must be simulated for horizontal and vertical test flights, operational space missions, atmospheri missions and abort modes were defined.

The dynamics requirements were established by defining the forces and moments that will act on the vehicle during the entire mission envelope which include, propulsion, gravity, aerodynamic effects, payload effects, docking effects, staging effects, ground reactions and the dumping of material overboard. The translational equations of motion requirements were established by defining the vehicles, satellites and payloads whose state vectors must be calculated and by defining the coordinate systems, relative equations of motion and accuracy of the calculations. A similar analysis was performed for the rotational equations of motion. Mass property and ephemeris requirements were also identified.

<u>Shuttle Vehicle Systems</u> - The Shuttle vehicle systems required for simulation were identified and described. The descriptive data generated in this effort was primarily based on the North American Shuttle proposal. The Shuttle vehicle and its system configuration is currently in a state of flux and therefore the descriptive data

contained in this report undoubtedly will become out of date as the Shuttle program progresses. However, for the purposes of this study, the data is more than adequate to define simulator requirements and a baseline design when it is tempered with the past experience of Apollo and Gemini programs. A cross correlation between the NR definition of systems and LRU's and this report is shown in Table 1-1 for reference purposes.

External Interfaces - The external interfaces of the Shuttle vehicle were identified and a preliminary type interface description established. Due to the fact that for every external interface there also exists an equivalent on-board system, the descriptive data on the workings of the interfaces is contained in the Shuttle Vehicle Systems section of the report and cross references are provided in this section.

Crew Procedures - The actual crew procedures for the Shuttle system will not be available for many years. As a result the study concentrated on identifying tasks by mission phase and crew member and identifying the probable interfaces between work stations. The data used for the analysis was a RTOP study by MacDonald-Douglas, conversations with George Franklin of NASA/MSC, past experience, and the requirements of the Shuttle vehicle & mission.

Crew Station - The latest available data at the time of the writing of this report was used to identify the configuration of the Crew Station. The shape of the interior cabin, the location of the work stations and the allocation of the C&D panels by work station were established. Detailed data on the interior composition of the cabin is not currently available. However, simulation requirements were identified based on past experience and accepted levels of fidelity for mission simulators.

Visual Cues - The visual scene content was established for each of the mission phases. Attributes of the scene elements, to the extent feasible, were established and will be further defined in the SMSR report. The vehicle window configuration is not defined at this time but the best data available was utilized. The accelerations, velocites and displacements were established to the extent possible. Some dynamics data was not available such as in the Abort phases of the mission. The missing information will be incorporated if it becomes available when the time frame and ground rules of the study or assumptions will be made.

<u>Aural Cues</u> - The aural cues requirements associated with the mission and vehicle systems were identified and described. Detailed data on the characteristics of each sound was not available and probably will not be until the vehicle test program is in progress. This factor can be circumvented by specifying flexibility into the simulator aural cue equipment.

This report will be updated at the end of the study based on data received as of January 1, 1972.

Reference to study data sources are included in the margins and the text in order to facilitate update of this report. The numerical references are correlated with the data listing defined by Table 1-2.

F-398-8-A

TABLE 1-1
SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: AVIONICS

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	/Remarks/Assumptions
Star Sensor	3	4.9	ITT Model used on Aero Bee but does not meet proposed specs. Specs. and data required.
Rate Sensor Package	3	4.9	Honeywell GG 1027 Model used on F-14 AFCS. Data and Specs. required
Angle of Attack Transducer	3	4.9	Honeywell HG 280 used on DC 10.
IMU	3	4.9	Singer model KT70 used on A7D/E.
IMU Power Supply	3	4.9	Singer model KT70 used on A7D/E.
TVC Monitor	2 (?)	4.9.	No Data Exact function not known.
Air Data Package	3 (?)	4.9. 4.9.	Honeywell Model HG280 used on DC10.
MPS TVC Drivers	3	4.3, 4.9.	No Data Available
Manual TVC/RCS Control	1	4.9.	Honeywell Model BG 286 used on Apollo SCS.
Aero Control Electronics Unit	?	4.9.	Honeywell AFCS used on F-14.
Horizon Sensor Assembly	3	4.9.	Barnes Model 15-163
OMS/TVC Driver Unit	3(?)	4.9.	No Data Available

TABLE 1-1

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: AVIONICS

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks /Assumptions
APS Driver/Monitor	3	4.9.	Honeywell Model BG.287 used on Apollo SCS.
Accelerometer Package	3	4.9.	Honeywell Model G.G.1026 used on F-14 AFCS
Aero Back-up Electronics	1	4.9.	No Data available
Subsystem Sequence Controller	2(?)	4.9	To be used for unmanned flights. No data available
Gyro Accelerometer Package	1	4.9.	No Data Available
Backup Optical Unit	1	4.9.	Apollo COAS
Throttle/Speed Brake Electronics	?		No Data
GN &C Computer	3(?)	4.1.8.3	IBM Model AP101 or Singer/Kearfott SKC2000.
Program I/O Processor	(?)		IBM SP1
FDAI/EDA	(?)		Honeywell JG 264/BG 285 used on Apollo SCS.
FCS Control Panel	(?)		Honeywell F-14

TABLE 1-1

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: ELECTRICAL POWER

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
BATTERY	2	4.1 ELECTRICAL POWER	NICKEL-CADIUM - 10 AMPHOUR - 28 VOLT
GENERATOR CONTROL UNIT	3	4.1 ELECTRICAL POWER	APU DRIVEN GENERATOR
TRANSFORMER RECTIFIER UNIT	3	4.1 ELECTRICAL POWER	150 AMP
REMOTE CONTROL CIRCUIT BREAKER	?	4.1 ELECTRICAL POWER	MAGNETIC LATCH - HERMETIC SEALED UNITS
REMOTE POWER CONTROLLER	4	4.1 ELECTRICAL POWER	MAGNETIC LATCH - HERMETIC SEALED UNITS
BATTERY CHARGES	1	4.1 ELECTRICAL POWER	CONSTANT CURRENT CHARGER - DUAL REDUNDANT OUTPUT
INVERTERS	4	4.1 ELECTRICAL POWER	30, 1250 VA, 115/200V, 400 HZ
SEQUENCERS	2	4.1 ELECTRICAL POWER	NO DATA AVAILABLE
CONTROL TRANSFORMER RECTIFIER	?	4.1 ELECTRICAL POWER	NO DATA AVAILABLE
FUEL CELL	3	4.1 ELECTRICAL POWER	7/10 KW RESTARTABLE - CRYOGENIC O2 and H2 30 VOLT OUTPUT
ALTERNATOR - GENERATOR	3	4.1 ELECTRICAL POWER	20/30 KVA APU DRIVEN SPRAY OIL COOLED WITH INTEGRATED GEARBOX
FUEL CELL HEAT EXCHANGER	3	4.1 ELECTRICAL POWER	

TABLE 1-1

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: MECHANICAL POWER

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
AUXILLARY POWER UNITS	4	4.2.1 AUXILIARY POWER UNITS	200 HORSEPOWER - USES HYDRAZINE
HYDRAZINE TANK	4	4.2.1 AUXILIARY POWER UNITS	NO DATA AVAILABLE
HELIUM TANK	4	4.2.1 AUXILIARY POWER UNITS	NO DATA AVAILABLE
HEAT EXCHANGER	4	4.2.1 AUXILIARY POWER UNITS	NO DATA AVILABLE
HYDRALIC PUMPS	8	4.2.2 HYDRALIC POWER SYSTEM	3000 PSI
ALTERNATOR	3	4.2.1 AUXILIARY POWER UNITS	400 HZ, 30, 20/30 KVA

TABLE 1-1

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: OPERATIONAL INSTRUMENTATION

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
PILOT VOICE RECORDER	1	4.11.1 RECORDERS	
SWITCH SCAN MULTIPLEXER	12	FIGURE 4.11-1	
CAUTION AND WARNING	2	FIGURE 4.11-1, 4.11.4	AUTONETICS - APOLLO TYPE (NEW ITEM)
CRASH RECORDER	1	4.11.1 RECORDERS	SUNSTRAND, ECHO SCIENCE, OR DAVOLL FERRY USE ONLY
SIGNAL CONDITIONING UNIT-DFI	17	4.11.2 SENSORS AND SIGNAL CONDITIONING	SAT/APOLLO AUTONETICS SCE
TIMING UNIT (MTU)	2	FIGURE 4.11-1	APOLLO CTE, GENERAL TIMC
LOOP RECORDER	1	FIGURE 4.11-1 4.11.1 RECORDERS	SUNDSTRAND, ECHO SCIENCE, OR DAVOLL (5 MINUTE PLAYBACK)
PCM RECORDER - PAYLOAD	1	FIGURE 4.11-1 4.11.1 RECORDERS	SUNDSTRAND, ECHO SCIENCE OR DAVOLL (MAINT. AND PAYLOAD)
OPER. TRANSDUCERS	2359 DFI 2803 DFI	FIGURE 4.11-1 4.11.2 SENSORS AND SIGNAL CONDITIONING	VARIOUS MAKES
PCM REMOTE UNIT DFI	1	FIGURE 4.11-1	SCI, TELEDYNE
PCM MASTER UNIT - DFI	2	FIGURE 4.11-1	DFI ONLY SCI, TELEDYNE
GROUND CHECKOUT DECODER	?	?	MAY NOT EXIST

398-8-A

TABLE 1-1

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

YSTEM: OPERATIONAL INSTRUMENTATION

QUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
GN&C COMPUTER 64K	2?	4.18.3	IBM MODEL AP101 OR SINGER/KEARFOTT SKC 2000
INPUT-OUTPUT BUFFER	?	4.18.2.9.3/ 4.18.2.9.4	SP-1 COMPUTER STRUCTURES SKYLAB POWER SUPPLY, AP1/SP1
MDE UNIT	?	4.19-4.19-7	IBM SP1
MAGNETIC TAPE READER	?	4.19.2	NO DATA AVAILABLE
TAPE CONTROL ELECTRONICS	?	4.19.2	NO DATA AVAILABLE
CRT DISPLAY UNIT	8?	4.19.2.1	IBM-F14 TYPE HEAD WITH ADDITION OF A READ/WRITE REFRESH BUFFER, A SYMBOL
			GENERATOR, ANALOG AND DIGITAL CONTROL LOGIC, D/A'S AND POWER SUPPLIES
DFI TIMING UNIT	1		NO DATA AVAILABLE
WIDEBAND RECORDER	1		NO DATA AVAILABLE
FREQUENCY MULTIPLEXER	3		NO DATA AVAILABLE
PCM RECORDER DFI	1		NO DATA AVAILABLE
PCM RECORDER MAINTENANCE	1		NO DATA AVAILABLE

TABLE 1-1

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: OPERATIONAL INSTRUMENTATION

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
PCM MASTER UNIT - OFI	2		NO DATA AVAILABLE
PAYLOAD DATA INTERLEAVER	2		NO DATA AVAILABLE
PCM REMOTE ACQUISITION UNIT	24		NO DATA AVAILABLE
DEDICATED SIGNAL CONDITIONER	6		NO DATA AVAILABLE
COMMAND ACQUISITION UNIT	1		NO DATA AVAILABLE

38-8-A

TABLE 1-1

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

STEM: D&C

UIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
RTICAL SPEED	2	Note: The SV&MSR did not address detailed D&C instruments due to the lack of firm data	BENDIX E-C, AAK-23/A24G-17A
RO ALTITUDE	2	NO DATA AVAILABLE	AEROSONICS, AAU-16/A
S/MACH	2	NO DATA AVAILABLE	BENDIX E-C, ASK-14/A24G-18
AI (3 AXIS)	2	NO DATA AVAILABLE	MODIFIED APOLLO CM FDAI
I	2	NO DATA AVAILABLE	BENDIX E-C, ACA AQU-4A
S/SAT	1	NO DATA AVAILABLE	NO DATA AVAILABLE
CELEROMETER	2	NO DATA AVAILABLE	NO DATA AVAILABLE
POSITION	1	NO DATA AVAILABLE	3 DISPLAYS - LEFT, RIGHT, NOSE
S PRESSURE	3	NO DATA AVAILABLE	DOUBLE POINTER
IS PC	1	NO DATA AVAILABLE	NO DATA AVAILABLE
IS FUEL	1	NO DATA AVAILABLE	NO DATA AVAILABLE
IS OX	1	NO DATA AVAILABLE	NO DATA AVAILABLE

TABLE 1-1

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: D&C

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
EVENT TIMER	2	NO DATA AVAILABLE	NO DATA AVAILABLE
HYD. PRESSURE	1	NO DATA AVAILABLE	NO DATA AVAILABLE
MPS PC	1	NO DATA AVAILABLE	NO DATA AVAILABLE
MPS LH2/LO2	1	NO DATA AVAILABLE	NO DATA AVAILABLE
EXT. TANK QUANTITY	1	NO DATA AVAILABLE	NO DATA AVAILABLE
ISS DISAGREE	2	NO DATA AVAILABLE	NO DATA AVAILABLE
CMD DISAGREE	2	NO DATA AVAILABLE	NO DATA AVAILABLE
DRIVER FAIL	2	NO DATA AVAILABLE	NO DATA AVAILABLE

TABLE 1-1

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: COMMUNICATION AND TRACKING

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
SGLS INTERROGATOR	2	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
VHF TRANSCEIVER	2	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
ATC TRANSPONDER	2	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
SGLS TRANSPONDER	2	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
SGLS DECODER	2	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
USB TRANSPONDER	2	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
SIGNAL PROCESSOR	2	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
AUDIO CONTROL CENTER	2	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
TACAN TRANSPONDER	3	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
COMMAND DECODER	2	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
RADAR ALTIMETER	3	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
WIDEBAND TRANSMITTER S-BAND	1	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE

TABLE 1-1

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: COMMUNICATION AND TRACKING

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
S-BAND ANTENNA	4	4.10 COMMUNICATIONS AND TRACKING	HELIX IN CAVITY (RHCP)
C-BAND ANTENNA	6	4.10 COMMUNICATIONS AND TRACKING	HORN (LP) FOR RADAR ALTIMETER
L-BAND ANTENNA	1	4.10 COMMUNICATIONS AND TRACKING	ANNULAR SLOT (VP) FOR TACAN AND ATC
UHF/VHF ANTENNA	3	4.10 COMMUNICATIONS AND TRACKING	HP DUAL CAVITY FOR ILS
VHF ANTENNA	2	4.10 COMMUNICATIONS AND TRACKING	HELIX IN CAVITY (RHCP)
VHF ANTENNA	1	4.10 COMMUNICATIONS AND TRACKING	TOP CAP (VP)
VHF ANTENNA	1	4.10 COMMUNICATIONS AND TRACKING	SPIRAL (VP)
L-BAND ANTENNA	2	4.10 COMMUNICATIONS AND TRACKING	HELIX IN CAVITY (RHCP) FOR TACAN
L-BAND ANTENNA SELECTOR	1	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
VHF ANTENNA SELECTOR	1	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
S-BAND ANTENNA SELECTOR	1	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
CCTV CAMERA (B&W)	4	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE

TABLE 1-1

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: COMMUNICATION AND TRACKING

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
TV CAMERA - COLOR	1	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
DFI TRANSMITTER	1	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
ILS RECEIVER	3	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
SECURE TERMINAL	4	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE

TABLE 1-1

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: AIRBREATHING PROPULSION

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
TURBOFAN ENGINE	4	4.6 AIRBREATHING PROPULSION SYSTEM	P&W F401-PW-400 MODIFIED
FUEL TANK	1,2,or3	4.6 AIRBREATHING PROPULSION SYSTEM	22,565 LBS FOR EACH TANK

TABLE 1-1

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: SOLID ROCKET MOTORS

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
MOTOR	2	4.7.1 MAIN SRM	SOLID PROPELLANT 156"
MOTOR	12	4.7.1 MAIN SRM	SOLID PROPELLANT 27K LBS @ 2 SECONDS
MOTOR	2	4.7.2 ABORT SOLID ROCKET MOTOR	SOLID PROPELLANT 385,000 LBS AVERAGE @ 21 SECONDS
MOTOR	2	4.7.3 DEORBIT SRM FOR EXTERNAL TANK	SOLID PROPELLANT 18,500 LBS @ 37 SECONDS

TABLE 1-1

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: EXTERNAL TANK

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
OXYGEN TANK - LIQUID	1	4.8.1 STRUCTURE	
HYDROGEN TANK - LIQUID	1	4.8.1 STRUCTURE	
BATTERY	2	4.8.4 AVIONICS	
ORDINANCE TIMING SYSTEM (DEORBIT AVIONICS)	1	4.8.4 AVIONICS	
RANGE SAFETY AVIONICS	1	4.8.4 AVIONICS	

TABLE 1-1

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: MAIN PROPULSION SYSTEM

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
MPS ENGINE INTERFACE UNIT	3	4.3 MAIN PROPULSION SYSTEM	NO DATA AVAILABLE
MPS CONTROLLER	3	4.3 MAIN PROPULSION SYSTEM	NO DATA AVAILABLE
FUEL PREBURNER	3	4.3 MAIN PROPULSION SYSTEM	NO DATA AVAILABLE
OXIDIZER PREBURNER	3	4.3 MAIN PROPULSION SYSTEM	NO DATA AVAILABLE
MAIN ENGINE	3	4.3 MAIN PROPULSION SYSTEM	470K VAC THRUST
ENGINE ACTUATOR	6	4.3.5 ENGINE ACTUATOR	NO DATA AVAILABLE

TABLE 1-1

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: REACTION CONTROL SYSTEM

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
REACTION CONTROL MOTOR	40	4.4 REACTION CONTROL SYSTEM	1000 LBS THRUST VAC
HELIUM PRESSURIZATION SPHERES	4	4.4.4 PRESSURIZATION SUBSYSTEM	4000 PSI TANKS
MONOPROPELLANT HYDRAZINE FUEL TANKS	8	4.4 REACTION CONTROL SYSTEM	25 INCH DIAMETER X 42 INCH LONG CYLINDERS

TABLE 1-1

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: ORBITAL MANEUVERING SYSTEM (OMS)

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
ENGINE	2	4.5 ORBITAL MANEUVERING SYSTEM	5000 LBS VAC
TANK OXIDIZER	2	4.5 ORBITAL MANEUVERING SYSTEM	12,200 LBS MAX. - NITROGEN TETROXIDE
TANK FUEL	2	4.5 ORBITAL MANEUVERING SYSTEM	12,200 LBS MAX. - MONOMETHYLHYDRAZINE
HELIUM TANK	2	4.5 ORBITAL MANEUVERING SYSTEM	3200 PSIA

SORTED BY INDEX NUMBER

DATA REFERENCES
SHUTTLE MISSION SIMULATOR STUDY

OCTOBER 20, 1972

DOC. SRCE	DOC. DATE	NUMBER	REV	DOCUMENT TITLE	DATE RECD	LOCN	SEQ NO.
GE	15MR72	MSC-03824		SS PHASE B EXTENSION FINAL REPORT-PAYLOAD IMPACT V2A	15JE72	HB	002
NA	25FE72	FM347234		PAN AMERICAN APPROACH TO SHUTTLE CREW SEL/TRNG/ASSIGN	15JE72	HB	003
NA	15MA72	MSC-04217	C	SPACE SHUTTLE GN+C DESIGN EQUATION DOCUMENT	15JE72	HB	004
GE	15MR72	MSC-03824		SS PHASE B EXTENSION FINAL REPORT-MASS PROPERTIES V3	15JE72	HB	005
GE	15MR72	MSC-03824		SS PHASE B EXTENSION FINAL REPORT-EXECUTIVE SUMMARYV1	15JE72	HB	006
MD	15MR72	MDC-E0558		TECHNICAL REPORT SYSTEM + ORBITER PART 2 VOL 1	15JE72	HB	007
MD	15MR72	MDC-E0558		TECHNICAL REPORT SYSTEM + BOOSTER PART 2 VOL 2	15JE72	HB	008
MD	15MR72	MDC-E0558		FINAL MASS PROPERTIES REPORT PART 4	15JE72	HB	009
MD	15MR72	MDC-E0558		DEVELOPMENT REQUIREMENTS PART 3	15JE72	HB	010
MD	15MR72	MDC-E0558		TECHNICAL REPORT-MMC ACTIVITY PART 2 VOL 3	15JE72	HB	011
NR	15MR72	MSC-03332		SS PHASE B FINAL REPORT-TECHNICAL SUM.ADD.A-BOOSTER	15JE72	HB	012
LC	15MR72	NAS826362		SPACE SHUTTLE CONCEPTS TECHNICAL REPORT VOL 2	15JE72	HB	013
MD	15MR72	MDC-E0558		EXECUTIVE SUMMARY PART 1	15JE72	HB	014
NR	15MR72	MSC-03333		SS PHASE B FINAL REPORT-MASS PROPERTIES STATUS REPORT	15JE72	HB	015
GE	15MR72	MSC-03824		SS PHASE B FINAL REPORT-TECHNICAL REPORT V2	15JE72	HB	016
NA	09FE72	EG13728		SPACE SHUTTLE GUIDANCE AND NAVIGATION REVIEW	15JE72	HB	017
BC	04JA72			STUDY OF MOTION SYSTEM REQ. FOR SIM. OF ADV. SPACECR.	15JE72	HB	018
NA	15MR72	MSC-06720		SOURCE DOCUMENTATION LIST VOL 2 CAT 2	15JE72	HB	019
NA		RFP		SPACE SHUTTLE PROGRAM REQUEST FOR PROPOSAL PHASE CD	15JE72	HB	020
NA	14JA72		A	SPACE SHUTTLE AVIONICS CONFIGURATION DEFINITION DATA	15JE72	HB	021
MM	DC71	MSC-05218		PREL. DES. OF SHUTTLE DOCKING AND CARGO HANDLING SYS.	15JE72	HB	022
NA	15MR72			DATA PKG FOR SHUTTLE TRAINING AIRCRAFT DEFINITION	15JE72	HB	023
NR	15MR72	MSC-03332		SS PHASE B FINAL REPORT-EXECUTIVE SUMMARY V1	15JE72	HB	024
NR	15MR72	MSC-03332		SS PHASE B FINAL REPORT-TECHNICAL SUMMARY V2	15JE72	HB	025
NA	15NV71	MSC-03690		SS ORBITER GN+C S/W FUNCT.REQ. VERTICAL FLIGHT OPNS.	15JE72	HB	026
NA	JA70	NHB8040.2		APOLLO CONFIG.MGT. MANUAL	07JL72	HB	027
NA	01DC71	MSC-04217	B	SHUTTLE GNC DESIGN EQNS VOL.1	15JE72	HB	028
NA	01DC71	MSC-04217	B	SS GN+C DESIGN EQUATIONS-PREFLIGHT THRU ORBIT INS. V2	15JE72	HB	029
NA	01DC71	MSC-04217	B	SS GN+C DESIGN EQUATIONS-ORBITAL OPERATIONS V3	15JE72	HB	030
NA	01DC71	MSC-04217	B	SHUTTLE GNC DESIGN EQUATIONS VOL 4 DEORBITAL ATM OPNS	15JE72	HB	031
NA	15JE72		B	PROGRAM PLAN C ULASKY	15JE72	HB	032
NA	15MR72	MSC-06720		SOURCE DOCUMENTATION LIST VOL 1 CAT 1	15JE72	HB	033
NA		INDEX		SPACE SHUTTLE DATA LIST	15JE72	HB	034
NR	12NV71	NAS910960		TECHNICAL REPORT PHASE B VOL 1	15JE72	HB	035
NR	12NV71	NAS910960		TECHNICAL REPORT PHASE B VOL 2	15JE72	HB	036
NR	25JE71	NAS910960		TECHNICAL SUMMARY ORBITER DEFINITION VOLUME 2 PART1	15JE72	HB	037
NR	25JE71	NAS910960		TECHNICAL SUMMARY ORBITER DEFINITION VOLUME 2 PART2	15JE72	HB	038
GE	12NV71	NAS911160		SPACE SHUTTLE LOW COST/RISK AVIONICS STUDY	15JE72	HB	039
GE	15DC71	NAS911160		SHUTTLE SYSTEMS EVALUATION ORBITER DATA VOLUME 3	15JE72	HB	040
NR	08MR72	NAS910960		SPACE SHUTTLE PHASE B FINAL AVIONICS REPORT	15JE72	HB	041
NA	21AP71	NAS826157		ENGINE DESIGN DEFINITION REPORT AVIONICS-PHASE CD	30JE72	HB	042
MD	01AP72	MDC-E0558		SIMULATION RESULTS REPORT	28JE72	HB	043
MD	30AP72	MDC-E0573		DISPLAYS + CONTROLS FUNCTIONAL REQUIREMENTS SPEC.	28JE72	HB	044

DOC. SRCE	DOC. DATE	NUMBER REV	DOCUMENT TITLE	DATE RECD	LOCN	SEQ NO.
NA	09AP72	MSC INDEX	MSC INFORMATION RETRIEVAL SYSTEM	27JE72	H	045
MD	01DC71	MDC-E0464	CREW INTERFACE DEFINITION STUDY PHASE 1	28JE72	H	046
AF	00AP66	FTCTR686	FACILITY DEFINITION STUDY FOR UNIV FLIGHT SIMUL/TRNR	15JE72	HB	047
MI	00JE72	E-2687	EVAL OF SYNC/ASYNC EXECUTIVE SYSTEM FOR SPACE SHUTTLE	06JL72	HB	048
NA	01MA71	MSC-02542	TYP.SHUTTLE MISSION PROFILES + ATT.TIMELINES V4	11JL72	H	049
NA	27AG71	T71-14939	REPRES. REENTRY MISSION PROF. FOR DELTA WING ORBITER	11JL72	H	050
NA	31JA72	NASW-2081	ECONOMIC ANALYSIS SHUTTLE SYSTEM VOL 2	11JL72	H	051
LC	15NV71	NAS826362	ALTERNATE CONCEPT + DEFINITION-SRM BOOSTERS PART 3	11JL72	H	052
LC	15NV71	NAS826362	ALTERNATE CONCEPT + DEFINITION-AVIONICS PART 4	11JL72	H	053
NA	20JE72	MSC-07034	FIRST VERTICAL FLIGHT TEST MISSION	14JL72	HB	054
NA	16JE72	MSC-07050	OPTIMUM SRM THRUST PROFILE-MINIMUM GLOW	14JL72	HB	055
NA	30MA72	MSC-07057	POST BLACK OUT GNC ANALYSIS OF ORBITER SPACE SHUTTLE	14JL72	HB	056
MD	15MR72	MDC-E0558	DESIGN DATA BOOK-PROGRAM AND SYSTEM BASELINE PARTS VI	14JL72	HB	057
MD	15MR72	MDC-E0558	DESIGN DATA BOOK-DRAWINGS VOL 2	14JL72	HB	058
MD	15MR72	MDC-E0558	DESIGN DATA BOOK-ORBITER AERO VOL 3	14JL72	HB	059
MD	15MR72	MDC-E0558	DESIGN DATA BOOK-BOOSTER AERO VOL 4	14JL72	HB	060
NA	11JL72	470 ICD14	MAIN ENGINE AVIONICS ICD-ROCKETDYNE	20JL72	HB	061
NR	14JE72	AIAA71659	ROCKETDYNES SPACE SHUTTLE MAIN ENGINE	20JL72	H	062
NA	MA71	MSC-04400	RECOMMENDED SPACE SHUTTLE COORDINATE SYSTEMS STANDARD	26JL72	HB	063
LG	26FF71	MSC02553	ADVANCED S/W TECHNIQUES FOR SHUTTLE DATA MAN. SYSTEM	25JL72	HB	064
SK	30NV70	SKC-2000	AEROSPACE DIGITAL COMPUTER-SKC 2000	07JL72	HB	065
NA	05AP72	MSCL07215	SOLID STATE TRANSUCER DEVELOPMENT%NEW HAND CONTROL(	28JL72	HB	066
NA	12MA72	MSC-07151	FLYING QUALITES REQUIREMENTS CLOSED LOOP FLY BY WIRE	26JL72	HB	067
NA	21DC70	A71-25530	SUPPLY + RESUPPLY OF STATIONS IN SYNCRONOUS ORBITS	25JL72	H	068
BA	15MR72	AIAA71313	INSTRUMENTAION REQUIRED FOR SHUTTLE MAINT + OPERATION	25JL72	H	069
NA	03AP72	MSC-06766	PERFORMANCE REQUIREMENTS-SHUTTLE FUNCTIONAL SIMULATOR	25JL72	H	070
NA	13JL72	CE2-72M86	MALFUNCTION STUDY (RELATIVE TO HFTS)	26JL72	HB	071
MD	30JE71	MDC-E0308	PHASE B TECHNICAL SUMMARY PART 11-1	07JL72	B	072
MD	30JE71	MDC-E0308	B TECHNICAL SUMMARY ORBITER PT11-2A	07JL72	B	073
MD	30JE71	MDC-E0308	B TECHNICAL SUMMARY ORBITER PT11-2B	07JL72	B	074
MD	30JE71	MDC-E0308	B TECH. SUMMARY BOOSTER PART 11-3	07JL72	B	075
MD	30JE71	MDC-E0308	B TECH. SUMMARY BOOSTER PT 11-3 APP	07JL72	B	076
NR	JL70	MSC-00706	SP.STA. C/D CREW TNG PLAN R DEFIN.	07JL72	B	077
NR	25JE71	MSC-03307	SHUTTLE B EXEC SUMMARY VOL.1	07JL72	B	078
NR	25JE71	MSC-03308	PROGRAM MGT.PLAN FOR PHASE C/D	07JL72	B	079
NR	25JE71	MSC-03327	EXTERNAL H2 TANK STUDY SUMMARY VOL1	07JL72	B	080
NA	17DC70	MSC-03793	SHUTTLE ENG.SIMULATION SURVEY	07JL72	B	081
GB	06JL71	MSC-03809	ALTERNATE CONCEPTS-EXEC SUM PART 1	07JL72	B	082
GB	06JL71	MSC-03810	ALT.CNCPTS.TECH.SUM.SHUT.DEF. P2V1	07JL72	B	083
GB	06JL71	MSC-03811	ALT.CNCPTS.ENG DEV.PLAN-ORBITER V2	07JL72	B	084
NA		MSC-04240	PRELIM.MISSION PROFILE-DESC.+LAND	07JL72	B	085
NA	27AP71	MSC-04243	PRE.ENTRY TGT PLANNING-ORBITER	07JL72	B	086
NA	24AG70	17618H002 0	ENTRY CORRIDORS FOR HI L/D (TRW)	07JL72	B	087
NA	06AP72		FLIGHT PERS.COMP.+SIMULATOR RES-INF	07JL72	B	088
NA	08JE72	C.CLASKY	PROGRAM PLAN-PROCEDURES DEVELOPMENT	28JE72	HB	089
SI	28JA72	ER-094	HFTS PROJECTED CONFIGURATION	JE72	HB	090
NR	19JE72		AVIONICS WEEK-WAR PHASE C/D CONFIG.	19JE72	B	091
NR			PRELIM NAR CONTROLS DISPLAY LAYOUT		B	092
SI	15JL72	SSPEHM31	SSPF DATA BOOK		B	093
NA	30AP71	MSC-04347	SHUTTLE TOUCHDOWN REQUIREMENTS		B	094

DOC. SRCE	DOC. DATE	NUMBER REV	DOCUMENT TITLE	DATE RECD	LOCN	SEQ NO.
AF	00MR72	HUMRR0772	TRANSFER OF INST TRAINING + SYNTH FLIGHT TRAINING SYS	28JL72	D	095
NA	30JE70	TMX-2101	FLIGHT TEST RESULTS FOR SPACE SHUTTLES (LIFT. BODIES)	26JL72	H	096
NA	11DC70	MSC-02542	TYPICAL SHUTTLE MISSION PROFILES + ATT TIME LINE VOL3	28JL72	HD	097
NA	07AG70	NAS910455	STUDY OF PERSONAL HYGIENE FOR FUTURE MANNED MISSIONS	28JL72	HD	098
NA	01OC71	MSC-C-005	GEN SPEC CONTROLS,FUNC DESIGN REQMTS MANNED SPACECRFT	28JL72	HD	099
NA	01NV71	MSC-D-007	GEN SPEC DISPLAYS,FUNC DESIGN REQMTS MANNED SPACECRFT	28JL72	HD	100
NA	30OC70	MSC-03779	HOUSEKEEPING CONCEPTS-WASTE CONTROL TASKS + SYS VOL I	28JL72	HD	101
NA	17MR71	MSC-04062	ORBITER FLIGHT TECH + LANDING PT SUBORBIT ABORT CNTRL	28JL72	HD	102
NA	00FE71	NAS9-4045	MAN ROLE INTEG VEHICULAR INFORMATION MANAGEMENT SYS	28JL72	HD	103
NA	25JE71	MSC-03305	ICD FOR SPACE SHUTTLE SYSTEM	28JL72	HD	104
NA	27JL71	MSC-04711	ORBITER FLIGHT TECHNIQUES FOR FLYBACK ABORTS	28JL72	HD	105
NA	11MY71	MSC-05148	PAYLOAD CG ANALYSIS FOR SHUTTLE,SIDE-BY-SIDE LOADING	28JL72	HD	106
NA	10JA72	MSC-05812	PAYLOAD CG ANALYSIS FOR SHUTTLE,END-TO-END LOADING	28JL72	HD	107
NA	15DC71	MSCEG7143	PILOTED SIMUL STUDY-UNPOWERED CROSSWIND LANDINGS	28JL72	HD	108
NA	00JI71	NAS911140	CHECKOUT SYS SUMMARY REPORT/UNIV CONTROL-DISPLAY CONS	28JL72	HD	109
NA	00DC71	NAS912420	TIFS SIMUL OF SSV LANDING APPROACH-MONTHLY REPORT	28JL72	HD	110
NA	15DC71	NAS912218	DEV OF OPTIMUM SIMUL OF SPACE STATION COMM TECHNIQUE	28JL72	HD	111
NA	00DC71	NAS912200	SSV ENTRY SIMULATION	28JL72	HD	112
NA	00DC71	NAS910950	STUDY ANALYSIS REPORT OF SSV MANUAL DOCKING SIMULATOR	28JL72	HD	113
NA	28SE71	AD444403	AIRCRAFT FLIGHT TEST INSTR ANALYSIS FOR SSV	28JL72	HD	114
NA	00FE72	NAS912420	TIFS SIMUL OF SSV LANDING APPROACH-MONTHLY REPORT	28JL72	HD	115
NA	15DC71	NASW-2129	INTEG SPACE PROGRAM + VEHICLE SYS ANALYSIS-STUDY B	28JL72	HD	116
NA	00AG71	NASW-2129	INTEG OPERATIONS/PAYLOAD/FLEET ANALYSIS VOL 2 PAYLOAD	28JL72	HD	117
NA	00AG71	NASW-2129	INTEG OPERATIONS/PAYLOAD/FLEET ANALYSIS VOL 5 MISSION	28JL72	HD	118
NA	00AG71	NASW-2129	INTEG OPERATIONS/PAYLOAD/FLEET ANALYSIS VOL 3 EXEC SM	28JL72	HD	119
NA	15JA72	MSCEG-723	SIMUL EVAL OF MANUAL CONTROL MODES-UNPOWERED DELTA/WG	28JL72	HD	120
NA	02JE71	NAS910960	SPACE SHUTTLE OPERATIONS REVIEW	28JL72	HD	121
NA	05JA71	NAS910960	SPACE STATION DOCKING/PAYLOAD XFER TRADE STUDY RESULT	28JL72	HD	122
TR	17FE72	724910432	DEF OF ORBITER FLIGHT ENVELOPE + INVESTIGATION FLT-TS	28JL72	HD	123
TR	25FE71	717251322	SSV COMMUNICATIONS COVERAGE V-POLAR ORBIT	28JL72	HD	124
NA	10JA72	MSCEG72-5	PROJECT SPACE SHUTTLE-SSV ENTRY SIMULATION	28JL72	HD	125
NR	02AP71	SV71-13	SSV CREW STATION REVIEW NO. 2	28JL72	HD	126
NA	25JE71	MSC-03310	SSV OPERATIONS PLAN FOR PHASE C/D VOL. 2 ORBITER	28JL72	HD	127
NA	00MA71	MSC-04411	SSV PAYLOAD HANDLING + DOCKING	28JL72	H	128
NA	19AP71	A71-36469	ORBITAL CARGO TRANSFER SYSTEM	28JL72	HD	129
NA	00JI71	NAS911948	SPACE STATION BUILDUP SIMULATION-FINAL REPORT	28JL72	HD	130
NA	10FE71	SSPD-27	SSV EVALUATION OF FOUR SIDESTICK CONTROLLERS ON SIMUL	28JL72	HD	131
NA	16MA72	MSC-04813 1	SSV PERFORMANCE CAPABILITIES-	01AG72	HD	132
NR	00JI72	22SV18329	DISPLAYS + CONTROLS MECHANIZATION SUMMARY	01AG72	HD	133
NA		RFP INFO	SSV PROPOSER DOCUMENTATION INFORMATION	01AG72	H	134
NR	00JI72	SK C+D	PRELIMINARY CONTROL + DISPLAY PANEL SKETCH	10AG72	HD	135
NR	19MA71	NAS910960	FLIGHT STATION + VISION PLOT-ORBITER	01AG72	HD	136
NR	01JE71	NAS910960	FLIGHT CONTROLS + DISPLAYS	01AG72	HD	137
NR	10JE71	NAS910960	CARGO DOCKING CONTROL STATION	01AG72	HD	138
NR	19FE71	NAS910960	MANIPULATOR ARMS-OPERATING STATION TV + LIGHTING	01AG72	HD	139
SI	02AG72		MINUTES-NASA/SINGER MEETING W C ULASKY SMS PROGRAM	02AG72	HD	140
SI	02AG72		MINUTES-NASA/SINGER MEETING W B SWINT MAIN ENGINE ICD	02AG72	HD	141
NR	25JE71	MSC-03310	SSV OPERATIONS PLAN FOR PHASE C/D VOL 3 BOOSTER	03AG72	H	142
AE	00AG71	NASACR	INTEGRATED OPERATION/PAYLOAD/FLEET ANALYSIS VOL 4	03AG72	H	143
AI	05JA71	A71-36642	PRELIM RESULTS MANNED CARGO XFER STUDY UNDER ZERO G	03AG72	H	144

DOC. SRCE	DOC. DATE	NUMBER REV	DOCUMENT TITLE	DATE RECD	LOCN	SEQ NO.
NA	00JE71	N70-40951	SPACE XPORTATION SYS TECH VOL 6 INTEGRATED ELEC/POWER	03AG72	H	145
NA	00JL72	DCPS-1	CREW PROCEDURES TASK DESCRIPTION-POST ABORT	03AG72	H	146
NR	04FE71	SV71-13	SSV CREW STATION REVIEW NO 2	03AG72	H	147
NR	17MA71	VA70-6001	CREW COMPARTMENT-DELTA ORBITER	04AG72	HB	148
NR	14JE71	VA70-3103	CREW COMPARTMENT-AND IVA TUNNEL STRUCTURE	04AG72	HB	149
SR	04NV70	N71-20624	GENERAL PURPOSE SIMULATOR SYSTEM STUDY-	04AG72	H	150
SD		PROPOSAL	SDC PROPOSAL MICROPROGRAM PROCESSORS IN AVIONICS	07AG72	HB	151
NA	24MR72	MSC-06747	GNC S/W FUNCTIONAL REQ. HOR. FLIGHT OPERATIONS	07AG72	HB	152
NR	21MR67		GRUMMAN OPTICAL ALIGNMENT SIGHT %COAST APOLLO SPEC	09AG72	HB	153
HC	00AG65	M-1043	DDP 516 GP COMPUTER USERS GUIDE	10AG72	H	154
HC	00AG65	M-978	DDP 516 GP COMPUTER PROGRAMMERS REFERENCE MANUAL	10AG72	H	155
MI	14JL70	23A 22-70	SSV REENTRY NAV. BARO ALTIMETER AND VOR/DME	10AG72	HB	156
MI	16FE72	23A 11-72	ENTRY + TERMINAL NAV USING MLS OR AILS + VOR/DME	10AG72	HB	157
MI	04OC71	23A 49-71	ENTRY + LANDING ORBITER USING PRS NAVAID	10AG72	HB	158
NA	31JL72	MSC-03690 C	SS ORBITER GNC S/W FUNCT. REQMNTS VERTICAL FLIGHT OPN	10AG72	HB	159
NA	27JE72	MSC-06900	SS BASELINE ACCOMODATIONS FOR PAYLOADS	10AG72	HB	160
NA	29NV71	LA112971- 1	SSV PROGRAM AVIONICS SYSTEM RECOMMENDATIONS	14AG72	HB	161
NA	11AP72	502-23-33	CENTRAL MULTIPROCESSOR + MAN-MACHINE TECHNIQUES	11AG72	HB	162
MC	04AG72	MSC-06744	DESIGN REQ. SPEC. VISUAL SIMULATION FOR SMS -PRELIM.[	11AG72	HB	163
NR		VE70-0629	CABIN STUDY NO 12A (DRAWING)	15AG72	HB	164
NR	DE71	VE70-0705	ORBITAL DISPLAYS + MAN FLT CONTROLS SUBSYS CONF.	01AG72	B	165
NR	12MA72	PROPOSAL	SPACE SHUTTLE PROGRAM TECHNICAL PROPOSAL VOL.3	17AG72	HB	166
NR	MA72	PROPOSAL	SS PROGRAM AERO DESIGN DATA BOOK VOL. 1 ORBITER	17AG72	HB	167
NR	MA72	PROPOSAL	SS PROGRAM AERO DESIGN DATA BOOK VOL. 2 MATED VEHICLE	17AG72	HB	168
NR	MA72	PROPOSAL	SS PROGRAM THERMODYNAMIC DESIGN DATA BOOK	17AG72	HB	169
NR	AP72	PROPOSAL	SS PROGRAM FLIGHT CONTROL DATA BOOK	17AG72		170
NR		PROPOSAL	SS PROGRAM EXTERNAL TANK	17AG72		171
HC		HDC-601	DIGITAL COMPUTER GENERAL DESCRIPTION	23AG72	H	172
HC		HDC-601	DIGITAL COMPUTER SOFTWARE SYSTEMS DESCRIPTION	23AG72	H	173
HC		HDC-601	PLATED WIRE MEMORY TECHNICAL DESCRIPTION	23AG72	H	174
SK	NV71	SKC-2000	SUBROUTINE LIBRARY REFERENCE MANUAL	22AG72	H	175
NA	17AP72	50M22157	SSME ACTUATOR SERVO LOOP SPECIFICATIONS-MSFC	22AG72	H	176
NA	10AP72	IEEE CONF	DIGITAL CONTROLLER FOR HIGH PRESSURE ROCKET ENGINE	22AG72	H	177
NR	12MA72	PROPOSAL	SS PROGRAM TEST PROPOSAL	24AG72	H	178
NA	27SE65	ISCPC	INTERPRETIVE SIMULATION APOLLO GUIDANCE COMPUTER	01SE72	HB	179
NA	18AL72	VISUAL	MINIMUM VISUAL REQUIREMENTS FOR APPROACH AND LANDING	06SE72	HB	180
NA	13AP72		SHUTTLE FLIGHT CREW NOMENCLATURE AND DUTIES-FRANKLIN	06SE72	HB	181
NR	30AL72	VL70U3013	SHUTTLE CABIN CONCEPTS-PER GEORGE FRANKLIN	06SE72	HB	182
NR	15MA72	PROPOSAL	SUPPORTING FUNCTIONAL ABSTRACTS	07SE72	H	183
NR	15JE72	PROPOSAL	ORAL DISCUSSIONS	07SE72	H	184
NA	26JL72	HAL MEMO	MINUTES HAL DEVELOPMENT MEETING 5	07SE72	HB	185
NA			LOWER PIVOT OF CONSOLE (PILOT SEAT TILT DRAWING)	07SE72	HB	186
IB		AP101/SP1	IBM OBC CANDIDATES FOR SHUTTLE	11SE72	HB	187
AI	20JL71	AIAA	SHUTTLE UTILIZATION FOR SORTIES AND AUTOMATED PAYLOAD	14SE72	H	188
NA	06JL72	MSC-07069	ABORT CAPABILITY ASSESSMENT PARALLEL SRM SHUTTLE	14SE72	H	189
NA	20DE71	MSC-05276	SHUTTLE INCREMENTAL VERTICAL FLIGHT TEST ASCENT STUDY	14SE72	H	190
NA	17AL72	MSC-07238	A REPRESENTATIVE REENTRY AND LANDING FOR SHUTTLE	14SE72	H	191
NA	10MA72	MSC-06869	ABORT CONTROL CONSIDERATIONS DURING 040C/EOH LAUNCH	14SE72	H	192
NA	16MA72	MSC-04026	COMPARATIVE ANALYSIS-SHUTTLE ASCENT TRAJ GRAVITY TURN	14SE72	H	193
NA	22FE72	MSC-05890	LANDING CHARACTERISTICS OF THE 040A ORBITER	14SE72	H	194

DOC. SRCE	DOC. DATE	NUMBER	REV	DOCUMENT TITLE	DATE RECD	LOCN	SEQ NO.
NA	06DE71	MSC-05239		LANDING CHARACTERISTICS OF 3 SHUTTLE CONFIGURATIONS	14SE72	H	195
NA	28JL72	MSC-07159		MANUALLY FLOWN FINAL APPROACH PATHS FOR ORBITER	14SE72	H	196
NA	18JE72	MSC-07044		ASCENT CONTROL CAPABILITY OF A PARALLEL SRM SHUTTLE	14SE72	H	197
NA	37AL72	MSC-07224		PRELIM. SHUTTLE ABORT STUDY USING SRM	14SE72	H	198
NA	06MR72	MSC-06002		OPTIMAL ASCENT FLYBACK TRAJECTORIES FOR SHUTTLE	14SE72	H	199
NA	15MR72	13M15000D		SPACE SHUTTLE ORBITER VEHICLE/ENGINE 470K VACUUM ICD	01SE72	H	200
NA	19MA72	TM X-2570		ASSESSMENT OF POTENTIAL BUFFET PROBLEMS SSV	14SP72	H	201
NR	000071	CP320R003		SPACE SHUTTLE MAIN ENGINE CEI SPEC	14SP72	H	202
NR	00JL70	SSE4.1A-4		SSE BASELINE CONFIGURATION CHANGE	14SP72	H	203
NR	00JL70	SSE4.1A-5		SSE EPL CAPABILITY	14SP72	H	204
NR	00JL70	SSE4.1A-8		SSE FUEL TANK PRESSURIZATION TAPOFF STUDY	14SP72	H	205
NR	00JL70	SSE4.1A-9		SSE SELECTION OF 1550 DEG F MAIN TURBINE OPER. TEMP	14SP72	H	206
NR	00JL70	SSE4.1A11		SSE DUAL VS SINGLE PREBURNER STUDY	14SP72	H	207
NR	00JL70	SSE4.1A12		SSE PRESTART PROPELLANT CONDITIONING SYSTEM	14SP72	H	208
NR	00JL70	SSE4.1A17		SSE TRANSLATING NOZZLE-ORBITER SSME	14SP72	H	209
NR	00AG70	SSE4.1424		SSE BASELINE ENGINE CONFIGURATION CHANGE	14SP72	H	210
NA	15MR72	13M15000	D	ORBITER VEHICLE/ENGINE 470K VACUUM ICD	14SP72	H	211
NA	27JE72	MSC-06900		BASELINE ACCOMMODATION FOR PAYLOADS	14SP72	H	212
MD	01AG72	MSC-04422		AUXILLIARY PROPULSION STUDY	14SP72	H	213
NR	01JA72	NAS912046		APS PROPELLANT THERMAL CONDITIONER STUDY REPORT 7	14SP72	H	214
NR	01DC71	NAS912046		APS PROPELLANT THERMAL CONDITIONER STUDY REPORT 6	14SP72	H	215
NR	01JL72	NAS912046		APS PROPELLANT THERMAL CONDITIONER STUDY REPORT 13	14SP72	H	216
MD	23JE72	MDCE0557		REUSABLE SURFACE INSULATION THERMAL PROTECTION-ADDEN	14SP72	H	217
NA	30MA72	TM X-2570		REUSABLE SURFACE INSULATION PRELIMINARY DATA	14SP72	H	218
NA	28DC70	TM V62123		NAV FOR SSV APPROACH + LANDING-PREC RANGE-MICRO SCAN	14SP72	H	219
NA	19MA72	TM X-2570		INFLIGHT AEROACOUSTIC ENVIRONMENT FOR SSV	14SP72	H	220
NR	21JL70	MSC-00718		SPACE STATION CREW OPERATIONS DEFINITION	20SP72	HB	221
AF	30AG72	PRELIM.		DOD SHUTTLE SYSTEM REQUIREMENTS	20SP72	HB	222
SK	SP72	SKC-2000		SKC-2000 ADVANCED DIGITAL COMPUTER	20SP72	H	223
NR	18JL72	RC1007		CONTROLLER SPACE SHUTTLE MAIN ENGINE	22SP72	HB	224
NR	18JL72	RC1010		COMPUTER PROGRAM REQUIREMENTS-CONTROLLER	22SP72	HB	225
NR	30MA72	RL00001	B	ENGINE BALANCE AND DYNAMIC MODEL	22SP72	HB	225
NR	15NV71	N72-23908		SPACE + PLANETARY ENVIRONMENT CRITERIA GUIDELINE-1971	22SP72	H	227
NA	11SP72	DRAFT		CREW/COMPUTER INTERFACE + DISPLAY FORMATS DEFINITION	22SP72	HB	228
NA				SPACE SHUTTLE PROGRAM REQUIREMENTS DOCUMENT-LEVEL 1	22SP72	HB	229
NA	26JA72	MSC-07070		ANALYSIS OF SHUTTLE ORBITER FERRY METHODS	22SP72	HB	230
NA	00AP72	STDN101.1		SPACECRAFT TRACKING + DATA NETWORK-BASELINE DOC.	29SP72	H	231
NR	12NV71	MSC-03329		ALTERNATE AVIONICS SYS STUDY + PHASE B EXTENSION	29SP72	H	232
NR	21JL70	MSC-00718		SPACE STATION CREW OPERATIONS DEF PHASE B	29SP72	B	233
XF				REAL TIME 6 DEG FREEDOM AIRCRAFT SIMULATION - SL-1	05OC72	B	234
NA	FE72	SD72-SA32		SPACE TUG DESIGN STUDY-VOLS 1,2,3-1,3-2,4,5	29SP72	H	235
NA	29JL71	MSC-04075		FUNCTIONAL PERFORMANCE REQ-SHUTTLE AVIONICS SYS	29SP72	H	236
RR				DIGITAL CONTROL SYSTEMS SELF-CHECK	05OC72	B	237
GE				SYSTEMS OPERATION SIMULATOR	05OC72	B	238
CP				ROLE OF MICRO PROGRAMMING IN FOURTH GENERATION COMPU	05OC72	B	239
IR				AEROSPACE SYSTEMS IMPLICATIONS OF MICROPROGRAMMING	05OC72	B	240
NA	10AG72	SSPFT2167		LONGITUDINAL + LATERAL-DIRECTIONAL DATA VL 70-000001	05OC72	HB	241
MT	06SP72	982-MC-02		MONTHLY PROGRESS REPORT-SIMUL COMPLEX STUDY	05OC72	HB	242
AF	03OC72	ZR1		DOD SHUTTLE MISSION SIMULATOR PLANNING DATA	05OC72	HB	243
GP	OLT	LIS440121		ICD FOR INTERP SIMUL LM GUIDANCE COMP/LMS	05OC72	HB	244

FORM 1411-3

PRINTED IN U.S.A.

DOC. SRCE	DOC. DATE	NUMBER REV	DOCUMENT TITLE	DATE RECD	LOCN	SEQ NO.
NA	21JF69	FHP-5-2	BLD NO 5 MISSION SIMULATION + TRAINING FACILITY	050C72	HS	245
			AIR TRAFFIC CONTROL SYS-DIGITAL SIMULATION FACILITY	050C72	HS	246
			A REAL TIME RADAR SIMUL USING APL DIGITAL COMP LINKS	050C72	HS	247
			A COMPARISON OF 3 TYPES OF MANUAL CONTROLS-3RD ORDER	050C72	HS	248
IN	FE72		INERTIAL SYS FUNCTIONAL + DESIGN REQ FOR ORBITER-R	050C72	HS	249
			REAL TIME SPACE VEHICLE + GSE SOFTWARE SIMULATOR-COD	050C72	HS	250
			TORQUE SENSITIVITY A-FUNCTION OF KNOB RADIUS + LOAD	050C72	S	251
MI	AG72	E-2708	ENERGY MANAGEMENT DURING SHUTTLE TRANSITION	050C72	HS	252
	AP72		SYSTEM 86 LIBRARY	050C72	HS	253
NA	JI72	MSC-07036	SIMULATION OF MANNED SPACE FLIGHT FOR CREW TRAINING	060C72	HS	254
ST	27SP72		PROGRAMMING ELEMENTS.SMS TRAINING-BURKE/VANBUCKEL/GRA	050C72	HS	255
MI	MR72		PROJECT INTREX	295P72	HS	256
IP			AP 101	100C72		257
			VISUAL STUDY MCDONNELL-DOUGLAS	100C72		258
			COMPUTER STUDY MCDONNELL-DOUGLAS	100C72		259
			SHUTTLE DEVELOPMENT SCHEDULE	100C72		260
			LMS CREW TRAINING SCHEDULE	100C72		261
GE	AP72	NAS911946	REMOTE MANIPULATOR DYNAMIC SIMULATION	100C72		262
MA	AP72	CR123570	SHUTTLE PROP SYS ONBOARD CHECKOUT + MONITORING DEV	100C72		263
CC	06MA72	CR61376	COMPUTER SYS SIMULATION + ANALYSIS	100C72		264
IN	16FE72	N7221203	ADV SFTWR TECH FOR DATA MGMT SYS VOL 3-PROG LANGUAGE	100C72		265
IN	16FE72	N7221205	ADV SFTWR TECH FOR DATA MGMT SYS VOL 2-SSV EXEC SYS	100C72		266
IN	16JF71	N7221204	ADV SFTWR TECH FOR DATA MGMT SYS VOL 1-SSV SW STUDY	100C72		267
MS	07MA72	CR123569	FLIGHT PROGRAM LANGUAGE REQMT	100C72		268
SF	- 71		ATC FOR THE SEVENTIES PART 1	100C72		269
SF	- 71		ATC FOR THE SEVENTIES PART 2	100C72		270
SF	- 71		ATC FOR THE SEVENTIES PART 3	100C72		271
AI	08MA71	A71-30381	PROG TECH FOR REAL TIME SYS FORM EXPERIENCE W EXPERMT	100C72		272
NA	DE70	TN-D-6066	DESIGN E OPER CHAR OF CENTRAL ONLINE DATA PROC-LANGLY	100C72		273
TN	AP71	TSC-FAA71	CONCEPTUAL NETWORK MODEL OF AIR TRANS SYS	100C72		274
MA	JA71	CR118310	VIRTUAL MEMORY SYS DESIGN	100C72		275
RA	JA71	CR118366	EXPERIENCE WITH EXTENDABLE COMPUTER SYS SIMULATOR	100C72		276
MT	MR71	MTR-1995	AIRCRAFT MOCKUP COMPUTER PROGRAM SPECIFICATIONS	100C72		277
AI	05MR68		NOMOGRAM FOR AIRCRAFT RUDDER PEAL DESIGN	100C72		278
AI	05MR69		OPTIMUM KNOB DIAMETER	100C72		279
AI	05MR69		DESIREABLE DIMENSIONS FOR CONCENTRIC CONTROLS	100C72		280
MI	JF69	CR100370	MANUAL CONTROL OF UNSTABLE VEHICLES-KINESTHETIC CUES	100C72		281
RA	AL69	RM-6027	ONLINE DEBUGGER FOR 65560 ASSEMBLY LANGUAGE PROGRAMS	100C72		282
MA	JA70	CR110445	METHOD FOR UNIFIED HARDWARE-SOFTWARE DESIGN	100C72		283

REF. KEY

4.0 SHUTTLE VEHICLE SYSTEMS

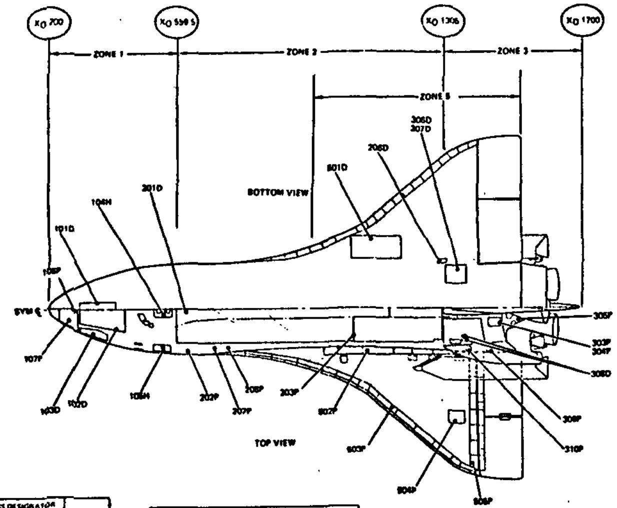
166 The shuttle vehicle is composed of three major units: the orbiter vehicle, the external fuel tank, and the two 156" external solid rocket motors. Figure 4.0-1 gives the relative location of the total vehicle major components. Note in this figure the Abort SRM's are shown attached to the shuttle vehicle body. In Figure 4.0-2 the shuttle vehicle does not have the ABORT SRM's in place and is in an orbital configuration. Inserts and cutaways provide general location of additional components of the shuttle vehicle. The payload shown is representative of one of the many possible configurations that may be accommodated in the shuttle payload bay.

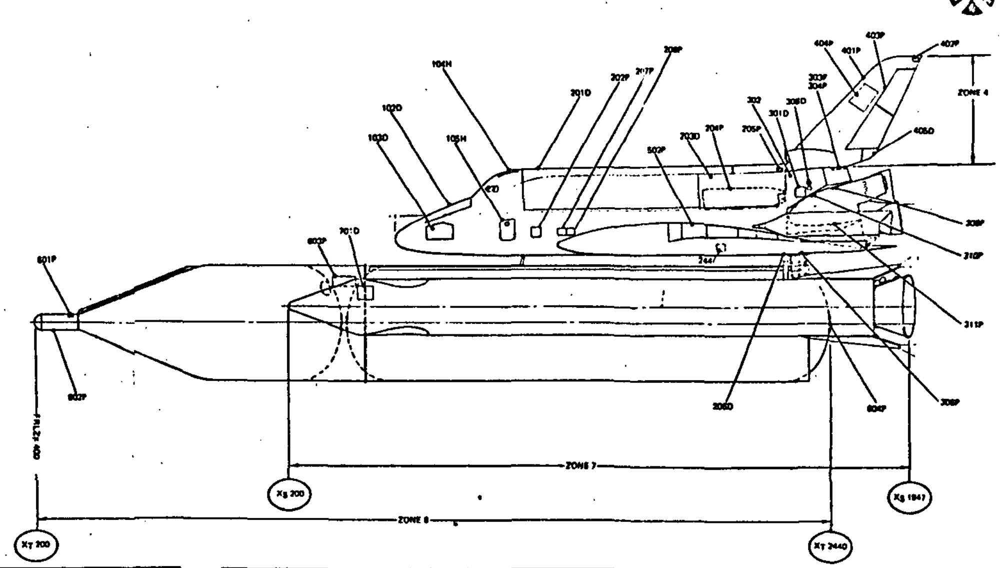
4.0.1 Rationale for Assumptions

Not Required

4.0.2 References

166 pages 3-3 and 2-105





ACCESS DESIGNATOR		
I	III	D
ZONE I	NUMBER SEQUENCE	TYPE ACCESS

LEGEND:
- H = HATCH – QUICK OPENING
- D = DOOR – QUICK OPENING
- P = PANEL – MULTIPLE FASTENERS
- X = NORMAL ACCESS ROUTINELY OPEN
- (X) = NORMAL ACCESS CLOSED ON LAUNCH PAD
- 105H = ALTERNATE ACCESS
- ● = REMOVED FOR OFF LINE MAINTENANCE
- # = INTERNAL ACCESS ROUTE

ACCESS NO	ITEM NO	NOMENCLATURE	SAFING ACCESS	M&R ACCESS	LAUNCH PAD ACCESS	LAUNCH PAD TPS PENE	LAUNCH PAD STANDS PSND
		ZONE 1 - FORWARD BODY					
101D		NOSE LANDING GEAR DOORS	X	X	(X) 105H	X	X
	1	AUDIO PHONE JACK					
	2	ECLSS GROUND COOLANT CONNECTOR					
	3	GROUND ELECTRICAL POWER CONNECTOR					
	4	ILS ANTENNA					
	5	NOSE LANDING GEAR					
	6	#LANDING GEAR – AIRLOCK ACCESS DOOR					
102D		DOCKING PORT ACCESS DOORS		X	(X) 105H	X	X
	11	DOCKING PORT HATCH					
	12	DOCKING UMBILICAL CONNECTOR					
	13	STAR TRACKERS					
	14	AIRLOCK COMPARTMENT					
	15	#MAIN DECK – AIRLOCK HATCH					
	16	HORIZON SENSORS LH AND RH					
103D		RCS THRUSTER ASSEMBLY DOOR		X	(X)		
	21	THRUSTERS					
	22	DRAIN HOLES					
	23	PROP SUPPLY DISCONNECTS					
104H		FLIGHT DECK HATCH	X	X	X		X
	31	OVERHEAD PANELS					
	32	MAIN DISPLAY PANELS					
	33	PILOTS SEAT LH AND RH					
	34	CENTER CONSOLE					
	35	CAUTION AND WARNING PANEL					
	36	SIDE CONSOLES					
	37	MANIPULATOR CONTROL STATION					
	38	#FLIGHT DECK – MAIN DECK ACCESS DOOR					
105H		MAIN DECK SIDE HATCH	X	X	X		X
	41	AVIONICS EQUIPMENT					
	42	#MAIN DECK – CARGO BAY HATCH					
	43	FOOD MANAGEMENT GALLEY					
	44	PAYLOAD MANAGEMENT PANEL					
	45	#MAIN DECK – LOWER DECK ACCESS DOOR					
	46	[illegible]					
	47	WASTE COLLECTOR ASSEMBLY					
	48	PAYLOAD [illegible]					
	49	[illegible]					
	50	WASTE [illegible] FILTERS					
	51	WASTE MANAGEMENT SEPARATORS					
	52	CREW EQUIPMENT STORAGE					
	53	[illegible] CANNISTERS					
	54	[illegible]					
	55	[illegible]					
	56	[illegible] HEAT EXCHANGER					
	57	[illegible]					
	58	ELECTRICAL AND HARDLINE COMM PANELS					
	59	POTABLE WATER TANKS					
106P		[illegible] PANEL		●	(X)	X	
	66	PROP FILL AND DRAIN DISCONNECT					
	67	HELIUM PRESS AND RELIEF DISCONNECT					
107P		RCS MODULE ATTACH ACCESS PANELS		●	(X)	X	
	69	ATTACH BOLTS		●	(X)	X	
		ZONE 2 - MIDBODY					
201D		CARGO BAY DOORS		X	(X)		
	71	CARGO [illegible]					
	72	CARGO MANIPULATORS					
	73	[illegible]					
	74	[illegible] NOMECHANISM					
	75	RADIATOR [illegible] MECHANISM					

ACCESS NO.	ITEM NO.	NOMENCLATURE	SAFING ACCESS	M&R ACCESS	LAUNCH PAD ACCESS	LAUNCH PAD TPS PENE	LAUNCH PAD STANDS PSND
202P		ECLSS ACCESS PANEL, LH AND RH		X	(X)	X	X (LH ONLY)
	81.	CREW COMPT – CARGO MODULE TRANSFER TUNNEL ADAPTER					
	82.	ECLSS FREON ACCUMULATOR					
	83.	ECLSS FREON PUMP MODULE					
	84	ECLSS WASTE WATER TANKS					
	85	ELECTRICAL AND HARDLINE COMM PANELS					
	86.	ECLSS INTERCHANGER					
	87.	FUEL CELL POWER PLANTS					
	88	ECLSS SUBLIMATOR					
	89	PGS SUPERCRITICAL LO_2 TANKS					
	90	PGS SUPERCRITICAL LH_2 TANKS					
	91.	L BAND ANTENNAS					
	92.	VHF/AM ANTENNAS					
203D		ABPS CLOSURE DOORS		●	(X) 301D	X	X
	101.	ABES NACELLES					
	102.	JP FUEL TANK AND ACCESSORIES					
	103	AIR BREATHER ENGINE DEPLOYMENT MECHANISMS					
204P		ABE NACELLE SIDE PANELS		●	(X)	X	
	111.	ENGINE START CARTRIDGE					
	112.	OIL TANK FILL POINT					
	113.	ENGINE ACCESSORIES					
205P		JP FUEL FILL DRAIN ACCESS PANEL (VERT)	X	X	X		X
	121.	FUEL FILL/DRAIN LINE CONNECTOR					
	122.	FUELING GROUND JACK					
206D		ORBITER/ETS MECHANICAL DISCONNECT COMPARTMENT DOOR, LH AND RH		X	X		
	126	DISCONNECT MECHANISM					
	127.	AFT ETS/JACKING ATTACH FITTING					
207P		EPS EQUIPMENT UMBILICAL (LH ONLY)		X	X		X
	131.	FUEL CELL H_2O VENT DISCONNECT					
	132	FUEL CELL REACTANT SERVICE AND VENT DISCONNECTS					
208P		CARGO SERVICE PANEL		X	X		X
	136	UMBILICAL DISCONNECTS					
		ZONE 3 – AFT BODY					
301D		AFT BODY UPPER COMPARTMENT ACCESS DOOR	X	X	X		X
	141	APU INSTALLATIONS					
	142.	AVIONICS EQUIPMENT RACK					
	143	ECLSS VAPOR CONTROL UNITS					
	144	ECLSS HEAT EXCHANGERS					
	145	MPS ENGINES					
	146	MPS COMPONENTS					
	147	AIR DUCTING					
	148	HYD WATER BOILERS					
	149	#UPPER – MID COMPARTMENT ACCESS HOLE					
	150	MPS PROP DUCTING					
	151.	OMS POD ATTACH POINTS					
	152	PNEUMATIC SUBSYSTEM HELIUM TANKS					
	153	RUDDER SPEED BRAKE MIXER BOX					
	154	APU EXHAUST DUCT LOWER SECTION					
	155	#MID – LOWER COMPARTMENT ACCESS HOLE					
	156	#MID COMPARTMENT, CARGO BAY ACCESS HOLE					
	157	#CARGO BAY FRAME CRAWLWAY					
302P		APU TAIL [illegible] BODY PANEL MODULE (2)		●	(X) 301D		

ACCESS NO.	ITEM NO.	NOMENCLATURE	SAFING ACCESS	M&R ACCESS	LAUNCH PAD ACCESS	LAUNCH PAD TPS PENE	LAUNCH PAD STANDS PSND
303P		AFT UMBILICAL PANEL – LH	X	X	X		X
	161.	GROUND ELECTRICAL POWER DISCONNECT					
	162	VEHICLE PURGE DISCONNECT					
	163	LH_2 TANK HELIUM PREPRESSURE DISCONNECT					
	164	LH_2 FILL AND DRAIN					
	165	SE FLYAWAY UMBILICAL COMM INSTR GNBC					
	166	VEHICLE STATIC GROUND JACK					
	167	LH_2 PRESSURE VENT DISCONNECT					
304P		AFT UMBILICAL PANEL – RH	X	X	X		
	171.	LO_2 OVERBOARD BLEED DISCONNECT					
	172.	LO_2 TANK HELIUM PRESSURE DISCONNECT					
	173	GH_2 ENGINE PURGE DISCONNECT					
	174	MPS HELIUM SUPPLY DISCONNECT					
	175	LO_2 FILL AND DRAIN DISCONNECT					
	176	SE FLYAWAY COMM INSTR GNBC					
305P		AFT BODY CLOSEOUT PANEL		X	(X)		
	181.	MPS HEAT SHIELD					
	182	LO_2 DUMP LINE					
	183	LH_2 DUMP LINE					
	184.	MPS ACCESS PANEL					
306D		ORBITER/ETS AND DUCTING DISCONNECT COMPARTMENT DOOR – RH		X	X		
	186.	LO_2 FEEDLINE DISCONNECT					
	187	LO_2 VENT PRESSURE DISCONNECT					
	188	ELECTRICAL HARNESS DISCONNECTS					
307D		ORBITER/ETS ELECTRICAL AND DUCTING DISCONNECT COMPARTMENT DOOR – LH		X	X		
	191.	LH_2 FEEDLINE DISCONNECT					
	192.	ELECTRICAL HARNESS DISCONNECTS					
	193	LH_2 RECIRCULATION LINE DISCONNECT					
	194.	LH_2 PRESSURE/VENT DISCONNECT					
308D		APU FILL DRAIN VENT ACCESS PANEL		X	X		X
	196	FILL/DRAIN/VENT RECEPTACLES					
309P		OMS/RCS POD PROPELLANT VENT ACCESS PANEL		●	(X)	X	X
	201	PROP VENT RECEPTACLE					
310P		OMS/RCS POD O_2 VENT ACCESS PANEL		X	X		X
	206	O_2 VENT RECEPTACLE					
311P		DELTA V KIT ACCESS PANEL		# X			
	211	CROSSOVER LINE DISCONNECTS					
		ZONE 4 – VERTICAL SURFACES					
401P		VERTICAL STABILIZER LEADING EDGE PANEL		X	(X)	X	
	216	STABILIZER LEADING EDGE SPAR					
402P		STABILIZER TIP PANEL		X	(X)	X	
	221	VHF ANTENNA					
403P		RUDDER SPEED BRAKE SEAL PANELS		X	(X)	X	
	226	RUDDER SPEED BRAKE HINGES					
	227	RUDDER SPEED BRAKE ELECTRICAL HARNESS					
	228	ACTUATOR RUDDER ATTACH POINTS					
	229	RUDDER SPEED BRAKE SEAL					
404P		RUDDER SPEED BRAKE ACTUATOR ACCESS PANEL		X	(X)	X	
	231	RUDDER SPEED BRAKE ACTUATORS					
405D		DECELERATION CHUTE DOORS		X	(X)	X	
	236	DECELERATION CHUTE					
	237	CHUTE INITIATOR					

ACCESS NO	ITEM NO	NOMENCLATURE	SAFING ACCESS	M&R ACCESS	LAUNCH PAD ACCESS	LAUNCH PAD TPS PENE	LAUNCH PAD STANDS PSND
		ZONE 5 – WINGS					
501D		MAIN LANDING GEAR DOORS	X	X	(X)	X	X
	241.	LANDING GEAR			301D		
	242.	ECLSS SE HEAT EXCHANGER (LH ONLY)	N/A		N/A	N/A	N/A
	243.	ABPS FUEL FILL AND DRAIN – HORIZONTAL ONLY (LH ONLY)					
	244.	#WHEEL WELL – FUS INNER AND OUTER OML	X		301D	X	X
	245.	ILS ANTENNA					
	246.	RADAR ALTIMETERS ANTENNAS					
502P		WING FUSELAGE ATTACH ACCESS PANELS		X	(X)	X	
	251.	WING FUSELAGE ATTACH POINTS					
503P		WING LEADING EDGE ACCESS PANELS		X	(X)	X	
	256.	LEADING EDGE SECTIONS ATTACH POINTS					
504P		ELEVON ACTUATORS ACCESS PANELS		X	(X)	X	
	261.	ELEVON ACTUATOR ASSEMBLY					
505P		ELEVON WING ACCESS PANELS		X	(X)	X	
	271.	ELEVON HINGES					
	272.	ELEVON CONTROL ELECTRICAL HARNESS					
	273.	WING NAVIGATION LIGHT HARNESS					
	274.	ELEVON ACTUATOR ATTACH					
	275.	S BAND ANTENNA AND HARNESS (LH ONLY)					
		ZONE 6 – ETS					
601P		DEORBIT MOTOR MOUNT ACCESS PANEL		X	N/A		X
	291	INITIATOR					
602P		DEORBIT MOTOR FAIRING			(X)		X
	296	DEORBIT MOTOR					
	297	#LO_2 TANK FORWARD INTERNAL ACCESS PANEL					
603D		TANK INTERFACE ACCESS PANEL			N/A		
	301	LO_2 TANK [illegible]					
	302	[illegible]					
	303	LO_2 ANTI VORTEX BAFFLE					
	304	LO_2 FEEDLINE TANK OUTLET CONN					
	305	BATTERY [illegible]					
	306	#LO_2 TANK MAINTENANCE ACCESS PANEL					
	307	LO_2 TANK [illegible] SENSORS					
	308	[illegible]					
	309	[illegible]					
	310	[illegible]					
	311	LO_2 VENT [illegible]					
	312	EXPLOSIVE CHARGE					
	313	LO_2 ANTI [illegible] RING LINE CONN					
	314	DETONATOR PROP DISPERSION					
	315	[illegible]					
	316	[illegible] ELECTRICAL HARNESS					
604P		LH_2 TANK ACCESS PANEL			(X)	N/A	
	321	LH_2 PROP [illegible] SENSORS					
	322	LH_2 FEEDLINE INNERTANK					
	323	GAS DIFFUSER					
	324	[illegible]					
	325	ENGINE CUT OFF SENSORS					
	326	LH_2 PRESSURE VENT LINE					
		ZONE 8 – SRM					
801D		NOSE CONE EXTENSION ACCESS DOOR					●
	331	RANGE SAFETY [illegible]					
	332	[illegible]					
	333	[illegible]					
	334	[illegible]					

FIGURE 4.0-1 SHUTTLE EQUIPMENT LOCATION

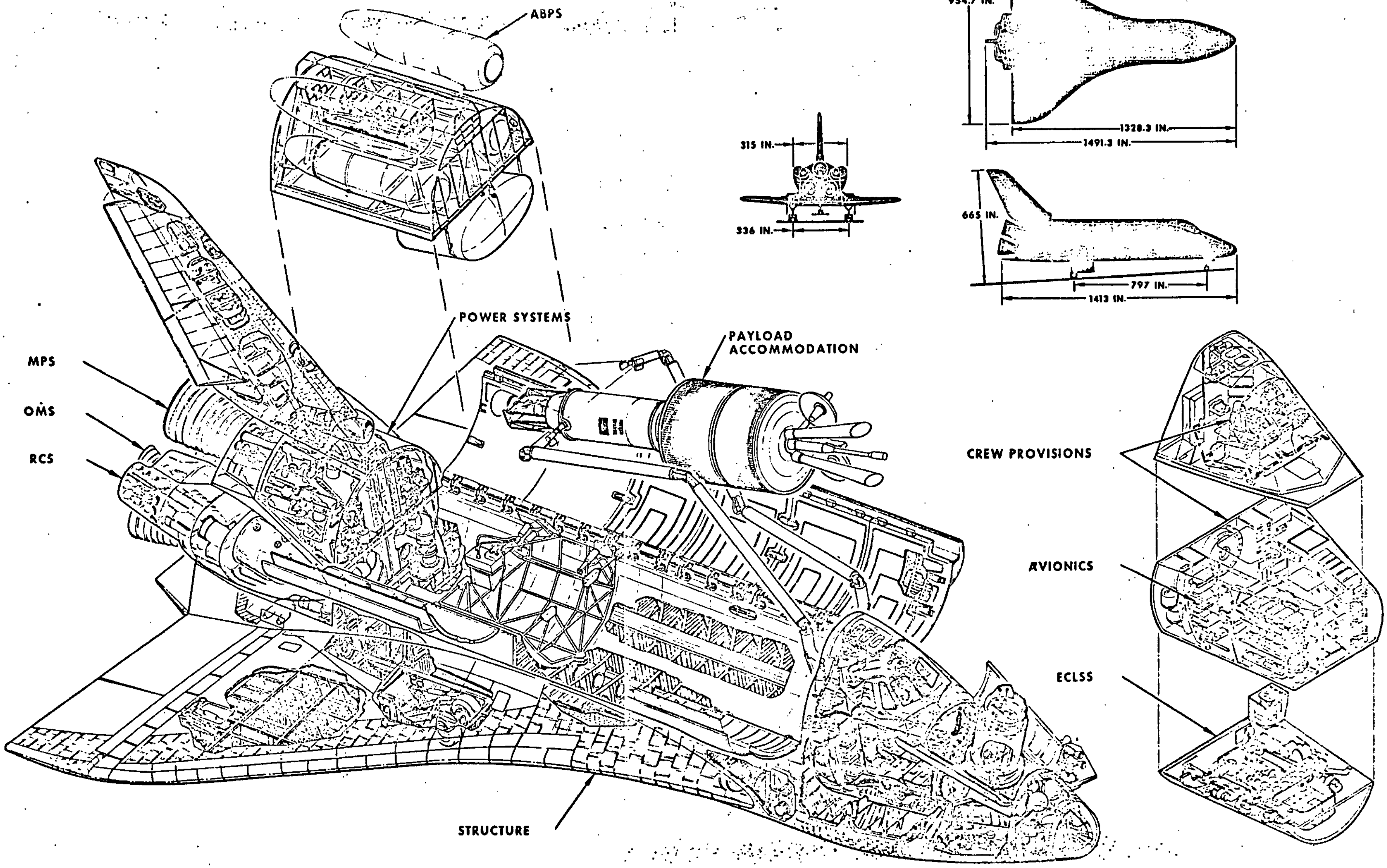


FIGURE 4.0-2 BASELINE ORBITER VEHICLE

4.1 ELECTRICAL POWER

166 The electrical power system for the shuttle vehicle is composed of fuel cells, batteries, auxiliary power generators (AC power), and air breathing engine generators (AC power). These power sources are distributed throughout the vehicle as both AC and DC supply to operate electrical pumps, motors, valves, and avionic subsystems throughout the mission. The restartable hydrogen-oxygen fuel cell powerplants were selected as the primary source for DC power. Projected life expectancy of these cells is approximately 5000 hours. Three of these fuel cells are provided, each having an output capability of 10KW at 29 volts ±5%. Heat and water generated by the fuel cell reaction process are input to the ECLS subsystem.

166 For the launch, landing, and during some mission operations the three 400 Hertz; 20/30 KVA; 120/208 VAC APU driven generators will provide additional power for peak loads. During ferry flights the electrical power requirements will be supplied by four ABPS driven AC generators.

166 Nickel-cadmium batteries provide the pyrotechnic and emergency power supply. The nickel-cadmium batteries use hermetically sealed cells to minimize maintenance requirements. A high-pressure safety relief vent on each cell provides the necessary safety for manned vehicles; however, the vents do not cycle under normal operating modes.

166 As a pyrotechnic and emergency power supply, two nickel-cadmium secondary batteries are employed. Each battery has nominal power of 10 amp-hours at 29 volts and 27.55 volts (nominal TBD% SOC at TBD° F. at 20 amps discharge.

166 A separate battery subsystem is provided for contingency power to meet

F.398.8-A

special requirements for the Development Flight Instrumentation (DFI). A load analysis has indicated a requirement to supply 962 watts for a total duration of 2.5 hours, which assumes a 100-percent duty during high activity periods and a 10 percent duty cycle during low activity periods. Three Apollo CSM silver-zinc entry batteries (40 amp-hr each) provide this power requirement.

166 Battery recharging will be accomplished by DC-DC convertors. Static inverters are provided for minor AC loads when startup of the APU is not warranted. Conversion of AC to DC will be accomplished when the fuel cells are inoperative.

166 Power distribution and control is accomplished by bus networks with switching and control logic for protection of buses and circuits from power source, bus load faults, and failures.

166 The orbiter electrical system provides an electrical power umbilical compatible with the space station docking port power umbilical and is capable of receiving space station power for orbiter emergency orbital docked-phase power requirements. In addition the orbiter provides a power umbilical and is capable of transferring power to the payload module. No electrical power transfer is provided between the orbiter and booster.

4.1.1. ELECTRICAL POWER DISTRIBUTION AND CONTROL (EPDC)

166 The orbiter EPDC is shown schematically in Figure 4.1-1 Primary 28 vdc power is distributed from the three 10 kw fuel cells through three central DC distribution center near the fuel cells, to seven local distribution boxes located near the load centers. There are two 3-phase, 400 cycle AC bus systems incorporated into the vehicle. The APU generators in the aft portion of the vehicle provide three independent AC power sources. The three

Space Division
North American Rockwell

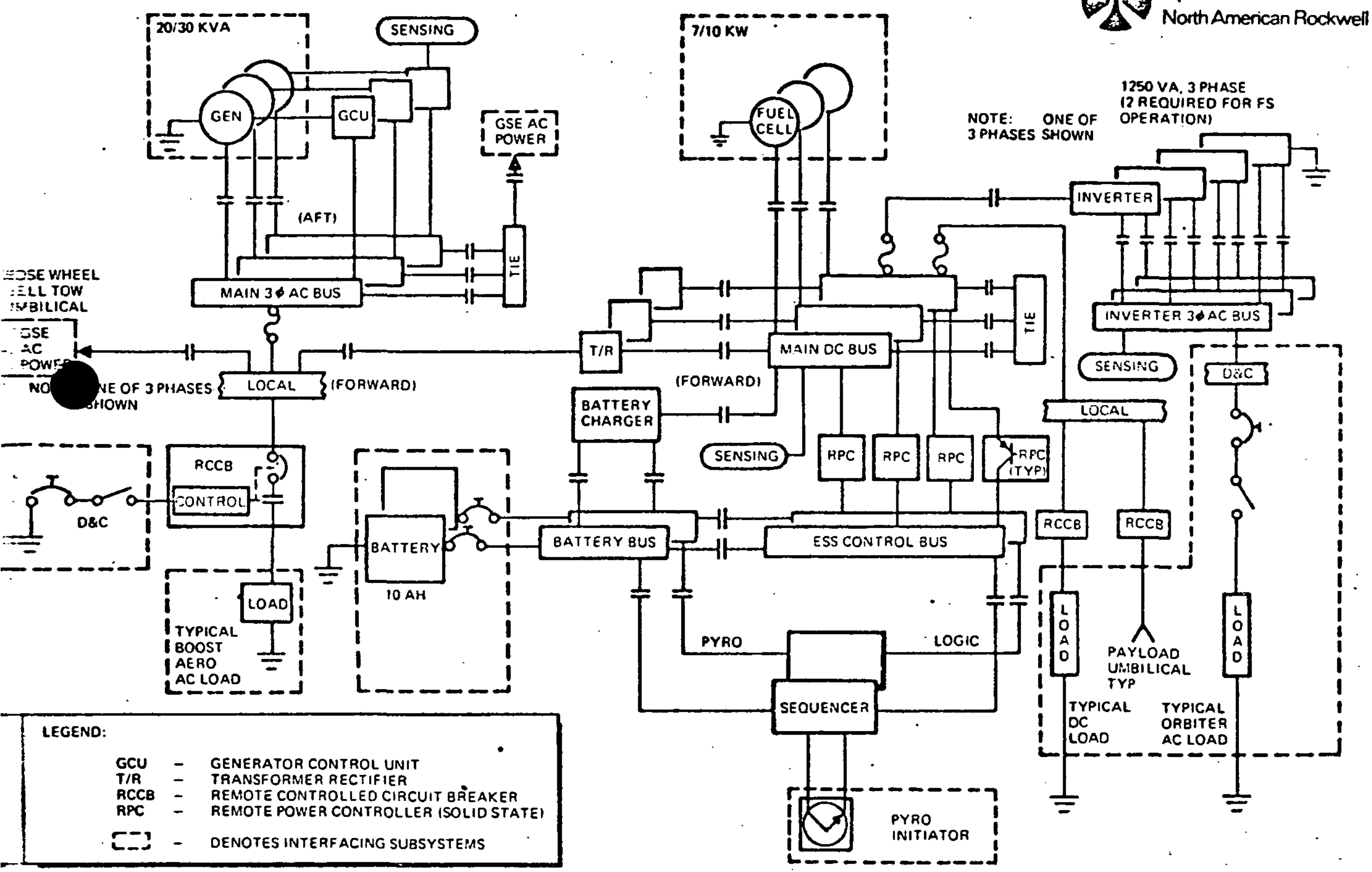


TABLE 4.1.1 *System Electrical Power Distribution and Control Schematic*

TABLE 4.1.1 *Equipment for EPDC*

Feature	Selected Approach	Equipment/Components
Power types		
DC	24-30 v at load	[Fuel cells,] 150-amp transformer-rectifier, [10 amp-hr batteries,] battery charger
Aero, boost and entry ac	115/200-v, 400-Hz generator (MIL-STD-704A)	[Generators,] generator control unit
Orbit ac	115/200-v, 400 Hz central inverter system (CSM limits)	CSM 1250-va 3-phase inverters (refurbish-reuse)
GSE	115/200-v, 400-Hz (MIL-STD-704A)	Ac power umbilicals
Bus redundancy	Three redundant	Magnetic latch, hermetic sealed power contactors
Redundancy management	Isolated (ac nonsynchronized—isolation maintained *from source through load)* Bus transfer after source failure Redundant loads powered *from redundant buses*	Magnetic latch, hermetic sealed power contactors
Load control	Hardwired electromechanical	Magnetic latch, hermetically sealed RCCB's RPC's, high-reliability relays
Power current return	Structure, multipoint ground (single-point ground for signal circuits)	
Sequencing	Conventional logic—with voting inputs and basic timing from GN&C computer	Relays and solid-state logic
Circuit sensing		
Main ac	Over-under voltage, Over-under frequency, Overload	*Generator control unit*
Dc	Undervoltage, overload	Solid-state sensors
Inverter	Over-under voltage, overload	Solid-state sensors
	Inverse time-current interruption	RCCB's, *thermal circuit breakers, fuses*
Fault	Current limited, timed interruption	MIL-Spec solid-state RPC's

[ ] Denotes non-avionics equipment; RCCB—Remote control circuit breakers; RPC—remote power controllers

buses are non-synchronous and should not be tied to a common bus except when GSE is providing power. The APU generator has controls for governing voltage and frequency; however, there is no provision for synchronization of the generators. The solid state 1250 VA-3-phase invertors driven from the main DC buses do have synchronization between phases. Note that in case of failure of any one unit, that the other invertors can be switched to provide redundant power. Inverter power is distributed from there to the local distribution boxes.

166 From the local distribution boxes, power is distributed to loads and controlled by remote power controllers. The interface for power control and data with the Displays and Control (D&C) system is by Data Control Management Acquisition Control and Test (DCM ACT) units and data bus. As shown, certain basic power control and data functions are hardwired directly to dedicated D&C equipment. Transformer-rectifier units near the main DC distribution center supply DC power from generator sources during ferry missions, GSE AC sources for ground checkout, and space station AC sources for emergencies during docked periods. Two batteries supply emergency power control and pyrotechnic DC loads. The batteries can be recharged from fuel cell energy by a regulated charger.

166 Redundant sequencers provide positive arming and safing of pyrotechnic devices and any other unilateral safety-of-flight electrical operations required of orbiter electrical subsystems.

166 Interior lighting is provided in the flight deck, Avionics Equipment Bay, IVA tunnel, cabin, and airlock compartments. Both floodlights and spotlights are provided. Exterior lights are provided for rendezvous, docking,

and payload maneuvering during orbital flight, as well as atmospheric flight, landing and taxi. General operating characteristics of the EPDC is shown in Table 4.1-1.

4.1.2 POWER DISTRIBUTION EQUIPMENT DESCRIPTION

4.1.2.1 GENERATOR CONTROL UNITS (GCU)

166 Generator Control Units (GCU) provide voltage regulation for the AC generators, and control the Generator Load Contactor (GLC) to close (upon command from the D&C) when generator voltage and frequency are normal and open when voltage or frequency exceed limits (or a feeder fault or generator overload is sensed by the generator current transformers). The GCU will also provide failure indications to the DCM system for failure isolation to generator, GCU feeder, or GLC.

166 The three 20/30 kva, 400-Hz generators used for orbital missions are sized to power the main propulsion engine accessories, the ABPS fuel boost pumps, the ECLSS vapor cycle machines, and other similar loads applied during ascent, entry, and landing. Frequency control is in accordance with MIL-STD-704A. Synchronization of APU generator frequencies is not controlled. The generator design is a smaller model of existing spray-oil cooled units, and was designed for integration with APU gearbox zero-g lube system.

4.1.2.2 POWER CONTACTORS

166 Main AC and DC power source and bus tie contactors are the electro-mechanical type, and will have auxiliary contacts for interlock and position indication. DC types will include overcurrent sensing and trip in some applications.

4.1.2.3 INVERTERS

166 There are four inverters; single-phase units with 1250-va, 115-v 400-Hz output rating. They are completely static, solid/state, and have

provisions for sensing and limiting overload current in either of two output terminals. Synchronization and phase relationship between units is provided by an internal clock oscillator in the master inverter unit for phase A, and 120-degree lock signals between inverters connected to succeeding phases. Bus 1 and Bus 2 inverter sets will also be synchronized.

4.1.2.4 TRANSFORMER-RECTIFIERS (TR)

166 The three transformer-rectifiers (T-R) are each rated at 28-v, 150-amp output and can deliver 225-amp for five minutes and 300-amp for five seconds. The T-R's contain no active regulator to control output voltage. The regulation of the transformer and rectifier over the normal load range provides output voltage within subsystem limits.

4.1.2.5 BATTERY/CHARGER

166 A solid-state battery charger uses fuel cell energy to recharge the 10-amp-hour nickel cadmium batteries after partial discharge for pyrotechnic loads and/or emergency power control. The charger is a constant current type with maximum voltage cutoff control, and will sense battery temperature to adjust cutoff voltage.

4.1.2.6 REMOTE POWER CONTROLLERS (RPC)

166 Two types of remote power controllers are used in the local power distribution boxes: solid-state and hybrid. Solid-state power controllers protect and control DC loads rated from 1 to 10 amp. They limit overload current to 150-percent RPC rating and control turn-on and turn-off time for transient current and voltage control. They also provide trip indication to D&C through the DCM system.

166 Hybrid power controllers control and protect DC loads rated above 10 amp and AC loads. This RPC type uses solid-state sensing and control and electromechanical contacts for power switching. The hybrid RPC's have

inverse current-time overload trip characteristics, and for AC are in both single and three-pole types.

166 Both solid-state and hybrid Remote Power Controller (RPC) are trip-free, and are controlled by a 5-v, 10-ma signal for on and 0-v open circuit for off. Reset after trip is accomplished in both types by removing the control signal. All RPC's have fuseable links as back up to the solid-state overload sensing and trip circuits, and the hybrid DC RPC's also have a current trip coil.

4.1.2.7 SEQUENCERS

166 Sequencer units contain a sequencer bus which is armed only during sequence events. Dual-redundant firing relays short protechnic devices during all periods except firing. Connection is made to sequencer buses on logic command from the DCM.

4.1.2.8 INTERIOR LIGHTING

166 Floodlighting is provided by fluorescent lamps and spotlighting by incandescent lamps. Integral D&C panel lighting is a part of D&C and is provided by that subsystem.

4.1.2.9 EXTERIOR LIGHTING

166 For exterior lighting during orbit, flashing rendezvous lights are provided for the space station rendezvous phase. Running lights and a docking light are provided for stationkeeping and docking. A spotlight on each payload manipulator arm provides payload illumination. For atmospheric flight phases, the rendezvous light is used as an anticollision light. Other FAA lighting requirements are met by position lights and fuselage lights. Night landing requirements are met with three landing lights mounted on the landing gear. The nose-gear landing light is also used as a taxi light.

4.1.2.10 FUEL CELL SYSTEM

166 The shuttle orbiter vehicle carries three 10 KW peak power fuel cells for a prime source of power during orbital missions. During ferry missions the fuel cells remain inactive. Figure 4.1-2 gives the expected fuel cell load.

166 Redundant cryogenic tanks provide the gaseous O_2 and H_2 at nominal pressures of 900 psia and 250 psia respectively.

166 *NOTE: To date manufacturer/type of fuel cell has not been selected, however, the two types under consideration have similar characteristics.

166 The hydrogen-oxygen fuel cells are the low-temperature restartable type and are either the contained electrolyte Pratt and Whitney or the ion-exchange membrane type power units. The reaction conditions are 60 psia pressure and 200 F using a catalyst for the reaction. Refer to Figure 4.1-3.

166 Voltage control will be achieved either by inherent characteristics or by voltage regulators.

166 Controlled gas regulators will provide near equal pressures to the fuel cell chambers to prevent flooding and contamination. Preheaters will be used to precondition incomming gas flows. Purging will be accomplished on a fixed watt-hour usage by integration devices and controls. Separate purge systems are provided for both reactants. Water vapor and gaseous reactant will be separated either by a static separator (droplet collector) operating at a differential pressure caused by circulation or by wicking. Product water is routed to the EC/LSS storage tanks or overboard disposal. Check valves and pressure relief regulators are provided on fuel lines and product lines to prevent directional flow problems. Heaters are provided

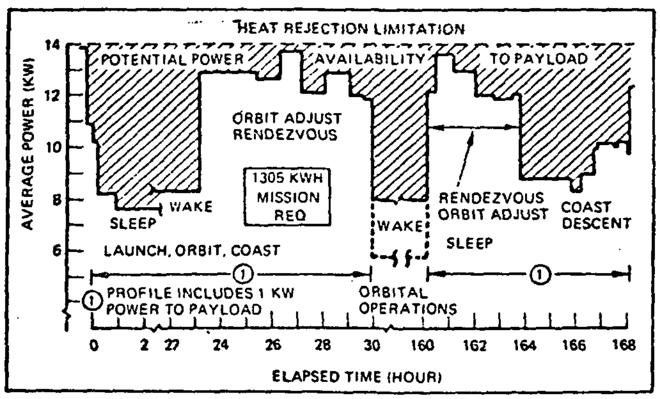


Figure 4.1.2 *Fuel Cell Powerplant Subsystem Power Profile (Phase Average)*

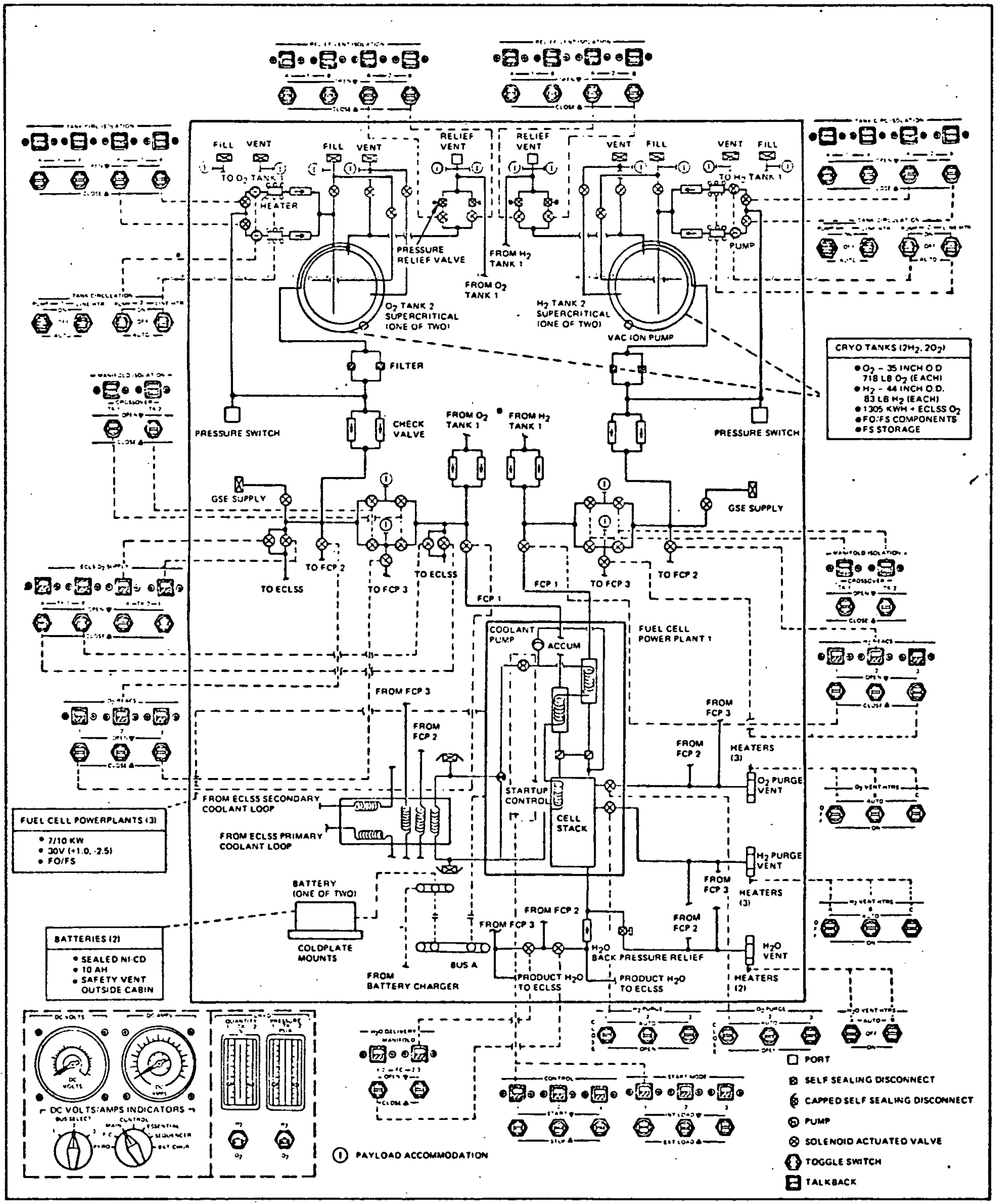


Figure 4.1.3 *Fuel Cell Powerplant Subsystem Schematic*

SD 72-SH-50-3

on all lines and ports where water products accumulate or discharge. Waste heat from the reactor unit is removed by circulating coolant through heat exchangers, or through the electrodes. Pumps, accumulators, pressure and temperature controls circulate the reactant and coolant and provide proper fluid volumes and thermal control within the power plant unit. The waste heat from the coolant loop is transferred to the ECLSS coolant loop or to freon heat exchangers during certain vehicle operating modes.

4.1.3 ELECTRICAL POWER OPERATIONAL CHARACTERISTICS

166 The interface to the electrical-power system for most power control and data transmission is by DCM Acquisition, Control, and Test (ACT) units. Central automatic power management functions are performed by the DCM central computer complex. Certain basic power control and data functions are hardwired directly to dedicated D&C equipment.

166 The AC generators are driven by APU's during boost, entry, cruise, and landing flight phases of the orbiter mission. The generators are connected to isolated AC buses normally and controlled by Generator Control Units (GCU).

166 The orbiter Transformer Rectifier units provide DC power from ground and space station AC sources and from the onboard generators during ferry missions. Batteries supply emergency power control and pyrotechnic DC loads. Orbiter batteries can be recharged from fuel cell energy. An orbiter central inverter system supplies AC loads which require 3 phase power during orbit flight phases from two redundant 3-phase AC buses, each supplied from a set of four single-phase inverters.

166 Power feeders, protected by current limiters, feed power distribution boxes from the central buses. Locally, solid-state and hybrid power controllers control and protect load circuits, interfacing with adjacent DCM-ACT units. Redundant loads are connected to redundant buses.

166 For failure isolation, failed power sources will be isolated from the bus, and the bus will be connected to another power source through bus tie contactors.

166 Table 4.1.2 lists the orbiter electrical power characteristics which are common to those in the booster. Figures 4.1-4 and 4.1-5 depict common orbiter-booster electrical voltage transient envelopes.

38

Table 4.1.2 Electrical Power Characteristics at Load Interface

Item	Main Bus	Sequencer Bus
DC POWER		
Voltage: Nominal	28	28
Steady-state limits	24-30	24-30
Transient limits	See Fig. 4.1.3	5-40 recovery in one second
Ripple voltage	(MIL-STD-704A)	4 p-p
Availability	All flight and ground operations	When bus armed
Power interruption	Single bus - see Figure Redundant Buses - none	See Figure
Negative return	Structure-multipoint ground	Wire-single point ground

Item	Main Bus	Inverter Bus (Orbiter)
AC POWER		
Phases	3, 120 degree ±4 degree	3, 120 degree ±4 degree
Voltage: Nominal	115/200	115/200
Steady-state limits	See Figure 4.1.4	115 +2 -4 average of 3 phases and single phase
Transient limits	See Figure 4.1.4	115 +15 -17 50 ms recovery
Wave shape	Sine (MIL-STD-704A)	Sine (MIL-STD-704A)
Frequency: Steady-state	400 ±20 Hz	400 ±2 Hz
transient	MIL-STD-704A	
Availability: Ferry missions	All flight and ground	All periods
Orbital missions	Boost, delta V, entry cruise, landing, ground	All periods
Power interruption	See Fig. 4.1.4	One minute
Neutral return	Structure-multipoint ground	Structure-multipoint ground

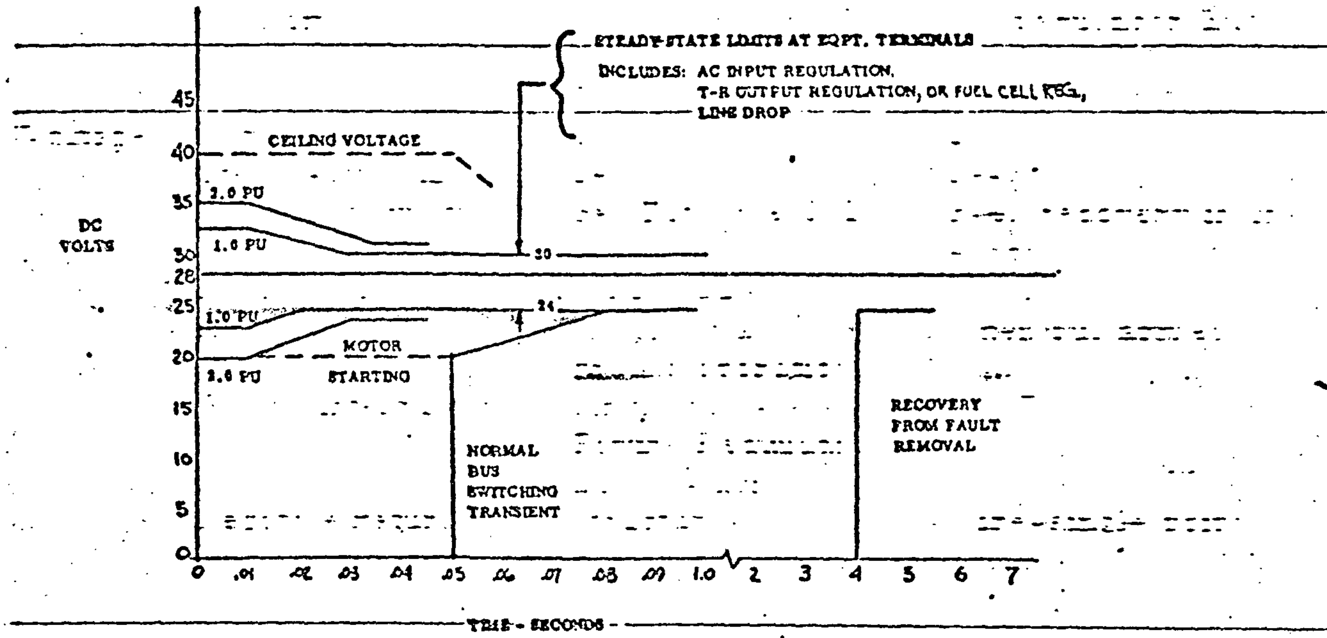


Figure 4.1.3 DC Transient and Steady-State Voltage Limits

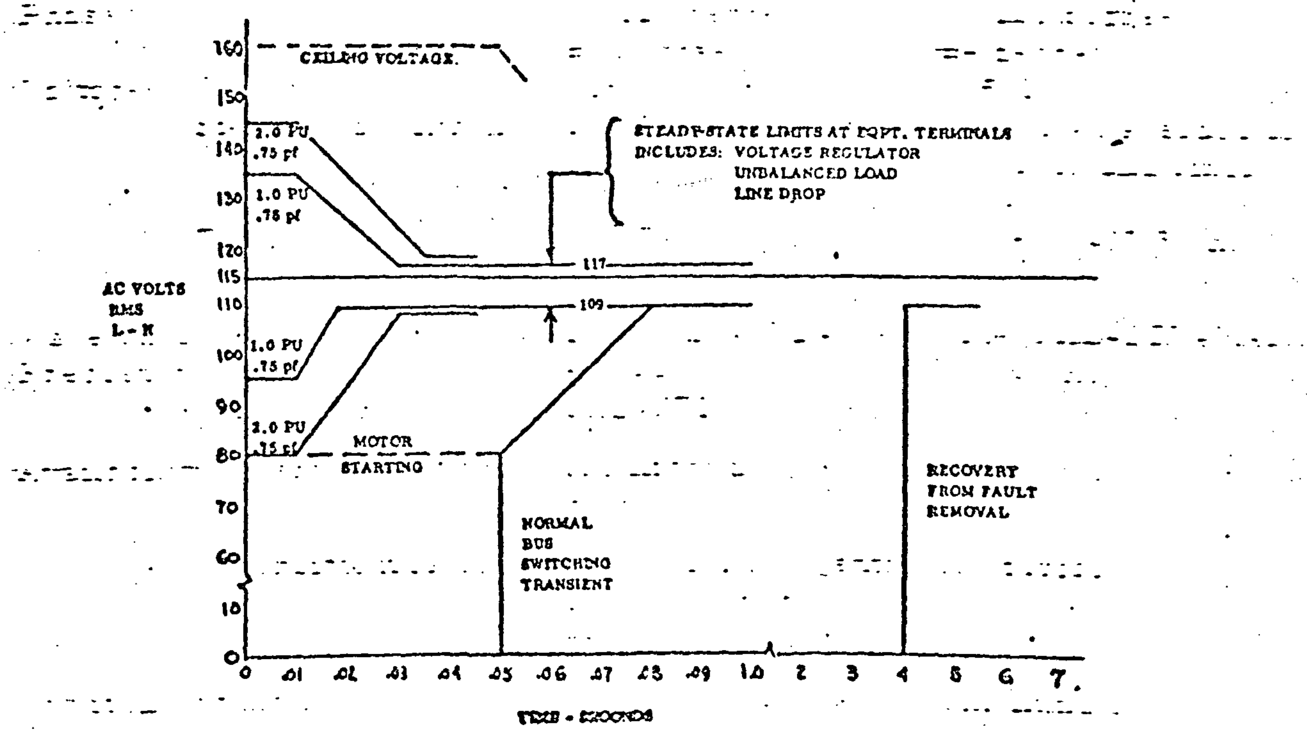


Figure 4.1.4 AC Transient and Steady-State Voltage Limits

REF. KEY

4.1.4 Functional Interfaces and Support Requirements

4.1.4.1 Data Control and Management (DCM)

166 The EPDC interface for power control; EPDC data; automatic, normal and emergency power management functions; and data recording will be through the DCM ACT units. The ACT unit control interface with the RPC's will be a continuous 5-v, 10-ma signal for on, 0-v open circuit for off. The ACT data interface with RPC trip signals, power contactor position, and EPDC analog data signals will be a 5-v 1-megohm input.

4.1.4.2 Displays and Controls (D&C)

166 The EPDC interface with D&C for normal crew display and control functions will be through the DCM. However, functions required for initial power-up (or power restoration) of DCM and integrated D&C will be hardwired between EPDC and D&C. Such functions include the application of GSE power, AC and DC bus switching, inverter system controls, fuel cell systems controls, and ECLSS electronics cooling controls. Where controls are required to be both manual hardwired through D&C and automatic through DCM, the circuit will be designed to permit D&C manual control to override the DCM signal. The D&C control interface with EPDC RPC's will be a 5-v, 10-ma signal for on, 0-v grounded for off, and open circuit for automatic through DCM (if required).

4.1.4.3 Electrical Power Generation (EPG)

166 The EPDC interface with the EPG will be at the fuel cell, AC generator, and battery terminals. Electrical power characteristics at the interface will be:

REF. KEY

1. Fuel cells: 29-v ±5-percent load range of 1.5 to 10 kw
2. AC generators: 120/208-v, 3-phase, load range of 0 to 30 kva, 400 ±20 Hz
3. Batteries: 27.55-v at 20 amp discharge with a TBD% SOC

166 The EPDC will nominally distribute power to the loads while maintaining power characteristics at the subsystem load interface as stated. The EPDC GCU will regulate the EPG AC generator voltage.

4.1.4.4 Electrical System Power Losses

166 The EPG will provide for EPDC power distribution conversion and control power losses as listed:

EQUIPMENT	POWER LOSS (watt)
DC distribution and control	108 w, plus 4 percent of DC load
AC distribution and control	66 w, plus 1.3 percent of AC load
Inverter distribution and control	26 w, plus 1 percent of inverter load
Inverter	25 percent of inverter load
Transformer-rectifier	17.6 percent of DC load
Interior lighting	275 w maximum
Exterior lighting, payload manipulation	800 w maximum
Exterior lighting, atmospheric running	275 w maximum
Exterior lighting, landing and taxi	3000 w maximum

4.1.4.5 Environmental Control/Life Support (ECLSS) Interface

166 EPDC equipment (except wire harnesses) and lighting will be located in the crew compartment - electronic equipment bay, near the

fuel cells and near the APU's. EPDC equipment is temperature-controlled by ECLSS to maintain the temperature limits listed below at the mounting surface while dissipating the power listed.

EQUIPMENT	LOCATION	NO. REG.	PER UNIT MAX. POWER LOSS (w)	TEMP. LIMITS (F°)
GCU	Near APU's	3	50	-65, +200
Inverter	Electronics bay	6	113	+40, +140
Battery charger	Electronics bay	1	40	-65, +150
Transformer rectifier	Electronics bay	3	740	-65, +150
Central dc distribution box	Electronics bay	2	60	-65, +200
Central ac distribtuion box	Near APU's	2	50	-65, +200
Inverter ac central distribution box	Electronics bay	2	10	-65, +200
Local Power distribution box	Forward crew Compartment	2	29	-65, +200
	D&C panel	2	40	-65, +200
	Airlock	2	5	-65, +200
	Electronics bay	2	98	-65, +200
	Near forward Cargo Bay	2	61	-65, +200
	Near Aft Cargo Bay	2	16	-65, +200
	Near APU's	2	165	-65, +200

4.1.4.6 Power Utilization Subsystems

166 The EPDC interface is at the using subsystem connector. Power characteristics as listed in Paragraph 4.1.4.3 is supplied to the interface for the loads. Power control and interconnecting wiring for all subsystems is provided by EPDC.

4.1.4.7 Support Equipment - GSE

166 The interface for ground electrical power to the orbiter will be through an external power umbilical connector located near the APU's. 120/208-v, 3-phase, 400-Hz power of 60-kva maximum can be furnished from the GSE to this interface.

4.1.4.8 Payload

166 An average of 1000-w DC (maximum 1.5 kw) power will be supplied by the orbiter to the payload power interface umbilical, located at the forward end of the cargo bay.

4.1.4.9 Space Station Interface

166 An average of 500-w DC (maximum 800 w) power will be supplied by the orbiter to the space station interface umbilical, located at the airlock docking port. The space station will be capable of supplying to the orbiter 7-kva maximum, 115/200-v, 3-phase, 400-Hz AC power through each of two umbilicals located at the airlock docking port.

4.1.5 Rationale for Assumptions

Not required.

4.1.6 References

166 Pages 3-85 to 3-93

166 Page 3-157

38 Pages 408-409

4.2 Mechanical Power

166 Auxiliary mechanical power is used by the shuttle vehicle for hydraulic operation of linear and rotary actuators, electrical power generation, shock dampers, and pressurization of tanks. Pressurization of fuel tanks is discussed in each section which has a requirement for individual fuel supply pressurization. Hydraulic power and generation of AC power is accomplished by Auxiliary Power Units (APU's) or the ABPS during ferry operations. Refer to Figure 4.2-1.

4.2.1 Auxiliary Power Units (APU)

166 Four independent 200 HP APU's provide power to a shaft driving an AC alternator, a lube pump, and three hydraulic pumps. The description of the AC alternator interface with the electrical power system is discussed in Paragraph 4.1.2. The lube pump supplies the APU and gearbox during operation. The hydraulic pumps provide the shuttle vehicle with primary hydraulic power for approximately 90 minutes during prelaunch, ascent, entry, and landing.

166 Electronic controls provide for speed control, turbine inlet temperature control, logic for APU startup and shutdown, instrumentation, and malfunction protection. Automatic shutdown of an APU will occur if turbine speed, turbine inlet temperature, and lube oil temperature or pressure exceed limits. Each APU is enclosed for both heat transfer reduction and fire protection. Redundant fire extenguishers are provided for each APU. Overheat sensors are provided to the caution and warning system.

166 A separate pressurization subsystem and elastomeric diaphragm tank assembly supply the hydrazine (N_2H_2) to each APU. Pressure modulation control is used to control the turbine speed and thereby frequency of the AC generator. The combustion of hydrazine is accomplished by a thermal decomposition chamber.

166 A period of ten minutes is required for preheating the decomposition chambers to ignition temperature. This period is compatible with prelaunch and entry timelines. Power requirements are 200 to 500 watts depending on the chamber configuration. While on-orbit, fuel temperature is maintained above 40° F using radiation heaters for fuel tanks and strip heaters for fuel lines.

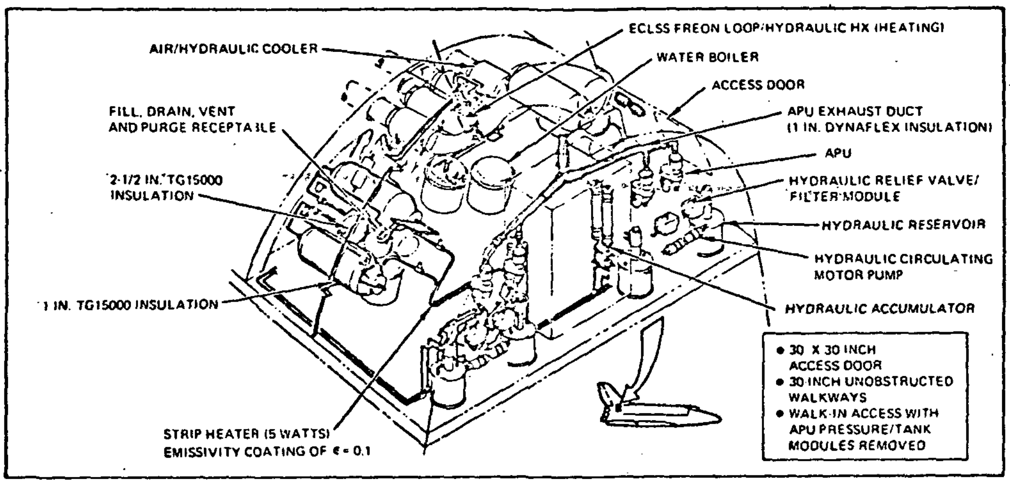
166 System cooling is provided by air coolers during prelaunch operations; by water boilers during boost, injection, and from start of entry to 20,000 feet; and by air coolers below 20,000 feet, including ferry operation. Ducted ram air is provided in flight, and electric-driven fans are used during ground operations. The use of air coolers below 20,000 feet instead of continuous water boiler operation maintains system temperatures below 275° F.

4.2.1.1 Rationale for Assumptions

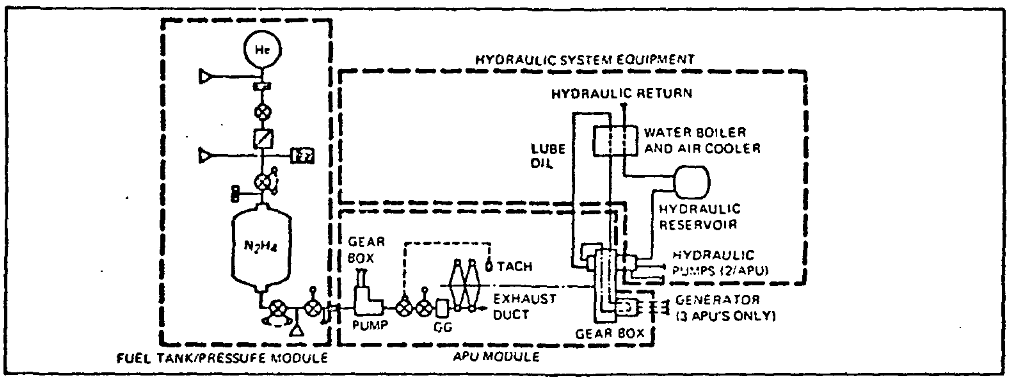
Not required.

4.2.1.2 References

166. Pages 3-86 to 3-89



Auxiliary Power Unit and Hydraulic Pumping Subsystems



APU Subsystem

FIGURE 4.2-1

4.2.2 Hydraulic Power System

166 The APU's drive multiple hydraulic pumps which provide supply redundancy. Check valves in the outlet lines of each pump prevent back flow to a failed pump. During atmospheric ferry flights, provision is made to provide hydraulic power from gear pumps on the ABES. During space operations when only light hydraulic power is required, electrical AC gear pumps will be used. Refer to figure 4.2.2-1

166 Hydraulic power is used to drive the linear actuators of the thrust vector control system of both the orbital maneuvering engines and the main engines, retraction of the main engine nozzle, and main engine control valves. Hydraulic power is also used for deployment of the ABE and operation of the payload and ABE access doors, hydraulic motor and linear actuator. All landing gear functions of braking, door operation, steering, and gear extension/retraction are hydraulic functions.

166 Accumulators are provided for extending or retracting the landing gear without flow from the system. The main gear in addition has accumulators for the braking system sufficient for multiple braking operations even in the event of fluid loss from the main hydraulic unit system.

166 Circulation pumps are used to operate low power requirement loads, such as the payload bay doors while in orbit, and to prevent viscosity changes in the hydraulic fluid caused by low temperatures. Heat exchangers are provided to maintain the hydraulic oil within its operating limits.

166 Caution and Warning displays to the crew are provided for low-temperature, high-temperature, low-fluid level, and low-pressure.

166 Distribution of power from the four independent hydraulic subsystems to the various flight control and utility subsystems is illustrated in FIGURE 4.2.2-1. Use of four independent subsystems provides safe flight and landing after a second failure regardless of the time of the second occurrence. All utility functions are isolated from the pumping source by valves which block the trunklines after each actuation. Four independent hydraulic systems are powered by variable displacement pumps driven by separate APU's. Bootstrap type reservoirs with air/oil separators for self-bleeding capability ensure fluid stiffness and service life.

166 The hydraulic subsystem is designed for operation at a nominal pressure of 3000 psi over a temperature range of —40°F to +275°F. The system uses MIL-H-83282 synthetic hydrocarbon fluid, thereby taking advantage of its superior high-temperature, improved fire resistance, and reduced vacuum vaporization characteristics.

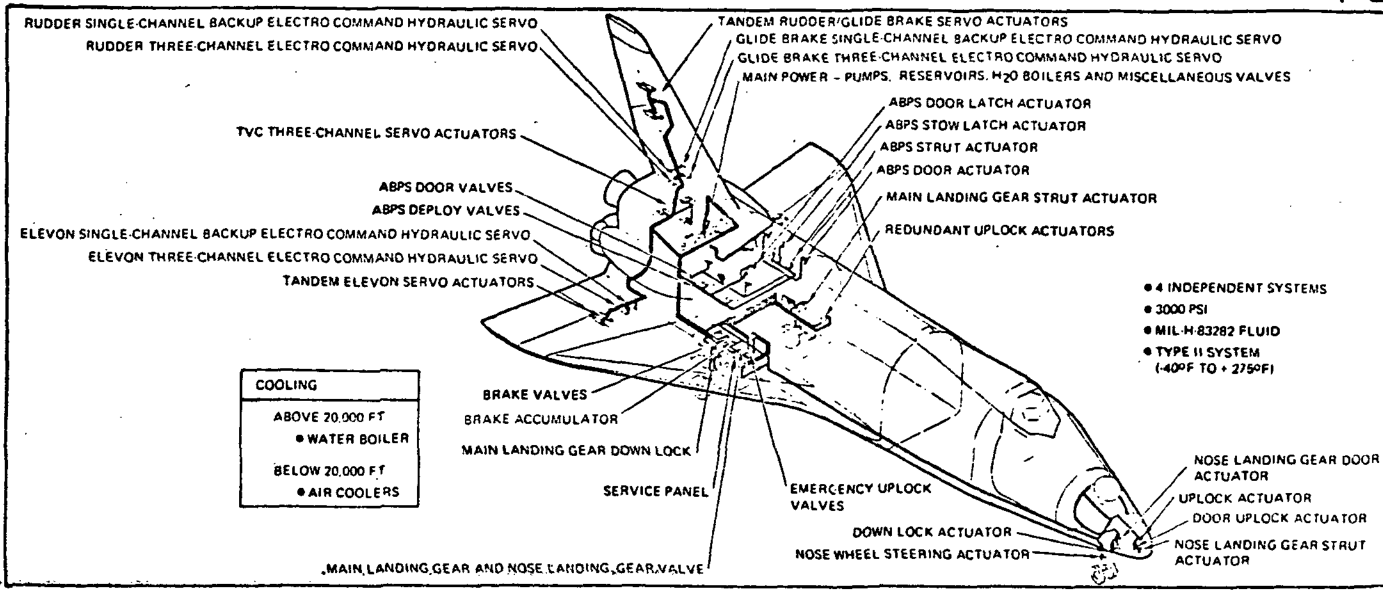
166 Fluid cleanliness is maintained by 5-micron (nominal) filters installed downstream of all case drain manifolds, all outlet manifolds, and in the return port of each reservoir. Filters are also installed upstream of all contamination-sensitive components such as servo valves.

4.2.2.1 Rationale for Assumptions

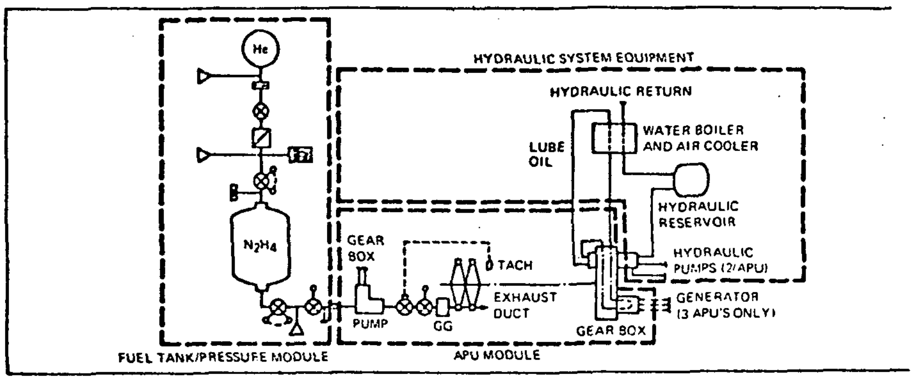
Not required.

4.2.2.2 References

166 pages 3-86 to 3-89



Orbiter Hydraulic Subsystem Arrangement



APU Subsystem

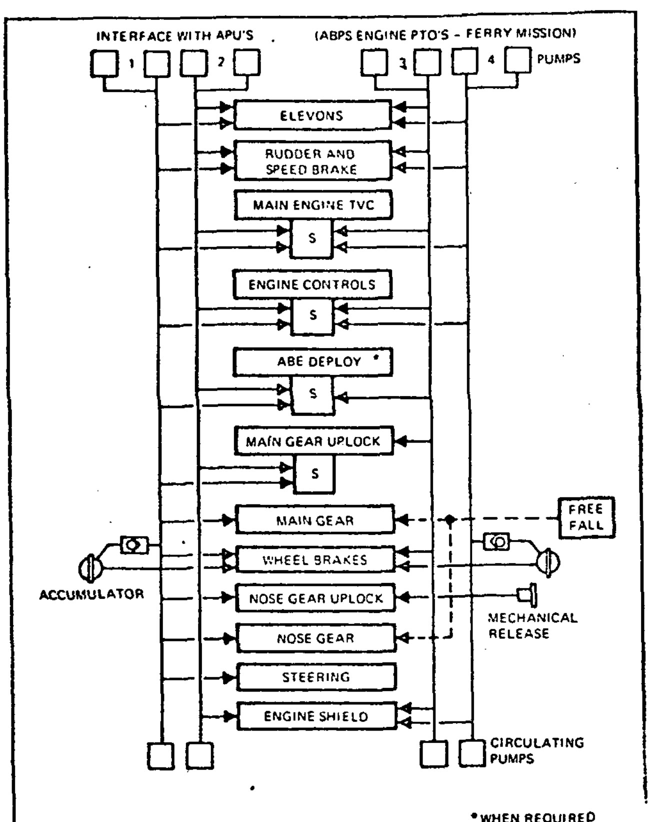


Figure 4.2.2-1 *Hydraulic Subsystem Configuration*

4.3 MAIN PROPULSION SUBSYSTEM (MPS)

166 The orbiter main propulsion subsystem (MPS), assisted by two booster solid rocket motors (SRM) during the initial phase of the ascent trajectory, provides the velocity increment and thrust vector control for insertion of the orbiter into a 50- by 100-nm orbit. MPS boost operation begins immediately before liftoff and terminates at orbit insertion.

166 The MPS consists of three liquid propellant rocket engines plus associated tankage, plumbing, valves, and controls. The engines operate on liquid oxygen and liquid hydrogen propellants contained in the orbiter external tank (ET), which is released following orbit insertion. The general arrangement of the mated MPS and ET is shown in Figure 4.3-1.

166 At normal power level (NPL), each engine operates nominally at a mixture ratio (LO_2/LH_2) of 6.0:1 and a chamber pressure of 3,000 psia to produce a vacuum thrust of 470,000 pounds with a fixed nozzle area ratio of 80:1. Nominal vacuum specific impulse (I_{sp}) for a single engine operating under these conditions is 455.2 seconds. Table 4.3-1 lists the total MPS propellant inventory for a 40,000 pound payload polar orbit. The same propellant loading will be used for reduced mission requirements or lighter payloads; however, in these cases the trajectory will be flown on a nonoptimum energy basis.

166 A schematic diagram of the integrated MPS and ET is presented in Figure 4.3-2. The ET contains three fuel and two oxidizer fluid lines interfacing with the orbiter at self-sealing disconnects on the bottom side of the orbiter aft fuselage. The three fuel disconnects are clustered on the left side mounted on a common carrier plate, and the two oxidizer disconnects are mounted on a similar carrier plate on the right side. The vehicle carrier plates are located

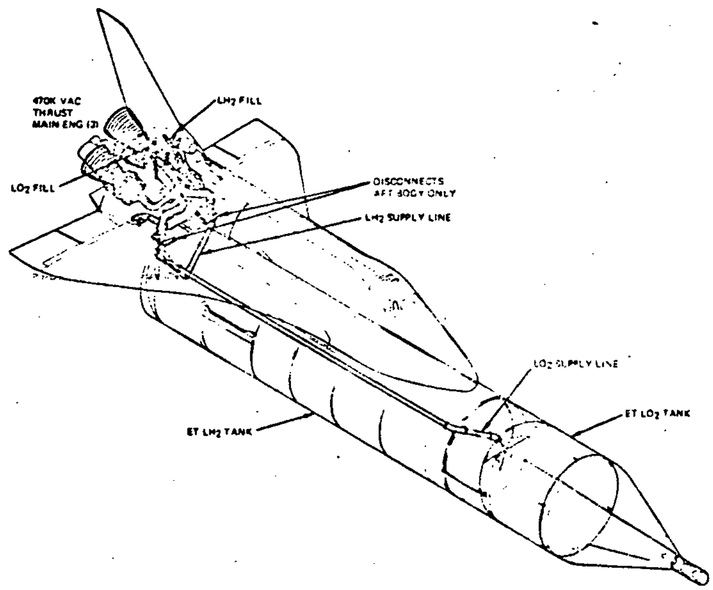


FIGURE 4.3-1

MAIN PROPULSION SUBSYSTEM

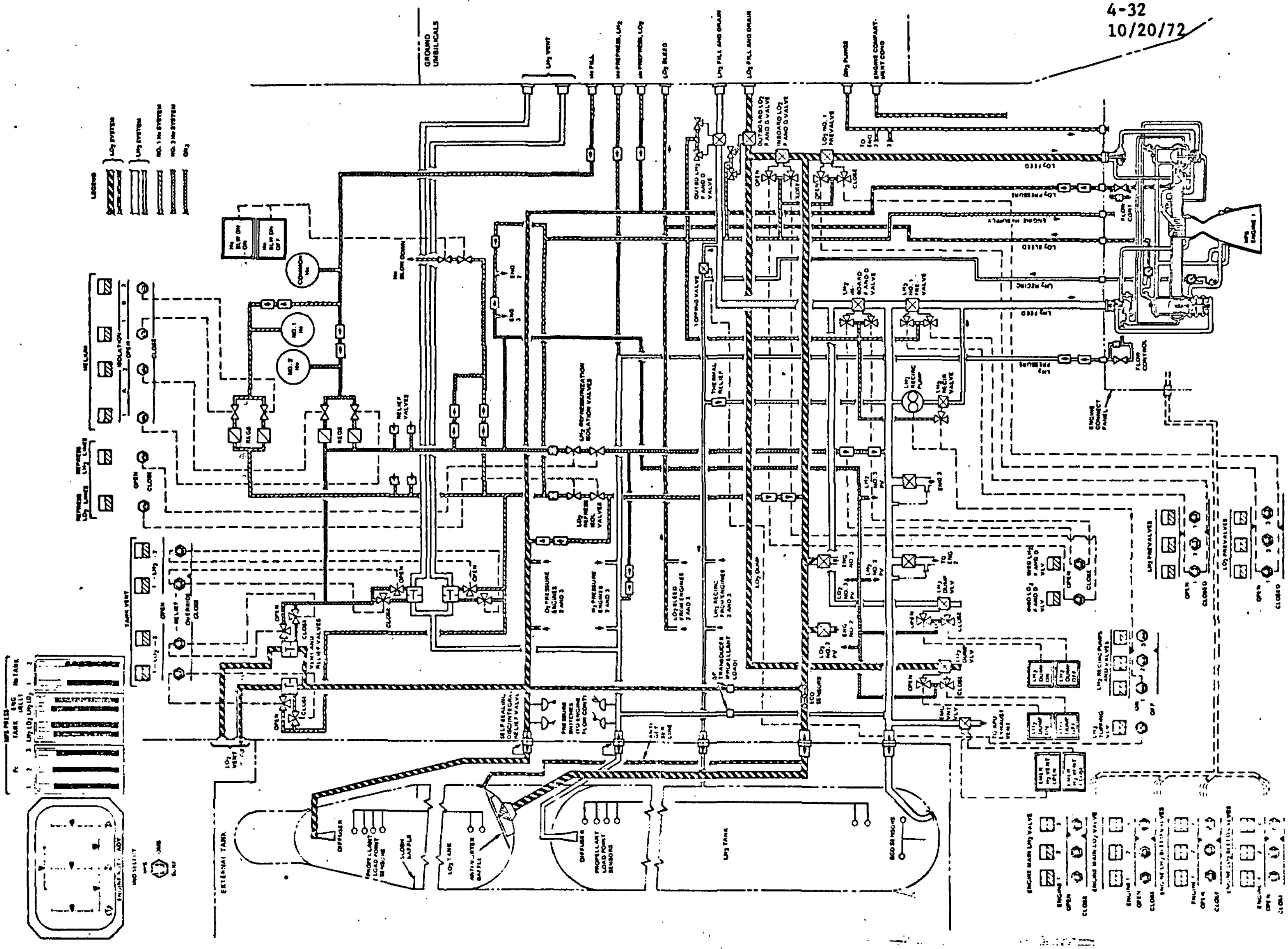


FIGURE 4.3-2

MAIN PROPULSION SUBSYSTEM SCHEMATIC

TABLE 4.3-1

ORBITER PROPELLANT INVENTORY

Item	LH_2 (lb)	LO_2 (lb)	Total
Ascent propellant (nominal)	241,428	1,448,572	1,690,000
1% ΔV performance reserve	1,011	6,064	7,075
Additional FPR (no PU)	143	857	1,000
Abort reserve (Polar Orbit)	1,143	6,857	8,000
Residuals			
Bias	1,500	-	1,500
Pressurant	1,375	3,370	4,745
Tanks and lines	889	778	1,667
Engines	75	945	1,020
Total Residuals	3,839	5,093	8,932
Total propellant at liftoff	247,564	1,467,443	1,715,007
Pre-liftoff use	250	2,130	2,380
Total propellant tanked	247,814	1,469,573	1,717,387

166 inside the vehicle moldline and are covered by flush closure doors following tank jettison to provide protection from heating during entry. The tank vent line disconnects each contain an integral safety relief valve that precludes tank overpressure when the tank-mounted disconnect is not engaged with the orbiter-mounted half. All other MPS fluid control valves, including tank vent and tank fill and drain valves, are located in the orbiter to minimize ET throwaway costs. Saturn S-II-type fluid line vibration damping mounts will be provided on the tank-mounted lines at critical locations. All tank-mounted fluid lines are designed for single mission (minimum weight and cost) application, whereas the orbiter-mounted fluid lines are designed for maximum reusability and employ vacuum jacket insulation. The installation arrangement of all orbiter-mounted fuel and oxidizer lines is portrayed in Figure 4.3-3.

166 The MPS fluid valves employ the same design concepts as the valves developed for the Saturn V program except for upgrading required to achieve the extended life (reusability) requirements of the orbiter. A 4,000 psi helium storage system with 750 psig regulation capability is provided in the orbiter for valve actuation and engine helium requirements. The schematic arrangement of the orbiter MPS helium system is included in Figure 4.3-3.

166 Propellant servicing is accomplished through fuel and oxidizer disconnects of eight-inch diameter located on the upper shoulder of the orbiter aft fuselage on the opposite sides of the vertical stabilizer. The eight-inch diameter fill and drain lines each contain two shutoff valves in series to assure closure at liftoff. LO_2 geysering in the ET feedline manifold is prevented during prelaunch operations by convection-induced flow of LO_2 from the insulated 17-inch diameter

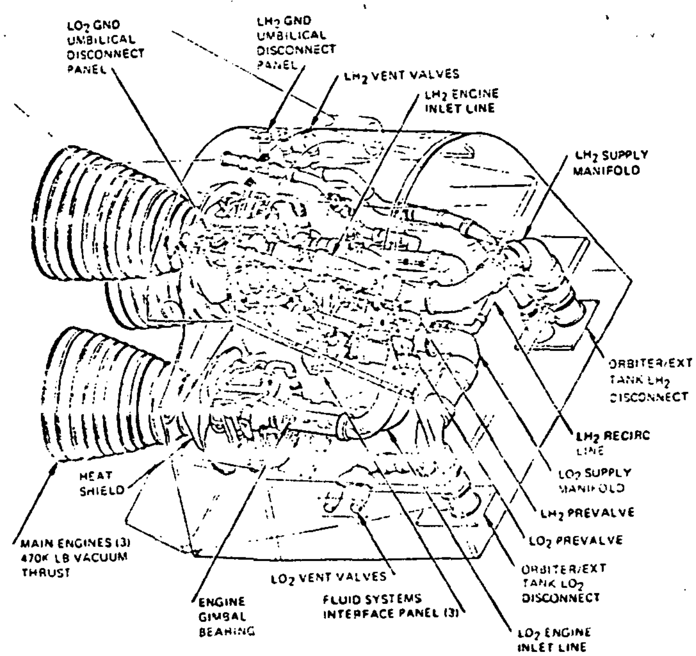


FIGURE 4.3-3

MAIN PROPULSION SUBSYSTEM INSTALLATION

166 manifold to the 4-inch diameter uninsulated antigeysering duct. Propellant loading is controlled by an orbiter-mounted signal conditioning unit supplying the ground loading equipment with signals from the ET-mounted warm wire point sensors. Differential pressure gauges installed in the orbiter between the propellant feed and vent lines provide intermediate loading level data between point sensors for both the fuel and oxidizer subsystems.

166 Predicted propellant loading tolerances are within ±0.6 percent for LO_2 and ±0.7 percent for LH_2. Tank manufacturing volumetric tolerances and repeatability of point sensor locations will be controlled to achieve the required flight-to-flight propellant loading accuracy. The propellant gauging system will be used for controlling tank loading operations and provides level indication. The in-flight engine mixture ratio control is preprogrammed before flight to a fixed value in the range of 5.8 to 6.2. The preprogrammed value for each flight will be based on predicted engine performance and predicted loading system tolerances to minimize the quantity of unused LO_2. Incorporation of an LH_2 bias (1,500 pounds) to assure depletion of the heavier LO_2 propellant is effective in reducing the magnitude of the residual propellant mass.

The following functions are provided to assure satisfactory engine start:

1. Delivery of subcooled propellants to each engine inlet (3 lb/sec LO_2, 1 lb/sec LH_2) to chill the engine to prescribed levels. This operation will start 15 minutes before engine start and terminates coincidently with engine start command.
2. Ground prepressurization of tank ullage with helium (LO_2 at 20 to 22 psia, LH_2 at 35 to 37 psia). This operation will be completed approximately 60 seconds before liftoff.

166 Subcooled LO_2 is maintained at the engine inlets by overboard bleed from the feedlines through the engines to ground disposal. During overboard bleed, LO_2 topping flow will be limited to a maximum temperature of 166° R to assure the presence of adequately subcooled propellant at the engine inlet. A comparable LH_2 overboard preconditioning system is not used because of the excessive pressure loss through the engines. Subcooled LH_2 is delivered to the engine inlets by electric-driven pumps which discharge conditioning flow downstream of closed prevalves through the engines and to the LH_2 tank via a recirculation manifold. During LH_2 recirculation, LH_2 topping flow is diverted to the recirculation return manifold to preclude warm LH_2 entering the pumps.

166 After completion of the engine conditioning cycle, the vehicle liftoff sequence begins by commanding engine start to NPL. When all three engines reach 50 percent power level and automatic self-check of engine operation is satisfactorily completed, ignition of the booster SRM occurs. When main engine thrust reaches approximately 90 percent, the vehicle thrust-to-weight ratio exceeds 1.0, and liftoff occurs. Throttling of engine thrust below 100 percent is commanded during orbiter ascent to limit vehicle acceleration to 3g. An emergency power level (EPL) of 109 percent may be commanded on the operating engines if premature thrust decay or engine shutdown is encountered on any of the three engines.

166 Tank prepressurization and hydrostatic head provide the required net positive suction pressure (NPSP) to the engine pump inlets during the starting transient. Following engine thrust buildup, tank ullage pressure is maintained by vaporized propellant pressurant extracted from the engines. An engine-supplied on-off control valve (controlled by redundant pressure switches sensing tank

166 ullage pressure) in parallel with a bypass orifice is used for in-flight tank pressure control. Predicted tank pressurization performance is shown in Figure 4.3-4.

166 The gimbal actuators and the engine control systems for the three engines
176 are supplied with hydraulic pressure from four redundant orbiter vehicle-mounted APU-driven hydraulic systems. Thrust vector control during boost is attained by gimbaling the MPS engines to the maximum pitch and yaw deflections of ±11 degrees. The engines are mounted in the orbiter aft fuselage in a triangular pattern with an angle of 6 degrees between the upper and lower engine centerlines in the pitch plane. The lower engines are installed with their null position canted inboard 3.5 degrees in the yaw plane; however, all engines are fired with parallel thrust vector alignment in the yaw plane. The engine spacing of 104 inches in yaw and 100 inches in pitch allows adequate clearance for maximum gimbal deflection on the two remaining engines after any one engine has failed in its null position. The gimbal actuators are mechanized to drive to a null position if two hydraulic systems are lost, if two hard-over servo valve failures occur, or if two electrical signals are lost. The hydromechanical logic circuit within the actuator assembly senses these failures and diverts the remaining hydraulic pressure supply to drive the actuator to the null position.

166 MPS engine control and data signals are processed through an engine
42 interface unit. The unit converts all engine control inputs (GN&C and manual) to be compatible with the engine controller serial digital data system. Engine serial data output is also appropriately converted for use by vehicle computer systems and for dedicated displays. Engine data input to the PCM recorder and

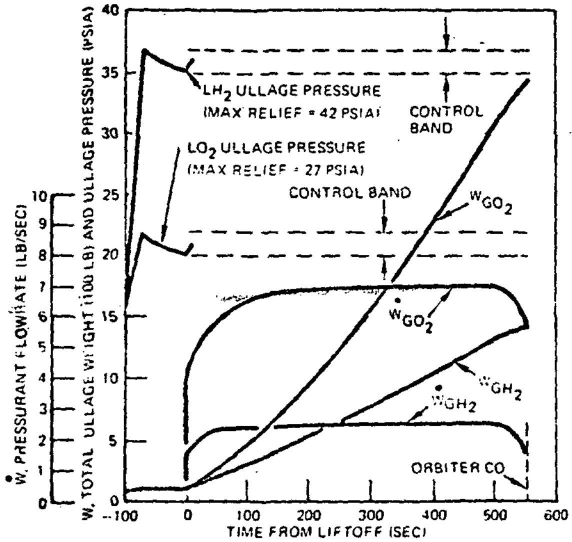


FIGURE 4.3-4

LO_2 AND LH_2 PRESSURIZATION REQUIREMENTS to telemetry will be transmitted as direct data output through separate engine instrumentation connectors per the vehicle/engine ICD. The engine interface unit is further described in Section 4.3.2.

166 A pogo suppression accumulator will be incorporated in each LO_2 propellant feedline at the engine inlet.

166 Engine cutoff will normally be provided by a signal from the Guidance, Navigation, and Control (GN&C) subsystem when orbit injection velocity is attained. The Flight Performance Reserve (FPR) propellant (Table 4.3-1) provides a 317 foot per second (1 percent) ideal delta velocity margin to assure orbit injection. In those missions where the FPR is expended, engine cutoff will be initiated by an LO_2 depletion signal from a cluster of five point sensors located in the orbiter LO_2 feed manifold. In the event an off-nominal operation condition results in LH_2 depletion, a cluster of five point sensors located in the bottom of the LH_2 tank provides an engine cutoff signal to preclude engine damage after shutdown.

166 Propellant dumping and depressurization of the tanks to 5 to 15 psi before tank release will be manually controlled. Dumping of residual liquids and venting of residual gases will be accomplished through overboard fuel and oxidizer dump lines which discharge aft of the vehicle. Liquid propellant also will be dumped through the engines concurrent with dump system operation to provide sufficient thrust to settle the propellants remaining in the tanks. The total velocity increments introducted by dumping will be limited to 30 ft/sec. LO_2 dumping will be initiated first followed by LH_2 dumping. GH_2 and GO_2 will be vented concurrently through the dump lines.

166 Propellant dumping through the engine will be a crew-controlled operation starting five seconds after main engine shutdown. Dumping will be initiated by opening the engine main valves and the vehicle prevalves. LH_2 and LO_2 will be dumped through both engines. The thrust obtained from dumping will retain propellants in the aft end of the propellant tanks and in the feed system. Dump cutoff control will be accomplished manually.

166 A short period after orbiter propellant dumping, the boiloff vent system is activated by opening the shutoff valves. Boiloff is vented overboard to maintain the ullage pressure within the 25.5- to 27.5-psia range. The redundant shutoff valve in series with the primary vent valve is left in the open position to permit emergency operation of the vent valve in the event of excess evaporation resulting from residual liquid splashing on hot structure. This redundant valve may be closed later in the mission when all of the residual liquid has been evaporated and/or the tank pressure decays at a level lower than the operating band of the boiloff vent system. The flow from the boiloff vent system is directed to the nonpropulsive vents.

166 Immediately following propellant dumping the engines are purged with helium, and the oxygen and hydrogen prevalves are closed. The engine main valves, closed in series with the vehicle valves, provide a double seal against tank leakage.

166 The AC electrical power and the hydraulic supply to the engines is discontinued two minutes after engine shutdown by stopping the APU and activating the cutoff circuits. At a convenient time following engine cutoff, all electrical power is removed from the engine interface to prevent overheating the engine control unit.

166 The main engine helium supply valve will also be closed if engine helium leakage continues after engine shutdown.

Figure 4.3-2 illustrates MPS controls and dedicated displays for critical performance parameters. Pilot override control capability is provided for critical MPS functions, e.g., engine shutdown and thrust level.

166 4.3.1 Engine

42 The orbiter main engine is a pump-fed, high chamber pressure engine utilizing a staged combustion concept and liquid hydrogen/liquid oxygen propellants. The 3,000-psi chamber pressure engine provides a vacuum thrust of 470,000 pounds with a nozzle expansion ratio of 80:1 and operates at an average propellant mixture ratio of 6:1.

166 The major components of the engine assembly consist of a main
42 chamber, nozzle, preburner, fuel and oxidizer turbopumps, and fixed low-pressure fuel and oxidizer boost pumps. Other components include an engine control unit, oxygen pressurant heat exchanger, fuel and oxidizer shutoff and control valves, a thrust mount, and a gimbal bearing block. Provisions also are included for a gaseous hydrogen pressurant bleed, fluid and electrical disconnect panels, and attachments for the TVC gimbal actuators.

166 During engine operation, propellants flow through the turbopumps, where
42 the propellants are pumped to high pressure for injection into the preburner and main combustion chamber. Prior to injection, the liquid hydrogen is used to cool the thrust chamber and preburner jackets. The preburner operates with propellants at a mixture ratio of approximately 1:1 to provide relatively cool fuel-rich combustion gases to drive the turbopumps. The fuel-rich turbine exhaust gases are injected into the main chamber with additional liquid oxygen to support main thrust chamber combustion.

4.3.2 Control/Monitor System

42 The control for engine start, main-stage, and shutdown is performed by sensing and monitoring engine performance, an electronic controller computes commands to modulate and sequence the values, and the integral spark ignition system. Refer to Figure 4.3.2-1. The engine monitoring function is provided to ensure the vehicle that the engine is functioning properly and to collect engine data for vehicle recording and post-flight engine maintenance analysis.

42 The sensors provide the engine data to the controller and vehicle for control, checkout and maintenance, and condition monitoring. Fifty-two engine parameters consisting of pressure, temperature, propellant volumetric flow, turbopump shaft speed, vibration, and position are measured. The sensor signal conditioning and analog-to-digital conversion electronics is located in the controller.

42 The controller performs the computations for engine control and sequencing, checkout, and monitoring. The computations are performed by digital computers within the controller assembly. The sensor signal conditioning and analog-to-digital conversion electronics are time shared among the sensors by multiplexing within the controller. The controller interfaces with the vehicle data bus, accepts and validates vehicle commands, and, with sensor data, computes propellant valve actuator commands for engine control. Controller output electronic circuitry transmits commands for positioning the valve actuators, switching hydraulic and pneumatic solenoid valves, and controlling spark ignition.

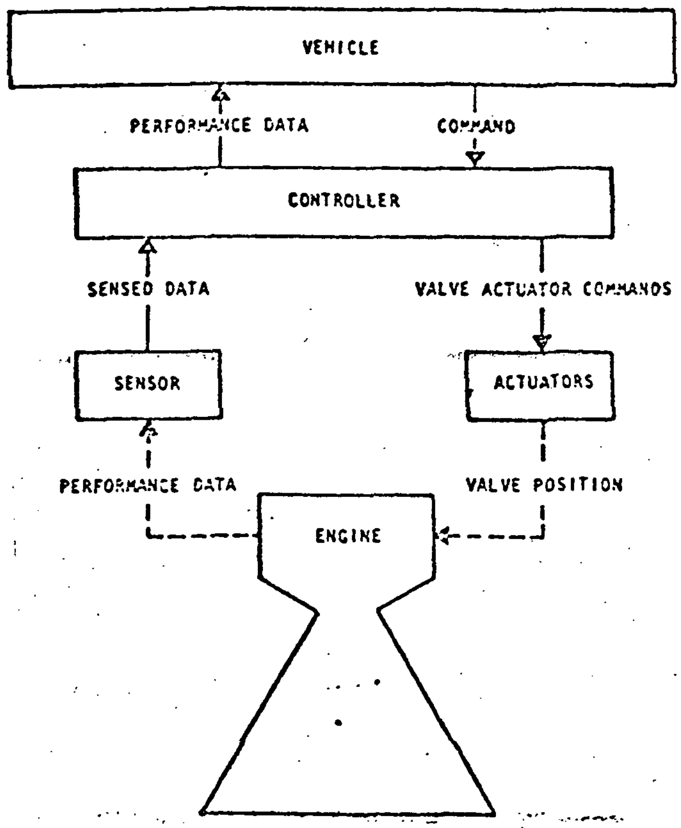


FIGURE 4.3.2-1

ENGINE CONTROL AND MONITOR

4.3.2.1 Engine Thrust/Mixture Ratio

42 To enable control of variable thrust and mixture ratio level, the system uses a simple two valve control. The control points selected have capability to fully control the engine thrust and mixture ratio over the full range of required engine operating conditions. The control modulates the areas of the oxidizer preburner oxidizer valve and the fuel preburner oxidizer valve to achieve independently the proper balance of propellant flows for thrust and mixture ratio control.

42 The control/monitor accepts data in flight and uses the data in the controller to compute the modulated control valve settings required to achieve vehicle commanded thrust and mixture ratio. This is achieved by the controller accepting vehicle thrust and mixture ratio commands and comparing them with monitored thrust and mixture ratio to determine a thrust and mixture ratio error. The thrust and mixture ratio errors determine the directions to change the two modulated control valve areas. The controller recomputes valve areas 50 times per second (every 20 msec). Each time the valve areas are adjusted the controller monitors engine performance and recomputes the error between vehicle commanded performance and engine montiroed performance. When the error in performance reaches allowable limits the controller maintains the valves at those areas which achieved the desired level of performance. The process of comparing commanded performance and monitored performance and adjusting control valve setting is a closed loop control system.

42 Two performance control loops and one limit control loop are used in the engine control. Engine thrust is computed (as a percent of vacuum normal power level) in the controller from sensed main combustion chamber pressure and corrected for mixture ratio (and dump coolant on the orbiter). The vehicle

commanded thrust is compared with the computed thrust to obtain an error signal. The thrust error signal is used to compute an area command for both the oxidizer preburner oxidizer valve and fuel preburner oxidizer valve to drive the thrust error to an allowable value.

42 Engine mixture ratio is computed in the controller from sensed propellant volumetric flows and corrected by propellant density computed from sensed propellant pressure and temperature. The vehicle commanded mixture ratio is compared with the computed mixture ratio to obtain an error signal. The mixture ratio error is used to compute a correction to **position** the fuel preburner oxidizer valve, which also drives the mixture ratio error to an allowable value.

42 Repeated computations of the thrust and mixture ratio control result in thrust and mixture ratio being controlled to within ±5,920 pounds (26,333.3 newtons) thrust and ±0.82 percent mixture ratio precision which is below the requirement of ±7,500 pounds (33,361.5 newtons) and ± 1 percent.

42 Preburner temperature limiting control is provided for protection of the turbine power system from overtemperature. The preburner limit control functions by the controller computing an override command for the thrust control loop to decrease thrust whenever a sensed preburner temperature rapidly approaches or exceeds 2,200 R (1,222.2 K). The limit control normally affects system operation only during maximum thrust rate transients. As the preburner temperature stabilizes below 2,200 R (1,222.2 K), the temperature limit override is reduced, allowing control to revert to the thrust control loop, and the desired mainstage level is attained. This control logic aids in maintaining long-life operation of the high-pressure turbopumps.

42 Closed-loop control is used to achieve repeatability of thrust buildup to full mainstage power from a low power level (10 percent NPL) during start. For engine start the initial control valve sequences produce a low power level prior to thrust buildup. The same closed-loop control valves used for mainstage are used for thrust buildup control. Closed-loop mixture ratio control is initiated when thrust buildup to mainstage is completed.

42 In addition to the two valves used in closed-loop performance control, three other sequenced or scheduled valves control propellant flow during engine operation. These are:

1. The main oxidizer valve which is timed-positioned scheduled during startup and shutdown.
2. The main fuel valve, which is sequenced open and closed during startup and shutdown.
3. The combustion chamber coolant valve, which is scheduled with thrust level to optimize chamber cooling.

42 Engine shutdown can be initiated from any power level, during transient of steady-state operation. It is accomplished by reducing thrust under closed-loop control at a rate of 4,800 pounds (21,351.4 newtons) per 10 msec. After a low thrust level has been attained, propellant valves are sequenced closed for a fuel-rich shutdown. Shutdown performance and repeatability meet requirements by use of closed-loop control during a major portion of the transient.

42 4.3.2.2 <u>Engine Monitoring</u>

Vehicle and engine protection is ensured by engine condition monitoring performed by the system. Selected engine parameters are sensed and monitored by the controller. Controller digital computer programs compare the parameters against limits stored in memory. If an engine limit is exceeded and if the vehicle limit control enable command has been invoked by the vehicle, the controller shuts the engine down. The vehicle also has a limit control inhibit command available to prevent against engine shutdown during critical periods of engine operation. This is done because the controller has no information with respect to the condition of the other booster engines, and it may be necessary to operate an engine above limits for a short time to ensure vehicle safety.

42 Parameters monitored for limit shutdown are fuel preburner over temperature, oxidizer preburner over-temperature, high-pressure fuel turbopump shaft overspeed, high-pressure oxidizer turbopump shaft over-speed, combustion chamber (high and low) pressure, and oxidizer heat exchanger failure.

42 If the safe limits stored in the controller memory are exceeded the controller automatically initiates a normal shutdown.

4.3.2.3 SSME CONTROLLER DATA FLOW

42 All data flow for the SSME (Figure 4.3.2.3-1) passes through the controller.

42 Data flow between the engine and vehicle consists of operational commands and data requests from the vehicle to the engine and data transmission from the engine to the vehicle. All vehicle data to the engine are received from the vehicle/engine data bus interface unit. The controller digital computer interface electronics route the data to or from the controller digital computers. Data flow in either direction is under vehicle control.

42 Included in the engine-to-vehicle data flow is engine status, other operational parameters, and maintenance data. The engine status includes engine operational phase, mode of operation, and self-test status. The operational parameters include actual thrust and mixture ratio. The data are transmitted in blocks of digital words at a maximum rate of 25 blocks of data per second and a maximum data bit rate of 10,000 bits/second.

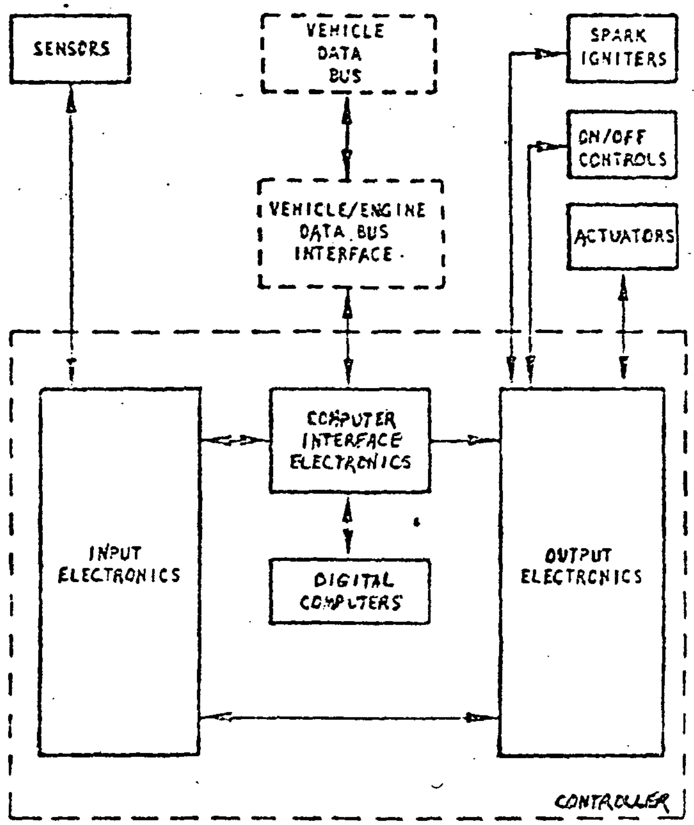


FIGURE 4.3.2.3-1

SSME DATA FLOW

42 Sensor data from 67 sensor assemblies (93 signals with redundant elements) for the control and monitor functions are sampled, signal conditioned, multiplexed, and converted from analog to digital form in the input electronics every 20 msec. These sensor data are routed by the digital computer interface electronics to the digital computers. During checkout, commands from the digital computer flow back through the digital computer interface electronics unit to activate sensor calibration circuits in the input electronics.

42 Data flow for sequencing the engine and controlling actuation devices is originated in the digital computers. Commands to initiate spark ignition, control solenoid valves, and modulate positions of valve actuators are computed in the digital computers. The digital interface electronics steer the data to the output electronics. The output electronics perform the digital-to-analog conversion and power amplification, and interface with the specific controlled devices. The commands are updated every 20 msec.

4.3.3 SSME Operation Details

4.3.3.1 SSME Sequence Schedule

4.3.3.1.1 Engine Start

42 Engine start operation is accomplished in two phases. The first phase of the start is the open-loop start normalization phase. The second phase is the closed-loop thrust buildup phase. Engine start to NPL is accomplished within 3.5 seconds. Simplicity and reliability of engine start control is obtained by accomplishing the start functions with the same control elements used for mainstage control and by establishing engine operation at a low power level prior to thrust buildup. Repeatability of thrust buildup is accomplished by bringing the system to full power under closed-loop control.

The start sequence (Figure 4.3.3-1) is initiated by opening the main fuel valve to provide priming of the engine fuel system and to establish fuel flow with tank-head pressure. The main fuel valve is full open in 0.4 second. Spark igniter operation is initiated in all units and maintained for 3.5 seconds into the start.

42 Opening of the three oxidizer valves is initiated 100 milliseconds after start. The main oxidizer valve ramps to a 55 ±2 per cent open position at a rate of 100 percent/sec. The fuel preburner oxidizer and oxidizer preburner oxidizer valves open to their initial opening of 2 percent of full area. Preburner ignition and main combustion chamber ignition follow in approximately 100 milliseconds. The four turbopumps are powered at a very low level by heated hydrogen until 0.5 second after start initiation. At 0.5 second, the fuel preburner oxidizer valve ramps to 12 percent open and thrust builds up to approximately 10 percent of NPL. This phase of engine start sequence is used to achieve approximately 10 percent NPL thrust to accommodate variations in the start transients due to tank pressures, booster-orbiter differences, environmental temperatures, etc. The positions of all valves are set to establish engine power at approximately 10 percent of NPL with an engine mixture ratio in the range of 1.0 to 2.0. Partial opening of the main oxidizer valve results in less oxidizer flow acceleration prior to main chamber priming, tends to increase preburner power level and temperature while at the low engine power levels, and reduces the pressure transients when the main chamber primes. When the main oxidizer injector primes, a thrust level of approximately 10 percent NPL is attained. During this phase when thrust is below 15 percent, the maximum thrust rate is approximately 13,000 pounds (57,826.6 newtons) per 10 msec period. This rate is the highest that has been experienced in simulation runs below 15 percent of NPL.

42 The flexibility of dual preburners allows fuel preburner flow buildup to occur prior to oxidizer preburner flow buildup to ensure a fuel-rich start and to maintain a satisfactory fuel pump flow coefficient with the largest pump inlet pressure unbalance [(20 psia (13,79N/cm^2)fuel, 225 psia (155.13 N/cm^2) oxidizer] that can exist.

42 The second phase of the start is initiated at 1.7 seconds by activating closed-loop thrust control. The thrust control is allowed to stabilize at 10 percent of NPL of 0.30 second. The thrust control compensation is proportional at this time. Integral compensation is added in proportion to thrust level until MPL is attained, when integral gain is at the nominal MPL mainstage value. At 2.0 seconds, the main oxidizer valve is ramped open and the thrust command ramp initiated. The thrust command ramp is set at 4,800 pounds (21,351.4 newtons) per 10 millisecond rate to ensure the fastest practical thrust rate increase without exceeding the 7,000-pounds (31,137.4 newtons) per 10 millisecond maximum allowable rate. During start, open-loop mixture ratio control is accomplished by positioning the fuel preburner oxidizer valve from a thrust controller cross-feed signal. The cross-feed gain is selected so that the engine mixture ratio is in a high specific impulse range (2.0 to 4.0 main combustor mixture ratio) during a major portion of the thrust buildup and is at approximately the nominal value of 6.0 at mixture ratio control activation. This maximizes the specific impulse during the start transient and minimizes closed-loop control activation transients.

42 At 3.25 seconds, closed-loop mixture ratio control is activated. At that time: (1) cross-feed gain from thrust to mixture ratio control is changed to the mainstage value, and (2) the mixture ratio control compensation output

is initialized for a smooth transition from scheduled to closed-loop mixture ratio control. At 3.5 seconds, thrust and mixture ratio are within mainstage tolerance.

4.3.3.1.2 Engine Shutdown

42 Engine shutdown can be initiated from any power level during transient or steady-state operation. Shutdown repeatability is obtained with closed-loop control. Shutdown may be initiated by vehicle command or controller-generated limit shutdown signal.

42 Engine shutdown is accomplished in two operating phases, illustrated in Figure 4.3.3-2 . The initial phase is a controlled thrust rate of 4,800 pounds (21,351 newtons) decrease per 10-msec period. This effective rate minimizes both the shutdown time and shutdown impulse while keeping the transient thrust change rate within the 7,000 pounds (31,137 newtons) per 10-msec maximum allowable thrust decay rate. The closed-loop shutdown is predictable and repeatable from any mainstage level. Closed-loop mixture ratio control is active during this phase of shutdown.

42 The second phase of the shutdown sequence is initiated when the thrust reference decreases below MPL and sequenced closing of the propellant valves is initiated. At the start of this phase, a 0.85 second main oxidizer valve closing ramp is initiated. At 0.3 second into the phase, the oxidizer preburner oxidizer valve and the fuel preburner oxidizer valve are commanded full closed at 150 percent/sec. At 0.65 second into the phase, main fuel valve closing is initiated. The combustor coolant valve is ramped close when thrust decreases below MPL. All oxidizer valves are closed before the main fuel valve, ensuring fuel-rich shutdown. Total impulse and propellant consumed below MPS are more than 100,000 pound-sec (444,820 newton-sec) and less than 480 pounds (216,7 kilograms), respectively.

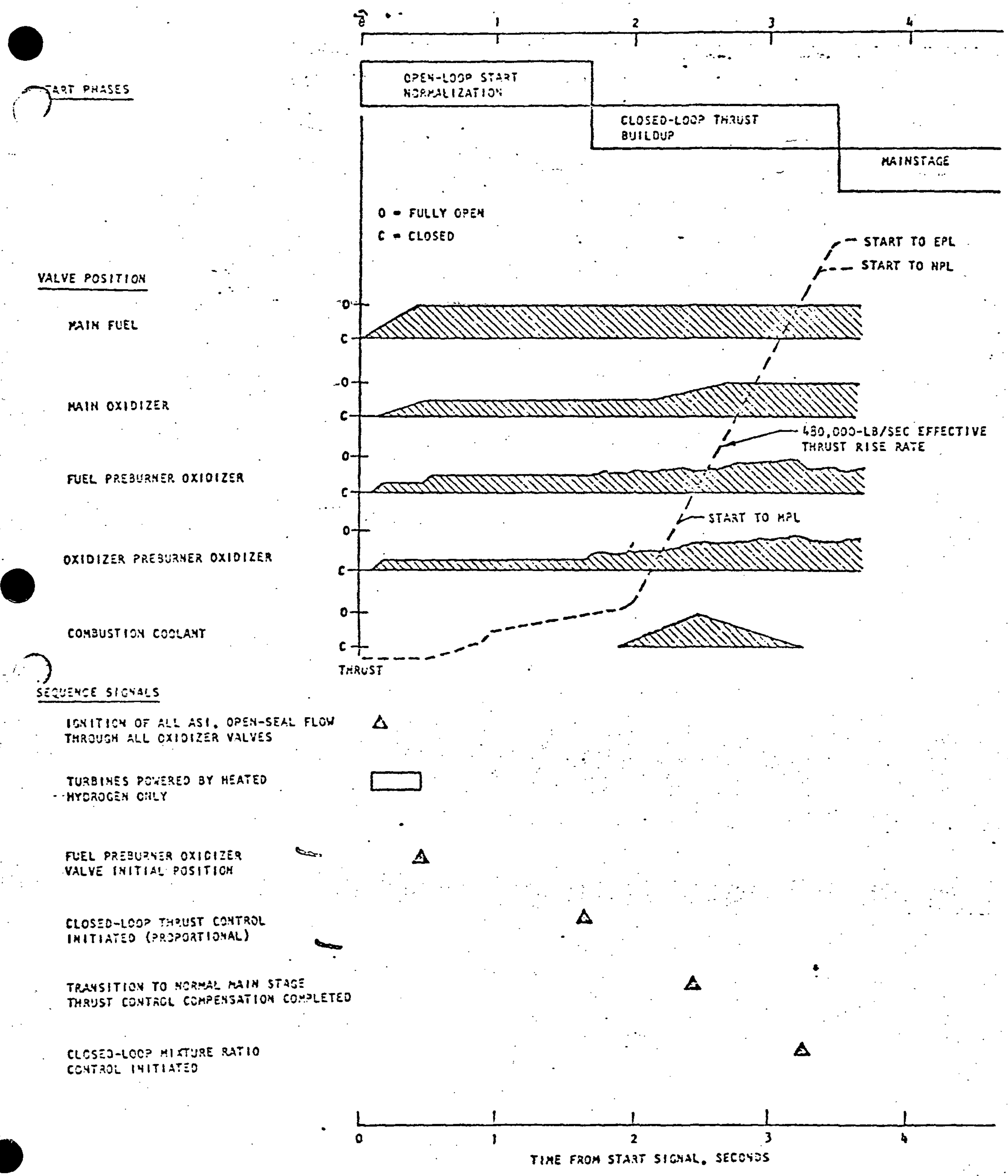


FIGURE 4.3.3-1

ENGINE START SEQUENCE

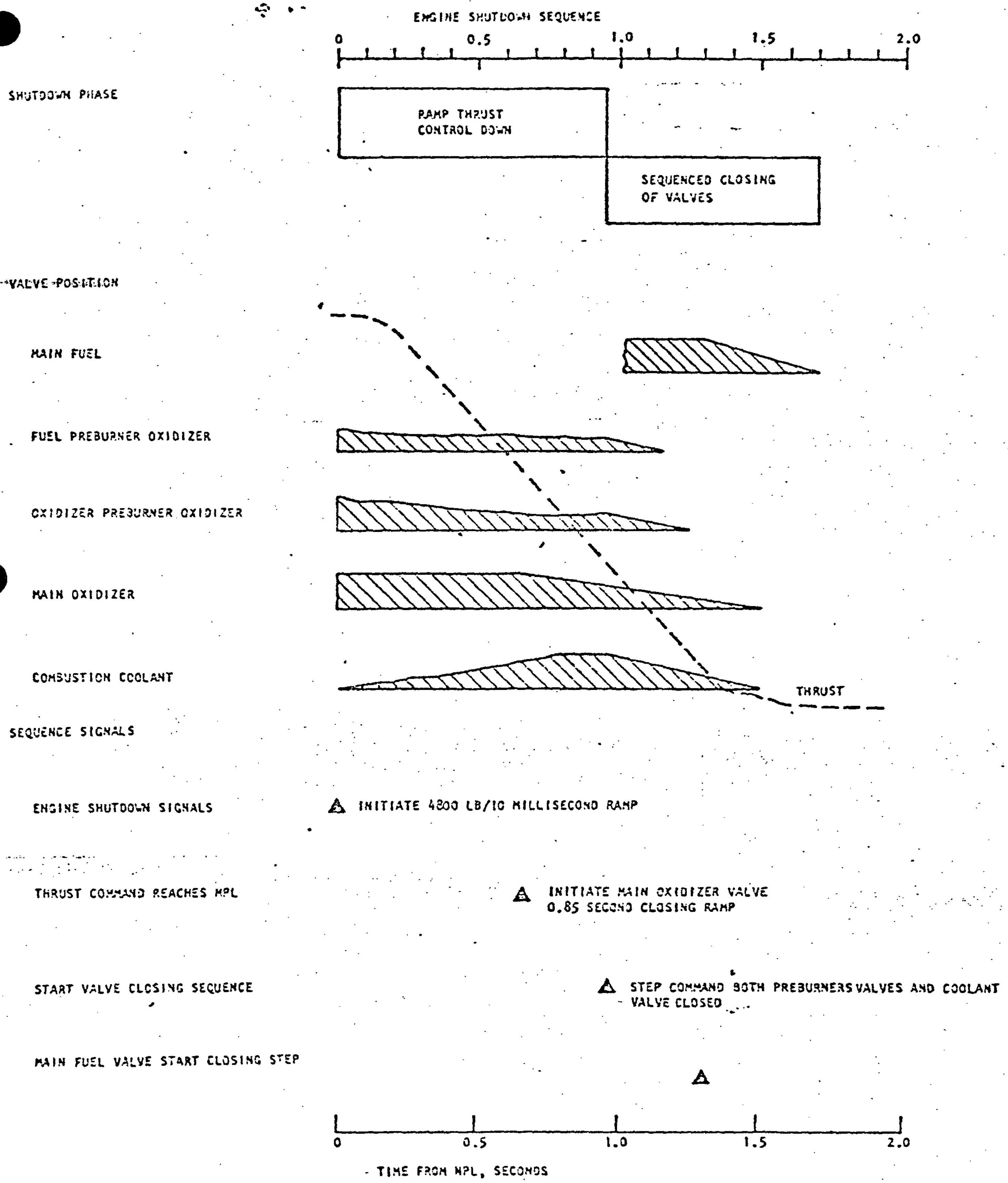


FIGURE 4.3.3-2

ENGINE SHUTDOWN SEQUENCE

42 The main fuel valve is full closed 1.05 seconds after thrust reference reaches MPL and all propellant flow has stopped except for residuals downstream of the main valves. Impulse from residuals will be approximately 15,000 lb/sec (66,723 newton-sec) and will have a duration of about 0.5 second.

4.3.3.2 SSME Flight Operations Monitoring

42 Continuous In-Flight Test

During engine operation, the controller continually performs a built-in test of the controller, sensors, and actuating devices. A summary of these built-in tests is provided in Table 4.3.3.2-1. During the start and shutdown sequences, the controller monitors the opening and closing sequencing of the propellant valve actuators and verifies that the power level of the engine increases and decreases in accordance with programmed limits. It also verifies proper timing of the ignition sequence.

42 As part of the shutdown sequence, the controller verifies that all control devices have been returned to a fail-safe condition and that the engine is safely shut down. Position sensors are monitored to verify retraction of the orbiter's extendible nozzle. The engine shutdown status is made available to the vehicle.

42 Additional monitoring during flight operation falls into five categories:

1. Performance Monitoring - Monitoring of engine thrust and mixture ratio to verify proper response to vehicle commands and to provide engine performance status to the vehicle.
2. Limit Control Override - Monitoring of preburner temperatures. If a temperature limit is exceeded, the controller throttles the engine power level to reduce temperature below the limit if limit control is enabled from the vehicle.

3. Limit Control Shutdown - Detection of preburner over temperature, turbopump overspeed, main combustion chamber overpressure and underpressure and oxidizer heat exchanger failure. The controller uses the limit control shutdown measurements to determine if the engine has exceeded a safe operating limit or shutdown condition. If any of these parameters exceed a safe operating limit, the engine is shut down provided the limit control is enabled from the vehicle.

   The preburner temperature limits used in limit control shutdown are higher than those used in limit control override. Table 4.3.3.2-2 summarizes the measurements and the limits which determine if the engine should be shut down.

4. Position Monitoring - Monitoring of actuator/valve positions for verification of proper operating sequences and interlocks.

5. Data Transmission - Transmitting engine status and performance data to the vehicle for recording of engine trends.

42 Propellant dumping through the engine may be sequenced during the post shutdown phase. The propellant valves are positioned by commands from the vehicle. The controller provides interlock logic to ensure oxidizer dumping prior to fuel dumping and to prevent both fuel and oxidizer valves from being open at the same time.

TEST ITEM	SUMMARY OF TEST	GROUND OPERATIONS CHECKOUT	FLIGHT OPERATIONS MONITORING
CONTROLLER			
DIGITAL COMPUTER	SAMPLE PROBLEM TESTS PROCESSOR AND MEMORY	X	X
	MEMORY PARITY CHECK ON EACH READ OPERATION	X	X
	WATCHDOG TIMERS VERIFY PROPER SEQUENCING THROUGH DIGITAL PROGRAM	X	X
	MEMORY SUM CHECKS	X	
COMPUTER INTERFACE ELECTRONICS	CIRCULATE DATA FROM MEMORY THROUGH DATA BUS INTERFACE ELECTRONICS BACK INTO MEMORY	X	X
	SIMULATE WATCHDOG TIMER OPERATION BY ALTERING DECAY TIME	X	X
INPUT ELECTRONICS	INPUT REFERENCE VOLTAGES TO CHECK ANALOG TO DIGITAL CONVERTER	X	X
	CHECK MULTIPLEXER SWITCHES WITH SENSOR TESTS	X	X
OUTPUT ELECTRONICS	INDIVIDUAL POWER SUPPLY VOLTAGES VERIFIED BY REGULATOR MONITOR CIRCUIT	X	X
SENSORS	STIMULATE SENSORS BY BRIDGE SHUNTING OR INDUCTIVE/CAPACITIVE COUPLING OF TEST SIGNALS	X	
	CHECK SENSOR SIGNALS FOR REASONABLENESS	X	X
	PERFORM COMPARISON TESTS ON REDUNDANT SENSORS		X
ACTUATORS	ISSUE STEP COMMANDS AND CHECK ACTUATOR POSITION AND RATE OF TRAVEL	X	
	COMPARE SERVOVALVE OUTPUTS WITH ELECTRONIC MODEL	X	X
	COMPARE ACTUATOR POSITION FEEDBACK SIGNALS WITH ACTUATOR COMMANDS	X	X
ON/OFF VALVES	ENERGIZE ON/OFF VALVES AND VERIFY SWITCHING BY POSITION AND/OR PRESSURE MEASUREMENTS	X	
	MONITOR NORMAL SWITCHING OPERATIONS BY POSITION AND/OR PRESSURE MEASUREMENTS	X	X
SPARK IGNITERS	ENERGIZE AND MONITOR SPARK RATE AND VOLTAGE	X	
	MONITOR NORMAL OPERATION		X

TABLE 4.3.3.2-1

BUILT-IN TEST METHODS

PARAMETER	SENSOR	NOMINAL/SHUTDOWN LIMIT	REMARKS
FUEL PREBURNER OVER TEMPERATURE	FUEL PREBURNER TEMPERATURE	2250 R	TEMPERATURE SPIKE ABOVE THIS LEVEL LIMITED TO 0.5 SEC
OXIDIZER PREBURNER OVER TEMPERATURE	OXIDIZER PREBURNER TEMPERATURE	2250 R	TEMPERATURE SPIKE ABOVE THIS LEVEL LIMITED TO 0.5 SEC
HIGH-PRESSURE FUEL TURBOPUMP	SHAFT SPEED	33,300 RPM	REDUCED DESIGN SAFETY FACTOR AT SPEEDS EXCEEDING THE LIMIT
HIGH-PRESSURE OXIDIZER TURBO-PUMP SHAFT OVER SPEED	SHAFT SPEED	28,700 RPM	REDUCED DESIGN SAFETY FACTOR AT SPEEDS EXCEEDING THE LIMIT
COMBUSTION CHAMBER PRESSURE (LOW)	CHAMBER PRESSURE	40 PERCENT NPL	REQUIRES THREE CONSECUTIVE SAMPLES BELOW THIS LEVEL*
COMBUSTION CHAMBER PRESSURE (HIGH)	CHAMBER PRESSURE	112 PERCENT NPL	REQUIRES THREE CONSECUTIVE SAMPLES ABOVE THIS LEVEL*
OXIDIZER HEAT EXCHANGER FAILURE	HIGH-PRESSURE OXIDIZER DISCHARGE PRESSURE AND HEAT EXCHANGER OUTLET PRESSURE	$\Delta P > 575$ PSIA	MALFUNCTION DETECTION TO PREVENT HOT GAS BACKFLOW TO VEHICLE
LOSS OF HIGH PRESSURE OXIDIZER TURBOPUMP INTER-MEDIATE SEAL PURGE	PURGE INLET PRESSURE	LESS THAN 40 PSIA	PREVENT COMMUNICATION OF OXIDIZER AND TURBINE HOT GAS

*CONTROLLER SWITCHES TO A HIGH SAMPLE RATE (200 SAMPLES/SECOND) TO VERIFY DATA IN TIMELY FASHION AND TO REDUCE TIME TO SHUTDOWN INITIATION.

TABLE 4.3.3.2-2

ENGINE LIMIT CONTROL SHUTDOWN PARAMETERS

4.3.3.3 Data Transmission

42 The engine status and maintenance data available for transmission to the vehicle via the vehicle/engine data bus are listed in Table 4.3.3.3-1. The data rate in samples per second is indicated for each parameter versus the mission phases. Data are transmitted in bytes consisting of 8 binary bits. The data are arranged in blocks which are transmitted to the vehicle at a maximum rate of 25 times per second. Engine data which are obtained at sample rates higher than 25 samples per second are stored by the controller for transmission in the next block of data. For example, each block of data transmitted at 25 times per second contains 4 samples of each parameter which is sampled at 100 samples per second. The first byte normally transmitted with each block of data is the engine status byte defined by Table 4.3.3.3-2.

The engine status byte identifies to the vehicle:

1. The engine phase of operation (3 bits).
2. The mode within the phase (3 bits).
3. The self test status (2 bits).

2	5	1	Octal number
010	101	01	Binary equivalent

For example, the engine status byte indicates:

1. That the start preparation phase was entered.
2. That the modes were completed to a valid "engine ready".
3. That the engine is satisfactory (no failures).

Since the engine is in the start preparation phase, the data available to the vehicle is that listed under the column "Start Preparation" in Table 4.3.3.3-1. All bytes are arranged in blocks to fit within the 10,000 bit/second limit.

DATA BYTE (8 BITS)	DATA BYTE NO.	MISSION PHASE: GROUND CHECKOUT	START PREPARATION	START	MAINSTAGE	SHUTDOWN	POST SHUTDOWN
		SAMPLES PER SECOND					
ENGINE STATUS	1	25	25	25	25	25	25
MIXTURE RATIO (2 BYTES)	2,3	-	-	5	5	5	-
THRUST (2 BYTES)	4,5	-	-	5	5	5	-
FAILURE IDENTIFICATION	6	25	25	5	5	5	25
PARAMETER VALUE OF DATA BYTE NO. 6	7	25	25	5	5	5	25
LOW PRESSURE FUEL TURBOPUMP							
DISCHARGE PRESSURE	8	-	25	25	25	-	
DISCHARGE TEMPERATURE	9	-	5	5	5	5	-
SHAFT SPEED	10	-	-	5	5	5	-
RADIAL ACCELEROMETER	11	-	-	1	1	1	-
FUEL FLOW RATE (LEAST SIGNIFICANT BYTE)	12	-	100	100	100	100	-
FUEL FLOW RATE (MOST SIGNIFICANT BYTE)	13	-	10	25	25	25	-
HIGH PRESSURE FUEL TURBOPUMP							
DISCHARGE PRESSURE	14	-	25	50	50	50	-
SHAFT SPEED	15	-	-	5	5	5	-
RADIAL ACCELEROMETER	16	-	-	1	1	1	-
FUEL PREBURNER PRESSURE	17	-	-	50	50	50	-
FUEL PREBURNER TEMPERATURE	18	-	-	1	1	1	-
LOW PRESSURE OXIDIZER TURBOPUMP							
DISCHARGE PRESSURE	19	-	25	25	25	25	-
SHAFT SPEED	20	-	-	5	5	5	-
RADIAL ACCELEROMETER	21	-	-	1	1	1	-
HIGH PRESSURE OXIDIZER TURBOPUMP							
DISCHARGE PRESSURE	22	-	25	50	50	50	-
DISCHARGE TEMPERATURE	23	-	5	5	5	5	-
BOOST STAGE DISCHARGE PRESSURE	24	-	-	50	50	50	-
SHAFT SPEED	25	-	-	5	5	5	-
RADIAL ACCELEROMETER	26	-	-	1	1	1	-
LOX FLOW RATE (LEAST SIGNIFICANT BYTE)	27	-	10	100	100	100	-
LOX FLOW RATE (MOST SIGNIFICANT BYTE)	28	-	10	25	25	25	-
OXIDIZER PREBURNER PRESSURE	29	-	-	50	50	50	-
OXIDIZER PREBURNER TEMPERATURE	30	-	-	5	5	5	-
MAIN COMBUSTION CHAMBER FUEL INJECTION PRESSURE	31	-	-	25	25	25	-
MAIN COMBUSTION CHAMBER PRESSURE (2 BYTES)	32,33	-	-	100	100	100	-
MAIN COMBUSTION CHAMBER COOLANT TEMPERATURE	34	-	5	1	1	1	-
OXIDIZER TANK PRESSURANT PRESSURE	35	-	25	50	50	50	-
MAIN COMBUSTION CHAMBER COOLANT PRESSURE	36	-	5	1	1	1	-
HYDRAULIC SYSTEM PRESSURE	37	1	1	1	1	1	1
EXTENDIBLE NOZZLE POSITION	38	1	25	1	1	1	25
MAIN FUEL VALVE POSITION	39	25	10	5	1	5	10
MAIN OXIDIZER VALVE POSITION	40	25	10	50	1	50	10
MAIN COMBUSTION CHAMBER COOLANT VALVE POSITION	41	25	10	1	1	1	1
FUEL PREBURNER OXIDIZER VALVE POSITION	42	25	10	50	50	50	10
OXIDIZER PREBURNER OXIDIZER VALVE POSITION	43	25	10	50	50	50	10
FUEL BLEED POSITION	44	25	10	1	1	1	1
OXIDIZER BLEED POSITION	45	25	10	1	1	1	1
LOX SYSTEM PURGE PRESSURE	46	25	25	-	-	-	-
FUEL SYSTEM PURGE PRESSURE	47	25	25	-	-	-	-
HIGH PRESSURE OXIDIZER TURBOPUMP LIFTOFF SEAL AND BLEED VALVE CONTROL PRESSURE	48	25	25	1	-	-	-
HIGH PRESSURE FUEL TURBOPUMP LIFTOFF SEAL AND BLEED VALVE CONTROL PRESSURE	49	25	25	1	-	-	-
INTERMEDIATE SEAL PURGE PRESSURE	50	1	10	-	1	1	-
TIME REFERENCE	51	-	-	1	1	1	1
FUEL PREBURNER LONGITUDINAL ACCELEROMETER	52	-	-	1	1	1	-
OXIDIZER PREBURNER LONGITUDINAL	53	-	-	1	1	1	-
ACCELEROMETER		-	-	1	1	1	-
MAIN COMBUSTION CHAMBER LONGITUDINAL ACCELEROMETER	54	-	-	1	1	1	-
CONTROLLER INTERNAL PRESSURE	55	1	1	1	1	1	1
CONTROLLER INTERNAL TEMPERATURE	56	1	1	1	1	1	1
SPECIAL DATA (IN RESERVE)	57	-	-	159	213	185	-

TABLE 4.3.3.3-1

DATA TRANSMITTED TO VEHICLE BY ENGINE MISSION PHASE

42 The order of transmission is fixed by the engine operating phase. The total data rates versus phase are listed in Table 4.3.3.3-5. Start and shutdown have the highest data rates.

42 Failure identification information is contained in data bytes 6 and 7 listed in Table 4.3.3.3-1. Table 4.3.3.3-4 is a list of failure modes and line replaceable units which will be identified to the vehicle. Byte number 51 in Table 4.3.3.3-1 is a timing pulse which is issued to correlate events with the time they occur.

4.3.3.4 Controller Built-in Test

42 The controller built-in-test is initiated by the executive program and is performed once in its entirety each time through the 20 msec major cycle of the executive program. It is performed continuously throughout all phases of ground and flight operation. The built-in-test starts with a complete checkout of the dual digital computers. Upon verification of the operability of the computer processors and memories, the checkout is expanded to the digital computer interface electronics, the input electronics, the output electronics, and the power supply electronics. The interrelationship of these major controller sections is shown in Figure 4.3.3.4-1. The input electronics conditions the sensor inputs and converts them to a digital format. The output interface electronics converts the digital computer outputs to the form required to drive output devices such as actuators, valves, and spark igniters. The computer interface electronics provides the interface between the dual digital computers and all computer inputs/outputs. The power supply electronics converts the electrical power supplied by the vehicle to the individual power supply voltages required by the controller.

4.3.3.4.1 Computer Checkout

42 The sample problem tests every instruction in the computer's repertoire and performs a complete test of every section of the processor and memory. Memory parity checking is used to check the data obtained from the memory. A parity bit is generated and stored during each memory write operation so that every word stored in memory including the parity bit has an odd number of ones. Upon each memory read operation, the parity of the word extracted from memory is verified to be odd.

42 The checkout of the memory is concluded by the testing of the parity hardware. Several known constants, some with odd and some with even parity, are fetched from the memory, and the parity checking hardware is monitored to verify that it can distinguish between odd and even parity. The parity-generating hardware is also checked by performing alternate store and fetch instructions using several constants.

42 During ground checkout, an additional test of the fixed portion of the memory is made by a memory sum check. This verifies the integrity of the computer program and assures that each location in fixed memory can be addressed.

4.3.3.4.2 Computer Interface Electronics Checkout

42 Upon successful completion of the total processor and memory checkout, the sample problem initiates the checkout of the computer interface electronics. Each of the dual digital computers has a dedicated set of computer interface electronics to perform the functions of:

PHASE (3 BITS)	
NO.	PHASE
0	NOT USED
1	GROUND CHECKOUT
2	START PREPARATION
3	START
4	MAINSTAGE
5	SHUTDOWN
6	POST-SHUTDOWN
7	(RESERVE)

PHASE	MODE	SELF-TEST STATUS
3 BITS	3 BITS	2 BITS
8 BITS		

SELF-TEST STATUS (2 BITS)	
NO.	ENGINE STATUS
0	NOT USED
1	ENGINE OK
2	COMPONENT FAILED
3	ENGINE LIMIT EXCEEDED

MODE (3 BITS)

NO.	GROUND OPERATIONS		FLIGHT OPERATIONS			
NO.	GROUND CHECKOUT	START PREPARATION	START	MAINSTAGE	SHUTDOWN	POST-SHUTDOWN
0	NOT USED	NOT USED	NOT USED	NOT USED	NOT USED	NOT USED
1	STANDBY OPERATING MODE	PURGE SEQUENCE NO. 1	FUEL ADMITTED	NORMAL CONTROL	THROTTLING TO MPL	STANDBY OPERATING MODE
2	GROUND CHECKOUT IN PROGRESS	PURGE SEQUENCE NO. 2	OXIDIZER ADMITTED	THRUST LIMITED	MPL TO ZERO THRUST	NOZZLE RETRACTED
3	GROUND CHECKOUT COMPLETE	PURGE SEQUENCE NO. 3	CLOSED LOOP THRUST CONTROL	FAIL SAFE MODE	VALVES CLOSED	PROPELLANT DUMP MODE
4	COMPONENT CHECKOUT IN PROGRESS	PURGE SEQUENCE NO. 4	CLOSED LOOP THRUST AND MIXTURE RATIO CONTROL	ENGINE LIMIT	SAFE SHUTDOWN COMPLETE	ABORT TURNAROUND MODE
5	(RESERVE)	ENGINE READY	FAIL SAFE MODE	(RESERVE)	FAIL SAFE SHUTDOWN MODE	(RESERVE)
6	(RESERVE)	(RESERVE)	ENGINE LIMIT	(RESERVE)	EMERGENCY LIMIT SHUTDOWN	(RESERVE)
7	(RESERVE)	(RESERVE)	(RESERVE)	(RESERVE)	(RESERVE)	(RESERVE)

TABLE 4.3.3.3-2

ENGINE STATUS

PHASE	BITS/SECONDS	
	WITHOUT SPECIAL DATA	SPECIAL DATA
GROUND CHECKOUT	2840	
START PREPARATION	4464	
START	8728	1212
MAINSTAGE	8296	1704
SHUTDOWN	8520	1480
POST SHUTDOWN	1176	

TABLE 4.3.3.3-3

TOTAL DATA RATE

1. Controlling flow of digital data to and from the computer, vehicle/engine data bus interface, and memory.
2. Monitoring computer operation to ensure fail safe computer monitoring and fail operational controller operation.
3. Providing a redundant time reference to the digital computer.

The control of digital data flow is through three separate electronic circuits:

1. Direct Memory Access Control - controls the direct data flow in and out of the memory.
2. Input Multiplexer Electronics - provides the data flow path between the controller inputs and the memory.
3. Data Bus Multiplexer Electronics - provides communication with the vehicle data bus.

42 These three circuits are checked out together with end-to-end test. Under direct memory access control, data are transferred via the direct memory access output into the data bus multiplexer electronics data register. This data are then transferred back into memory via the input multiplexer electronics. The input data are then compared to the initial data to verify the correct operation of the direct memory access control, data bus multiplexer electronics and the input multiplexer electronics. The interface between the data bus multiplexer electronics and the vehicle data bus is automatically checked every time a command from the vehicle is received and validated: the controller responds by transmitting engine status.

42 Two watchdog timers are used with each computer to ensure fail safe computer monitoring and fail operational controller operation upon a watchdog timer failure. The watchdog timers verify that each computer is

1. CONTROLLER CHANNEL 1
2. CONTROLLER CHANNEL 2
3. GASEOUS NITROGEN SYSTEM PURGE CONTROL VALVE
4. HELIUM FUEL SYSTEM PURGE CONTROL VALVE
5. EMERGENCY SHUTDOWN CONTROL VALVE CHANNEL 1
6. EMERGENCY SHUTDOWN CONTROL VALVE CHANNEL 2
7. BYPASS CONTROL VALVE
8. LIFTOFF SEAL AND BLEED VALVE CONTROL VALVE (FUEL)
9. LIFTOFF SEAL AND BLEED VALVE CONTROL VALVE (OXIDIZER)
10. INTERMEDIATE SEAL PURGE CONTROL VALVE CHANNEL 1
11. INTERMEDIATE SEAL PURGE CONTROL VALVE CHANNEL 2
12. SYSTEM PURGE PRESSURE SENSOR
13. HELIUM FUEL SYSTEM PURGE PRESSURE SENSOR
14. LIFTOFF SEAL AND BLEED VALVE PRESSURE SENSOR (FUEL)
15. LIFTOFF SEAL AND BLEED VALVE PRESSURE SENSOR (OXIDIZER)
16. INTERMEDIATE SEAL PURGE PRESSURE SENSOR
17. FUEL PREBURNER IGNITER NO. 1
18. FUEL PREBURNER IGNITER NO. 2
19. OXIDIZER PREBURNER IGNITER NO. 1
20. OXIDIZER PREBURNER IGNITER NO. 2
21. MAIN COMBUSTION CHAMBER IGNITER NO. 1
22. MAIN COMBUSTION CHAMBER IGNITER NO. 2
23. MAIN FUEL VALVE ACTUATOR CHANNEL 1
24. MAIN FUEL VALVE ACTUATOR CHANNEL 2
25. MAIN OXIDIZER VALVE ACTUATOR CHANNEL 1
26. MAIN OXIDIZER VALVE ACTUATOR CHANNEL 2
27. COMBUSTION CHAMBER COOLANT VALVE ACTUATOR CHANNEL 1
28. COMBUSTION CHAMBER COOLANT VALVE ACTUATOR CHANNEL 2
29. FUEL PREBURNER OXIDIZER VALVE ACTUATOR CHANNEL 1
30. FUEL PREBURNER OXIDIZER VALVE ACTUATOR CHANNEL 2
31. OXIDIZER PREBURNER OXIDIZER VALVE ACTUATOR CHANNEL 1
32. OXIDIZER PREBURNER OXIDIZER VALVE ACTUATOR CHANNEL 2
33. EXTENDIBLE NOZZLE ACTUATOR CHANNEL 1
34. EXTENDIBLE NOZZLE ACTUATOR CHANNEL 2
35. FUEL BLEED VALVE POSITION SENSOR
36. OXIDIZER BLEED VALVE POSITION SENSOR
37. EXTENDIBLE NOZZLE RETRACTED POSITION SENSOR CHANNEL 1
38. EXTENDIBLE NOZZLE RETRACTED POSITION SENSOR CHANNEL 2
39. EXTENDIBLE NOZZLE EXTENDED POSITION SENSOR CHANNEL 1
40. EXTENDIBLE NOZZLE EXTENDED POSITION SENSOR CHANNEL 2
41. LOW-PRESSURE FUEL TURBOPUMP DISCHARGE PRESSURE/ TEMPERATURE SENSOR NO. 1 CHANNEL 1
42. LOW-PRESSURE FUEL TURBOPUMP DISCHARGE PRESSURE/ TEMPERATURE SENSOR NO. 1 CHANNEL 2
43. LOW-PRESSURE FUEL TURBOPUMP DISCHARGE PRESSURE/ TEMPERATURE SENSOR NO. 2 CHANNEL 1
44. LOW-PRESSURE FUEL TURBOPUMP SHAFT SPEED SENSOR CHANNEL 1
45. LOW-PRESSURE FUEL TURBOPUMP SHAFT SPEED SENSOR CHANNEL 2
46. LOW-PRESSURE FUEL TURBOPUMP RADIAL ACCELEROMETER SENSOR
47. FUEL FLOWRATE SENSOR NO. 1 CHANNEL 1
48. FUEL FLOWRATE SENSOR NO. 1 CHANNEL 2
49. FUEL FLOWRATE SENSOR NO. 2 CHANNEL 1
50. FUEL FLOWRATE SENSOR NO. 2 CHANNEL 2
51. FUEL PREBURNER TEMPERATURE SENSOR NO. 1
52. FUEL PREBURNER TEMPERATURE SENSOR NO. 2
53. FUEL PREBURNER PRESSURE SENSOR CHANNEL 1
54. HIGH-PRESSURE FUEL TURBOPUMP DISCHARGE PRESSURE SENSOR CHANNEL 1
55. HIGH-PRESSURE FUEL TURBOPUMP DISCHARGE PRESSURE SENSOR CHANNEL 2
56. HIGH-PRESSURE FUEL TURBOPUMP SHAFT SPEED SENSOR CHANNEL 1
57. HIGH-PRESSURE FUEL TURBOPUMP SHAFT SPEED SENSOR CHANNEL 2
58. HIGH-PRESSURE FUEL TURBOPUMP RADIAL ACCELEROMETER SENSOR
59. FUEL PREBURNER LONGITUDINAL ACCELERATION SENSOR
60. LOW-PRESSURE OXIDIZER TURBOPUMP DISCHARGE PRESSURE SENSOR CHANNEL 1
61. LOW-PRESSURE OXIDIZED TURBOPUMP DISCHARGE PRESSURE SENSOR CHANNEL 2
62. LOW-PRESSURE OXIDIZER TURBOPUMP SHAFT SPEED SENSOR CHANNEL 1
63. LOW-PRESSURE OXIDIZER TURBOPUMP SHAFT SPEED SENSOR CHANNEL 2
64. LOW-PRESSURE OXIDIZER RADIAL ACCELEROMETER SENSOR
65. HIGH-PRESSURE OXIDIZER TURBOPUMP DISCHARGE PRESSURE/ TEMPERATURE SENSOR NO. 1 CHANNEL 1
66. HIGH-PRESSURE OXIDIZER TURBOPUMP DISCHARGE PRESSURE/ TEMPERATURE SENSOR NO. 1 CHANNEL 2
67. HIGH-PRESSURE OXIDIZER TURBOPUMP DISCHARGE PRESSURE/ TEMPERATURE SENSOR NO. 2 CHANNEL 1
68. HIGH-PRESSURE OXIDIZER TURBOPUMP BOOST STAGE IMPELLER DISCHARGE PRESSURE SENSOR
69. HIGH-PRESSURE OXIDIZER TURBOPUMP RADIAL ACCELEROMETER SENSOR
70. OXIDIZER PREBURNER TEMPERATURE SENSOR NO. 1
71. OXIDIZER PREBURNER TEMPERATURE SENSOR NO. 2
72. OXIDIZER PREBURNER PRESSURE SENSOR NO. 1
73. OXIDIZER PREBURNER LONGITUDINAL ACCELEROMETER SENSOR
74. OXIDIZER FLOWRATE SENSOR NO. 1 CHANNEL 1
75. OXIDIZER FLOWRATE SENSOR NO. 1 CHANNEL 2
76. OXIDIZER FLOWRATE SENSOR NO. 2 CHANNEL 1
77. OXIDIZER FLOWRATE SENSOR NO. 2 CHANNEL 2
78. OXIDIZER TANK REPRESSURIZATION PRESSURE SENSOR CHANNEL 1
79. OXIDIZER TANK REPRESSURIZATION PRESSURE SENSOR CHANNEL 2
80. COMBUSTION CHAMBER FUEL INJECTION PRESSURE SENSOR CHANNEL 1
81. COMBUSTION CHAMBER PRESSURE SENSOR NO. 1 CHANNEL 1
82. COMBUSTION CHAMBER PRESSURE SENSOR NO. 1 CHANNEL 2
83. COMBUSTION CHAMBER PRESSURE SENSOR NO. 2 CHANNEL 1
84. COMBUSTION CHAMBER COOLANT OUTLET TEMPERATURE SENSOR CHANNEL 1
85. COMBUSTION CHAMBER COOLANT OUTLET PRESSURE SENSOR CHANNEL 1
86. HYDRAULIC SYSTEM PRESSURE SENSOR CHANNEL 1
87. HYDRAULIC SYSTEM PRESSURE SENSOR CHANNEL 2
88. CONTROLLER INTERNAL TEMPERATURE SENSOR CHANNEL 1
89. CONTROLLER INTERNAL TEMPERATURE SENSOR CHANNEL 2
90. CONTROLLER INTERNAL PRESSURE SENSOR CHANNEL 1
91. CONTROLLER INTERNAL PRESSURE SENSOR CHANNEL 2
92. OXIDIZER INLET PRESSURE NOT READY
93. OXIDIZER INLET TEMPERATURE NOT READY
94. FUEL INLET PRESSURE NOT READY
95. FUEL INLET TEMPERATURE NOT READY
96. HYDRAULIC SYSTEM PRESSURE NOT READY
97. MAIN FUEL VALVE POSITION NOT READY
98. MAIN OXIDIZER VALVE POSITION NOT READY
99. FUEL PREBURNER OXIDIZER VALVE POSITION NOT READY
100. OXIDIZER PREBURNER OXIDIZER VALVE POSITION NOT READY
101. COMBUSTION CHAMBER COOLANT VALVE POSITION NOT READY
102. FUEL PREBURNER TEMPERATURE OUT OF LIMITS
103. OXIDIZER PREBURNER TEMPERATURE OUT OF LIMITS
104. HIGH-PRESSURE FUEL TURBOPUMP SHAFT SPEED OUT OF LIMITS
105. HIGH-PRESSURE OXIDIZER TURBOPUMP SHAFT SPEED OUT OF LIMITS
106. COMBUSTION CHAMBER PRESSURE OUT OF LIMITS
107. OXIDIZER TANK REPRESSURE OUT OF LIMITS
108. THROUGH 255 ARE SPARES

NOTE: CHANNEL INCLUDES SENSOR, HARNESS, CONNECTORS AND CONTROLLER INPUT ELECTRONICS.

TABLE 4.3.3.3-4

FAILURE IDENTIFICATION

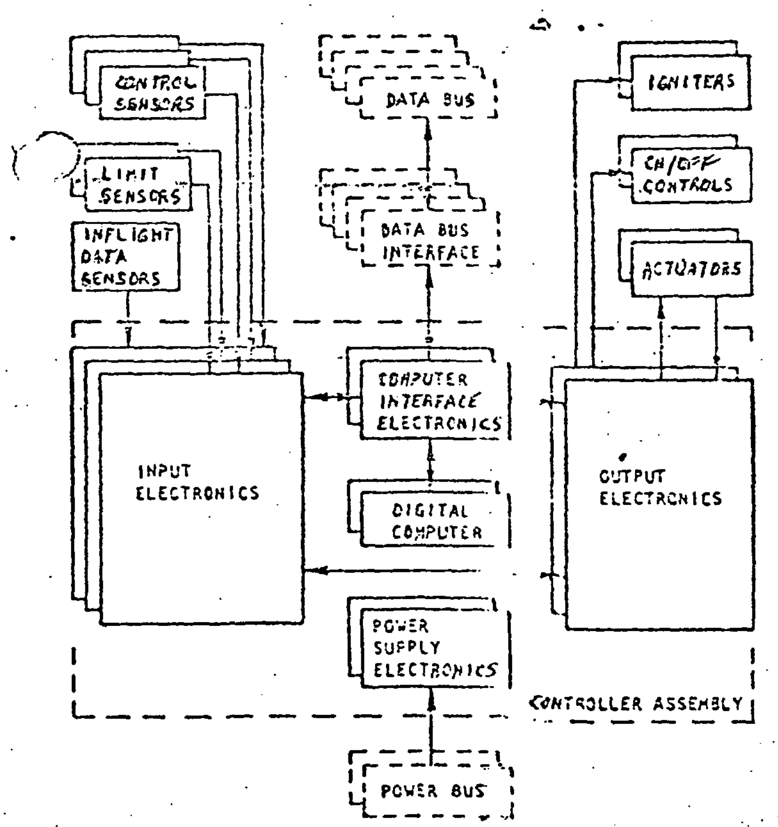


FIGURE 4.3.3.4-1

ENGINE CONTROLLER FUNCTIONAL RELATIONSHIPS progressing through its program and is executing program instructions per a predetermined schedule. The watchdog timer is a simple timing circuit that checks the time it takes the computer to progress between checkpoints in the computer program. Imbedded within the program are instructions requiring the computer to reset each timer, at different points in the program's execution, in a sequential fashion. Each subroutine includes an instruction to reset only one of the watchdog timers, thus ensuring detection of a computer failure that prevents the computer from returning to the executive program on schedule. If either of the computers fails to reset either of its watchdog timers, the watchdog timer output switches to zero.

42 Each computer monitors the watchdog timer output of the other to determine operational status. This status is made available to the vehicle by the controlling computer (initially Channel 1 computer).

42 The output switch, which determines which computer is in control, is controlled by the Channel 1 computer watchdog timer output. Channel 1 computer maintains engine control as long as it remains failure free. Upon failure of Channel 1 computer, its watchdog timer output goes to zero, thus switching engine control to the Channel 2 computer.

42 All processor and memory failure modes cause the watchdog timer output of the corresponding computer to switch to zero. The watchdog timer output is used to command a HALT instruction in a failed computer. The HALT instruction prevents the computer processor from executing any further instructions to ensure that the failed computer does not output actuator commands or data to the vehicle data bus.

42 The proper operation of the watchdog timers is checked during ground checkout. The decay time of the watchdog timers is speeded up by switching in checkout circuitry. The watchdog timer output of each computer is monitored to verify that the watchdog timers will switch to zero if not reset within a specific time interval. All the failure modes of the watchdog timer test circuitry tend to shorten and not legnthen the timer delay times, thus ensuring a shutdown of the corresponding computer upon a failure.

42 Two real time clocks are used with each computer to provide fail-safe computer operation. During each controller test cycle, the two clocks are monitored to verify that they agree. If the clocks do not agree, the malfunction is detected and the corresponding computer channel is commanded to a HALT instruction. The watchdog timers run down causing control to switch if the failure is in Channel 1 computer. If both clocks are stopped but still agree, the failure is detected by the watchdog timer. In this case, the processor will not return to the executive program in time to reset one of the watchdog timers and the timer output will go to zero.

4.3.3.4.3 Input Electronics Checkout

42 Checkout of the input electronics is achieved by the sample problem program. The input electronics processes analog and pulse rate data from the engine-mounted sensors. These data are converted into a digital format for inputting to the digital computer memory via the computer interface electronics. The input electronics consists of low-level and high-level multiplexer gates, amplifiers, demodulators, pulse rate converters and analog to digital converters.

42 Internal reference voltages are applied at specific low- and high-level multiplexer input gates. These voltages have been incorporated to aid in checkout. The reference signals are monitored at the output of the analog to digital converters by the computers to verify operation of amplifiers and the analog to digital converters. These reference voltages are also used by the computers to measure bias and offsets in the input channels. This information is used by the computers to provide a continuous recalibration of the input channels by automatic software changes. The remaining portion of the input electronics checkout is accomplished during sensor input monitoring and processing discussed in 4.3.3.5.

4.3.3.4.4 Output Electronics Checkout

42 The output electronics converts the digital computer commands to voltages for controlling the valve actuators, solenoid valves, and igniters. All of the controller outputs are continuously verified by the computers. The controller output voltages are fed back to the input electronics and are processed through the multiplexer gates and analog to digital converters, in the same fashion as the sensor inputs, and are stored in memory. The controller output data are then compared to the commanded data and verified to be correct. This test provides a complete checkout of the output electronics. Upon detection of a malfunction of the controlling output channel, engine control is switched to the second output channel.

4.3.3.4.5 Power Supply Electronics Checkout

42 Built-in test circuits continuously monitor the controller supply electronics. Five internal controller power supplies, each capable of operating from either vehicle power bus, provide internal controller power distribution.

The five power supplies provide power to the triple redundant input electronics, to the dual redundant digital computers/interface electronics, and to the dual redundant output electronics.

42 Each power supply includes circuitry in the regulators that monitor regulator performance and turn off the power supply upon detection of a malfunction. The operational status of all power supplies is continuously monitored as part of the controller self-test. The completion of the power supply electronics checkout completes the controller built-in test. The engine controller has been completely checked out during the controller built-in test with the exception of the sensor input multiplex gates which are checked as part of the sensor built-in test.

4.3.3.5 <u>Sensor Built-in Test</u>

42 Computer Channels 1 and 2 both conduct all tests. Computer Channel 2 conducts all tests, storing results in memory in case computer Channel 1 fails requiring computer Channel 2 to assume control.

4.3.3.5.1 <u>Sensor Ground Checkout</u>

42 All sensors except nonflight data sensors are functionally tested during the automatic ground checkout.

4.3.3.5.2 <u>Sensor Inflight Monitoring</u>

42 Three levels of sensor redundancy are used in the engine system:

1. Triple redundant sensors are used for performance control.
2. Dual redundant sensors are used for engine ready checks and limit control.
3. Nonredundant sensors are used for acquisition of maintenance data for transmission to the vehicle and recording.

42 Each of the triple redundant sensors used in performance control are initially checked for reasonableness by comparing their outputs with minimum and maximum limits. The sensors are designed so that a failure will cause the sensor output to fall outside of the reasonableness limits. The reasonableness limits for the seven performance control parameters are given in Table 4.3.3.5.2-1. As an illustration of a reasonableness test during mainstage engine operation, if the value of main combustion chamber pressure is not between 1280 psia (848.1 N/cm^2) and 3,450 psia (2378.7 N/cm^2), a reasonableness failure of the sensor is indicated.

42 After the reasonableness test, the triple redundant performance control signals are compared to each other. If all three channels have passed the reasonableness check and agree within a comparison failure limit stored in the computer memory, the average value of the three sensor channels is used to update the sensor information stored in memory.

42 Comparison failure limits for the seven performance control sensors are given in Table 4.3.3.5.2-1. The comparison limit that is used to detect a failed main combustion chamber pressure sensor is a difference between sensor signals of 4.0 percent of full scale. The engine control will still meet operational repeatability requirements if the differences are within this limit. The normal 3σ deviation that can be expected between good sensor channels is approximately 1 percent of full scale. Therefore, the probability of having nuisance failures using the 4 percent of full-scale failure limit for the main combustion chamber pressure sensor channels is negligible.

42 If only two of the three sensor channels pass the above tests, the average value of the two agreeing channels is formed and this value is used to update the memory. If the same sensor channel does not pass the above tests three times in succession, it is assumed that the sensor channel has failed and that channel is not used in future processing. An alternate sensor value (computed from other sensed parameters, Table 4.3.3.5.2-2) is used to replace the failed sensor channel.

42 The performance control operates with two of the original sensors and an alternate computed sensor value as before. The sensor tests are continued with the new set of triple redundant sensor value as before. The sensor tests are continued with the new set of triple redundant sensors. Upon a second failure, the performance control continues to operate using the value of the good sensor and the alternate computed value. Thus a fail operational capability is provided for second failures of performance control sensors without an increase in sensor count.

42 A third sensor channel failure is detected either by the reasonableness tests or comparison failure limits test. A third failure results in computer Channel 1 being shut down. If the failure is in the sensors and not the signal conditioning electronics, computer Channel 2 will also have detected the failures and shut down. If the failure were in the signal conditioning electronics dedicated to Channel 1 computer, the Channel 2 computer will continue to operate and control the engine.

42 The dual redundant sensors used for engine ready and the engine limit control are checked for reasonableness by comparing their outputs with minimum and maximum limits similarly to the performance control sensors. If either

sensor channel indicates that the engine is within operating limits, the engine is safe and is allowed to operate normally. If both outputs from a set of dual redundant sensors indicate that a limit has been exceeded in three successive samples, an engine failure is assumed and the engine is shutdown.

42 The nonredundant sensors used only for acquiring maintenance data are not tested in flight operation. These sensors are noncritical and do not affect control functions.

4.3.3.6 Actuator/Valve Built-in Test

42 The controller performs failure monitoring of the actuators and valves by use of actuator/valve position sensors. The modulating-type hydraulic actuators contain dual redundant servovalves to provide a fail operational/fail safe capability. A penumatic shutdown system provides actuator fail safety in case of loss of hydraulic power or loss of electronic control to the actuator. Pressure ladder-sequenced penumatic actuators and motors provide this shutdown capability to the valves and retraction capability to the orbiter nozzle. The controller contains provisions to detect failures and to switch out failed elements.

42 Monitoring of the actuators is accomplished by use of an analog electronic servovalve model. The model receives the same commands as the servovalve and electrically models the normal servovalve output spool position. The outputs of the servovalve and the electronic model are compared. If a servovalve failure is detected, the failed servovalve is isolated from the actuator and a redundant standby servovalve is switched in to control the actuator. Monitoring of the standby servovalve is accomplished in a similar manner.

CONTROL PARAMETER	REASONABLENESS FAILURE LIMITS* MINIMUM	MAXIMUM	COMPARISON FAILURE LIMITS
MAIN COMBUSTION CHAMBER PRESSURE	1230 PSIA	3450 PSIA	4 PERCENT FULL SCALE
HIGH-PRESSURE OXIDIZER PUMP			
DISCHARGE PRESSURE	1750 PSIA	6140 PSIA	4 PERCENT FULL SCALE
DISCHARGE TEMPERATURE	170 R	200 R	1.5 R
FLOWRATE	580 LB/SEC	1575 LB/SEC	0.1 PERCENT**
LOW-PRESSURE FUEL PUMP			
DISCHARGE PRESSURE	45 PSIA	290 PSIA	4 PERCENT FULL SCALE
DISCHARGE TEMPERATURE	36 R	50 R	0.5 R
FLOWRATE	85 LB/SEC	250 LB/SEC	0.1 PERCENT**

*APPLICABLE IN 40 PERCENT TO 112 PERCENT OF NPL RANGE OF THRUST

**COMPARISON BETWEEN DIGITIZED PULSE RATE SIGNALS

TABLE 4.3.3.5.2-1

PERFORMANCE CONTROL SENSOR-REASONABLENESS AND COMPARISON FAILURE LIMITS

PERFORMANCE CONTROL PARAMETER	ALTERNATE SENSOR VALUE COMPUTED FROM
MAIN COMBUSTION CHAMBER PRESSURE	MIXTURE RATIO AND TOTAL FUEL AND OXIDIZER FLOWRATES
LOW-PRESSURE FUEL TURBOPUMP DISCHARGE PRESSURE	LOW-PRESSURE FUEL TURBOPUMP SPEED AND FUEL FLOWRATE
LOW-PRESSURE FUEL TURBOPUMP DISCHARGE TEMPERATURE	CONSTANT
LOW-PRESSURE FUEL TURBOPUMP FLOWRATE	MAIN COMBUSTION CHAMBER PRESSURE AND OXIDIZER FLOWRATE
HIGH-PRESSURE OXIDIZER TURBOPUMP FLOWRATE	MAIN COMBUSTION CHAMBER PRESSURE AND FUEL FLOWRATE
HIGH-PRESSURE OXIDIZER TURBOPUMP DISCHARGE PRESSURE	LOW-PRESSURE OXIDIZER TURBOPUMP DISCHARGE PRESSURE AND HIGH-PRESSURE OXIDIZER TURBOPUMP SPEED AND MAIN OXIDIZER FLOWRATE
HIGH-PRESSURE OXIDIZER TURBOPUMP DISCHARGE TEMPERATURE	CONSTANT

TABLE 4.3.3.5.2-2

ALTERNATE PERFORMANCE CONTROL SENSORS

42 The controller also compares the actuator position feedback signals with the actuator position command to detect actuator failures not detected by monitoring of the servovalves. Upon a failure of both servovalve channels, or a failure detected by monitoring actuator position, the penumatic shutdown assumes control. This monitoring procedure is used during both ground and flight operation.

42 Built-in test of the penumatic purge valves is provided by measurements of penumatic pressure. Indicated failures are isolated by the electrical stimulation test of the pressure sensor to distinguish between sensor failures and valve failures.

4.3.3.7 Spark Igniter Built-in Test

42 Each igniter circuit provides an output which is monitored for spark rate and voltage level. This allows verification of igniter operation during both ground checkout and flight operation.

4.3.4 Instrumentation Parameter List

42 The engine is equipped with sensors to provide data acquisition during engine flight and non-flight operations. Dual sensing elements are provided for pressure, cryogenic temperature, flow and pump speed pickups.

42 Parameters for data acquisition are listed in Table 4.3.4-1. This list was developed from studies covering:

1. The dynamic simulation for selecting the control system.
2. Verification of the operational readiness condition of the engine through checkout.
3. Requirements for engine readiness prior to start.
4. Establishing performance and trend analysis of the engine and subsystems.

The table shows the usage of sensor outputs for achieving the following functions:

1. Performance Control.
2. Limit Control.
3. Position Indication.
4. Engine Ready.
5. In-flight Maintenance Recording.
6. Non-flight Data Acquisition.

The non-flight data are provided at a connector panel attached to the controller.

4.3.4.1 Alternate Performance Control Parameters

42 Alternate performance control sensors are listed in Table 4.3.4.4-1. Engine performance is maintained within specification requirements with the use of these alternate values after the first and second sensor output failure.

SUBSYSTEM AND PARAMETER	REQUIREMENTS	SENSOR QUANTITY AND TYPE	SENSOR OUTPUTS	FUNCTIONAL USE: PERFORMANCE CONTROL	LIMIT CONTROL	ENGINE READY	POSITION INDICATION	MAINTENANCE RECORDING	ENGINE CHECKOUT	NON-FLIGHT DATA
LOW-PRESSURE FUEL TURBOPUMP										
DISCHARGE PRESSURE	RANGE, 0 TO 400 PSIA; ACCURACY ±2 PERCENT FULL SCALE; RESPONSE, 0 TO 100 HZ; PROCUREMENT SPECIFICATION, RC7003	2; DC; INTEGRAL	A_1; B_1; A_2; B_2	A_1; B_1; A_2	--	--	--	$\frac{A_1+B_1+A_2}{3}$	--	B_2
DISCHARGE TEMPERATURE	RANGE, -423 TO +700 F; ACCURACY, ±2 PERCENT OPERATING RANGE; RESPONSE, 0.2 SECOND TIME CONSTANT; PROCUREMENT SPECIFICATION, RC7003	2; DC; INTEGRAL	A_1; B_1; A_2; B_2	A_1; B_1; A_2	--	--	--	$\frac{A_1+B_1+A_2}{3}$	--	--
SHAFT SPEED	RANGE, 0 TO 20,000 RPM; ACCURACY, ±1 RPM; PROCUREMENT SPECIFICATION, RC7005	1	A; B	--	--	--	--	A OR B	--	--
RADIAL ACCELERATION	RANGE, 10 TO 6000 HZ, 0 TO 300 GRMS; ACCURACY, ±0.2 MV/G; PROCUREMENT SPECIFICATION, RC7006	1	A	--	--	--	--	A	--	--
FLOWRATE	RANGE, 0 TO 18,000 GPM; ACCURACY, ±20 GPM; RESPONSE, 300 RAD/SEC; PROCUREMENT SPECIFICATION, RC7005	2; PU	A_1; B_1; A_2; B_2	A_1; B_1; A_2	--	--	--	A_1 OR B_1 OR A_2	--	--
HIGH-PRESSURE FUEL TURBOPUMP AND PREBURNER										
DISCHARGE PRESSURE	RANGE, 0 TO 7500 PSIA; ACCURACY, ±2 PERCENT FULL SCALE; RESPONSE, 0 TO 100 HZ; PROCUREMENT SPECIFICATION, RC7001	1; DC	A; B	--	--	--	--	A	--	B
PREBURNER PRESSURE	RANGE, 0 TO 6000 PSIA; ACCURACY, ±2 PERCENT FULL SCALE; RESPONSE, 0 TO 100 HZ; PROCUREMENT SPECIFICATION, RC7001	1; DC	A; B	--	--	--	--	A	--	B
PREBURNER TEMPERATURE	RANGE, 0 TO 2300 F; ACCURACY, ±2 PERCENT OPERATING RANGE; RESPONSE, 0.5 SECOND TIME CONSTANT; PROCUREMENT SPECIFICATION, RC7004	2	A_1; A_2	--	A_1; A_2	--	--	$\frac{A_1+A_2}{2}$	--	--
SHAFT SPEED	RANGE, 0 TO 45,000 RPM; ACCURACY, ±1 RPM; PROCUREMENT SPECIFICATION, RC7005	1; DC	A; B	--	A; B	--	--	A OR B	--	--

*DC = DUAL CONNECTORS; PU = PICKUP; A, B = SENSOR OUTPUTS; SUBSCRIPTS = SENSOR NO. 1 AND SENSOR NO. 2 OF REDUNDANT SETS

TABLE 4.3.4-1

INSTRUMENTATION PARAMETER LISTS

SUBSYSTEM AND PARAMETER	REQUIREMENTS	SENSOR QUANTITY AND TYPE	SENSOR OUTPUTS	FUNCTIONAL USE: PERFORMANCE CONTROL	LIMIT CONTROL	ENGINE READY	POSITION INDICATION	MAINTENANCE RECORDING	ENGINE CHECKOUT	NON-FLIGHT DATA
RADIAL ACCELERATION	RANGE, 0 TO 6000 HZ, 0 TO 300 GRMS; ACCURACY ±0.2 MV/G; PROCUREMENT SPECIFICATION, RC7006	1	A	--	--	--	--	A	--	--
PREBURNER LONGITUDINAL ACCELERATION	RANGE, 100 TO 15,000 HZ, 0 TO 500 GRMS; ACCURACY, ±0.2 MV/G; PROCUREMENT SPECIFICATION, RC7006	2	A_1; A_2	--	--	--	--	A_1	--	A_2
LOW-PRESSURE OXIDIZER TURBOPUMP										
DISCHARGE PRESSURE	RANGE, 0 TO 600 PSIA; ACCURACY ±2 PERCENT FULL SCALE; RESPONSE, 0 TO 100 HZ; PROCUREMENT SPECIFICATION RC7001	2; DC	A_1; B_1 A_2; B_2	--	--	A_1; A_2	--	$\frac{A_1+A_2}{2}$	--	B_2
SHAFT SPEED	RANGE, 0 TO 8000 RPM; ACCURACY ±1 RPM; PROCUREMENT SPECIFICATION, RC7005	1	A; B	--	--	--	--	A OR B	--	--
RADIAL ACCELERATION	RANGE, 10 TO 6000 HZ, 0 TO 300 GRMS; ACCURACY, ±0.2 MV/G; PROCUREMENT SPECIFICATION, RC7006	1	A	--	--	--	--	A	--	--
HIGH-PRESSURE OXIDIZER TURBOPUMP AND PREBURNER										
DISCHARGE PRESSURE	RANGE, 0 TO 6000 PSIA; ACCURACY ±2 PERCENT FULL SCALE; RESPONSE, 0 TO 100 HZ; PROCUREMENT SPECIFICATION, RC7003	2; DC INTEGRAL	A_1; B_1; A_2; B_2	A_1; B_1; A_2	--	--	--	$\frac{A_1+B_1+A_2}{3}$	--	B_2
DISCHARGE TEMPERATURE	RANGE, -423 TO +700 F; ACCURACY ±2 PERCENT OPERATING RANGE; RESPONSE, 0.2 SECOND TIME CONSTANT; PROCUREMENT SPECIFICATION, RC7003	2; DC INTEGRAL	A_1; B_1; A_2; B_2	A_1; B_1; A_2	--	A_1; A_2	--	$\frac{A_1+B_1+A_2}{3}$	--	--
BOOST STAGE IMPELLER DISCHARGE PRESSURE	RANGE, 0 TO 8000 PSIA; ACCURACY ±2 PERCENT FULL SCALE; RESPONSE, 0 TO 100 HZ, PROCUREMENT SPECIFICATION, RC7001	1; DC	A; B	--	--	--	--	A	--	B
PREBURNER PRESSURE	RANGE, 0 TO 6000 PSIA; ACCURACY ±2 PERCENT FULL SCALE; RESPONSE 0 TO 100 HZ; PROCUREMENT SPECIFICATION, RC7001	1; DC	A; B	--	--	--	--	A	--	B

TABLE 4.3.4-1

INSTRUMENTATION PARAMETER LISTS (Continued)

SUBSYSTEM AND PARAMETER	REQUIREMENTS	SENSOR QUANTITY AND TYPE	SENSOR OUTPUTS	FUNCTIONAL USE: PERFORMANCE CONTROL	LIMIT CONTROL	ENGINE READY	POSITION INDICATION	MAINTENANCE RECORDING	ENGINE CHECKOUT	NON-FLIGHT DATA
PREBURNER TEMPERATURE	RANGE, 0 TO 2300 F; ACCURACY, ±2 PERCENT OPERATING RANGE; RESPONSE, 0.5 SECOND TIME CONSTANT; PROCUREMENT SPECIFICATION, RC7004	2	A_1;A_2	--	A_1;A_2	--	--	$\frac{A_1+A_2}{2}$	--	--
SHAFT SPEED	RANGE, 0 TO 35,000 RPM; ACCURACY, ±1 RPM; PROCUREMENT SPECIFICATION, RC7005	1; DC	A; B	--	A; B	--	--	A OR B	--	--
RADIAL ACCELERATION	RANGE, 10 TO 6000 HZ; 0 TO 300 GRMS; ACCURACY, ±0.2 MV/G; PROCUREMENT SPECIFICATION, RC7006	1	A	--	--	--	--	A	--	--
PREBURNER LONGITUDINAL ACCELERATION	RANGE, 100 TO 15,000 HZ, 0 TO 500 GRMS; ACCURACY, ±0.2 MV/G; PROCUREMENT SPECIFICATION, RC7006	2	A_1;A_2	--	--	--	--	A_1	--	A_2
OXIDIZER TANK PRESSURANT PRESSURE	RANGE, 0 TO 6000 PSIA; ACCURACY ±2 PERCENT FULL SCALE; RESPONSE, 0 TO 100 HZ; PROCUREMENT SPECIFICATION, RC7001	1; DC	A; B	--	A & B	--	--	$\frac{A+B}{2}$	--	--
FLOWRATE	RANGE, 0 TO 6300 GPM; ACCURACY, ±5 GPM; RESPONSE, 300 RAD/SEC; PROCUREMENT SPECIFICATION, RC7005	2; PU	A_1;B_1; A_2;B_2	A_1;B_1; A_2	--	--	--	A_1 OR B_1 OR A_2	--	--
MAIN COMBUSTION CHAMBER										
COMBUSTION CHAMBER PRESSURE	RANGE, 0 TO 3500 PSIA; ACCURACY, ±2 PERCENT FULL SCALE; RESPONSE, 0 TO 100 HZ; PROCUREMENT SPECIFICATION, RC7001	2; DC	A_1;B_1; A_2;B_2	A_1;B_1; A_2	--	--	--	$\frac{A_1+B_1+A_2}{3}$	--	B_2
FUEL INJECTION PRESSURE	RANGE, 0 TO 4000 PSIA; ACCURACY ±2 PERCENT FULL SCALE; RESPONSE, 0 TO 100 HZ; PROCUREMENT SPECIFICATION, RC7001	1: DC	A; B	--	--	--	--	A	--	--
COOLANT OUTLET PRESSURE	RANGE, 0 TO 5000 PSIA; ACCURACY, ±2 PERCENT FULL SCALE; RESPONSE 0 TO 100 HZ; PROCUREMENT SPECIFICATION, RC7001	1; DC INTEGRAL	A; B	--	--	--	--	A	--	--
COOLANT OUTLET TEMPERATURE	RANGE, -423 TO+700 F; ACCURACY, ±2 PERCENT OPERATING RANGE; RESPONSE, 0.2 SECOND TIME CONSTANT; PROCUREMENT SPECIFICATION, RC7002	1; DC INTEGRAL	A; B	--	--	--	--	A	--	--

TABLE 4.3.4-1

INSTRUMENTATION PARAMETER LISTS (Continued)

SUBSYSTEM AND PARAMETER	REQUIREMENTS	SENSOR QUANTITY AND TYPE	SENSOR OUTPUTS	FUNCTIONAL USE: PERFORMANCE CONTROL	LIMIT CONTROL	ENGINE READY	POSITION INDICATION	MAINTENANCE RECORDING	ENGINE CHECKOUT	NON-FLIGHT DATA
LONGITUDINAL ACCELERATION	RANGE, 100 TO 15,000 HZ, 0 TO 500 GRMS; ACCURACY, ±0.2 MV/G; PROCUREMENT SPECIFICATION, RC7006	2	A_1;A_2	--	--	--	--	A_1	--	A_2
HYDRAULIC										
HYDRAULIC SYSTEM PRESSURE	RANGE, 0 TO 4000 PSIA; ACCURACY, ±2 PERCENT FULL SCALE; RESPONSE, 0 TO 100 HZ; PROCUREMENT SPECIFICATION, RC7001	1; DC	A; B	--	--	A; B	--	$\frac{A + B}{2}$	A; B	--
FLOW CONTROL VALVES POSITIONS										
MAIN FUEL VALVE	RANGE, 0 TO 90 DEGREES, ACCURACY, ±1 PERCENT	2	A_1;A_2	--	--	A_1;A_2	A_1;A_2	A_1 OR A_2	A_1;A_2	--
MAIN OXIDIZER VALVE										
SERVOVALVE		2	A_1;A_2	--	--	--	A_1;A_2	--	--	--
ACTUATOR	RANGE, 0 TO 90 DEGREES	2	A_1;A_2	--	--	A_1;A_2	A_1;A_2	A_1;A_2	A_1;A_2	--
COOLANT VALVE										
SERVO VALVE		2	A_1;A_2	--	--	--	A_1;A_2	--	--	--
ACTUATOR	RANGE, 0 TO 90 DEGREES	2	A_1;A_2	--	--	A_1;A_2	A_1;A_2	A_1 OR A_2	A_1;A_2	--
FUEL PREBURNER OXIDIZER										
SERVO VALVE		2	A_1;A_2	--	--	--	A_1;A_2	--	--	--
ACTUATOR	RANGE, 0 TO 90 DEGREES	2	A_1;A_2	--	--	A_1;A_2	A_1;A_2	A_1 OR A_2	A_1;A_2	--
OXIDIZER PREBURNER OXIDIZER										
SERVO VALVE		2	A_1;A_2	--	--	--	A_1;A_2	--	--	--
ACTUATOR	RANGE, 0 TO 90 DEGREES	2	A_1;A_2	--	--	A_1;A_2	A_1;A_2	A_1;A_2	A_1;A_2	--
FUEL BLEED VALVE	RANGE, 0 TO 0.133 INCH	1	A	--	--	--	--	A	A	--
OXIDIZER BLEED VALVE	RANGE, 0 TO 0.133 INCH	1	A	--	--	--	--	A	A	--

TABLE 4.3.4-1

INSTRUMENTATION PARAMETER LISTS (Continued)

SUBSYSTEM AND PARAMETER	REQUIREMENTS	SENSOR QUANTITY AND TYPE	SENSOR OUTPUTS	FUNCTIONAL USE: PERFORMANCE CONTROL	LIMIT CONTROL	ENGINE READY	POSITION INDICATION	MAINTENANCE RECORDING	ENGINE CHECKOUT	NON-FLIGHT DATA
PNEUMATIC CONTROL SYSTEM										
GN_2 PURGE PRESSURE	RANGE, 0 TO 1000 PSIA; ACCURACY ±2 PERCENT FULL SCALE; RESPONSE, 0 TO 100 HZ PROCUREMENT SPECIFICATION, RC7001	1; DC	A; B	--	--	--	--	A	A	--
THRUST CHAMBER FUEL JACKET PURGE PRESSURE	RANGE, 0 TO 1000 PSIA; ACCURACY, ±2 PERCENT FULL SCALE; RESPONSE 0 TO 100 HZ; PROCUREMENT SPECIFICATION, RC7001	1; DC	A; B	--	--	--	--	A	A	--
HIGH-PRESSURE OXIDIZER TURBOPUMP LIFTOFF SEAL AND OXIDIZER BLEED VALVE PRESSURE	RANGE, 0 TO 1000 PSIA; ACCURACY, ±2 PERCENT FULL SCALE; RESPONSE, 0 TO 100 HZ; PROCUREMENT SPECIFICATION, RC7001	1; DC	A; B	--	--	--	--	A	A	--
HIGH-PRESSURE FUEL TURBOPUMP LIFT-OFF SEAL AND FUEL BLEED VALVE PRESSURE	RANGE, 0 TO 1000 PSIA; ACCURACY, ±2 PERCENT FULL SCALE; RESPONSE, 0 TO 100 HZ; PROCUREMENT SPECIFICATION, RC7001	1; DC	A;B	--	--	--	--	A	A	--
HIGH-PRESSURE OXIDIZER TURBOPUMP INTERMEDIATE SEAL PRESSURE	RANGE, 0 TO 1000 PSIA; ACCURACY, ±2 PERCENT FULL SCALE; RESPONSE, 0 TO 100 HZ; PROCUREMENT SPECIFICATION, RC7001	1; DC	A; B	--	A; B	--	--	$\frac{A+B}{2}$	A; B	--
CONTROLLER										
INTERNAL PRESSURE	RANGE, 0 TO 50 PSIA; ACCURACY, ±2 PERCENT FULL SCALE; RESPONSE 0 TO 100 HZ; PROCUREMENT SPECIFICATION RC7001	1; DC	A; B	--	--	--	--	A OR B	A; B	--
INTERNAL TEMPERATURE	RANGE, -320 TO +300 F; ACCURACY, ±2 PERCENT OPERATING RANGE; RESPONSE, 1 SECOND TIME CONSTANT; PROCUREMENT SPECIFICATION, NAS-27315	2	A_1;A_2	--	--	--	--	$\frac{A_1+A_2}{2}$	--	--
EXTENDIBLE NOZZLE										
SERVOVALVE POSITION		2	A_1;A_2	--	--	--	A_1;A_2	--	--	--

TABLE 4.3.4-1

INSTRUMENTATION PARAMETER LISTS (Continued)

SUBSYSTEM AND PARAMETER	REQUIREMENTS	SENSOR QUANTITY AND TYPE	SENSOR OUTPUTS	FUNCTIONAL USE						
				PERFORMANCE CONTROL	LIMIT CONTROL	ENGINE READY	POSITION INDICATION	MAINTENANCE RECORDING	ENGINE CHECKOUT	NON-FLIGHT DATA
ACTUATOR POSITION		2	$A_1;A_2$	--	--	--	$A_1;A_2$	A_1 OR A_2	$A_1;A_2$	--
NOZZLE RETRACTED		2	$A_1;A_2$	--	--	--	--	A_1 OR A_2	$A_1;A_2$	--
NOZZLE EXTENDED		2	$A_1;A_2$	--	--	$A_1;A_2$	--	A_1 OR A_2	$A_1;A_2$	--

TABLE 4.3.4-1

INSTRUMENTATION PARAMETER LISTS (Continued)

PERFORMANCE CONTROL PARAMETER	REDUNDANT SENSORS	ALTERNATE VALUE COMPUTED FROM
MAIN COMBUSTION CHAMBER PRESSURE	2	MIXTURE RATIO AND TOTAL FUEL AND OXIDIZER FLOWRATES
LOW-PRESSURE FUEL TURBOPUMP DISCHARGE PRESSURE	2	LOW-PRESSURE FUEL TURBOPUMP SPEED FUEL FLOWRATES
	2	NONE
LOW-PRESSURE FUEL TURBOPUMP FLOWRATE	2 MAGNETIC PICKUPS	MAIN COMBUSTION CHAMBER PRESSURE AND OXIDIZER FLOWRATE
LOW-PRESSURE FUEL TURBOPUMP DISCHARGE TEMPERATURE	1 TURBINE FLOWMETER	
HIGH-PRESSURE OXIDIZER TURBOPUMP FLOWRATE	2 MAGNETIC PICKUPS 1 TURBINE FLOWMETER	MAIN COMBUSTION CHAMBER PRESSURE AND FUEL FLOWRATE
HIGH-PRESSURE OXIDIZER TURBOPUMP DISCHARGE PRESSURE	2	LOW-PRESSURE OXIDIZER TURBOPUMP DISCHARGE PRESSURE; HIGH-PRESSURE OXIDIZER TURBO-PUMP SPEED MAIN OXIDIZER FLOWRATE
HIGH-PRESSURE OXIDIZER TURBOPUMP DISCHARGE TEMPERATURE	2	NONE

TABLE 4.3.4.1-1

ALTERNATE PERFORMANCE CONTROL SENSORS

4.3.5 Engine Actuator

176 The gimballing of the main engine is accomplished by an electro-mechanical servo-actuator. The redundancy of components give the actuator fail operation capability after first failure and fail-safe after the second failure. In the fail safe mode, the actuator returns to neutral. With the triple redundant feature of the hydraulic servo valves, the actuator can accept three separate independent input signals. The servo actuator has a closed loop hydro-mechanical feedback loop. The developed force, direction and rate of position change of the actuator piston are determined by the electro-hydraulic servo valve.

176 The input hydraulic pressure is 3,000 psig, and the return pressure is nominally 45 psig. Using 3,000 psig fluid at 38 + 5 - 20° C, the servo actuator has a piston rod maximum velocity of 5.25 inches per second with a load of 60,000 pounds.

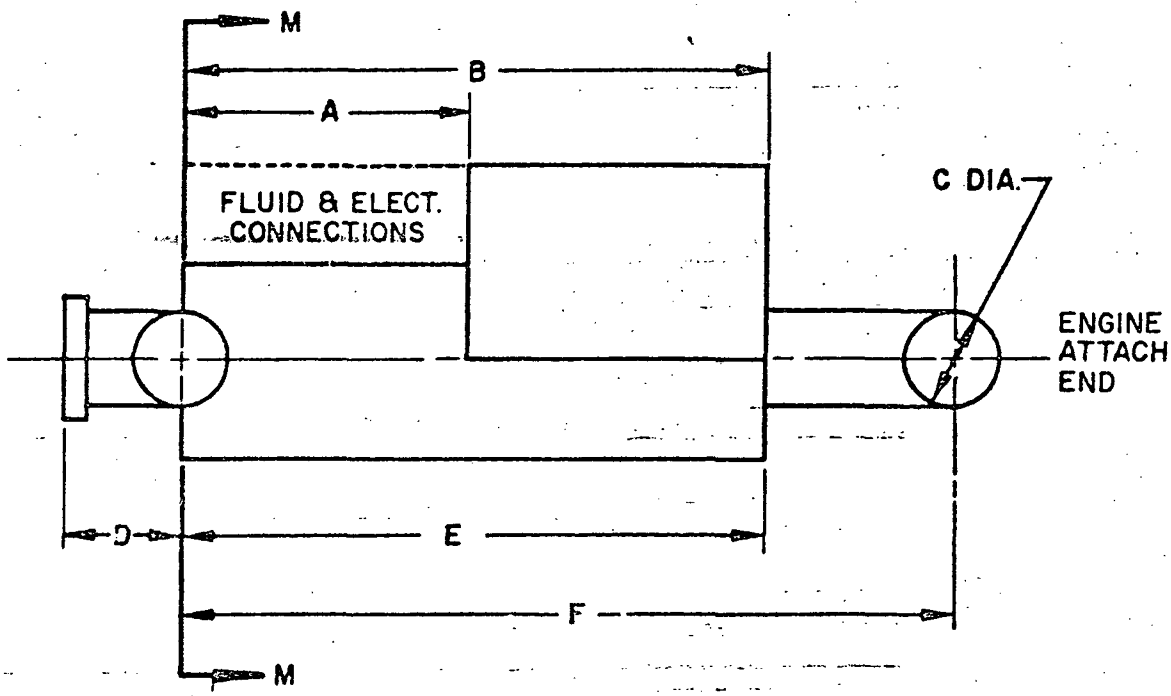
176 The servo has a linear accuracy of 1 ma thru full stroke with a hysteresis of less than 1.5 ma. Threshold signal for piston movement is less than 0.5 ma. With no electrical signal to the servo and with 3,000 psig hydraulic supply, the position of the piston shall be within 0.055 in of the reference midstroke position.

4.3.5.1 Operating Characteristics

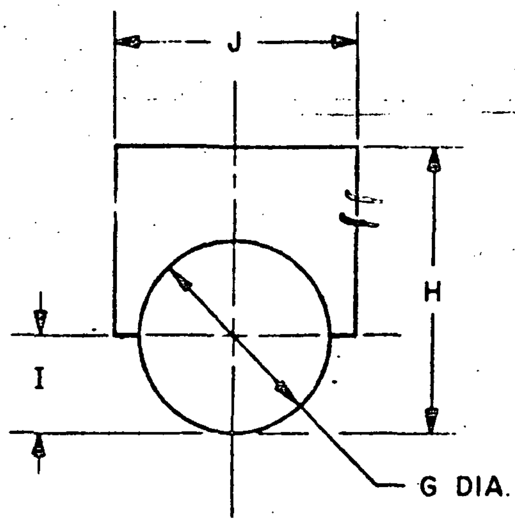
176 The servo system has the following general characteristics:

Frequency response - The system frequency response of amplitude ratio and phase shift is within the limits shown on Figure 4.3.5-3 and Figure 4.3.5-4. This is an engine position to current input ratio driven by a current source. The nominal phase shift at 1.0 Hz shall be 17°. This applies to inertia loads only.

ACTUATOR ENVELOPE-MAXIMUM DIMENSIONS



NOTE:
1. ACTUATOR SHOWN IN NULL POSITION
2. ALL DIMENSIONS IN INCHES.



SECTION M-M

DIMENSIONS	
A	18.5
B	35.5
C	3.0
D	4.5
E	35.5
F	43.0
G	8.0
H	14.0
I	4.0
J	9.0

FIGURE 4.3.5-1

176 Transient response - The response of the load or engine position to input signals to the servoactuator as a function of time falls within the limits of Figure 4.3.5-5. There are no continuing oscillations, limit cycling, or hunting in the steady state condition. This requirement applies with inertia loads only.

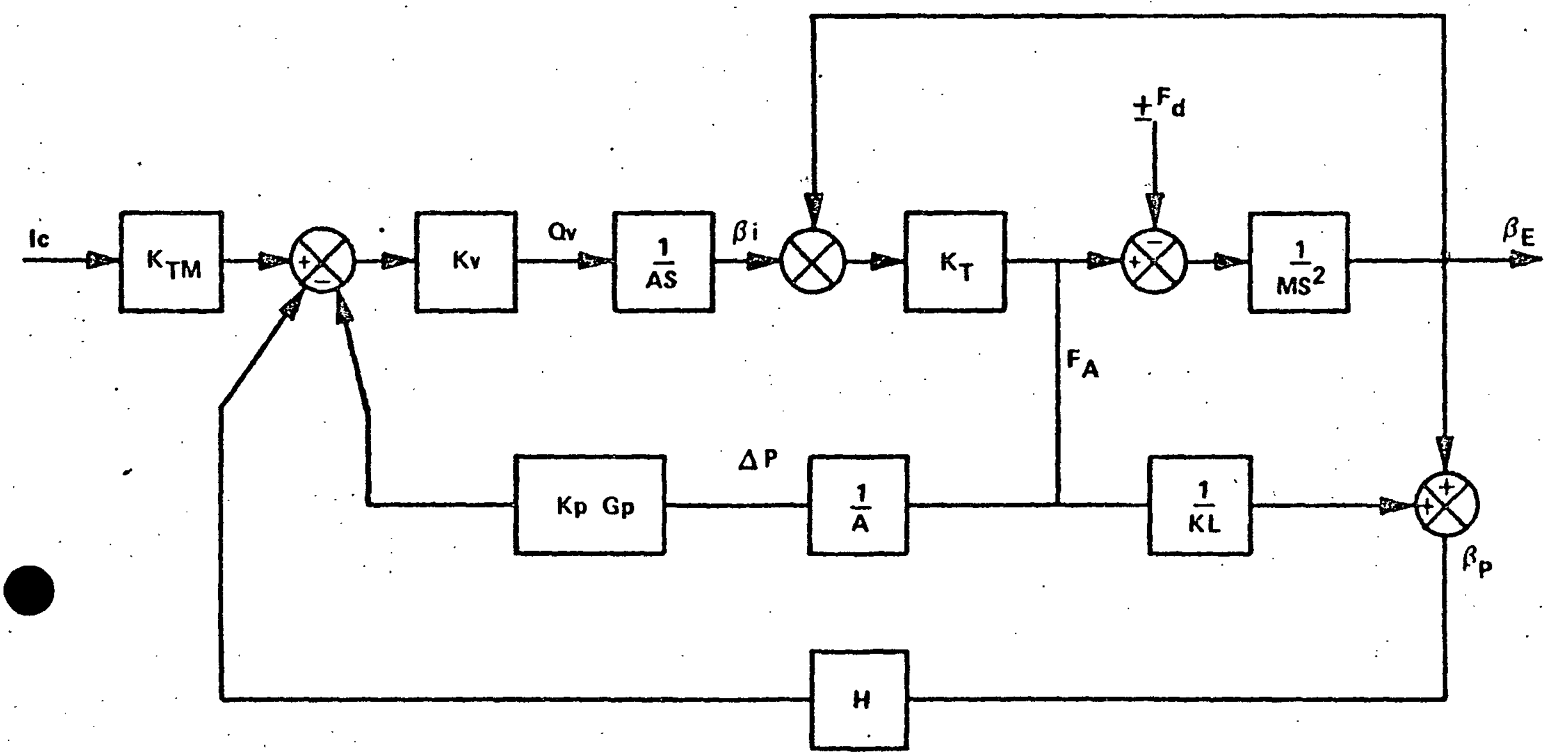
176 System stability - With system characteristics as given in Figure 4.3.5-2, and the actuator bias loads from 0 to 60,000 pounds, the load does not have continuing oscillations, hunting or limit cycling in excess of 0.006 inch peak to peak position during a 10 second period. Hunting or long time drift during a one minute period does not exceed the threshold of 0.5 ma.

176 Position loop gain adjustment - Provisions shall be incorporated into the actuator to allow adjustment of the position open loop gain $\frac{(KvH)}{A}$ from 12 sec $^{-1}$ minimum to 22.3 sec $^{-1}$ maximum. This gain adjustment mechanism should be field serviceable and should not have to be returned to the vendor facility for installation.

176 Servovalve - The servovalves are of the flow control type utilizing mechanical feedback from the final stage to the first stage. External to the servovalve and removeable mechanics, hydraulic circuits are provided to stabilize the reasonant load with bias loads up to 60,000 pounds as shown in Figure 4.3.5-2. Any additional steady state compliance due to these circuits is minimized.

176 Power consumption - The power used by the servovalve torque motor with 50 ma applied shall be no greater than 0.3 watts at 77 degrees F.

176 Actuator - The actuator is of the linear, double acting, equal area type with a net piston working area of 30 square inches.



LINEAR BLOCK DIAGRAM - SPACE SHUTTLE TVC
(SEE TABLE 1 FOR VALUE OF PARAMETERS)

FIGURE 4.3.5-2

SYSTEM PARAMETERS

SYMBOL	UNITS	VALUE	DESCRIPTION
KTM	in-lb/ma		Torque motor gain.
Kv	cis/in-lb		Valve flow gain.
A	in^2	30	Piston area
K_T	Lbs/in	219000	Total spring rate
K_L	Lbs/in	240000	Spring rate - Structural tie points
M	Lbs/sec^2 in	170.0 (Orbiter)	Engine equivalent mass
$K_L = \frac{KvH}{A}$	Sec^{-1}	22.30	Position loop gain
$K_{PL} = \frac{(K_V K_T K_P)}{A^2}$		33.6 Orbiter	Pressure feedback open loop gain
K_P	In-lb/psi		Pressure feedback gain
G_P		p^s/p^s+1	Pressure feedback shaping network
T_P	Sec	1.0 Orbiter	Time constant - Shaping network
H	In-lb/in		Position feedback gain
Q_v	cis		Valve output flow
βi	in		Ideal actuator piston position
β_P	in	Max = 5.5 in. = 10.5 deg.	Actuator piston position
β_E	in		Engine Position
Ic	MA	No less than 50 MA max	Input command current
Fd	lbs		External disturbance force
F_A	lbs		Actuator output force
P	psi		Differential pressure
Ps	psi	3000	System Supply Pressure
S	sec^{-1}	L	Laplace operator

TABLE 4.3.5-1

176 Length and stroke - The length of the actuator is as specified in Figure 4.3.5-1. The maximum stroke shall be ±5.50 inches. (10½ deg). The design allows for a minimum stroke of ±3.9" (±7½$^{\circ}$) by the installation of a stroke limiting mechanism internal to the actuator. This stroke limiting mechanism may be field installed.

176 Piston bypass valve - The piston bypass valve is manually operated, which when operated, relieves fluid locking of the piston. The valve is pressure actuated to close at a system pressure of 200 psig maximum.

176 Lock - A removable mechanical lock capable of holding the actuator rigidly in its midstroke position is provided. The lock when properly adjusted establishes the reference midstroke position.

176 Potentiometer - A linear, dual element, potentiometer with center taps is provided for actuator piston position indication. The potentiometer is internally mounted, directly coupled to the actuator rod.

176 Stroke - The mechanical stroke of the potentiometer is 12.0 inches, including 0.025 inch of over travel at each end. The electrical stroke is 11.0 ± 0.02 inches.

176 Electrical characteristics - The electrical characteristics are as shown in Figure 4.3.5-6. The sum of the pading resistance, which is 500 ±50 ohms, and the contact resistance is 575 ±125 ohms for each element as measured between wiper and element. The resistance between the center tap and the element is 150 ohms maximum for each element.

176 Excitation - The excitation voltage of the potentiometer is 60 ±0.25 vdc.

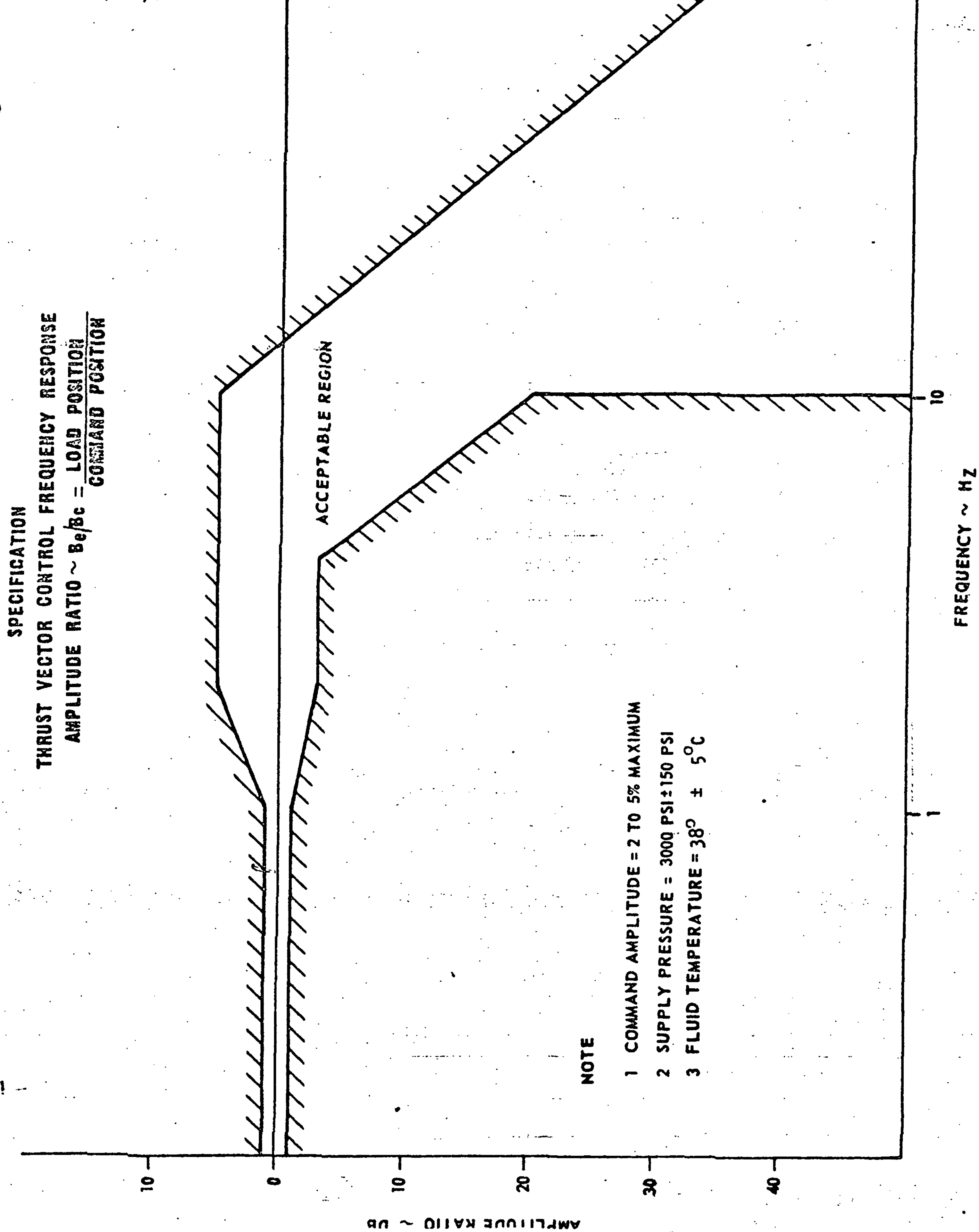

SPECIFICATION
THRUST VECTOR CONTROL FREQUENCY RESPONSE
AMPLITUDE RATIO ~ $B_e/B_c = \frac{\text{LOAD POSITION}}{\text{COMMAND POSITION}}$
ACCEPTABLE REGION
NOTE
1 COMMAND AMPLITUDE = 2 TO 5% MAXIMUM
2 SUPPLY PRESSURE = 3000 PSI ± 150 PSI
3 FLUID TEMPERATURE = 38° ± 5°C
FREQUENCY ~ Hz
1
10
100
10
0
10
20
30
40
AMPLITUDE RATIO ~ DB

FIGURE 4.3.5-3

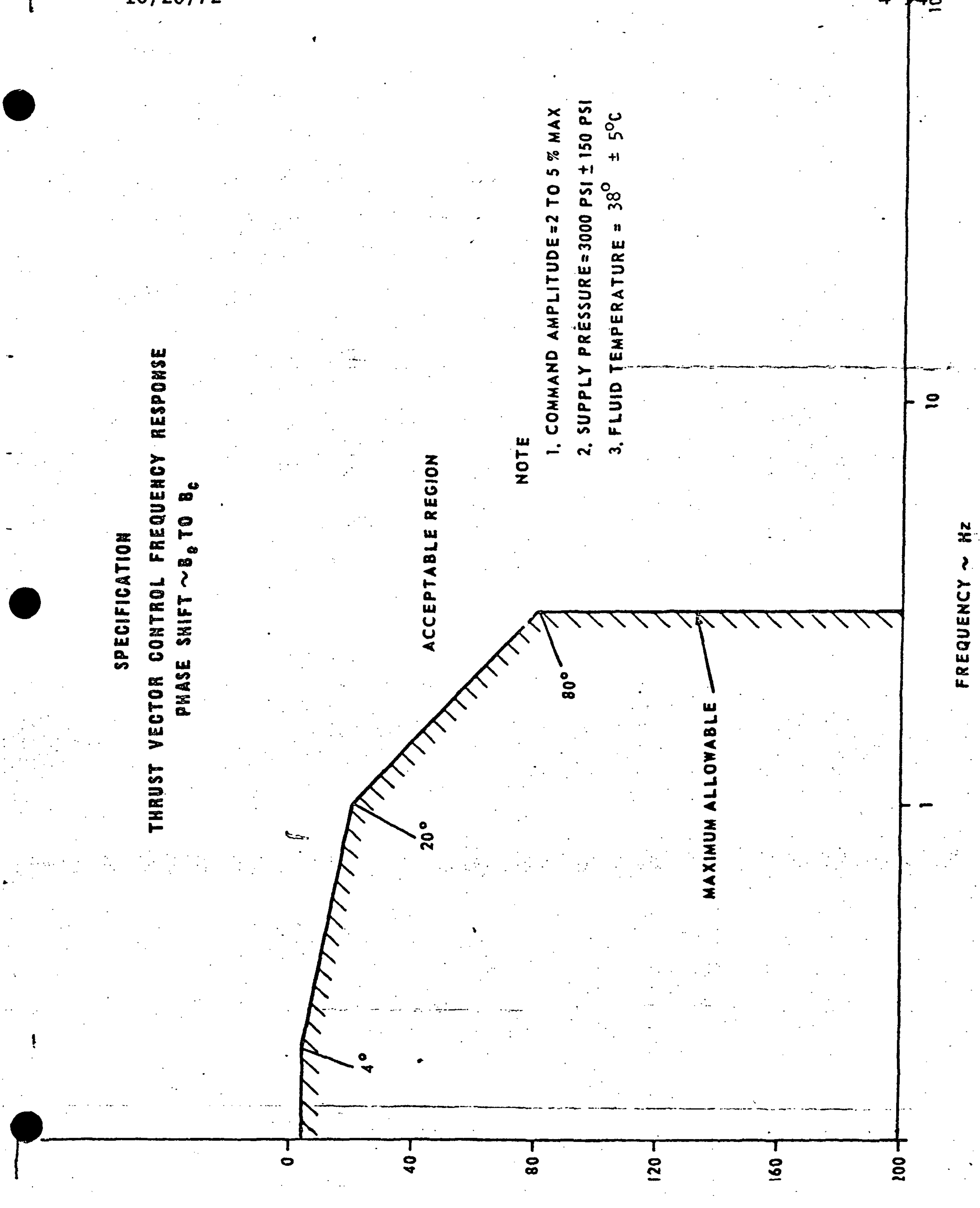

SPECIFICATION
THRUST VECTOR CONTROL FREQUENCY RESPONSE
PHASE SHIFT ~ B_e TO B_c
ACCEPTABLE REGION
NOTE
1. COMMAND AMPLITUDE = 2 TO 5 % MAX
2. SUPPLY PRESSURE = 3000 PSI ± 150 PSI
3. FLUID TEMPERATURE = 38° ± 5°C
4°
20°
80°
MAXIMUM ALLOWABLE
0
40
80
120
160
200
1
10
100
FREQUENCY ~ Hz

FIGURE 4.5.3-4

THRUST VECTOR CONTROL TRANSIENT RESPONSE
STEP CHANGE IN INPUT CURRENT

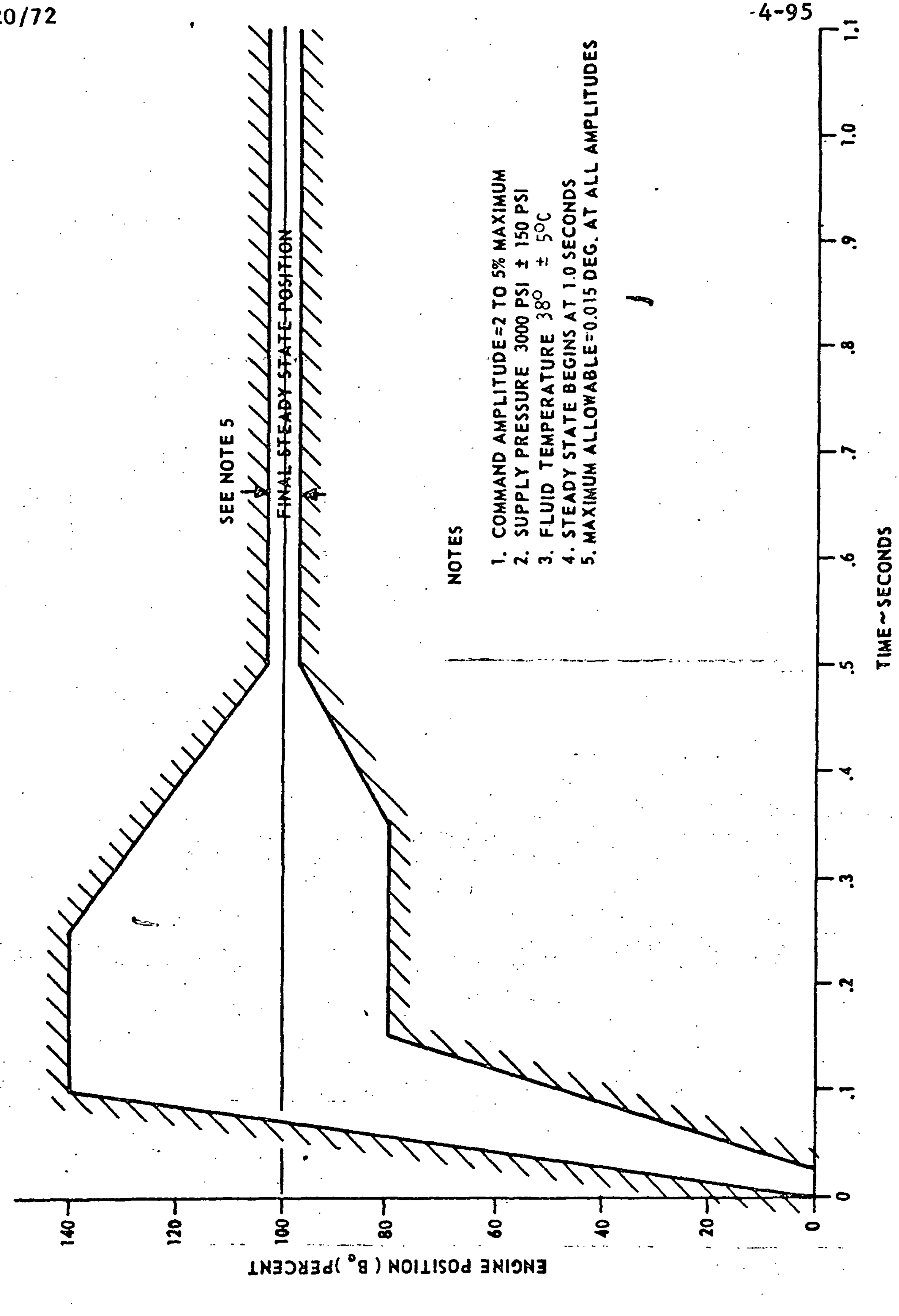
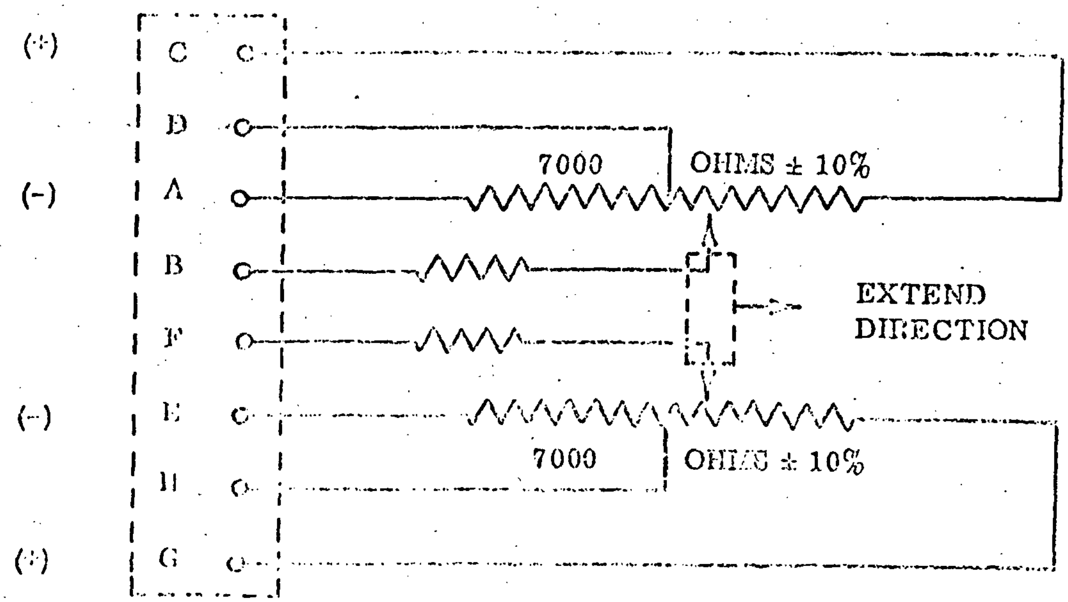


FIGURE 4.3.5-5

MSC FORM 422-8 (VERTICAL) (NOVEMBER 1962) CONTINUATION SHEET

REVISIONS			
SYM	DESCRIPTION	DATE	APPROVAL



NOTE:
1. E_o = OUTPUT VOLTAGE
2. E_s = SUPPLY VOLTAGE

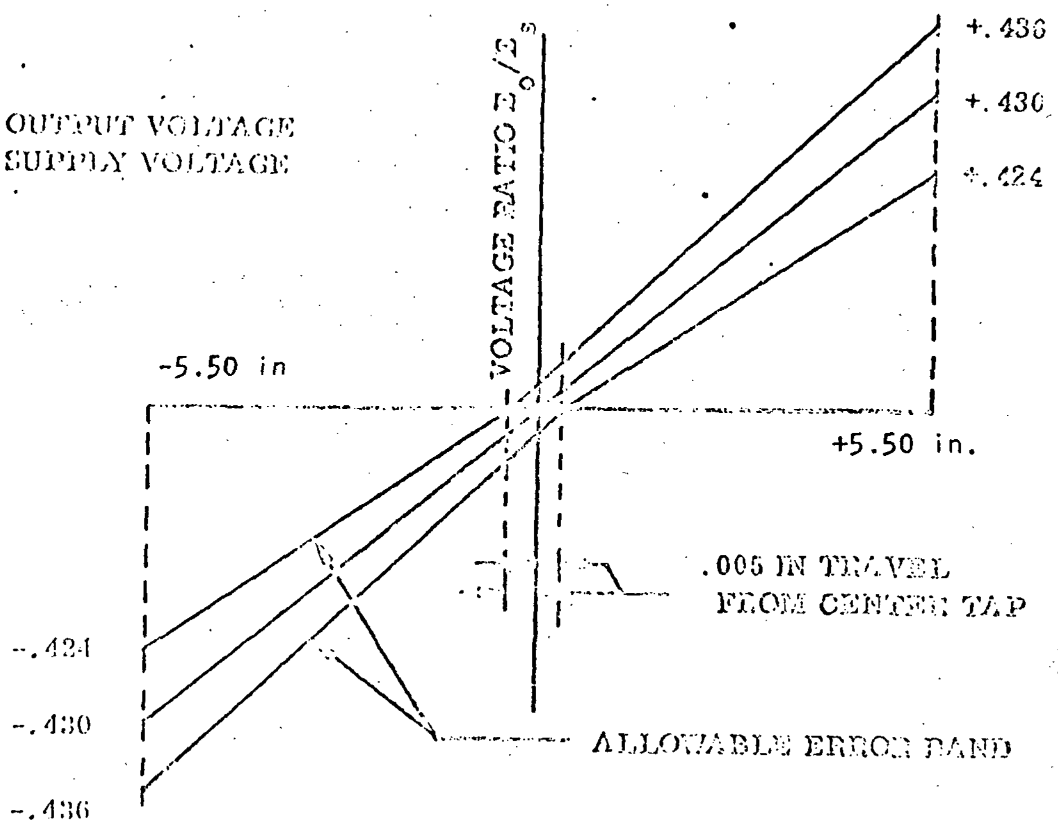


FIGURE 4.3.5-6

4.3.7 RATIONALE

Not Required

4.3.8 REFERENCES

166	Pages 3-55 thru 3-62
176	Pages 1 thru 33
20	Pages IV-20 thru IV-21
42	All the document

REF KEY

4.4 REACTION CONTROL SUBSYSTEM

166 The orbiter reaction control subsystem (RCS) consists of three self-contained monopropellant hydrazine propulsion subsystem modules installed as shown in FIGURE 4.4-1. The RCS provides attitude control and three-axis translational capability during both orbital and entry phases of the mission. The RCS may provide a backup for propellant acquisition during OMS engine operation.

4.4.1 CONFIGURATION

166 The RCS configuration employs 40 monopropellant thrusters operating at a rated vacuum thrust of 1000 pounds to provide a fail-operational/fail-safe attitude control and translation capability. Monopropellant hydrazine, stored in positive expulsion tanks, is used as the fuel. Tank pressurization is provided by regulated helium stored at ambient conditions. The RCS thrusters are installed in three independent removable modules located in the orbiter nose section and in each of the aft OMS pods. All thrusters, tanks, and components are completely interchangeable in all modules. The forward module contains 16 thrusters; each aft module located in the OMS pod contains 12 thrusters. Manifold isolation and purge valves provide multiple redundancy against loss of propellant in the event of a thruster valve open failure. Multiple redundant valves and regulators provide propellant tank pressure control.

166 During docking operations, the forward RCS module thruster deployment panel is reoriented 45 degrees toward the closed position to preclude exhaust plume impingement on either the space station or a deployed payload. This automatically disables the normally upward firing thrusters and modifies the functions of the downward and outward firing thrusters to then provide the required maneuvering forces. In this

manner, all forward module plume centerlines are maintained at a minimum of 45 degrees from any potentially sensitive surface.

166 The propellant storage and expulsion system uses the ethylene propylene terpolymer (EPT-10) diaphragm tank concept.

166 The polar mission propellant requirement defines RCS tank size and distribution between fore and aft modules. Propellant quantities for the three reference missions are shown in TABLE 4.4-1.

4.4.2 THRUSTER DESCRIPTION

166 The RCS thrust chamber assembly uses a stand-off tube to minimize heat soak-back to the propellant valve. This valve is a fuel-operated valve controlled by an electrically operated valve. A cavitating venturi is incorporated in the valve, thereby providing constant fuel flow to the thruster and permitting performance evaluation based on monitored chamber pressure variation. This concept minimizes catalyst bed overloading at each pulse, thereby minimizing cold start problems. The injector assembly consists of a body made of Hastelloy X with a low momentum Rigimesh injector design. The radial out-flow catalyst bed provides an increasing cross-sectional area to propellant flow, thereby minimizing the bed loading and decomposition gas velocities for maximum catalyst life. The inner section of the catalyst bed consists of 25 to 30 mesh Shell 405 catalyst to initiate the decomposition of the N_2H_4. The outer section of the bed uses a coarse 14 to 18 mesh low-cost catalyst which improves the overall catalyst bed performance and life characteristics. The thrust chamber and nozzle are fabricated from a single Hastelloy B precision investment casting. An electrical heater (12 watts per thruster) is attached to the injector head and maintains a minimum catalyst bed temperature of 150° F to eliminate cold starts.

4.4.3 PROPELLANT TANKAGE

166 A usable propellant capacity of 3130 pounds is provided in the forward module tanks, and a usable capacity of 1878 pounds is available in each of the aft modules. Five tanks are located in the forward module, and three are contained in each of the aft modules. Propellant tankage and lines are maintained within the acceptable operating regime by module thermal control. This is accomplished by electrical heaters (on-orbit) and insulation (on-orbit and entry).

166 The propellant tankage system is designed to supply propellants to the thruster inlets at 290 psia minimum. Manual isolation valves are provided to isolate each tank assembly from the pressurization system if required. A manual disconnect is used to facilitate filling and draining the propellant tanks in each module.

166 The RCS EPT-10 tank diaphragm is used to provide positive fuel expulsion during all operating conditions including zero g and re-entry. A capillar barrier at the outlet prevents diaphragm damage or pulgging of the propellant outlet port and permits most of the propellant to be expelled in the event of diaphragm failure. Tank diaphragm failures can be detected during turnaround by individually pressurizing each tank, monitoring the flow from the propellant fill and drain dump fitting and sampling the helium pressurant.

4.4.4 PRESSURIZATION SUBSYSTEM

166 Four identical 4000 psi helium spheres provide the helium required for propellant tank pressurization. Two spheres are mounted in the forward RCS module, and one is contained in each of the two aft modules. The pressurization subsystem is designed to use primary and secondary regulators with operating bands of 310 to 325 and 320 to 335 psi, respectively, with a minimum 600 psia inlet pressure. Parallel pressure relief valves

F-398-8-A

designed to accommodate full helium flow (equivalent to four thrusters operating at steady-state conditions) are provided to limit propellant tank pressure to the design limit pressure of 435 psia. The pressure/volume/temperature (PVT) propellant gauging system developed and used successfully on the Apollo CSM RCS will be employed in the orbiter RCS.

166 Upon completion of RCS operations, purge isolation valves will be activated to provide a low-pressure helium purge to cool and decontaminate the thrusters. Post-landing operations will include thruster and valve assembly purge and drying with heated low-pressure nitrogen to ensure removal of residual propellants.

4.4.5 RCS OPERATION

166 The thruster engines receive firing commands from the Guidance, Navigation and Control computer or by indirect entry via manual translation or rotational hand controllers. Data from the RCS system is supplied to the onboard computer system for display to the crew of propellant and engine conditioning status. Electrical power and instrumentation conditioning for T/M is provided through redundant power systems. Manual crew controls are provided for control of activation and monitoring functions of the RCS operating system

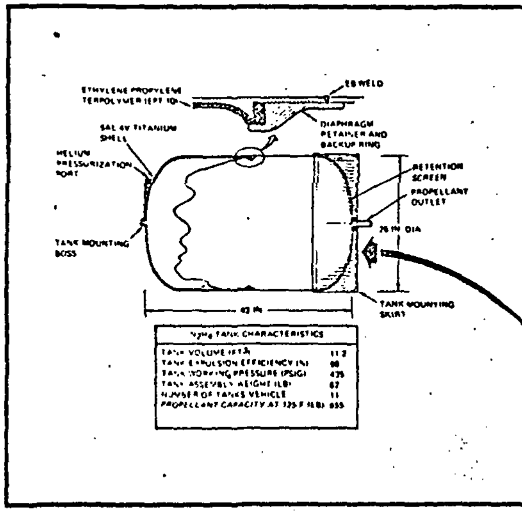
4.4.6 RATIONALE

Not required

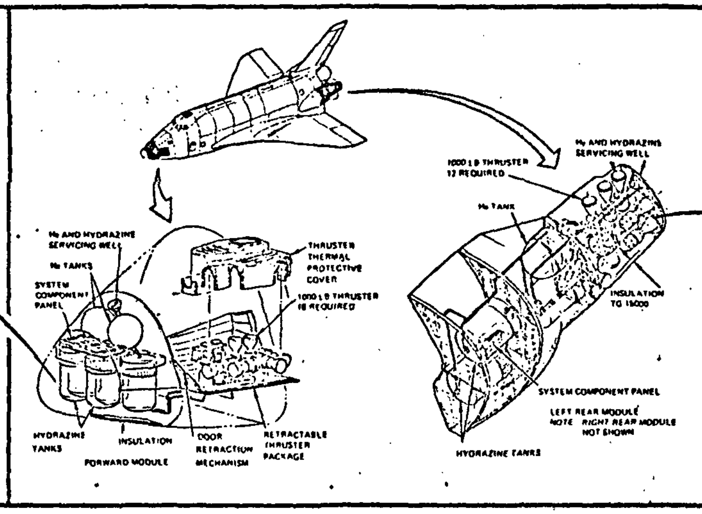
4.4.7 REFERENCES

166 pages 3-63 thru 3-68

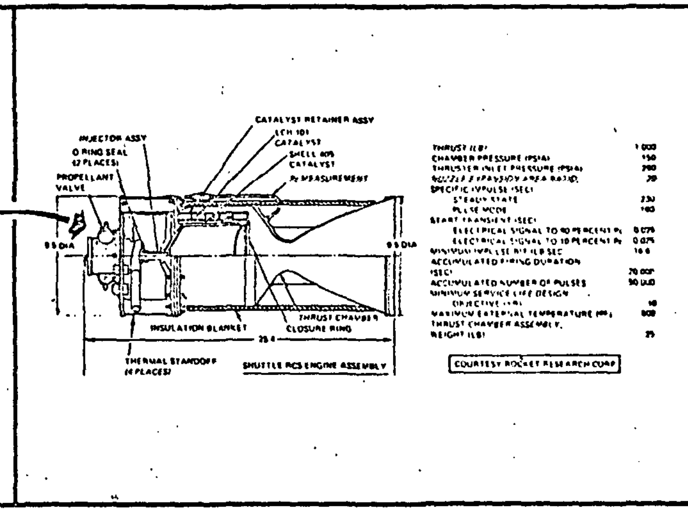
20 pages IV-20 thru IV-21



Proposed N_2H_2 Tank Design and Characteristics



Reaction Control Subsystem and Locations



Proposed RCS Thruster and Its Characteristics

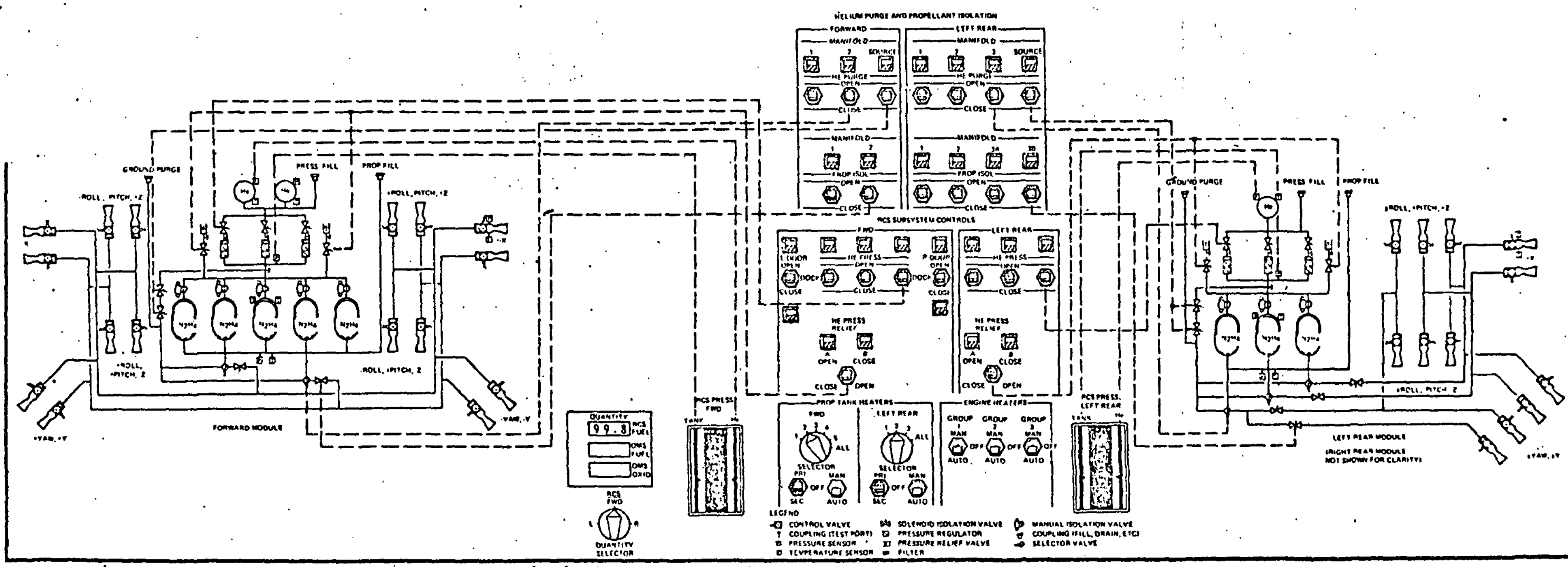


Figure 4.4-1 Reaction Control Subsystem Schematic

4-102
10/20/72

REF. KEY

4.5 ORBITAL MANEUVERING SUBSYSTEM (OMS)

166 The OMS provides the propulsive thrust to perform orbit circularization, orbit transfer, rendezvous, and deorbit. The OMS tankage is sized to provide propellant capacity for Mission 1, which retains a 65,000-pound payload throughout the mission.

166 The OMS is capable of burning all of its allocated propellant in either a single long burn or a series of multiple burns spread at random over the mission duration.

4.5.1 OMS CONFIGURATION

166 The propellant quantity required for the design mission will be provided in two pods, one located on each side of the aft fuselage. Each pod contains a high-pressure helium storage bottel, tank pressurization regulators and controls, a fuel tank, an oxidizer tank, and a pressure-fed recket engine. The OMS employs nitrogne tetroxide (N_2O_4) as the oxidizer and monomethylhydrazine (MMH) as the fuel. Additional propellant storage is provided by three self-contained pressurant/propellant supply kits which maybe located in the cargo bay. The OMS ΔV/payload capability and the number of auxiliary OMS kits required are shown in FIGURE 4.5.1 as a function of the booster-MPS liftoff capability for the three design mission inclinations.

166 Schematics of the basic subsystem and auxiliary subsystem are shown in FIGURE 4.5.1. Each pod system is fail-operational/fail-safe, except for the engine bipropellant valve, which is fail-safe. A pod crossfeed line, employed in conjunction with individual engine isolation valves, assures availability of all propellant for deorbit with the remaining single engine following a multiple failure in the bipropellant valve of one engine.

4.5.2 ENGINE

166 The OMS engine is a 5,000-pound thrust reusable pressure-fed rocket engine. The nominal characteristics for this engine are listed in FIGURE 4.5-1

166 The thrust chamber is regeneratively cooled using engine fuel flow with supplementary film-cooling. The coolant jacket extends aft from the injector to the radiation-cooled nozzle extension attach flange. The thrust chamber is fabricated of stainless steel. The injector is a flat-faced, nonbaffled design utilizing acoustic cavities to achieve dynamic combustion stability. The bipropellant engine valve is a series-parallel redundant valve with a pneumatic actuation package. The basic valve design is similar to that used in the Apollo CSM service propulsion subsystem engine valve, except for the use of a cam mechanism to lift the seals away from the ball prior to ball rotation, thus minimizing the amount of seal rubbing on the ball surface.

166 The design uses the basic gimbal technique employed on the Apollo CSM service propulsion subsustem engine. Electro-mechanical actuators incorporating redundant drive mechanisms will be employed.

4.5.3 TANKAGE

166 The propellant and helium tanks are fabricated from 6Al-4V titanium alloy. The fuel and oxidizer tanks are identical and contain a maximum usable propellant load of 12,200 pounds per pod. The helium tanks will be identical to the Apollo CSM service propulsion subsystem helium tanks with a nominal servicing pressure of 3200 psia at 70°F.

166 Analysis indicates that lateral sloshing in the OMS tanks, which might occur during boost, does not represent a significant dynamic problem. Dynamic coupling of the sloshing forces and moments with the vehicle stabilization and flight controls can occur only at frequencies at which the disturbance torques generated are not significant. Therefore, the OMS propellant tanks do not have baffles.

166 Liquid propellant will be retained in the engine supply lines through use of screened retention reservoirs containing devices to prevent premature surface dip and vapor ingestion at the tank outlets at propellant depletion. The reservoirs refill by hydrostatic vapor expulsion during OMS firings and require no auxiliary settling thrust.

166 The auxiliary tankage and pressurization components located in the cargo bay are identical to hardware used in the OMS pods. Six propellant tanks (three oxidizer and three fuel) and three helium tanks may be installed. A single pressurization control assembly is employed for the auxiliary tnakage. The auxiliary tankage kit is designed so that either one, two, or three sets of propellant and helium tanks can be installed as required by a particular mission.

166 Quantity gauging is provided during OMS engine firings by a set of 10 dual point sensor assemblies. Quantity indications between point sensors, necessary for OMS status assessment, are provided by an integrating unit similar to the Apollo CSM service propulsion subsystem backup gauging system, using engine burn time and nominal propellant flow rates. Low-level warning is provided in each tank by an independent point sensor assembly. Backup and zero-gravity gauging is provided by pressure/volume/temperature type

gauging through use of the on-board performance monitor and CRT display.

4.5.4 PRESSURIZATION

166 The fuel and oxidizer tanks are pressurized through parallel flowpaths, thus eliminating the possibility of mixing fuel and oxidizer vapors coused by leakage and/or diffusion through the isolation check valves in the helium pressurization lines. Check valves are employed to prevent liquid propellant migration.

4.5.5 INSTALLATION

166 The OMS is installed in two removable pods with propellant crossfeed lines that remain with the vehicle. The auxiliary tankage in the cargo bay also feeds propellant to the engines in each pod through the crossfeed lines.

4.5.6 ENVIRONMENTAL CONTROL

166 Thermal control for the OMS is provided by insulation and electric heaters to maintain the internal pod environment between 40° to 125°F on-orbit with a maximum of 150°F allowed as a result of entry heating. The insulation is installed to allow subsystem thermal isolation as a unit. This allows high heat capacitance elements to thermally stabilize the low mass elements.

4.5.7 MAINTENANCE

166 The engine propellant isolation valve located between the tanks and the engine brpropellant valves provide for engine removal without purging of the propellant tanks. The engine valves also can be checked out with the engine installed and the tanks fully loaded .

166 Manual shutoff valves installed upstream of the propellant tanks isolate the pressurization system from residual propellant vapor during ground operations. This valve also allows extensive checkout and replacement of pressurization system components without applying unnecessary pressure cycles

to the propellant tanks. The propellant tanks will be purged only when maintenance or replacement of the tanks or associated components is required. The manual shutoff valve control will be designed so that the vent receptacle cover cannot be installed with the valve closed.

4.5.8 OPERATIONS

166 OMS preflight checkout and servicing is accomplished in the hypergolic serivicing facility. Following propellant servicing, the helium storage tanks and propellant tanks are then pressurized to 1,000 and 50 psig, respectively, to meet safety standards for normal working areas. The loaded pods are subsequently mated to the orbiter, after which the electrical and instrumentation interface will be checked out with the tank and engine isolation valves closed. Final filling of the helium bottles will be accomplished at the launch pad. Auxiliary system checkout and servicing is identical to this procedure. An alternate capability will provide loading propellants at the launch pad after installation of the external pods and auxiliary system.

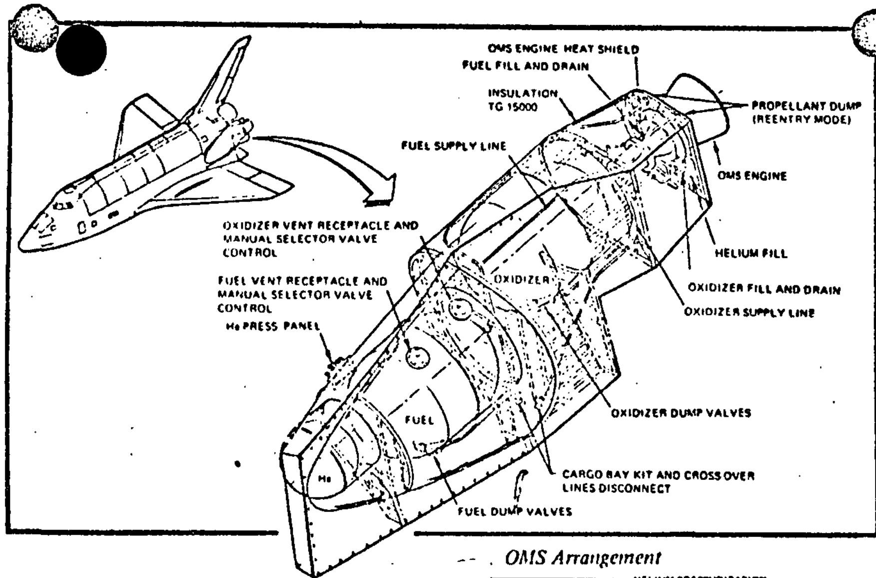
166 The OMS is capable of operation at any time after launch, including simultaneous operation with the MPS engines. During all operating modes, the engines are normally fired simultaneously with the TVC maintaining parallel thrust vectors. During single-engine operation, the RCS will provide roll control.

166 Auxiliary tankage propellant, when installed, will be consumed first. Propellant is retained in the pod tanks by closing the helium isolation valves in the pods. When a low-level warning is received from the auxiliary sump tank, the pod helium isolation valves are opened, and the auxiliary helium isolation, propellant isolation, and crossfeed valves are closed automatically.

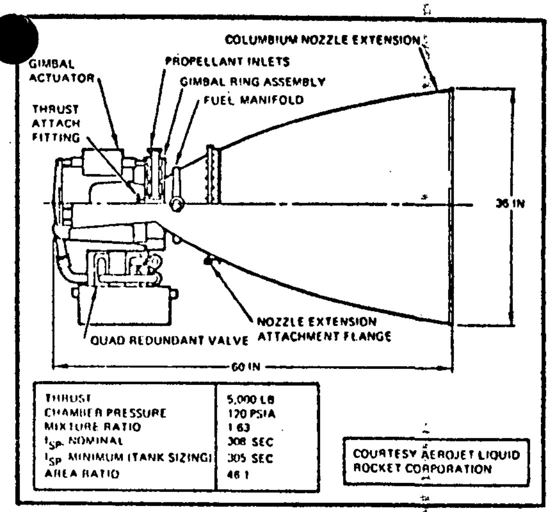
Following the deorbit burn, the engine propellant isolation valves are closed and the downstream propellant lines and engines are purged using residual helium. The engine valves remain open until just prior to reentry to vacuum-dry the purged assemblies and thereby remove remaining propellants and vapors. The engine valves are then closed to allow repressurization of the line upstream of the engine valves as a means of preventing air ingestion after entry.

166 A dump system is provided for dumping OMS propellant in the horizontal flight condition. Residual propellants are dumped during entry following a normal mission; however, the system is sized to dump all OMS propellant in the pods prior to landing in the event of a low-altitude abort. For other abort modes, the OMS propellant will be burned through the engines.

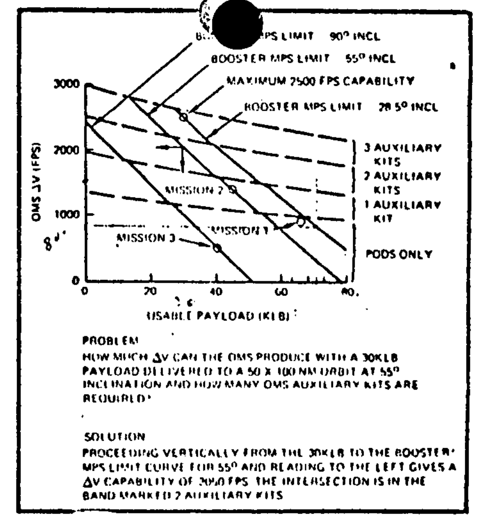
F-398-8-A



OMS Arrangement



OMS Engine Characteristics



OMS ΔV and Payload Capability

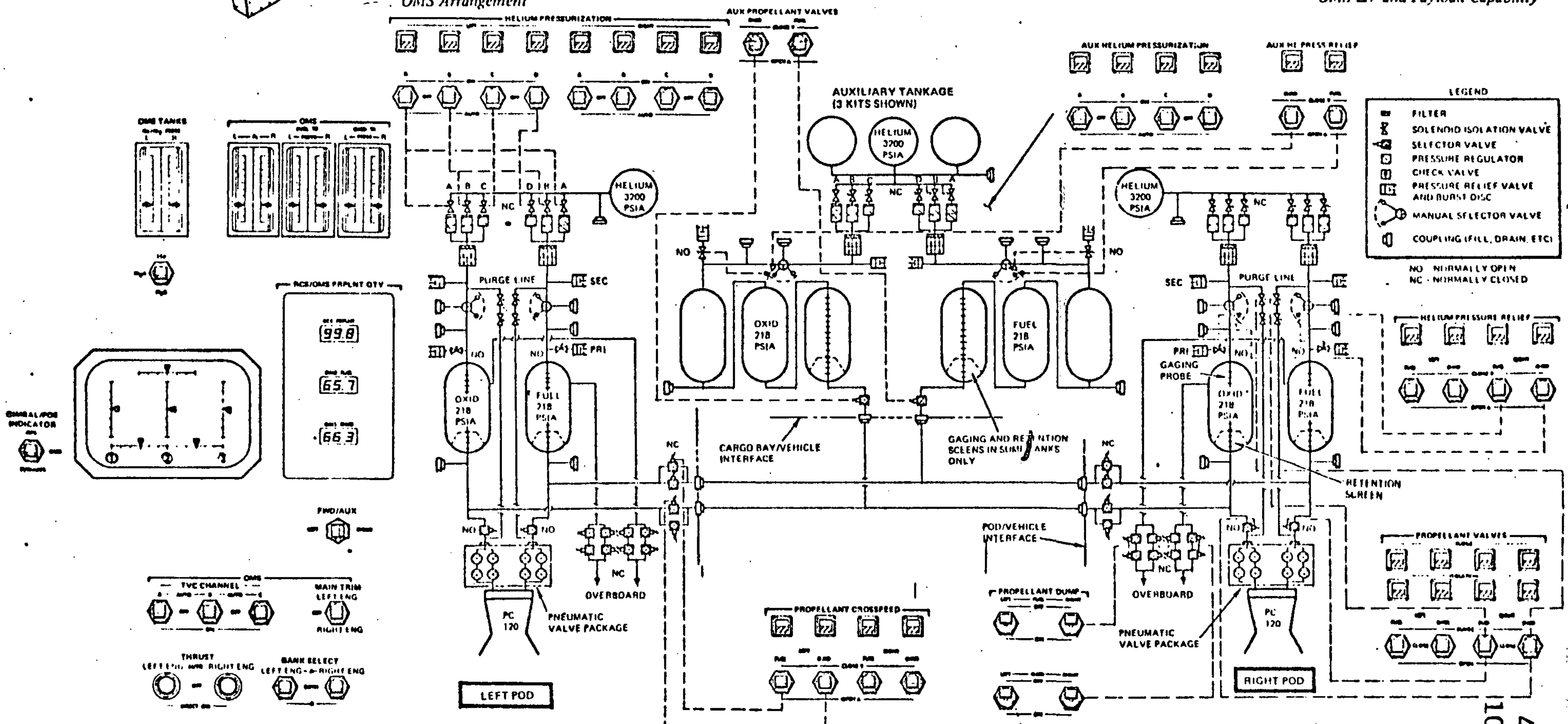


Figure 4.5-1 Orbit Maneuvering Subsystem Schematic

4.5.9 RATIONALE

Not Required

4.5.10 REFERENCES

166	Pages 3-68 thru 3-75
20	Pages IV-20 thru IV-21

REF. KEY

4.6 AIRBREATHING PROPULSION SUBSYSTEM (ABPS)

166 The ABPS primary purpose is to provide loiter flight capability upon return from a space mission. With modifications, the ABPS also provides self-ferry capability from alternate landing sites to the launch site. The self-ferry configuration is used to conduct the horizontal flight test program.

166 The orbital ABPS module, incorporating two engines, deployment mechanisms, a fuel tank, and supporting subsystems, may be installed in the aft section of the cargo bay. The weight of this module is charged against total payload capability; the module is removable for missions requiring maximum payload capability. The ferry system consists of the orbital module plus two additional engines, a ferry fuel tank, and supporting structure. A secondary electrical and hydralic power generation system is included in the ferry system.

4.6.1 CONFIGURATION

166 The orbital ABPS module consists of the structural, the engine/nacelle/pylon, and the fuel tank assembly modules as shown in FIGURES 4.6.1. The vehicle scar weight associated with installation of the integrated module is that necessary for instrument and control wiring, vehicle structural "hard points," ferry mode vehicle power interface (hydraulic and electrical), and provisions for installation of the horizontal flight fuel fill and drain adapters. When the ABPS is installed, the two aft sections of the segmented cargo bay doors are removed.

166 The structural module includes provisions to accommodate the orbiter structural interface, nacelle thrust and airloads, nacelle deployment, tank

166 mounting, and manipulator arm stowage. The nacelles are deployed by rotary motion about the nacelle pylon longitudinal axis using hydraulically driven power hinges. Segmented deployment doors actuated by linear hydraulic actuators remain closed throughout launch and orbital operations for environmental control purposes. The doors are automatically sequenced open during the deployment cycle and are then closed following engine depolyment.

166 Each nacelle module houses an afterburner turbofan engine modified to accommodate the predicted launch and orbital environment. The engine is a derivative of the Pratt and Whitney F401-PW-400 engine now under development. Supporting subsystems include a dual element fire detection subsystem, a fire extinguishing sybsystem (dual bottle/dual shot per nacelle), electric thrust control subsystem, an air start assist system, and a thermal control system. The electric thrust control subsystem is an adaptation from the concept being developed under the B-1 program with minor modifications required for engine and cockpit quadrant arrangement differences. A cartridge air start system is proposed with either cartridge or pneumatic capability provided for ground starts.

4.6.2 FUEL TANK

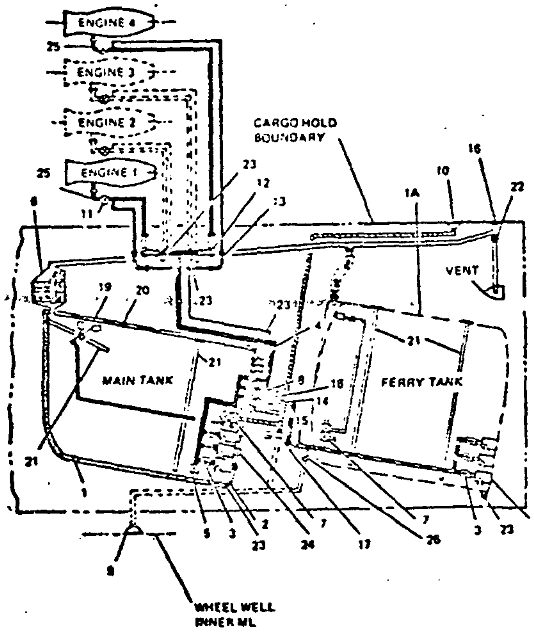
166 The fuel tank module includes a main fuel tank and the associated tank-mounted units of boost pumps, shutoff valves, vent valves, quantity gauging, and thermal control provisions. The tank may contain 22,565 pounds of fuel for the early orbital development flights. For operational flights with scientific payload, this tank will be off-loaded to the 13,681-pound quantity required for engine air-starting and 15 minutes of loiter flight at 10,000 feet.

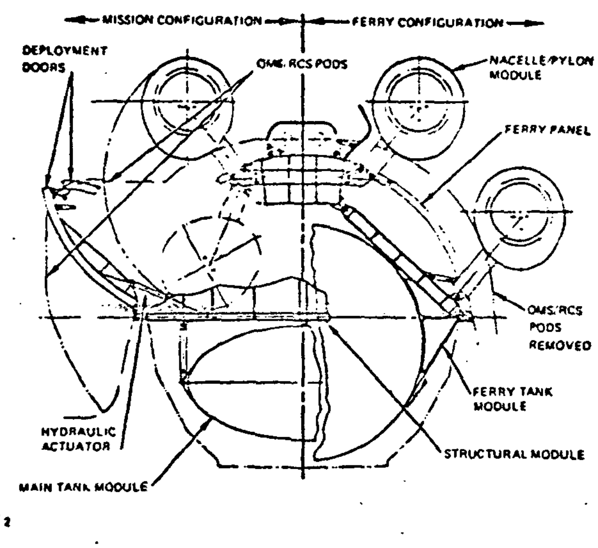
166 Tank fuel flow is provided by three electrically driven "plug-in" type boost pumps supplying a common maniford that connects to both engine nacelles. An atmospheric air-venting system is provided that uses series/parallel vent valves to maintain on-orbit tank pressurization and fuel isolation. Refueling-defueling capability and capacitance gauging systems are provided for both the vertical and horizontal vehicle attitude. Dual level control valves are employed for capacity control in either attitude.

166 The thermal control subsystem employs electric heaters and insulation to maintain the bulk fuel, the engine, and support subsystems (starters, fire extinguishing, thrust control, lube oil, etc.) within design temperature limits. A steady-state temperature range of $-65^{\circ}F$ to $275^{\circ}F$ is specified for the engine with a $-350^{\circ}F$ limit on transients; the lube oil in the storage tank is limited to $-25^{\circ}F$ minimum; and a range of $-30^{\circ}F$ to $+160^{\circ}F$ is stipulated for the fuel.

4.6.3 FERRY CONFIGURATION

166 To provide the required ferry thrust, two additional F401-PW-400 engine nacelle assemblies are installed in a fixed position on the ABPS structural module lower shoulder. The vehicle APU system will not be used for power generation during ferry or horizontal flight test operations; therefore, a secondary power generation kit providing hydraulic and 400-Hz electrical power capability will be installed and driven by each of the four engines. An uninsulated ferry tank will be installed aft of the orbital fuel tank to accommodate an additional 44,905 pounds of fuel. This capacity was selected to provide maximum fuel (67,470 pounds) for flight test up to

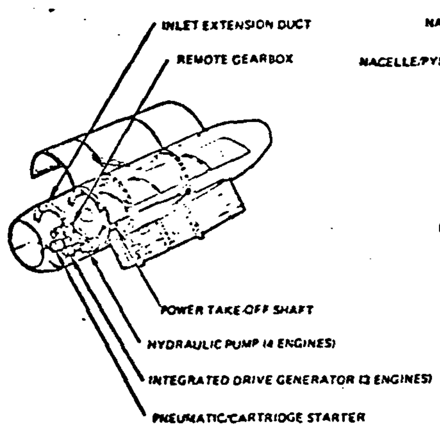




AIRBREATHING PROPULSION SYSTEM MODULE

FUEL SYSTEM SCHEMATIC DIAGRAM

	COMPONENTS	NO. REQUIRED
1.	MAIN TANK	1
1A.	FERRY TANK	1
2.	BOOSTER PUMP (PLUG IN TYPE)	6
3.	PUMP DISCHARGE CHECK VALVE	6
4.	FEED SYSTEM PRIME VALVE	3
5.	SUCTION FEED CHECK VALVE	1
6.	VENT VALVE	1
7.	REFUEL LEVEL CONTROL VALVE	2
8.	MAIN TANK ISOLATION VALVE	3
9.	FILL AND DRAIN ADAPTER AND CAP – HORIZONTAL SERVICING	1
10.	FILL AND DRAIN ADAPTER AND CAP – VERTICAL SERVICING	1
11.	ENGINE ISOLATION (FIREWALL VALVE)	4
12.	DEPLOYMENT ROTARY JOINT (FEED)	2
13.	DEPLOYMENT COUPLING HOSE (PRIME)	2
14.	DEFUEL VALVE (MAIN TANK)	1
15.	FERRY TANK FUEL VALVE	1
16.	VENT OVERBOARD ADAPTER AND CAP	1
17.	MANUAL VALVE (FILL AND DEFUEL)	1
18.	MANUAL VALVE (TRANSFER)	1
19.	INSULATION BLANKET	1
20.	HEATER (DUAL ELEMENT)	1
21.	FUEL GAGING CAPACITANCE PROBE	4
22.	MANUAL VALVE (VENT ISOLATION)	1
23.	DRAIN VALVE	5
24.	PUMP PRESSURE SWITCH	3
25.	ENGINE CONNECTION HOSE	1
26.	FILL & DEFUEL VALVE (FERRY TANK)	2



SECONDARY POWER GENERATION KIT FERRY CONFIGURATION

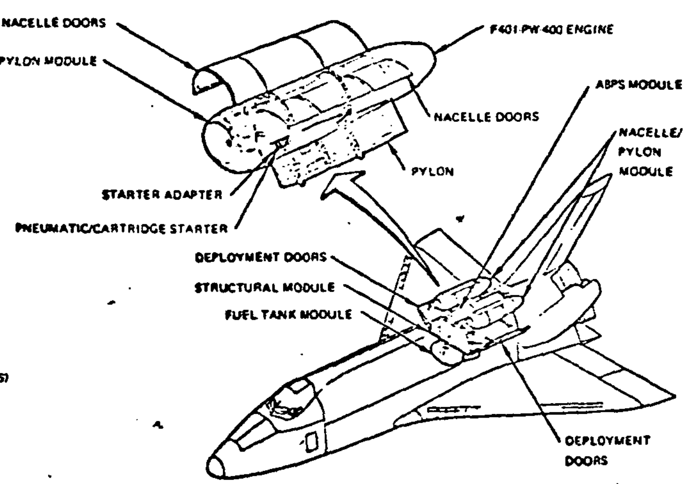


Figure 4.6-1 *ABPS Configuration*

166 the vehicle gross weight limit of 232,000 pounds. Capacitance gauging and single point refueling provisions will be provided for the ferry tank in the horizontal attitude only. The ferry tank will be off-loaded to provide a total fuel loading of 52,000 pounds for the 400-nm ferry range capability.

4.6.4 OPERATION

166 The APES is purged of all fuel and lube oils prior to orbital flight. This is necessary since in orbit the engines encounter vacuum environment.

166 The vehicle uses pneumatic helium pressure for the main propellant and engine valve activation, engine pump seals, purging and inerting all propulsion lines, engines land fuel tank prepressurization.

166 Operationally, the ABPS remains inactive until reentry except for thermal control system functions. After entry, the nacelles are deployed, and the engine oil and fuel systems are primed. For early development flights, assisted air starts will be initiated at 40,000 feet- the upper limit of the air-start envelope. The design fuel load for operational flights is based on delaying air start initiation to 25,000 feet, which will provide sufficient time during descent for engine start and operational check prior to initiating the planned 15-minute loiter flight at 10,000 feet altitude.

AC The APES engine is provided with a manual power lever to control the engine thrust from IDLE to maximum. Below the IDLE position the power lever will actuate fuel cutoff valves.

C Control of all engine operations is initiated by crew members. Engine thrust level is governed by a fuel controller. The fuel controller combines input signals from the throttle lever position, engine RPM sensor, compressor inlet temperature and pressure sensors, and the compressor discharge pressure sensor to position the fuel metering valve.

B Instrumentation monitoring is required on fuel flow rates, temperatures, pressure; turbine inlet and exhaust temperatures; valve state; lubricating oil quantity, pressure and temperature; and burner pressure.

D The Caution and Warning System is provided inputs from the ABES system during ferry flights and the re-entry flight when activated by crew members.

4.6.5 RATIONAL for ASSUMPTIONS

A. Throttle controls for the APES are visible on instrument panel configurations, however, detailed descriptions are not yet available.

B. Reference 36, page 4-673 indicates APES instrumentation by quantity of measurements only. At present it is assummed these measurements pertain to standard turbofan parameters monitored in existing aircraft. Also since the engine is deployable, it is also assummed there are indications presented to the crew for the condition of engine deployed.

C. There is no reference detailed to the level of components of systems for the APES. It is assummed the APES is manually controlled, i.e. no computer throttle control and that the turbofan engine has a fuel controller similar to ~~a~~ standard aircraft.

D. There is no reference to date of interfaces of the Caution and Warning System, however, this interface is required for operation.

4.6.6 REFERENCES

20 pages IV-20 to IV-21

166 pages 3-75 to 3-80

4.7 SOLID ROCKET MOTORS

66 The shuttle vehicle is equipped with 156" solid rocket motors (SRM) providing thrust forces for main vehicle liftoff, shuttle vehicle abort separation and external tank retro-deorbiting. Each of these engine groups is discussed in detail in the following paragraphs.

4.7.1 MAIN SRM

66 Two 156-inch diameter SRM's are attached to the orbiter external tank and burn in parallel with the orbiter MPS engines to provide ascent propulsive thrust up to staging. In addition to the rocket motors, the booster assembly contains an aft skirt launch support structure, forward skirt and external tank attach structure, separation rocket motors, recovery system, aerodynamic fairing, an electrical power and distribution system, and a malfunction detection instrumentation system.

66 The SRM assemblies transmit thrust through a structural skirt at the forward end of the motor into the external tank subsystem intertank structure. Total vehicle support on the launch pad is provided by structural skirts on the aft end of the motors. The motor nozzels are fixed at a cant angle of 11 degrees to the motor centerline in the yaw plane. Thrust termination ports are provided in the forward end of the motors for use under abort conditions. FIGURE 4.7.1.1 shows a baseline motor envelope,while preliminary motor characteristics are presented in FIGURE 4.7.1.2.

66 A segmented case design was selected to afford a maximum degree of flexibility in selecting a fabrication site and for ease of transportation

and handling. The motor nozzle uses a composite ablative inner liner and steel outer shell with provisions for attachment to the motor case aft closure.

166 Each SRM composed of polybutadiene acrylonitrile (PBAN) provides an initial sea-level thrust of approximately 3.52 million pounds. The motor ballistic provides a high initial thrust/weight with decreasing thrust to limit maximum q followed by an increasing thrust to minimize g loss while maintaining vehicle acceleration below the 3-g limit. Predicted ballistics for a motor using proven propellant grain geometries were used for trajectory analysis, FIGURE 4.7.1.2.

166 A conventional self-contained pyrogen ignition system with appropriate redundancies and safe and arm provisions will be utilized. Final selection of initiator type (exploding bridgewire versus hot wire) will be based on an assessment of overall Shuttle system ordinance requirements.

166 Thrust termination is provided for abort modes by means of two symetrical blowout ports formed in each forward motor by the ignition of linear shaped charges (FIGURE 4.7.1.3). Exhaust stacks are provided to direct the gas discharge through the forward attach structure and away from the orbiter. The stacks also provide some gas expansion to achieve slightly higher thrust. The ports are sized to provide approximately 10 percent positive SRM net thrust throughout all phases of burning to enhance separation. A malfunction detection system is used in conjunction with the thrust termination system to provide an early warning of an impending catastrophic SRM failure. As a minimum, the system will contain pressure and temperature sensors to detect abnormal pressures and/or overheating conditions.

166 The SRM/external tank subsystem separation system (FIGURE 4.7.1.4)
171 operates in normal and abort modes at varying SRM residual thrust levels and oblique vehicle flight paths. The SRM separation system uses auxiliary rockets to provide relative separating motion between the SRM cases and the orbiter/tank. Forward and aft recket thrusters provide the desired separation trajectory, while the aft thrusters are rotated to counter any SRM residual thrust. The thrusters are located to minimize plume impingement on the orbiter. This system prevents reaction loading on the external tank subsystem. Three rockets are located forward and three aft to provide safe separation with single rocket failure for normal staging ($F_{vac_{avg}} = 27K$ pounds, $t_{burn} = 2$ seconds). The aft separation rockets are installed at a greater angle than forward rockets to account for residual thrust of skewed main rocket nozzle.

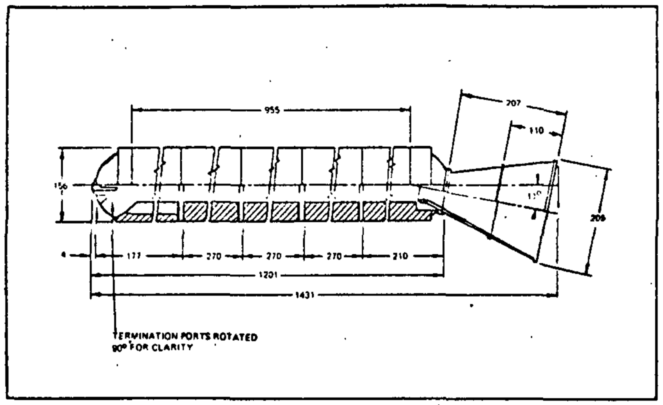


Figure 4.7.1.-1 Solid Rocket Motor Configuration

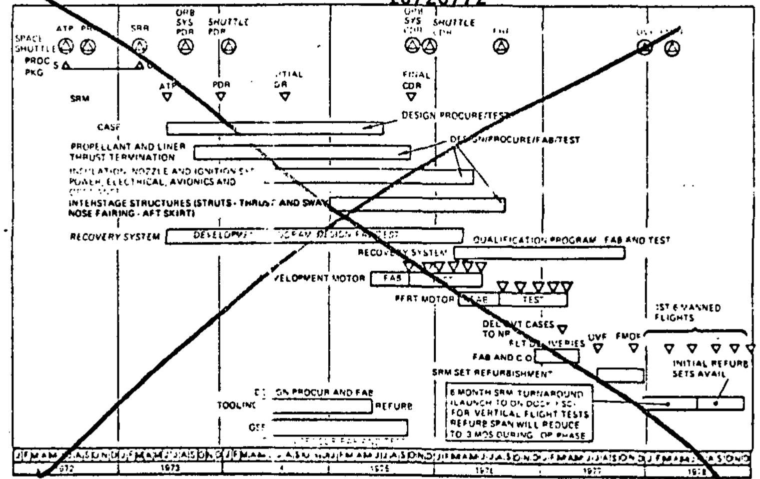


Figure 3-75. SRM Development Schedule

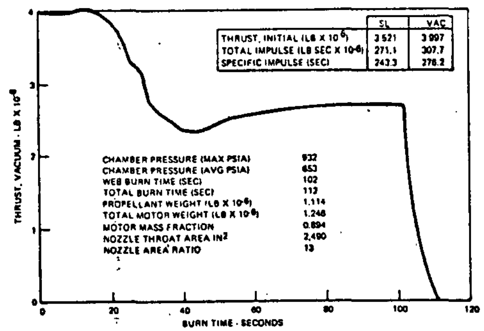


Figure 4.7.1-2 SRM Thrust Time Profile and Characteristics

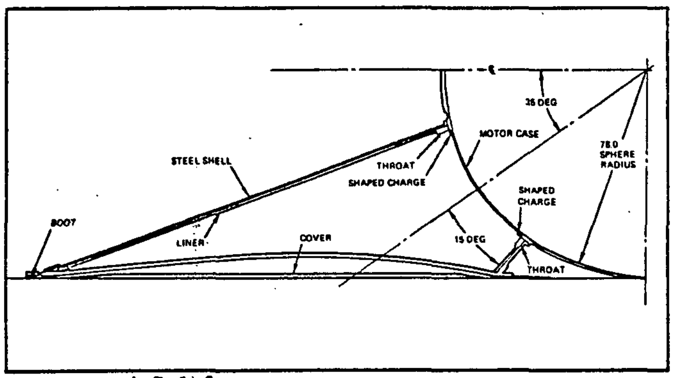


Figure 4.7.1-3 SRM Thrust Termination Port Design

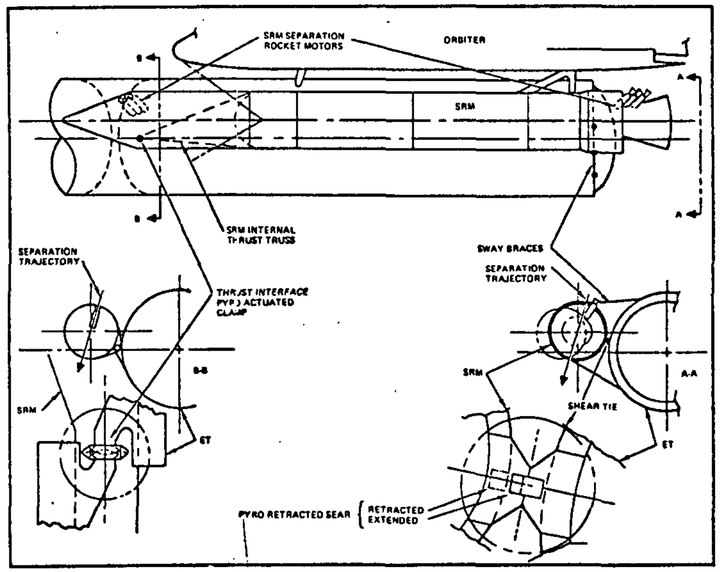


Figure 4.7.1.-4 SRM Installation and Separation System Concept

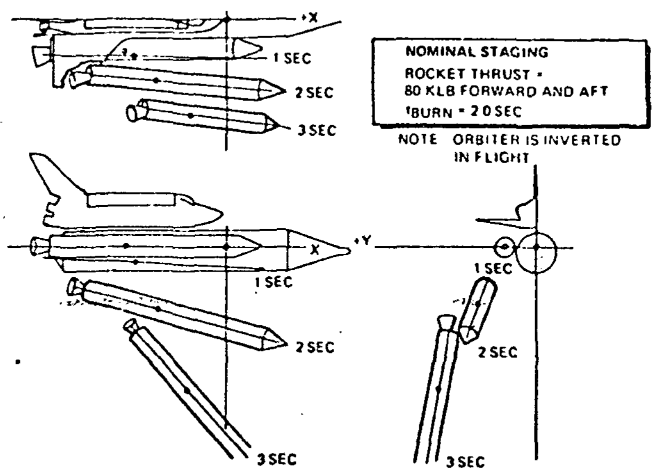


Figure 252. Successful SRM Separation Under Nominal Conditions

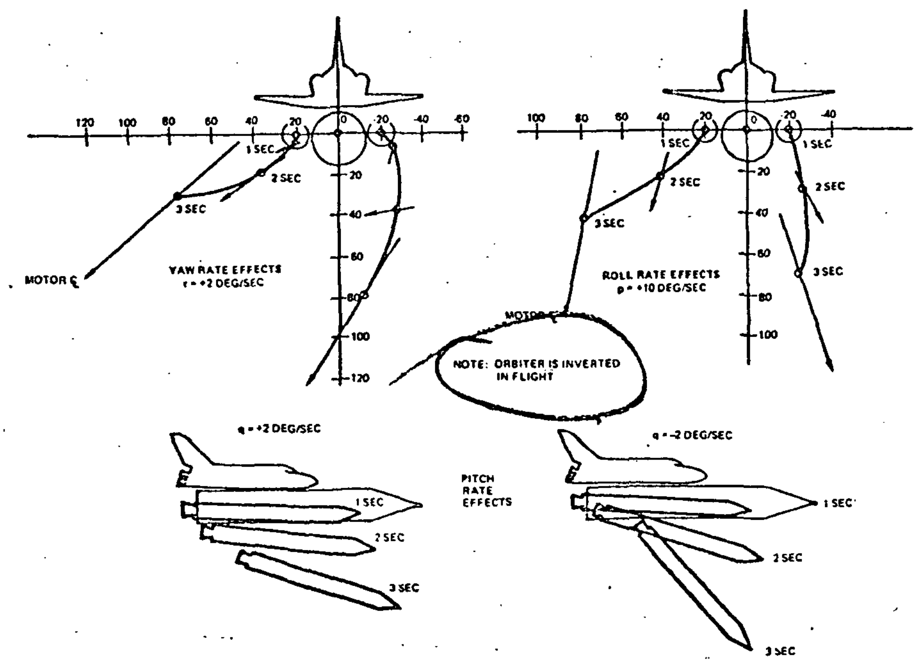


Figure 253. Successful SRM Separation Under Off-Design Conditions

FIGURE 4.

4.7.2 ABORT SOLID ROCKET MOTOR (ASRM)

166 The abort solid rocket motor subsystem, consisting of two solid rocket motors attached to the orbiter aft fuselage, provides the rapid start and high thrust necessary to successfully accomplish orbiter separation from the booster SRM's and external tank subsystem in the event of an abort between 0 and 30 seconds from liftoff.

166 In an aborted mission the abort rocket motors are fired simultaneously, reaching peak thrust in 600 milliseconds, and burn for approximately 21 seconds with an average thrust of 385,000 pounds. The predicted thrust-time history is shown in FIGURE 4.7.2.1 At ignition, a 2.45 thrust/weight ratio is provided to accelerate the orbiter (with 65,000 pounds payload) away from the booster and external tank. After burnout, at approximately 12,000-foot altitude, the abort solid rocket motors are jettisoned. In a normal mission, the motors are jettisoned unused 30 seconds from liftoff.

166 The ASRM's are separated during a nominal mission (unused) or following an abort firing (expended). The nominal separation is accomplished by placing the SRM's at the rear of the configuration so that a simple release mechanism allows the vehicle to accelerate away. The nominal acceleration at this point in this trajectory gives adequate separation safety margins. The ASRM's will land approximately two nm downrange. The more difficult ASRM separation problem occurs when the abort rockets have fired to separate and maneuver the orbiter away from the tank and booster. For orbiter stability reasons, the ASRM's are separated immediately after burnout with the orbiter and ASRM's experiencing deceleration forces.

166 The abort solid rocket motors require separation under two conditions of aerodynamic loading and mass. In a normal mission, the unused abort solid rocket motors are jettisoned at a dynamic pressure of 300 to 400 lb/ft^2, which provides separation forces in an upward and outward direction with respect to the orbiter. In an abort mode, the motor weight is 8,100 pounds at burnout and the dynamic pressure is 700 to 800 lb/ft^2. Relative acceleration studies indicate complete separation and clearance between the abort solid rocket motors and orbiter in less than 1 second. Bolt-on fittings are provided to meet orbiter maintenance time schedule. Separation occurs at the clamp thrust interface, leaving only the attachment fittings on the orbiter. These fittings will be structurally loaded during orbiter structural testing. A triple redundant explosive separation clamp sized for worse-case conditions will assure separation.

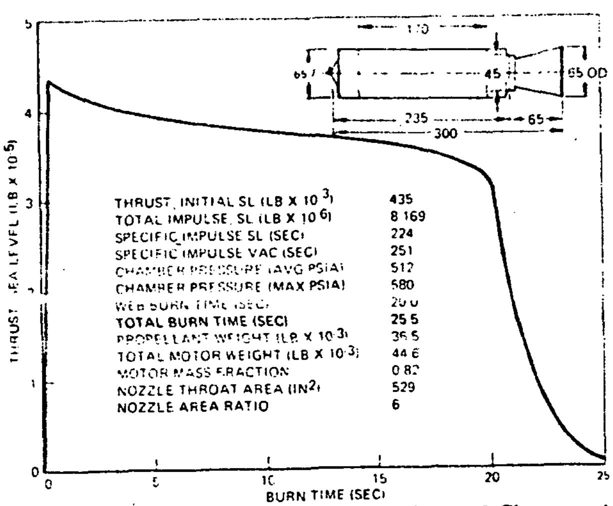


Figure 4.7.2-1 *ASRM Thrust Time Profile and Characteristics*

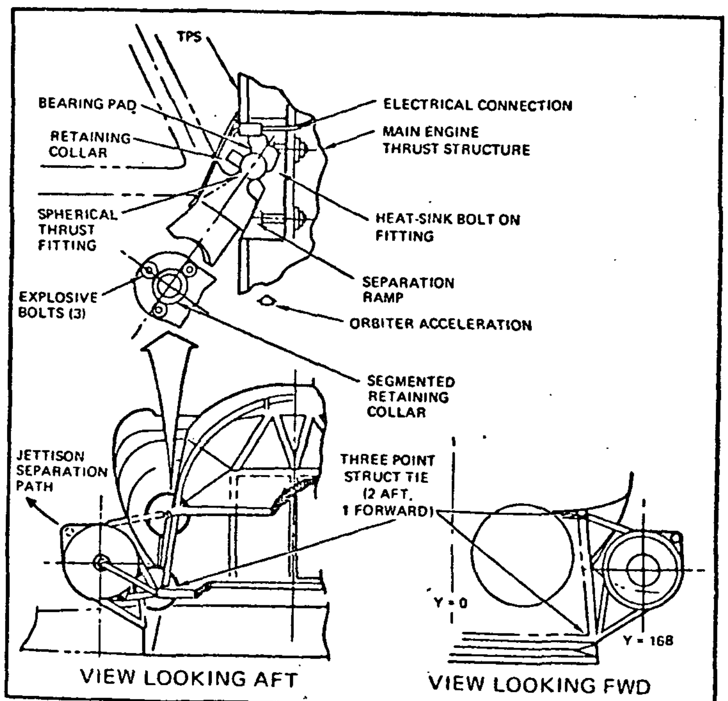


Figure 4.7.2-2 *ASRM Installation and Separation System Concept*

4.7.3 DEORBIT SRM FOR EXTERNAL TANK (ET)

166 Deorbit of the ET requires a 300-fps retrograde velocity increment to
171 effect ET fragment impact in the prescribed footprint in the Indian Ocean. This ΔV is achieved with a forward-firing solid rocket motor installed in the ET forward fairing and firing through a fragible nose cap. The motor is armed and fired by the ET avionics battery/sequencer subsystem following ET-orbiter separation. The deorbit motor design, as dictated by the ΔV requirement, has an average vacuum thrust of 18,500 pounds and provides a total impulse of approximately 686,000 lb-sec during its 37-second burn time. The preliminary motor design is approximately 37 inches in diameter, 76 inches in length, and has a total loaded weight of 2,600 pounds. The configuration contains 2,400 pounds of propellant and is based upon the modification (5-percent increase in propellant quantity) of an existing spaceflight-qualified rocket motor.

Table 4.7.3.1 DEORBIT SRM VACUUM PERFORMANCE

Thrust Avg.		18,500 lb.
Thrust Max		21,000 lb.
Burn Time Avg.		37.1 sec
Total Impulse		686,350 lb.-sec.
Specific Impulse		286.5 sec.
Pc Avg.		520 PSIA
Nozzle Area Ratio		40
Throat Area		19.35 in.
Throat Dia.		4.95 in.
Exit Dia.	I.D.	31.2 in.
C_f		1.837
γ		1.18
Case Dia.		37 in.
Length, Overall		75.3 in.
Wt. Loaded		2,610 lb.
Propellant Wt.		2,400 lb.
Burnout Wt.		210 lb.

The relative location of the ET Deorbit SRM is shown in FIGURE 4.7.3.1

F-398-8-A

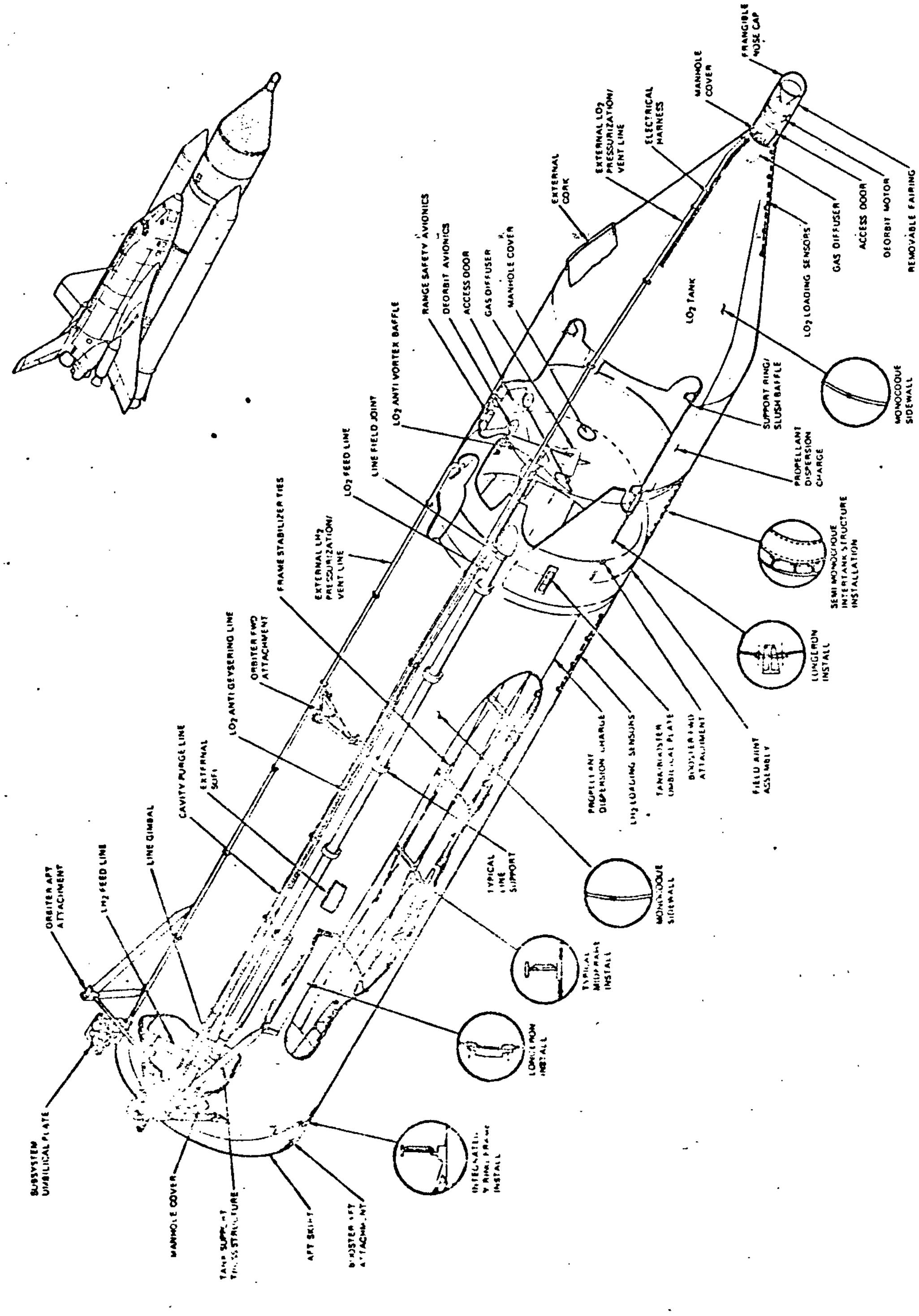


Figure 4.7.3-1 External Tank Design

4.7.4 RATIONAL

Not required

4.7.5 REFERENCES

166 Pages 3-151 to 3-153

171 Pages 2-59to 2-66

allow a 15-percent-over-mission load condition on the landing system by increasing the main gear tire pressure to 250 psi and maintaining static tire deflection at 32 percent. The ferry/horizontal flight test condition provides the design criteria for the maximum tire load caused by the down-load due to elevon deflection for nose wheel liftoff. Main gear tire deflection at this condition is approximately 50 percent. The ferry condition also defines the criteria for turnover, spacing, and structural sideload caused by turns during taxiing. When designed to these conditions, the landing system is capable of drift landings with peak ground wind speeds to 35 knots from any azimuth.

Additional cost savings during the implementation and operational phases of the orbiter results from landing/deceleration system maintainability and servicing features. The main gear, nose gears, and deceleration chute are fully accessible for structural and system inspections and servicing by access through the gear doors and vertical stabilizer speed brake. Using standard access equipment, the main and nose gears can easily be inspected in the preventive maintenance apportionment time of 2.25 hours. With the speed brake open, the deceleration parachute backup structure is exposed and can be inspected visually within 0.50 hour. In order to meet the allocated hours for servicing, the deceleration chute is a module package which, for installation, requires only mechanical and electrical connecting. Gear lubrication and main and nose gear shock strut service is also accomplished through the open gear doors. Jacking provisions are provided at each strut to accommodate wheel/tire replacement, without removal of other components, in 1.2 hours. The main gear brakes are replaceable as a unit, are open on the upper aft side for ease of inspection, and can be replaced within the allocated maintenance time.

Tow attachments to allow towing on slopes to 5 percent are provided on the front and rear of each gear, and are positioned to prevent towing equipment interference.

During the development of the landing and deceleration system, full consideration will be given to meeting the orbiter maintainability and turnaround time requirements, discussed further in Section 2.4.

SD will define, develop, certify, and deliver for the orbiter a landing and deceleration system which provides for all phases of shuttle operations. This results in the minimum launch weight impact and simplest direct acting system, using existing equipment where possible to minimize cost and schedule time and risk.

The use of existing equipment requires on-orbit conditioning of the main and nose gear wheel wells to maintain the exposure temperature within —60°F to 275°F. This is accomplished by use of a combination of electric heaters and passive insulation, described in Section 3.1.3. Deceleration parachute thermal protection also is described. The landing/deceleration system will be developed and space-rated by conducting environmental testing where necessary. Certification by similarity to existing hardware will provide cost savings to the program. Thermal vacuum testing of nonmetallic components, such as tires and lubricants, is proposed. Landing gear simulators are proposed for the conduct of operational and endurance tests. Static ultimate load and fatigue tests will be conducted as part of the structural test program. Final qualification testing of the deceleration chute, nose gear steering, and brake systems will be accomplished during taxi as part of the horizontal flight test program. The landing system test requirements and program are discussed in detail in Volume 5.

The orbiter mission will expose landing system components to long-term space environment for the first time, requiring space rating of this equipment. SD will apply its Apollo CSM experience and analysis capability on long-term space environmental exposure affects to the development of certified equipment and to a reasonable certification program.

The landing/deceleration system proposed is based on the Phase B study. No problem areas not previously solved by SD on Apollo CSM or by NR on its many aircraft systems are anticipated.

3.1.3 THERMAL PROTECTION AND CONTROL (WBS 1.3.1.3)

The thermal protection and control system consists of two elements. One, the thermal protection system (TPS), is external to the primary structural shell of the vehicle. It maintains the airframe outer skin within acceptable temperature limits during the vehicle mission. Where internal vehicle special compartments or areas require additional thermal control, the external TPS is augmented by a second element; the internal thermal control system (TCS). Overall vehicle thermal design also is included in the ECLSS, discussed in Section 3.5, the purge and vent subsystem, discussed in Section 3.1.1.5; and other subsystems. TPS and TCS design, along with the integrated vehicle thermal analysis, are discussed in this section.

SD 72-SH-50-3

3.1.3.1 TPS DESCRIPTION AND FEATURES

The baseline TPS (FIGURE 3-25) consists of: (1) ceramic reusable surface insulation (CRSI) (ceramic panels with an external waterproof coating on a strain-isolation foam pad) directly bonded to the airframe in areas exposed to surface temperature between 650°F and 2500°F; (2) elastomeric reusable surface insulation (ERSI) directly bonded to the airframe in areas exposed to temperatures below 650°F, and (3) reinforced carbon-carbon (RCC) material in the wing leading edge and body nose cap in areas exposed to temperatures above 2500°F.

3.1.3.2 CERAMIC RSI

Candidate ceramic insulation material systems basically consist of mullite or silica. SD's rationale supporting the use of mullite as the baseline TPS material is presented in the Discussion Item 2 response. SD plans to review the RSI material candidates after ATP and make the final selection by PRR. Four basic elements of the mullite CRSI system are reviewed as follows:

- Mullite panels — A low-density insulative composite material formed by coating a matrix of mullite fibers rigidized with an aluminum-boria silica refractory glass binder. The panel and pad (FIGURE 3-25) dimensions are determined by the thermal/structural analyses
- PD-200 pad — A chemically foamed methyl-phenyl silicone elastomeric material. This pad provides strain isolation of the CRSI from the aluminum structure and accommodates local surface irregularities. Its outer surface design temperature is 650°F, determined by the allowable bond temperature. The inner surface design temperature is 350°F. This temperature was selected to yield the lowest TPS and aluminum primary structural weight. Early analyses indicate that by increasing the bottom and chine pad thickness by 0.2 inch, the ablative characteristics will provide a ceramic panel loss fail safe entry. This will be substantiated by further analyses and tests.
- SR-2 coating — A waterproof ceramic coating fired at 2500°F on the top and sides of the panel. This coating is chemically compatible with and similar in expansion coefficient to the mullite insulation. The coating provides the necessary thermal control optical characteristics, rain erosion protection, and abrasion resistance for ground handling and atmospheric flight.
- RTV-560 adhesive — A silicone elastomer room-temperature-cured adhesive system used for both panel and pad bonding.

The RSI system is structurally sized considering the effects of airframe structure with a general thermal/structural analysis method for multi-material orthotropic elastic bodies. It is based on finite element, direct stiffness methods for computing displacement stresses and strains thru the substructure adhesive, pad, panel, and coating. This analytical technique has been verified and correlated with test results.

Panel-to-panel gaps (0.12 to 0.25 inch) are sized to avoid CRSI panel compressive loads at maximum expansion during entry. The gaps are partially filled with a low-density-quartz expandable gasket to thermally protect the substructure at the base of the joint. The height of the gasket is determined by thermal analysis to preclude material thermal degradation. A panel self-venting system is provided which allows venting to the boundary layer pressure through the panel gaps. It consists of a local interruption in the panel to PD-200 bond line (adjacent to the panel lower outer edge.) A silicone primer, applied to the lower panel surface, provides a water barrier while allowing venting of internal gases. This venting concept has been test-verified. Two CRSI test prototypes, mounted on simulated airframe structure and configured to two critical areas of the baseline system, have been successfully tested to withstand 100 orbiter thermal environment cycles. During the test series the prototypes were also subjected to a dynamic/acoustic energy spectrum of 163 db for the equivalent of 25 missions. See Section 2.2.4 for dynamic acoustic design requirements.

3.1.3.3 ELASTOMERIC RSI

An important feature is the use of an ERSI (ESM1004X) as the primary TPS on the orbiter upper surfaces where lower temperatures (<650°F) are experienced. Using an elastomer instead of a ceramic results in a TPS weight reduction of 3500 pounds. It is a flexible, open-cell structure material possessing good low-temperature flexural properties, and is attached to the airframe in coated sheets with RTV-560 bond. The ESM1004X is coated with an elastomeric silicone resin (for waterproofing) pigmented with titanium dioxide and carbon black (for thermal control). It is an impact-resistant, easily repairable material which will minimize the susceptability to handling damage.

4.8 External Tank Subsystem (ET)

166 The ET is a single assembly with integrated LO_2 and LH_2 tankage and structure. The ET is mounted in parallel below the orbiter and between the two SRM boosters. The configuration consists of structure, thermal protection system, main propulsion system tankage components, avionics, and mechanical components (Figure 4.8-1).

4.8.1 Structure

166 The ET structural design is illustrated in Figure 4.8-1. Both propellant tanks are constructed of 2219 aluminum monocoque skins with support frames. The design employs explosive bulge forming to form the individual bulkhead/cone gores. The skins and frames are butt fusion-welded together to provide reliable sealed joints.

166 The LO_2 tank aft skirt and the LH_2 tank forward and aft skirts use 2024 integral machine-milled skin/stringer structure stabilized with attached hydro-formed and chem-milled frames.

166 The ET to orbiter structural attachment (Figure 4.8-2) consists of one forward and two rear connections, through truss structures mounted to the LH_2 tank support frames and longerons. The mechanical release components are installed in the orbiter. The attachment between the ET and each SRM booster consists of one forward ball joint connection at the intertank longeron/frame juncture plus two links and a slide at the LH_2 tank aft skirt frame.

166 A field joint is provided at the intertank skirts by circumferential tension bolts. This joint provides tank assembly and tank shipment benefits.

166 Doors in the forward fairing and the LO_2 tank aft skirt provide access to interior installed equipment. In addition, manhole doors in each tank

166 provide manufacturing and field operations access for installing and maintaining equipment installed within the tank.

166 Saturn S-II type linear pyrotechnic explosive is installed along the sides of each tank for range safety propellant dispersion.

166 The ET is axially supported aft of the LO_2 tank by attachment of the canted SRM booster thrust cones to the ET ball joints on the intertank pitch axis.

4.8.2 Thermal Protection System (TPS)

166 Spray-on foam insulation (SOFI) is applied to the complete outer surface of the LH_2 tank, including the sidewalls and the end bulkheads. Sheet cork and the other high-density ablators are bonded directly to the outer surface of local structural areas (Figure 4.8-3). Sheet cork ablator is also bonded to a fiberglass substructure that is locally supported from the LH_2 tank aft bulkhead. The TPS coverage is minimized by using the heat sink provided by the ET sidewalls, SOFI, and propellants.

4.8.3 Deorbit Motor

166 Deorbit of the ET requires a 300-fps retrograde velocity increment to effect ET fragment impact in the prescribed footprint in the Indian Ocean. This ΔV is achieved with a forward-firing solid rocket motor installed in the ET forward fairing and firing through a frangible nose cap. The motor is armed and fired by the ET avionics battery/sequencer subsystem following ET/orbiter separation. The deorbit motor has an average vacuum thrust of 18,500 pounds and provides a total impulse of approximately 686,000 lb-sec during its 37-second burn time. The motor design is 37 inches in diameter, 76 inches in length, and has a total loaded weight of 2600 pounds. The configuration contains 2400 pounds of propellant.

4.8.4 Avionics

166 The ET installed avionics consist of two redundant primary batteries, sequencers and range safety receiver/decoders and antennas, development and operational flight instrumentation, and all interconnecting harnesses.

4.8.4.1 Instrumentation

166 The instrumentation used for propellant loading and tank pressure control during the mission consists of pressure transducers, temperature transducers, and liquid level sensors. Quad-redundant sensors are used throughout except for the optical type liquid level point sensors. The latter are mounted at the following volumetric levels at the walls in each tank: 2%, 3%, 10%, 20%, 30%, 40%, 50%, 60%, 70%, 80%, 90%, 97%, 98%, 99%, and 101% - with four sensors each located at the 1% and 100% levels. The 100% level is defined as the rated propellant load volume - exclusive of the 3% ullage volume. These sensors will be used during the fill operation to control the propellant fill rate. The liquid level sensors are used as part of the propellant utilization control system working in conjunction with engine controls.

Optical type liquid level point sensors located in the feed lines are used as control sensors during the LH_2 and LO_2 dump sequencing. These sensors are referred to as depletion sensors.

Tank pressure will be controlled to within $\pm$ 1.5 psi of the desired level during each phase of the mission by means of pressure control units located on the vehicle. After orbiter engine shutdown, these units will control the dumping of liquid residuals and will vent the tanks to 5 psi prior to separation.

4.8.4.2 SEPARATION

166 Since there is no communications link between orbiter and tank after separation, a simple combination timer, distance calculator, and attitude and attitude rate measuring unit is installed in the tank. This unit generates the signal to fire the retrorocket under normal operation mode and delays this signal in case of a malfunction where the distance/attitude condition for a normal firing time could create a hazard to the orbiter. It also generates the sequencing signals for the tumbling-enforcing jet system.

The geometric relations between orbiter and tank deploy tapes which remain attached to the orbiter during the separation maneuver and the relative velocity of their ejection from the tapereel tank provide for the direct determination of distances, attitude, and attitude rates.

The ET ordnance timing system performs the sequencing function which accomplishes retrorocket ignition. Prior to separation all power, control, and system monitor functions come from and are located in the orbiter. The post-separation system is fully autonomous (including its own power supply).

4.8.4.2.1 Electrical Power

166 No separate power system is required in the ET during operation up to separation. All power up to 2 sec prior to separation, and the power requirements for the ordnance system operation after separation are satisfied by its own batteries. The only additional requirement remaining after separation is for the operation of the timer/separation condition sensor unit. This low power requirement is provided by two dedicated batteries incorporated in the ET system.

166 Figure 4.8-4 illustrates the system to provide a separation sequence that accommodates both perpendicular and axial separation trajectories and meets all modes of separation from pad-abort to nominal mission orbital separation. Through use of the orbiter's existing ACPS (for orbital separation) and aerodynamic forces (for suborbital separation) no other active displacement system is required for ET separation anywhere in the mission.

4.8.4.2.2 Interface

166 Orbiter/ET interfaces requiring separation are grouped in two subsystems (LO_2 and LH_2) and three structural disconnect mechanisms. All of the subsystem's propellant transfer disconnects are composed of linear actuated poppet valves attached to umbilical plates on both the ET and orbiter sides of the interfaces. As the umbilical plates separate, the poppets are closed when the valves are disconnected. Because of differential motion between the ET, orbiter, and propellant lines resulting from thermal structural deflections, there is no rigid tie between the ET's umbilical and the ET. Retracting bungees are installed between the ET and its umbilical plates to retract and retain the ET's umbilical plates after release from the orbiter. Electrical plugs and receptacles are likewise attached to, and separated by, the orbiter and ET umbilical plates.

166 Three structural interfaces are arranged in a triangular pattern with two thrust interfaces aft. Side loads are reacted only at the left aft and forward attachments. All three interfaces react vertical loads. The geometry of the aft interfaces are such that the orbiter's MPS thrust forces are transferred to the ET by canted links approximately aligned with the MPS thrust axis. The forward structural attachment is comprised

166 of an A-frame truss canting aft from the ET with its apex having a pivoted shallow probe engaging a socket within the orbiter's lower mold-line. A stud within the probe engages jaws of a restraining latch attached to the orbiter. The canted links provide controlled displacement of the orbiter away from the ET during pad-abort. Perpendicular (orbital and suborbital) ET separation is readily accommodated. Mechanically actuated doors will subsequently cover all of the orbiter's aft ET interfaces. The forward interface will have a heat-sink and therefore will not require a door cover.

4.8.5 RATIONALE FOR ASSUMPTIONS

Not required

4.8.6 REFERENCES

166 - Pages 3-151 thru 3-154

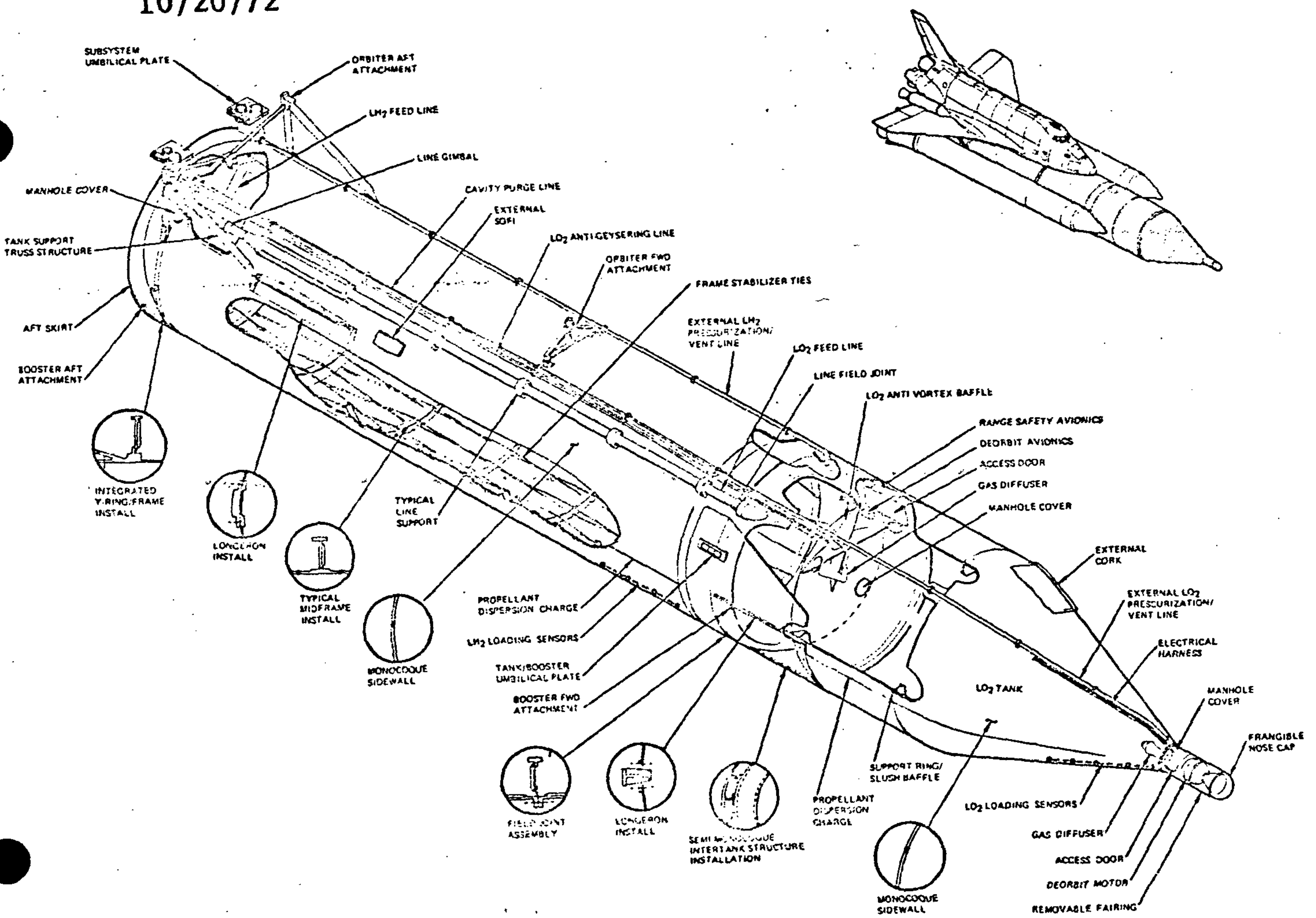


Figure 4.8-1 *External Tank Designed for Low Cost*

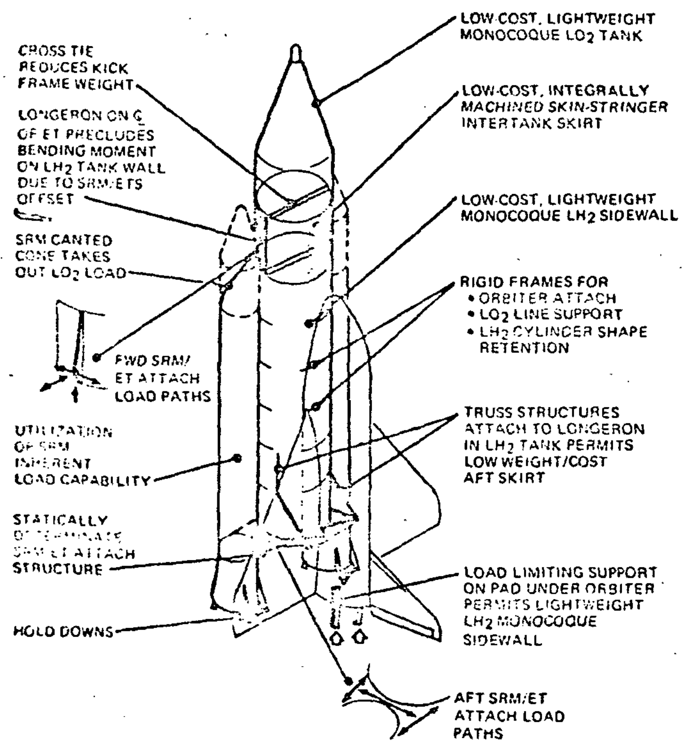
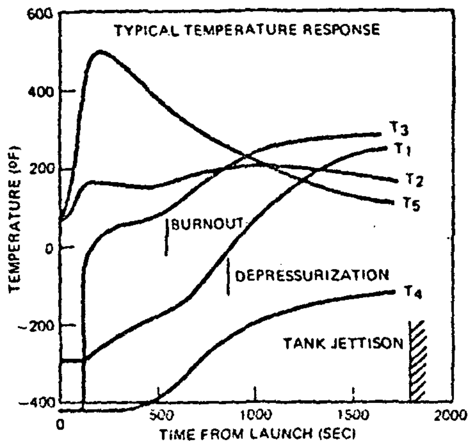
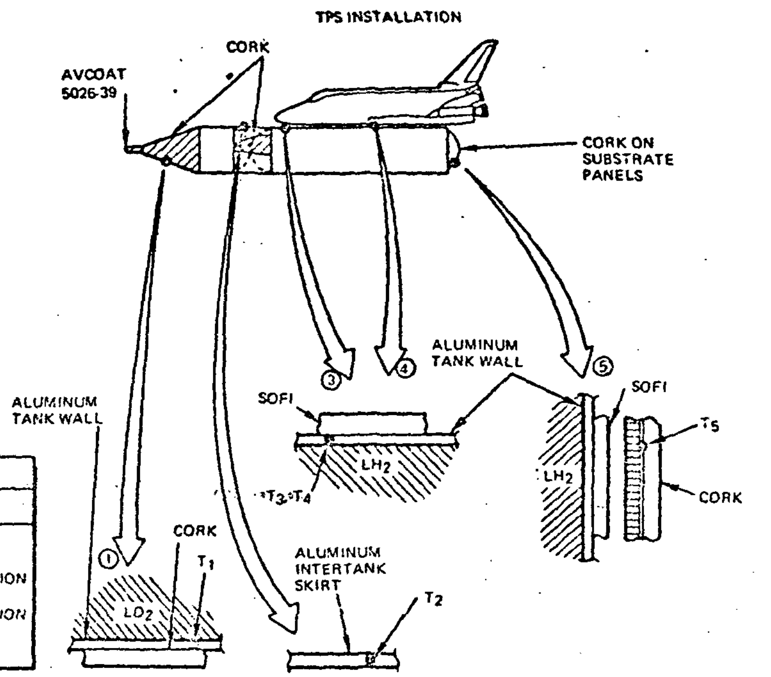


Figure 4.8-2 *External Tank Load Distribution*





THERMAL DESIGN CRITERIA			
LOCATION	TALLOW (°F)	LIMITATION	CRITICAL TIME
1	250	BOND	UP TO TANK JETTISON
2	400	STRUCT.	UP TO TANK JETTISON
3	210	STRUCT.	UP TO TANK DEPRESSURIZATION (0.17 ALUMINUM SKIN)
4	150	STRUCT.	UP TO TANK DEPRESSURIZATION (0.155 ALUMINUM SKIN)
5	500	BOND/STRUCT	UP TO BURNOUT

Figure 4.8-3 *External Tank Thermal Protection*

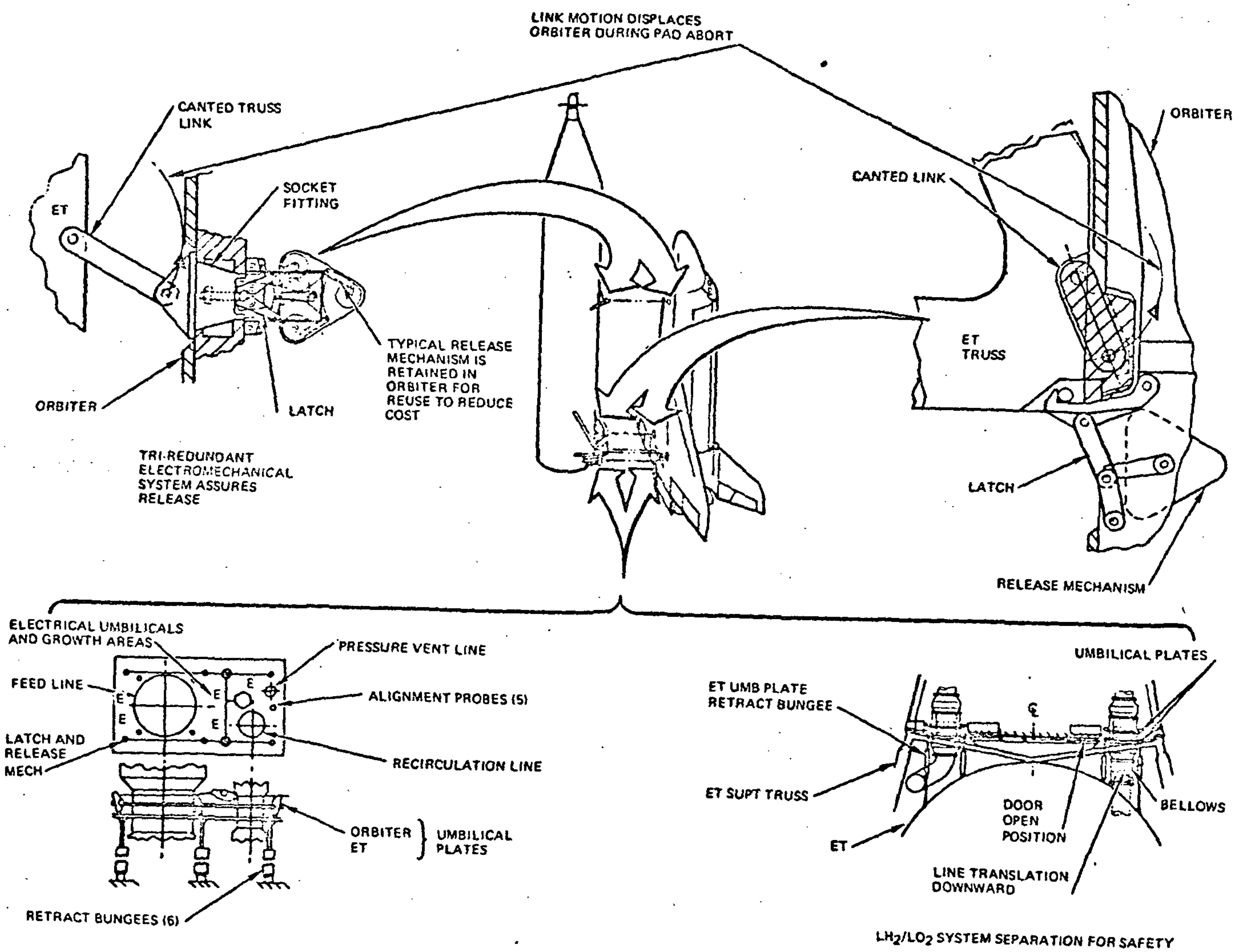


Figure 4.8-4 *Mechanical Separation* **Components**

Ref
Key

4.9 Guidance, Navigation, and Control (less computer)

The shuttle flight control system, given guidance commands
and data on vehicle dynamics from sensors, provides RCS firing commands,
ØMS gimbal actuator commands, MPS gimbal actuator commands, and aero-
surface actuator commands. The flight control system is divided into
primary and backup control systems. The primary GN&C subsystem
1 provides control for all flight phases, with both automatic and manual
modes. The primary system is present in three redundant strings, inter-
2 connected only at IMU output and output force servos and jet drivers.
During quiescent on-orbit periods, only one string is used. Flight
1 control loops used with the main engine TVC, ØMS TVC, and the RCS are
closed through the GN&C computer. Aerosurface control uses the computer
and the Aerodynamic Stability Augmentation System. (ASAS) An additional
manual aerodynamic control mode bypasses the computer and uses only
the ASAS. The ASAS is an F-14 type conventional analog system which
employs body mounted rate gyros and accelerometers. Gain scheduling is
provided by inputs from an air data system. The entry and approach/
3 landing primary control systems are illustrated on the following page:

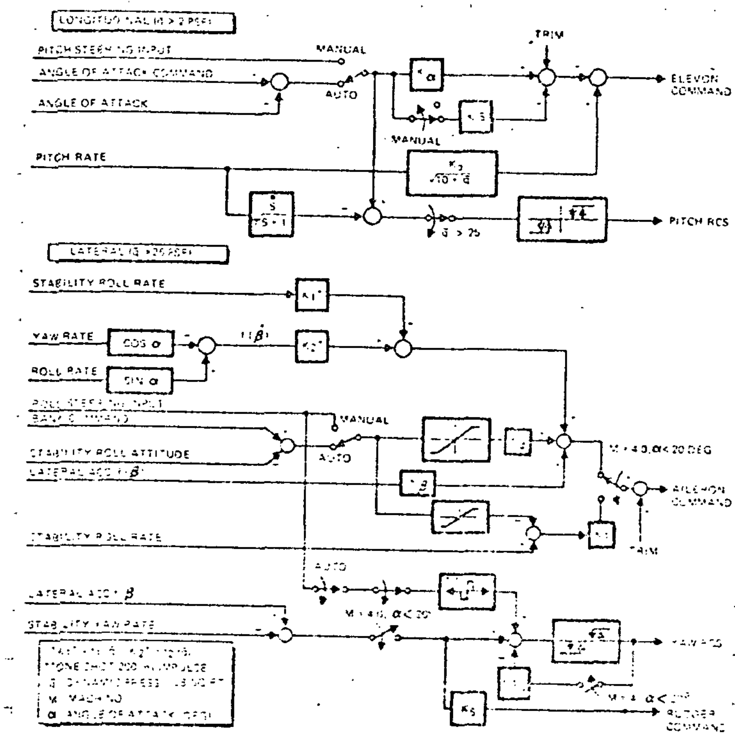


Figure 2-92. Entry Primary Flight Control System

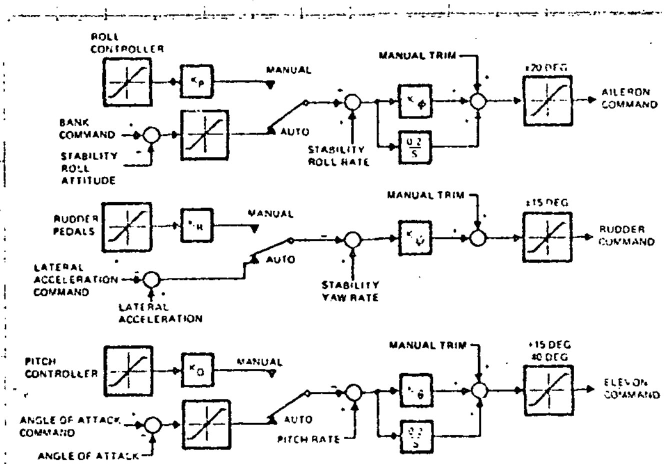
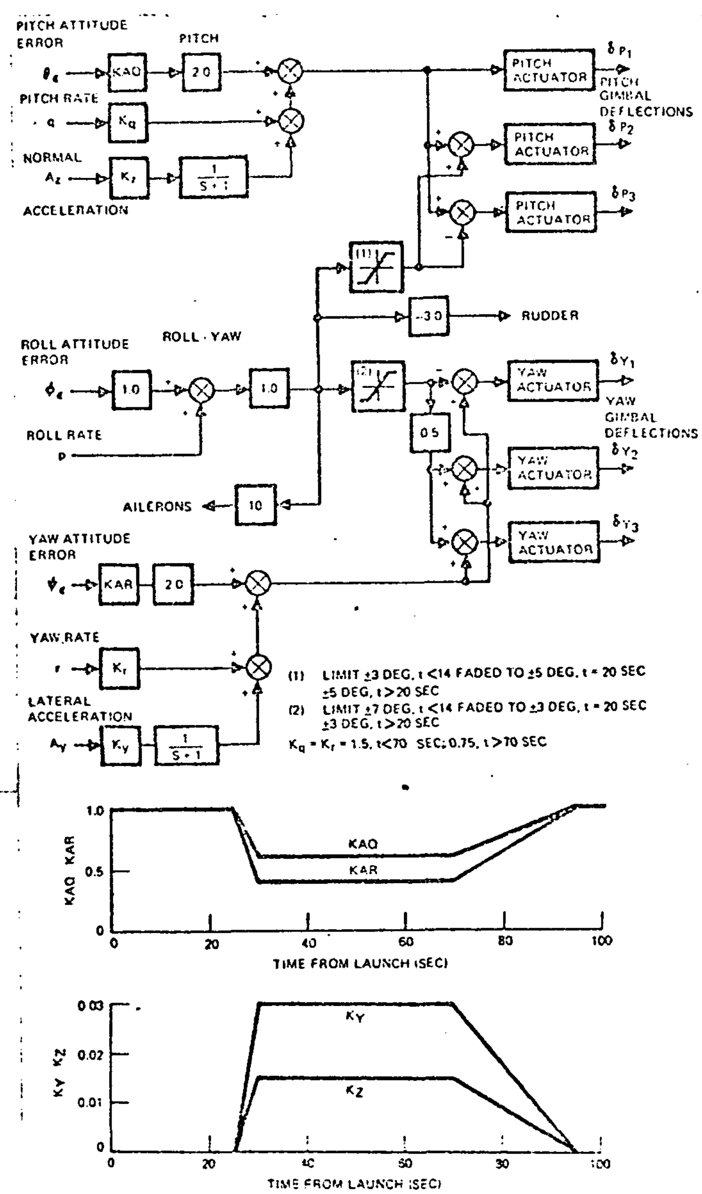


Figure 2-97. Approach/Landing Primary Flight Control

-398.8-A

Ref Key

4 The load relief control scheme used during periods of high dynamic pressure during boost uses both main engine TVC and aerosurface control. It is illustrated below:



Ref Key

5 On-orbit (non-thrusting) flight control effects attitude and translation maneuvers using RCS jets. Cross-axis coupling is eliminated by RCS jet select logic. Key features are:

1. independently selectable three-axis hold capable of accepting guidance commands.

2. independently selected three-axis translation control

3. selectable attitude accuracies of $\pm$ 0.5, $\pm$ 10, and $\pm$ 45 degrees, and minimum attitude rate of 0.1 $\frac{deg}{sec}$

4. minimum overshoot response to commanded inputs provided by rate feedback, and RCS select logic providing minimum propellant consumption and inhibiting opposing jet firings.

6 Attitude control during ØMS firing is achieved by gimballing the ØMS engines. In the case of single ØMS engine operation, RCS will provide roll control.

1 A backup GN&C subsystem is provided, which provides safe return capability for all flight phases. It is separate from the primary subsystem, using dedicated sensors and electronics. Backup control inputs are manual. Control modes are summarized below:

7

Mode	Flight Phase	Characteristic
Command	• Boost/insertion (TVC) Orbital (OMS TVC,RCS), Entry, Aero, Landing	• Crew initiates guidance and control modes and monitors—control automatic through GN&C computer
Control stick steering	• Orbital (OMS TVC or RCS), Entry, Aero, Landing	• Manual control and guidance displays through GN&C computer • Rate command, attitude hold, and RCS minimum impulse; RCS translation—acceleration command
Manual	• Aerodynamic	• Manual control through ASAS Rate command and damp
Manual backup	• [illegible]	• [illegible] computer. Rate command and damp • [illegible]

Ref
Key

1 Crew steering information is provided on a cockpit CRT. Display
8 information is processed through a Modular Display Electronics (MDE)
9 unit. The backup entry and approach/landing control uses a simple SAS rather than direct control to improve handling qualities. The scheme is shown below:

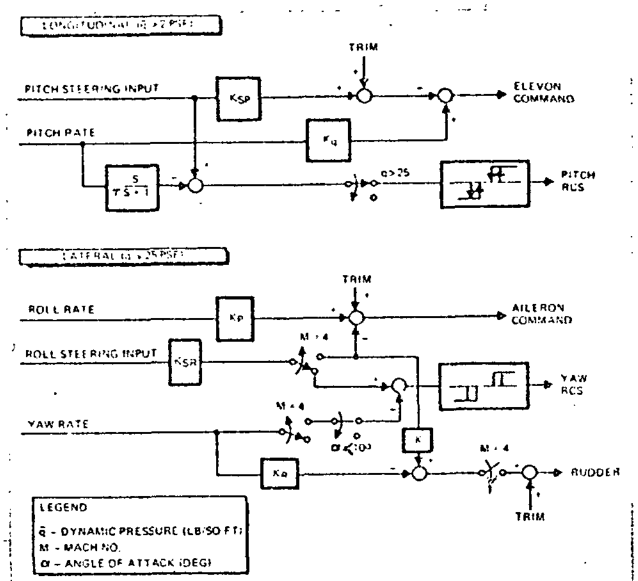


Figure 2-94. Entry Back Up Flight Control System

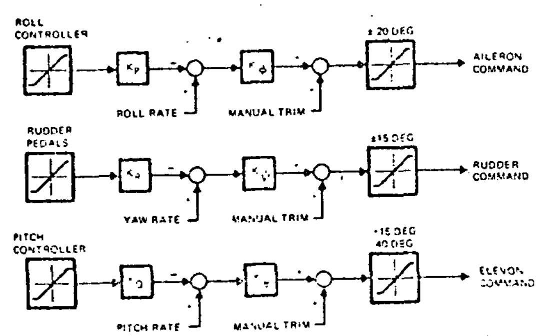
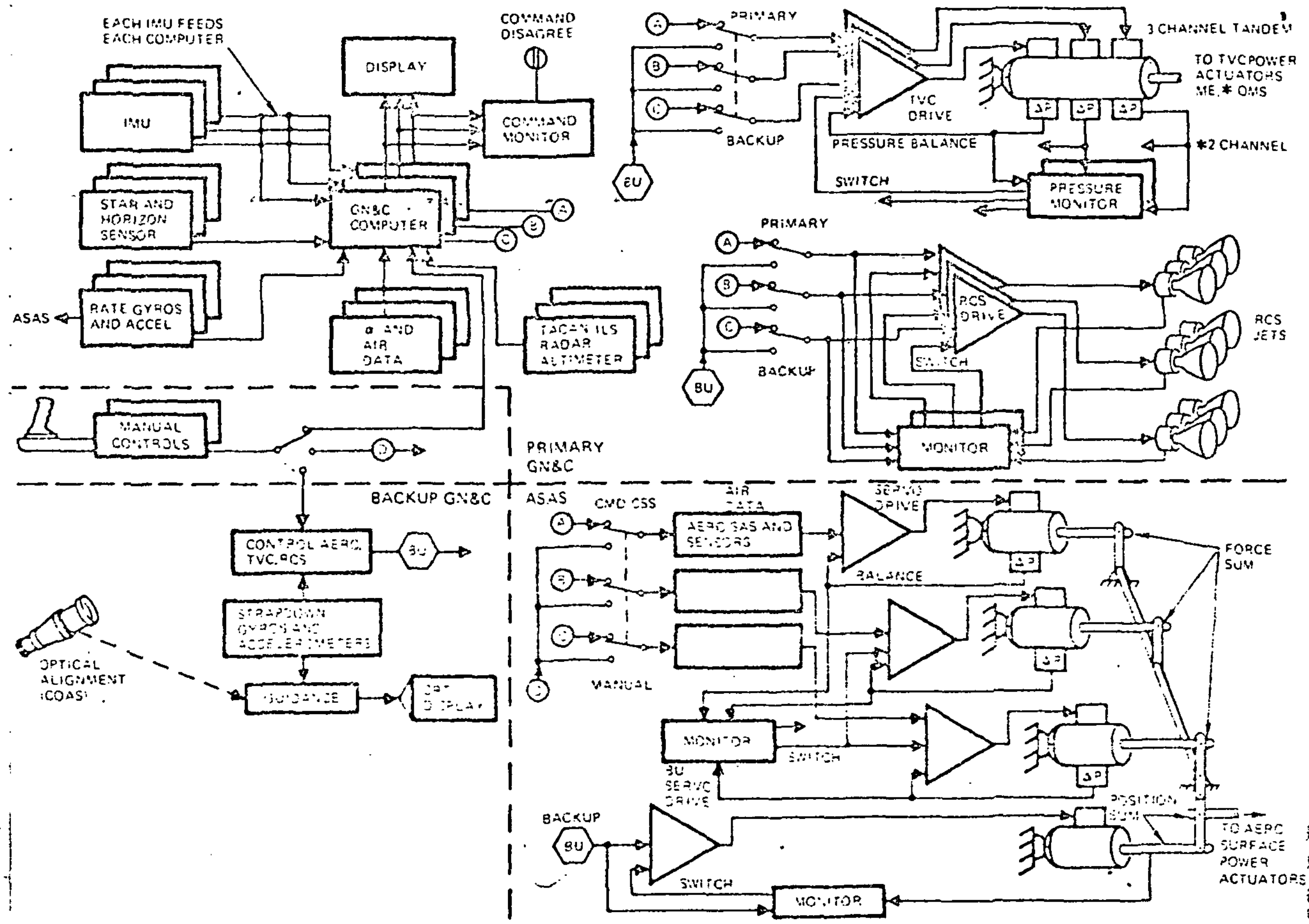


Figure 2-98. Approach/Landing Back Up Manual Flight Control System

Ref
Key

7 The full GN&C system is illustrated below:

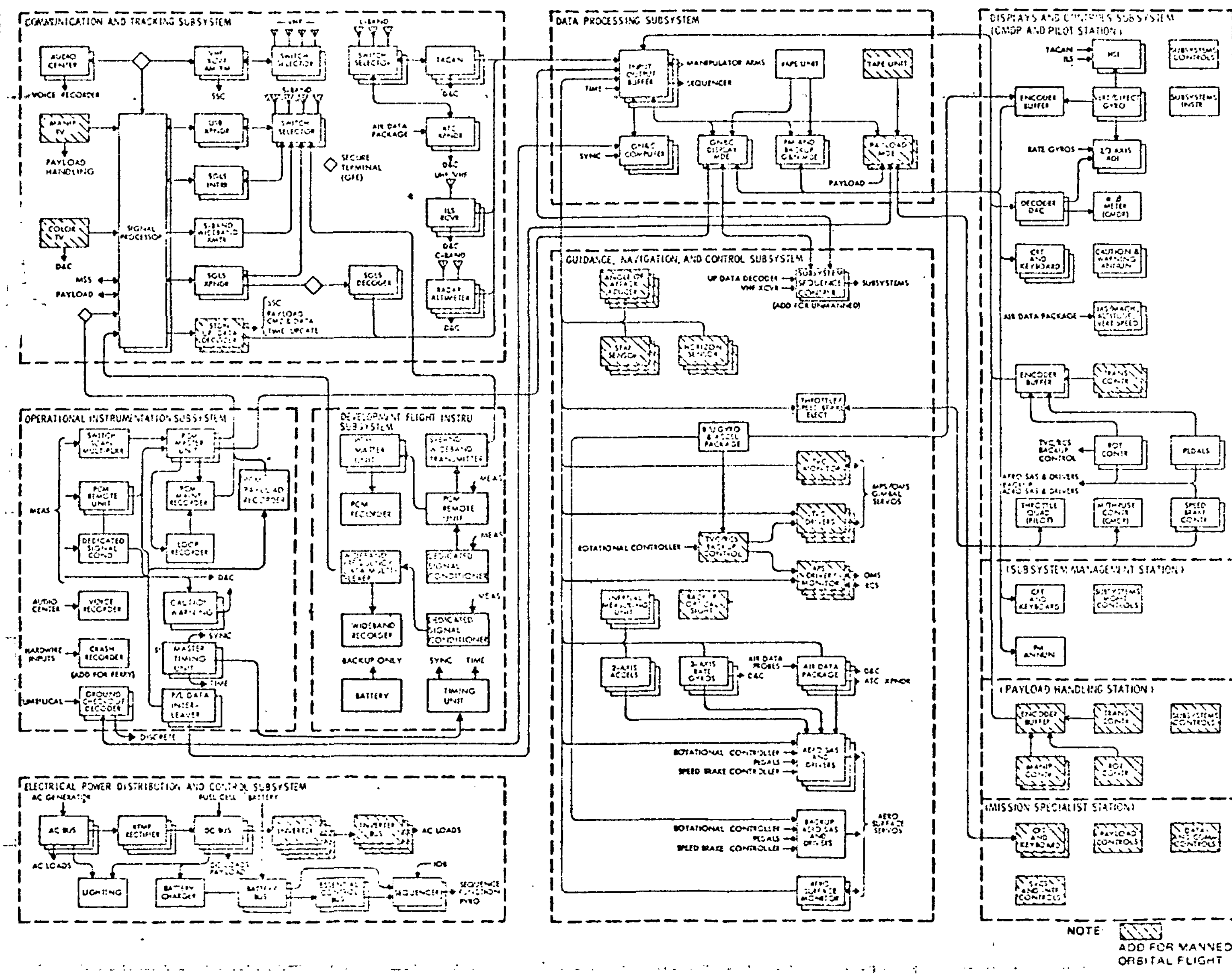


10 Control System, Line Replaceable Units (LRU's), are as follows (not including GN&C computers or their interface elctronics):

TVC Monitor
MPS TVC Drivers
ØMS/TVC Driver Unit
APS Driver/Monitor
Throttle/Speed Brake Electronics
Manual TVC/RCS Control
Aero-Control Electronics Unit
Aero-Backup Electronics

Ref
Key

11 Interfaces with other systems are illustrated below:



Ref
Key

2,7 Provision is made throughout the system for failure recovery. Redundancy management and failure provision are summarized in the tables below:

Table 3-24. Phased Buildup and Redundancy Management Influence GN&C Packaging

Function	Unit Packaging	Rationale
Jet select logic and drivers	• Logic in GN&C computer • Drivers in 3 units, dual power supplies	• Allows flexibility for redundancy management for dual failure • Provides spatial diversity, minimizes number of drivers to provide FO-FS
Backup GN&C	• Dedicated gyro, accelerometer units • Dedicated aero-control electronics units • Dedicated RCS, TVC unit • Separate computation in MDE processor	• Allows backup GN&C to be functionally independent • Delays RCS/TVC implementation to first vertical flight. FHF qual integrity retained by avoiding rework for FVF • Delays implementation to FVF • Protection against primary system single point software failures
TVC gimbal drivers	• Three units, dual supplies. Each drives one redundant actuator in each engine, primary or backup control	• No gimbal loss after two failures. Delays implementation to FVF
Aero-control electronics	• Three units. Each dedicated to one level of redundant actuators on each surface, primary, or backup control	• Fail operational after one failure. Detection capability remains for transfer to backup. Delays driver avionics cost.

FVF—first vertical flight; MDE—modular display electronics; FHF—first horizontal flight

Table 3-26. GN&C Redundancy Management Considerations

Consideration	Trade Studies	Implementation and Solution
GN&C redundancy level	Redundancy level versus program costs (aborts, hardware, operations, weight, power, and cooling)	FO-FS best. Triplex primary system with unlike backup path
Hardover aerosurface/ TVC failures cause vehicle loss (allowable detect and correct times 0.5 sec, much less than pilot reaction capability)	Position sum actuators, operate-standby actuators, and force sum actuators	Use force sum actuators with simple pressure monitors to switch per conventional aircraft practice to "fail soft"
Protection against single-point GN&C software failure	Cost of exhaustive software C/O against cost of unlike separate 4th string	Use unlike "get home safe" GN&C 4th string
False failure detections caused by string command divergence from control law integrations and guidance sensitivities	Cross-strapping IMU's and computers or cross-strapping IMU's only	Cross-strap IMU: allows simple mechanization

Table 3-27. GN&C Failure Recovery Sequences

First Failure		Second Failure		
Element	Detection and Correction Sequence	Element	Detection and Correction Sequence	GN&C Status
IMU	A	IMU	E	Safe
		Navigation aids, computer	B	Operational
		Drivers, servos, sensors	C	Operational
		Backup channel	D	Safe
Navigation aids computer	B	IMU	A	Operational
		Navigation aids, computer	F	Safe
		Drivers, servos, sensors	G	Safe
		Backup channel	D	Safe
Primary drivers, servos, and sensors	C	IMU	A	Operational
		Navigation aids, computer	F	Safe
		Drivers, servos, sensors	G	Safe
		Backup channel	D	Safe
Backup	D	IMU	A	Safe
		Navigation aids, computer	B	Safe
		Drivers, servos, sensors	C	Safe

Detection and correction sequence

A. Detection: Software compare in GN&C computers
Correction: Software midvalue select in GN&C computers

B. Detection: Hardware compare in command monitor and in pressure-driver monitor
Correction: Manually disengage failed channel, or pressure-driver monitors automatically disengage failed channel

C. Detection: Hardware compare in pressure-driver monitors
Correction: Fail-soft by force fight-automatically disengage failed channel

D. Detection: Servo monitor (aero), CRT (backup guidance), BITE
Correction: None required for space since normally disengaged; auto disengage for aero

E. Detection: Software compare in GN&C computers
Correction: Manual switch to backup automatic (software freeze on last inertial data point on discompare); if isolated by BITE, option to revert to primary

F. Detection: Hardware compare in command monitor and in pressure-driver monitor
Correction. Manual switchover to backup GN&C, or pressure-driver monitors automatically switch to backup GN&C

G. Detection: Hardware compare in pressure-driver monitor
Correction: Fail-soft by force fight and automatically engage backup channel; if isolated by BITE, option to revert to primary channel

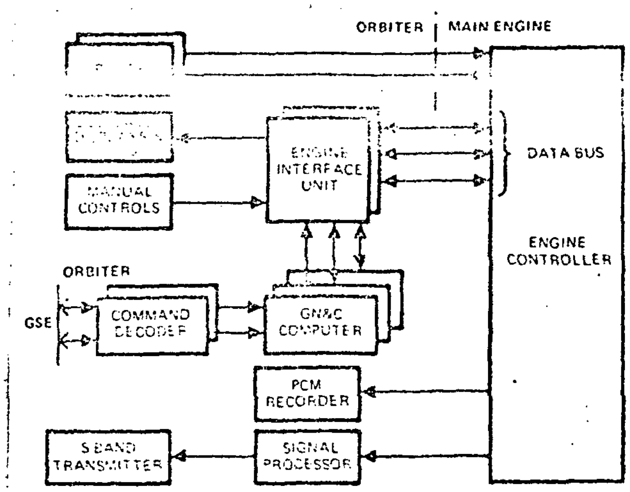
Note: The detection and switching techniques used cover the different combinations of failures; the major ones are described above.
BITE—[illegible]

Table 3-28. Redundancy Management Key Features

- Redundant strings interconnected only at IMU output and at the output force servos and jet drivers to maintain design simplicity. IMU interconnection prevents divergence in guidance computations and allows detection of slow degradation failures. Servo and jet driver interconnection (via monitor) required for "fail-soft."
- RCS engines divided into 3 groups electronically, each with dedicated GN&C computer and capable of doing all required maneuvers.
- Hydraulic pressure monitors in the main engine TVC and aerosurface actuators automatically disengage a failed channel for first failure and switch to backup for second. OMS TVC is not time-critical and does not require a monitor.
- When the backup system is engaged, there are no significant transients requiring initial corrective pilot inputs. Backup is rate command system that damps out failure transients automatically with rotational controller centered.
- Steering display command monitor detects guidance loop discrepancies for display; no automatic action is initiated by this monitor.
- Built-in test and self-test furnish LRU status to crew but are not used to automatically manage redundancy.
- After a second failure, if the failed string can be isolated using BITE, control may be restored to the remaining good string at crew option.

Ref Key

12 Main engine control signals are processed through the Engine Interface Unit, described in 4.3.2. Main engine gimbal actuators are mechanized to drive to a null position if two hydraulic systems are lost, two hard-over servo valve failures occur, or if two electrical signals are lost. Hydraulic pressure monitors in the main engine TVC
2 actuators automatically disengage a failed channel upon the first failure and switch to backup upon the second. Main engine control interface is illustrated below (one per engine):



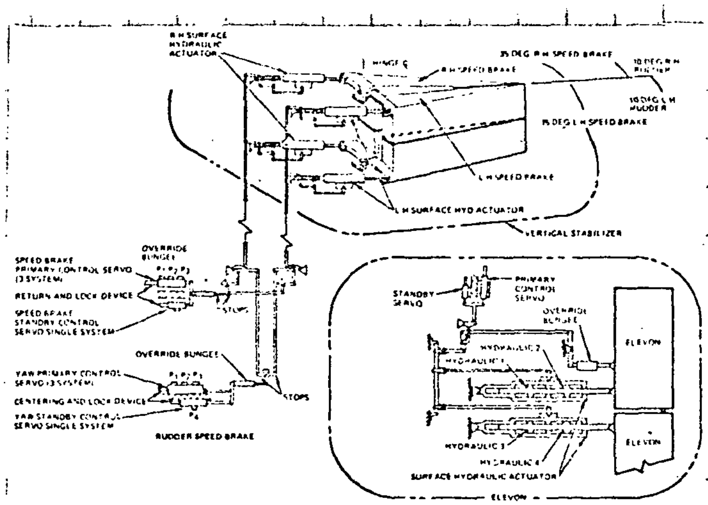
6 ØMS basic gimbal design technique is the same as that employed on the Apollo CSM. It employs electro-mechanical actuators incorporating redundant drive mechanisms. RCS thrusters will be activated by a fuel
14 operated valve, controlled by an electrically operated valve. RCS engines are electronically divided into three groups, each with dedicated
2 GN&C computer and capable of doing all required maneuvers.

Ref Key

15 The aero-surface control system receives its input from the three-channel primary control servo and a single-channel backup servo. These inputs are summed mechanically and coupled by linkage to the power actuator. Load limiting/override bungees are provided to prevent excessive loads in this linkage system for both power-on and the power-off droop condition. Two dual tandem irreversible surface actuators are connected to each of the pinned together elevons. The rudder panels serve dual functions being used for aero-control and speed brakes. Two dual tandem actuators are connected to each side of the rudder panel to provide this control. Hydraulic power for actuation is provided by the four independent vehicle systems to individual sections of the dual tandem actuator. Aero-surface hinge moment and rate requirements for reentry and landing are provided with any two hydraulic power sources operating.

2 Hydraulic pressure monitors in the aerosurface actuators automatically disengage a failed channel upon the first failure, and switch to backup

16 upon the second. The aero-surface control mechanisms are illustrated below:



Ref Key

17 Elevon deflection ranges from -40° to +15° (negative is trailing edge up). Rudder deflection is ± 15°, while speed brake deflection ranges up to 70°. Sensor LRU's, excepting those covered in section 4.10., are:

- IMU (including power supply)
- Star Sensor
- Horizon Sensor Assembly
- Rate Sensor Package
- Accelerometer Package
- Air Data Package
- Angle of Attack Transducer
- Gyro/Accel Package
- Backup Optical Unit

18,7 Three Inertial Measurement Units (IMU's) will be carried. The IMU's will be all-attitude, forced air cooled, gimballed devices, each weighing 47 lbs., and requiring 120 watts of power. IMU outputs are cross-strapped,

19 feeding each IMU output to each computer. Each IMU will output inertial

7 acceleration and attitude. IMU error sources are tabulated below:

Component	Characteristic	Equipment Capability (1σ)	Error Budget (1σ)
Accelerometer	Bias (μg)	25	100
	Scale factor (%)	0.0025	0.0150
	Input/Output axis misalign (sec)	10	40
Gyro	Scale factor (%)	0.04	0.10
	G-insens drift (°/hr)	0.03	0.05
	Mass unbalance input axis (°/hr/g)	0.1	0.1
	Mass unbalance spin axis (°/hr/g)	0.03	0.10
	Anisoelastic (°/hr/g²)	0.003	0.100
Gimbal	Readout (two-speed) (sec)	25	72

1,7.20 IMU's will be eapable of fine alignment using star trackers and/or horizon

18 sensors, and coarse alignment using the backup optical unit. Fine

7 alignment accuracy will be within 1 arc-minute. Three star trackers

18 will be carried. Star trackers will use image-dissector type detectors hard mounted to the spacecraft structure. They will be capable of observations within a fixed field of view; about 17° by 17°. Line-of-sight

Ref Key

motion within the field of view will be electronically tracked. Capability to track sunlit or beacon lit rendezvous targets will be included to provide rendezvous bearing. Star tracker on-axis random
7 error and bias will be within 30 arc-seconds (1σ). Three horizon sensors will be carried. Angular error (limited by horizon definition) will be within 6 arc-minutes (1σ). A Crewman Optical Alignment Sight (CØAS)
1,18,21 will be included. It is a manual sighting device mounted in the left window, oriented along the spacecraft +x axis. It will be capable of repeated removal, stowage, and remount without elaborate calibration. Sufficient accuracy will be provided to allow backup rendezvous tracking, IMU coarse alignment, and backup system alignment. The device will be similar to the Apollo and Gemini instruments. Instrument, alignment,
7 and limit cycle error is expected to be within 11 arc-minutes (1σ).
7,18 Three body-mounted rate gyros will be employed in each of the three GN&C Strings. Rate sensors will be conventional spring-restrained, single degree of freedom rate gyros. Individual rate gyro operational and status monitor signals will be transmitted for each gyro. These gyros will provide body rate data for normal control purposes, and will be located in the best location (s) to aid in suppressing body-bending deflections and load alleviation. Two body mounted accelerometers, mounted one each in the yaw and pitch axes, will be employed in each of the three GN&C Strings. These instruments will be of a spring-loaded, seismic mass type. Air data is provided by vehicle nose pressure
1 ports at high altitude and by redundant probes deployed at lower altitudes.

Ref
Key

A DC-10 type digital air data computer will be aboard. A strapdown

7 gyro/accelerometer package will be provided for the backup GN&C

system. Backup sensor accuracies are tabulated below:

	[illegible]	Equipment	Error
Backup sensor			
Gyro	Scale factor (%)	0.05	0.065
	Bias (°/hr)	0.25	0.50
Accelerometer	Scale factor (%)	0.05	0.10
	Bias (μg)	400	500

Ref
Key

RATIONALE FOR ASSUMPTIONS

not applicable

References

1.	166	p. 3-97
2.	166	p. 3-101
3.	166	pp. 2-70, 2-73
4.	166	p. 2-37
5.	166	p. 2-80
6.	166	pp. 3-69, 3-70
7.	166	p. 3-99
8.	166	p. 3-103
9.	166	pp. 2-71, 2-73
10.	184	p. G4-4
11.	166	p. 3-121
12.	166	p. 3-60
13.	166	p. 3-122
14.	166	pp. 3-63, 3-64
15.	166	p. 3-23
16.	166	p. 3-24
17.	167	p. 2-3
18.	21	pp. 28,29
19.	4	p. 9.2-3, 29, p. 9.2-17
20.	17	sect. 28, p. 8; 30 p.9.6-96
21.	17	sect. 27, p. 4

4.10 Communications and Tracking

There are three general systems to be used in conjunction with the Shuttle Mission. These are the Spacecraft Data and Tracking Network (STDN), the Space Ground Link System (SGLS), and the FAA Air Traffic Control Network (ATC).

Primarily the communication system uses existing equipment. The equipment of the STDN network is compatible to the equipment installed in the SGLS network. The SGLS network will provide secure voice and data networks for classified missions. The ATC equipment gives the shuttle landing capability at almost any large airport.

The Shuttle orbiter vehicle contains transponders for tracking and navigation, telemetry television command, and voice. Tracking, telemetry command and voice link (TTCV), television transmission, and payload data/voice channels are compatible to the Spacecraft Tracking Data Network in current operation. S-band equipment is provided for the Satellite Control Facility (SCF) TTCV by the Space-Ground Link System (SGLS). In addition, S-band equipment is provided for voice and data crypts and normal payload data and voice relay to ground. Navigation, tracking/landing aids, and communication equipment is provided for flight which is compatible to the Air Traffic Control (ATC) system currently in operation.

The communications system also controls antenna selection, audio and video processing equipment, and signal distribution controls. The subsystem flow diagram, showing general functional requirements and functional redundancy is shown in Figure 4.10.2-1.

10/20/72

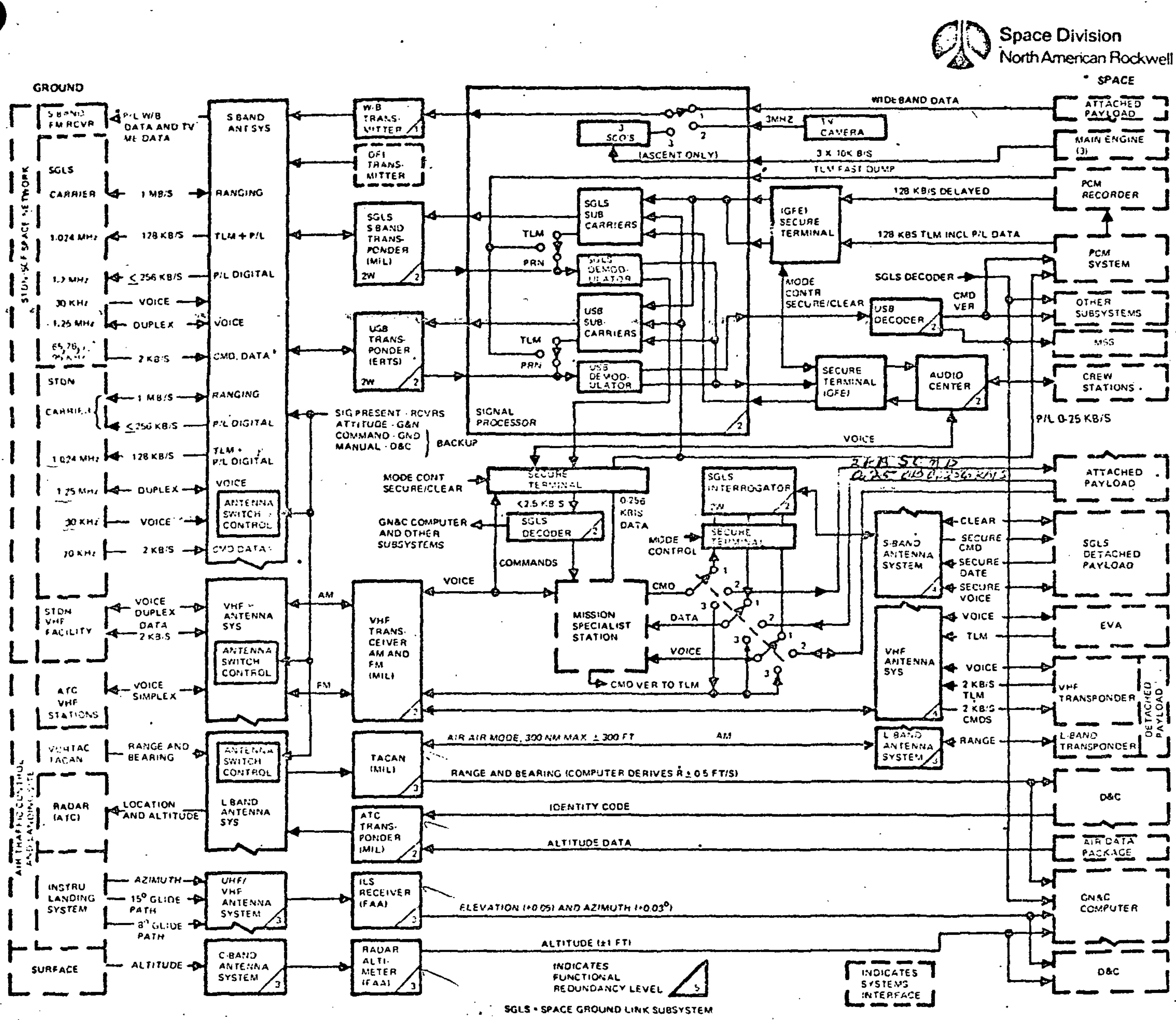


FIGURE 4.10.2-1

COMMUNICATION AND TRACKING SIGNAL FLOW

The only major change required for any currently operational communication system is to the glide slope transmitter located at the primary and alternate landing sites following a space operations.

The orbiter vehicle is equipped with antenna systems oriented for communication coverage at specific vehicle attitudes. Selectable antennas are used for increased geometric pattern coverage where required. Figure 4.10.2-2 shows the relative location of the antennas and the type (polarization) of the patterns. Attenuation of the radiated patterns is modified by body geometry as shown for each frequency band. Selection of the strongest signal/antenna is accomplished by automatic switching logic.

4.10.1. S-Band

The Shuttle vehicle uses three S-band frequencies for communication. There are two transmission frequencies (air to ground) of 2272.5 ($\pm$2.5) mHz and 2287.5 ($\pm$2.5) mHz. The receiving frequency is 2106.4 ($\pm$2) mHz.

The on-board receivers have a nominal receiver sensitivity of 97 db. On-board transmission line losses are estimated at -10 db from the receiver/transmitter to the antenna. The on-board transmitters have a TBD watt power capability. With this power the air-to-ground link has complete coverage with no signal attenuation where line-of-sight between the ground station and shuttle exists.

S-Band OMNI Antennas

There are four right hand circularly polarized, flush mounted helix antennas, located at intervals about the circumference of the orbiter. The omni antennas provide near-earth communications. The antennas are utilized individually and are selected by S-Band Antenna

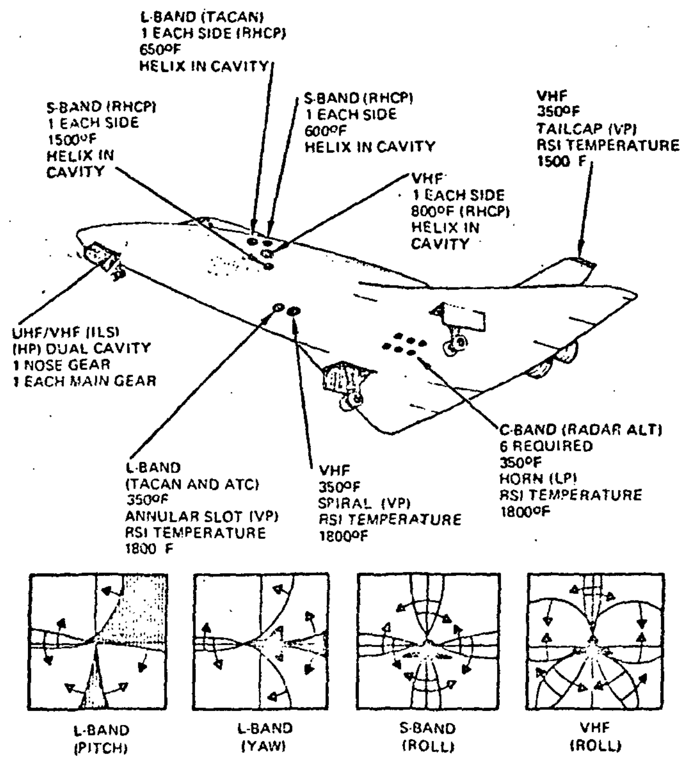


FIGURE 4.10.2-2

COMMUNICATION COVERAGE in all ATTITUDES

switches. Figure 4.10.2.1-1 is a typical antenna pattern for any one of the 4 OMNI antennas.

MSFN Unified S-Band Stations

Table 4.10.2.1-1 gives the Call Letters, Names, and Lat.-Long. locations of the S-Band ground stations. Goldstone, Madrid, and Canberra stations are equipped with 85 foot "Dish" antennas with 51 db gain. The remaining stations have 30 foot antennas, with 43 db gain. Power output of all the stations is variable from 1 to 10KW.

4.10.1.1 S-Band Voice

Duplex (simultaneous two-way) voice operated relay (VOX) voice communication is provided for crew members headsets. Normal voice duplex communication is provided by transmission of a phase modulated 2287.5 mc carrier -1250 kc subcarrier, and reception of a phase modulated 2106.4 mc carrier -30 KC subcarrier. Backup voice communication for down voice is provided by transmission of a direct phase modulated 2287.5 mc carrier. Backup voice communication for up voice is provided by using the up-data 70 kc subcarrier of the 2106.4 mc carrier.

4.10.1.2 S-Band T/M

Digital telemetry data will be by pulse code modulation (PCM). The high data bit rate of ≤ 256 Kbs is provided by direct phase modulation of the 2287.5 mc carrier. The low data rate of 128 Kbs is provided by transmission of a 2287.5 mc carrier-phase modulated by a 1024 kc subcarrier.

Analog data transmission is provided by a 2272.5 mc carrier using subcarrier frequencies of 65, 76, and 95 Khz. All data for these subcarriers undergoes crypto secure processing prior to multiplexing.

On-board data recorders are provided for PCM data for later transmission to a ground receiving station.

4.10.1.3 Command Data

The Shuttle has provisions for accepting digital encoded commands. The up-data link for this use has a maximum data rate of 2 Kbs using the 2106.4 mc carrier, phase modulated by a 70 kc subcarrier.

4.10.1.4 Video

Television transmission is provided by an analog video signal directly frequency modulating a 2272.5 mc carrier.

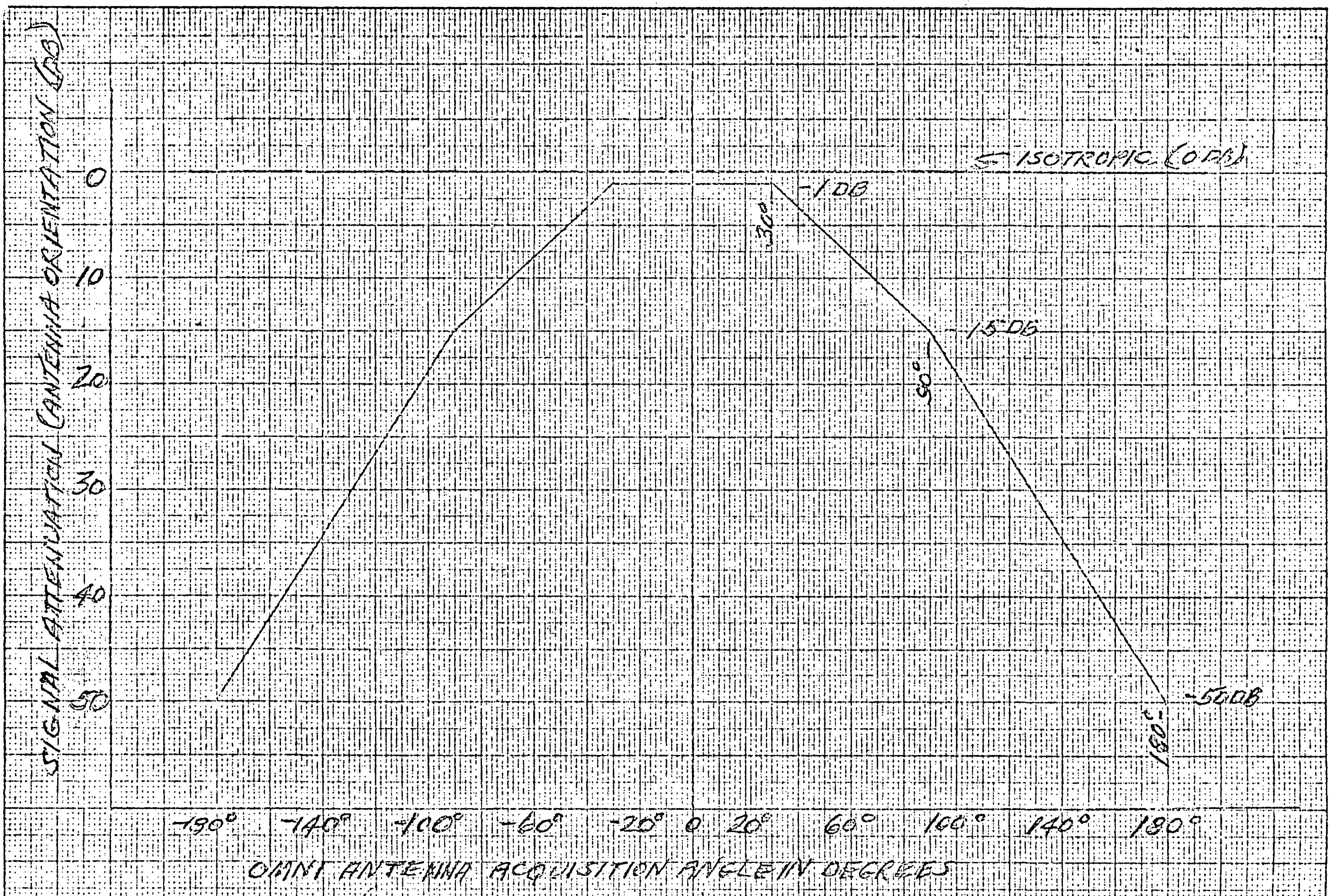
4.10.1.5 Wide Band Data Link

A wide band transmission link is provided for payload or main engine data (as applicable) in lieu of the video channel. Note: Lower response payload data will be transmitted as a part of the standard T/M data.

4.10.1.6 Range Measurement

The S-band equipment provides the capability for Doppler tracking and pseudo random noise (PRN) ranging by receiving and transmitting in phase coherence on the 2106.4 mc carrier.

10 X 10 TO THE CENTIMETER 46 1512
18 X 25 CM. MADE IN U.S.A.
KEUFFEL & ESSER CO.



10/20/72

10/20/72

Ship 1	Insertion ship S-band
MIL-71	Merritt Island S-band
BDA-02	Bermuda S-band
CYI-04	Grand Canary S-band
ACN-75	Ascension S-band
CRO-08	Carnarvon S-band
GWM-24	Guam S-band
HAW-12	Hawaii S-band
MAD-23	Madrid deep space
HSK-25	Canberra deep space
GDS-28	Goldstone deep space (85 ft)
PKS-96	Parkes deep space
TEX-16	Texas S-band
MAR-95	Goldstone deep space (210 ft)
STN-00	Stanford, California, S-band

10/20/72

TABLE 4.10.2.1-1

STDN station characteristics

Geodetic latitude, LATR, deg
Longitude, LONR, deg
Altitude, ALTR, ft
Range capability, SRANGE, n. mi.

Keyhole, FTINDC: 0 = none
1 = north-south
2 = east-west

```
RADAR =MAD-23  X-Y USB-85, LATR =  40.454991, LONR = 355.830159
 ALTR = 2503.276, SRANGE = 850000.0, FTINDC = 2.0
RADAR =HSK-25  X-Y USB-85, LATR = -35.583494, LONR = 148.976421
 ALTR = 3749.993, SRANGE = 850000.0, FTINDC = 2.0
RADAR =GDS-28  X-Y USB-85, LATR =  35.341594, LONR = 243.125015
 ALTR = 3021.648, SRANGE = 850000.0, FTINDC = 2.0
RADAR =PKS-96  H-D USB21n, LATR = -32.998769, LONR = 148.263517
 ALTR = 1322.176, SRANGE =2100000.0, FTINDC = 0.0
RADAR =MAR-95  H-D USB21n, LATR =  35.425958, LONR = 243.110517
 ALTR = 3169.286, SRANGE =2100000.0, FTINDC = 0.0
RADAR =STN-00  A-E USB15n, LATR =  37.409722, LONR = 237.822083
 ALTR =  525.000, SRANGE =1500000.0, FTINDC = 0.0
RADAR =MIL-71  X-Y USB-3n, LATR =  28.508272, LONR = 279.305142
 ALTR = -177.165, SRANGE = 300000.0, FTINDC = 1.0
RADAR =BDA-02  X-Y USB-3n, LATR =  32.351250, LONR = 295.340814
 ALTR = -141.075, SRANGE = 300000.0, FTINDC = 1.0
RADAR =CYI-04  X-Y USB-3n, LATR =  27.764536, LONR = 344.346945
 ALTR =  587.269, SRANGE = 300000.0, FTINDC = 1.0
RADAR =ACN-75  X-Y USB-3n, LATR =  -7.954794, LONR = 345.671554
 ALTR = 1728.999, SRANGE = 300000.0, FTINDC = 1.0
RADAR =CRO-08  X-Y USB-3n, LATR = -24.906577, LONR = 113.724182
 ALTR =   16.404, SRANGE = 300000.0, FTINDC = 1.0
RADAR =GWM-24  X-Y USB-3n, LATR =  13.310575, LONR = 144.735427
 ALTR =  374.015, SRANGE = 300000.0, FTINDC = 1.0
RADAR =HAW-12  X-Y USB-3n, LATR =  22.126307, LONR = 200.333450
 ALTR = 3749.993, SRANGE = 300000.0, FTINDC = 1.0
RADAR =TEX-16  X-Y USB-3n, LATR =  27.653750, LONR = 262.619990
 ALTR = -127.952, SRANGE = 300000.0, FTINDC = 1.0
RADAR =SHIP 1  INSERTION , LATR =  27.000000, LONR = 311.000000
 ALTR =    0.000, SRANGE =   23400.0, FTINDC = 0.0
```

4.10.1.7 Rationale

The system described in Reference 166 uses basically the same system as the Apollo Command Module Communication and Instrumentation System.

4.10.1.8 Reference

166 Pages 3-109 to 3-113

1. Communication and Data System - Spec
   North American Aviation MC901-0712
   29 Jan. 1968

2. CMS-SU-21-Section A Apollo Master
   Command Module Simulator, Communication
   and Instrumentation System Specifications

4.10.2 VHF System

166 The Shuttle Orbiter has two VHF amplitude modulated carriers of 296.8 mc
1 and 259.7 mc. Transmission or reception on either frequency is provided
2 by using two receivers and two transmitters. Four antennas are provided for selection by automatic switching logic. Vertical polarized antennas are provided at the orbiter tailcap and belly. Right Hand Circular Polarized (RHCP) RF patterns are provided on each side of the fuselage by helix cavity antennas.

166 The system provides a duplex loop suitable for simultaneous voice communication and also provides a low rate data transmission and command link of 2 Kbs for detached payload communication.

166 The orbiter vehicle has two VHF transceivers that are frequency modulated.
1 The channel frequency is selectable for use of voice communication with
2 Air Traffic Controllers during approach and landings. This system shares the same antennas as the duplex system. A triplexer functions to allow simultaneous operation of three transmitters or receivers off of a common antenna. RF losses by the triplexer are less than 1.5 db.

166 Each of the STDN stations has a VHF system using directable narrow-beam
1 antennas fed from the main dish. This system gives air-to-ground commun-
2 ication at orbital altitudes under any vehicle attitude when line-of-sight conditions exist.

166 The ATC system uses the VHF band for control of air traffic during the orbiter re-entry. The orbiter-ground link for ATC is established

F-398-8-A

following re-entry. Prior to this time, the vehicle will not be in line-of-sight, is blacked out from re-entry heat, or is out of range/ reception because of distance.

4.10.2.1 Rationale

The system described in Ref. 166 uses basically the same system as the Apollo Command Module Communication and Instrumentation System.

4.10.2.2 Reference

166 Pages 3-109 to 3-113

1. Communication and Data System - Spec.
   North American Aviation MC901-0712
   dated 29 Jan. 1968
2. CMS-SU-21 Section A, Apollo Master
   Command Module Simulator, Communications
   and Instrumentation System Specifications

4.10.3 UHF System

166 A UHF transponder gives the orbiter voice communication capability with the Ground Controlled Approach (GCA) of the airport for low visibility landings. The transponder has selectable channels that are preset prior to takeoff. This system has a maximum operational range of 50 miles. The orbiter antenna system is located in the three-wheel wells. The ground-based system uses an omni-directional antenna such as a corner reflector discone or top-hat antenna.

4.10.3.1 Rationale

The system described in Ref. 166 uses basically the same system as the Apollo Command Module Communication and Instrumentation System.

4.10.3.2 Reference

166 Pages 3-109 to 3-113

4.10.4 Audio Control Center

166 Each crew station has audio control equipment to accomplish audio
1 signal amplification and switching to select intercommunication between the crew station, to select communication RF links, to select recording equipment, to relay audio signals received. Each station has provisions for connecting a second headset for emergency operations. Automatic volume control is provided to balance the output for strong/weak signals. Standard Push-to-Talk (PTT) and voice-operated relay (VOX) mike circuits will actuate the transmitters to a "Transmit" condition.

166 Two communication panels provide redundant controls over the power
1 system-communication system interface. Access to the channel selector controls is provided between the pilot-copilot positions. Access to the crypto secure terminals is provided at the crew station aft positions.

4.10.4.1 Rationale

The system described in Ref. 166 uses basically the same system as the Apollo Command Module Communication and Instrumentation System.

4.10.4.2 Reference

166 Pages 3-109 to 3-113

1. Communication and Data System - Spec.
North American Aviation - MC901-0712
dated 29 Jan. 1968

4.10.5 TACAN

116 TACAN is a phase measuring high speed rotating ground based beacon providing
4 highly accurate azimuth determination in combination with a ground based
21 transponder providing distance measuring. TACAN covers the frequency range
1 of 962 megahertz to 1213 megahertz, with 1 megahertz channel spacing. Bearing
information (aircraft to station) is accomplished by a phase comparison of two equal frequencies. One frequency is fixed in phase aligned to true north while the other is rotated. The phase difference at the aircraft position establishes the relative bearing to the station. The Distance Measuring Equipment (DME) has two modes of operation. The airborne interrogator pulses at a relatively high frequency in search mode. Once lock-on is achieved with the ground based transponder, the pulse repetition frequency is reduced. Identification messages are transmitted each 30 seconds in morse. The ability to communicate between the aircraft and the ground station is a function of line-of-sight range and altitude. Accuracy of bearing information is within $\pm$ 1.5°. Accuracy of the DME is $\pm$ 0.2 percent of the distance measured. A "cone of confusion" exists directly over the station for bearing information but does not exist for the DME. The system will provide range and bearing information to the Guidance, Navigation, and Control subsystem and to the Display and Control subsystem. The TACAN will operate in the L-Band of the radio frequency spectrum. A self test feature will provide a visual indication of "GO/NO-GO" status of the functional units. An aural identification signal will be provided to the intercom equipment. Controls will be provided to allow the crew to select the desired channel. All modes will be simulated including multiple location of ground stations and search prior to lock-on. Radiation patterns of the ground station will be simulated including the radio horizon, maximum range, and cone of confusion. Accuracy bounds will

F-398.8.A

approximate those of the real-world equipment with resolution and update rates which will cause no noticeable discontinuities in either the displays or the on-board computer.

For in-orbit use, the TACAN system will be employed as a range and range-rate system. This will be accomplished by installing a TACAN ground transponder in the vehicle to be tracked or to which rendezvous is being made. This usage of TACAN will be very similar to its current use in air-to-air refueling operations.

4.10.5.1 References

4 pp. 9-12c-8

21 pp. 56-57

1) Radio Navigation Systems for Aviation etc., Bauss (1963).
Reference Data for Radio Engineers, Federal Telephone (1953).
166 pp. 3-109 to 3-113

4.10.6 Radar Altimeter

166 The radar altimeter system measures altitude relative to the local ground. The system operates in the C-Band of the radio frequency spectrum and is used in the range 0 to (TBD) feet. Outputs of the system are to the Guidance, Navigation, and Control computer and to the Control and Display Subsystem. A model of the ground terrain is required with highly accurate models of the prime landing sites nominal approaches. Off-nominal and secondary landing sites will be approximations requiring lower accuracy.

4.10.6.1 Rationale

Not required.

4.10.6.2 References

166 Pages 3-109 to 3-113

4.10.7 ATC Transponder

The Air Traffic Control (ATC) transponder will be used to identify and track the orbiter during post-entry phases of the mission. The ground control station will transmit interrogation signals to the transponder in the orbiter which will respond in the L-Band with altitude and identification signals to the ground station. The transponder will be compatible with continental United States and international operations requirements. The L-Band antenna switch will control interface of the ATC transponder and TACAN with one of the two antennas. Signal attenuation presents a non-linear altitude versus range limitation in the L-Band. The ATC transponder altitude limitation should be programmed as a variable similar to the TACAN coverage. The radio horizon can be represented as:

$$r = r_0 \frac{\cos \alpha}{\cos (\alpha + \theta)}$$

r = radio horizon for a given Nav-aid measured from the earth's center.

r_0 = earth radius

α = elevation angle constraints (default value of zero)

θ = central angle between Nav-aid position and shuttle positions

$\theta = \sin^{-1}(\bar{U}_R \text{ shuttle} \times \bar{U}_R \text{ Nav-Aid})$

The test for visibility with respect to the radio horizon between shuttle and the Nav-aid is:

$r < r$ shuttle $\rightarrow$ visible

$r > r$ shuttle $\rightarrow$ not visible

The index of refraction in the lower atmosphere (to about 10 miles) decreases with height. Radio frequencies above 200 mhz. follow curved paths slightly bent toward the earth. By replacing the real earth radius with one of 4/3 the true radius (5284 miles) the coverage can be considered straight lines.

A good approximation for the distance to the radio horizon is $d = \sqrt{2h}$

$h \rightarrow$ altitude in feet

$d \rightarrow$ radio horizon distance in miles

Over a smooth earth, line of sight is maintained if the distance in miles between antennas is less than $\sqrt{2h}$.

4.10.7.1 Rationale

Not required.

4.10.7.2 References

166 Pages 3-109 to 3-113

4.10.8 Instrument Landing System (ILS)

4 The ILS system is a fixed directional beacon with a directional (localizer)
21 and elevation (glideslope) beam about a reference path in the plane of the
166 runway centerline. The DME is used in conjunction with ILS to provide range
to go information. The localizer coverage is approximately $\pm$ 35° at 25 N.M. ground range and $\pm$ 360° for 10 N.M. ground range. The glideslope coverage is $\pm$ 10°. Elevation coverage is approximately 20° for both localizer and glideslope. The coverage range is 25 N.M. slant range for the localizer and 10 N.M. slant range for the glideslope. The beam is considered linear in a $\pm$ 4° range for the localizer and $\pm$ 0.8° range for the glideslope with respect to the reference path. The display is a "fly-to" error presentation. The localizer has a single null while the glideslope contains multiple nulls. The simulation will provide two glide slopes with each independently adjustable. The steep slope will be typically 13° centered with ground intercept at 5000 ft. from the end of the runway and the shallow will be typically 3° centered with the ground intercept at 1000 ft. from the end of the runway. Both the localizer and glide slope will be independently adjustable and located at the landing site. False nulls in the glide slope will be simulated.

The ILS system will be used for horizontal flight test only.

4.10.8.1 References:

4 pp. 9.12c-5 thru 9

21 pp. 3-244

166 Pages 3-109 to 3-113

4.10.9 GCA Radar

166 Each major airport in the continental U.S. is equipped with a Ground Control Approach (GCA) radar system. This radar is a skin-track pulse system which provides cross track, distance from touchdown, and height above glidepath to the GCA operator. The operator, using the UHF communication system, can direct aircraft down through overcast conditions to a Category II landing. GCA radar will control the final approach from 4 to 7 N miles to touchdown.

4.10.9.1 Rationale

All commercial and military aircraft use GCA for Cat. II landing in overcast conditions.

4.10.9.2 References

Instrument Flying Handbook, AC 61-27A,
Dept. of Transportation, Federal Aviation
Administration
166 Pages 3-109 to 3-113

4.10.10 Air Route Surveillance Radar

166 The FAA within continental U.S. has a series of overlapping radar systems that provides a map-like presentation of all aircraft within a defined zone. The radar system is composed of many skin-tracking radar installations or sectors. Each installation has a maximum range of 200 miles or less dependent on aircraft altitude (line of sight). These many installations are united into one control center for all traffic control. By means of electronically generated range marks and azimuth indicators, the controller can locate each radar target with respect to the radar installation, or can locate one radar target with respect to another. From direct reading counters on the controller's display panel, the controller determines the bearing and range of one aircraft target with respect to another. A video presentation not only gives him aircraft position, but their relation to other runways, navigation aids, and hazardous ground points in the area.

4.10.10.1 Rationale

FAA will control all air traffic into commercial or military air terminals.

4.10.10.2 References

Instrument Flying Handbook, AC 61-27A,

Dept. of Transportation, Federal Aviation Administration

166 Pages 3-109 to 3-113.

4.10.11 Precision Ranging System (PRS)

158
166
The PRS system may be used by the shuttle from post-blankout at an altitude of approximately 140,000 ft. to touchdown. The system consists of a shuttle based interrogator and multiple transponder beacons located at ground stations. As many as 10 PRS beacons could be interrogated by the shuttle during the approach and landing sequence.

158
166
This navigation system is based on using an inertial measurement unit (IMU) and a precision ranging system (PRS) as the primary sensors. The IMU measures changes in vehicle velocity due to non-gravitational forces on the vehicle. The PRS provides measurements of range and range rate from the vehicle to ground transponders. Range measurements are made twice per second using a phase comparison of frequencies. Range rate is measured at the same time using Doppler Shift. This system of touchdown has an accuracy of approximately 12 ft. altitude, 24 ft. crosstrack, 13 ft. down range, .6 ft./sec. altitude, 1.2 ft./sec. crosstrack, .9 ft./sec. down range.

4.10.11.1 Rationale

Not required.

4.10.11.2 References

158 All of the Document.

166 Pages 3-109 to 3-113

-398-8-A

4.10.12 Microwave Landing System (MLS)

157 The MLS will be used during approach and landing for range, azimuth, and elevation angle information with respect to the runway. The MLS is a sophisticated landing NavAid being developed jointly by DOD, DOT, and NASA and is scheduled for limited production in 1976. The system uses the Ku band. The airborne antennas will provide azimuth coverage of ± 45 degrees, and elevation coverage of ± 30 degrees referenced to to the Orbiter longitudinal axis. Four transmitter antennae will be used in the MLS. The range, azimuth, and long range elevation will operate in the C band while the flare elevation antenna will operate in the Ku band. The C band transmits data in a 20 N.M. range by 60° azimuth by 20° elevation angle. The Ku band flare antenna transmits elevation data in a 2 N.M. range by 8° azimuth by 8° elevation.

Accuracy of the system is estimated at 1 ft. altitude, 7 ft. in cross-track, .5 ft./sec. altitude rate, and .6 ft./sec. in crosstrack. These estimates are based on using a measurement update every 4 seconds.

4.10.12.1 References

157 Kriegsman, B.
"Entry and Terminal-Phase Navigation for SSV Orbiter Using MLS or AILS and VOR/DME," MIT Draper Lab 23A STS Memo No. 11-72, Feb. 16, 1972.

158 Kriegsman, B. and Gustafson, D.
"Entry-and-Landing Navigation Study for SSV Orbiter Using a PRS Navaid," MIT Draper Lab 23A STS Memo No. 49-71 (Ref. 1), Oct. 4, 1971.

Gustafson, D. and Kriegsman, B.
"SSV Re-entry Navigation Studies Using Barometric Altitude and VOR/DME Measurements," MIT Draper Lab 23A STS Memo No. 22-70, July 14, 1970.

4.10.12.2 Rationale

Not required.

REF. KEY

4.11 Operational Instrumentation

166 The orbiter vehicle is equipped with two types of instrumentation dependent on the mission objectives. Development Flight Instrumentation (DFI) will provide the required data for in-flight and post-flight evaluation for the initial flights. Once the developmental flights have been completed, the instrumentation system will be converted to the Operational Flight Instrumentation (OFI).

166 The data collected from sensors is conditioned for display and monitoring on-board the vehicle, provided to computers for CRT display, recorded, telemetered to the ground, and also provided to group support before and after launch.

166 Table 4.11-1 shows the division of measurements per system for both OFI and DFI usage. Figure 4.11-1 shows the general concept of the data interface requirements in both flight phases. Table 4.11-2 gives the OFI and DFI sample rate.

4.11.1 Recorders

166 There are five categories of recorders on-board the vehicle. These are the maintenance recorder, a short-burst (5 min.) loop recorder, a wide-band data recorder, a crash (FAA) recorder, and a voice playback unit. The recorder functions are straightforward with the following special features. The playback of the maintenance recorder is not required for flight crew training. This recorder is played back following a mission landing. The loop recorder is a cued recorder which is controlled manually or automatically by GN&C or C&W systems. This recorder will be used by the crew in problem analysis during flight. There is no requirement for the wide-band recorder. The crash recorder is not required for simulation. This

Figure 4.11-1 *OFI and DFI Measurement Summary*

Operational Instrumentation																Development Flight Instrumentation											
Meas Quantities											* In-Flight Meas Use					Meas Quantities									** Meas Use		
Pressure	Thermal	Dynamic	Electrical	Event	Rate	Position	Quantity	Load	Miscellaneous	Total	CRT Displays	Dedicated Displays	C&W	Perf Monitor	Subsystem	Pressure	Thermal	Dynamic	Electrical	Event	Position	Load	Miscellaneous	Total	PCM TM/Rec	Wideband Rec	Wideband TM
6	–	–	–	–	–	–	–	–	–	6	–	–	–	–	Aero-surfaces	195	253	–	–	–	–	–	–	448	448	–	–
–	6	–	–	–	–	–	–	18	–	24	–	–	–	–	Vehicle struct	–	72	286	–	–	–	198	–	556	270	286	57
20	277	–	–	–	–	–	–	–	–	297	–	–	–	–	Thermal prot	20	831	–	–	–	–	–	–	851	851	–	–
67	28	21	–	19	18	27	–	–	–	180	22	37	3	20	MPS	19	11	–	–	22	–	–	–	52	52	–	–
14	21	–	–	110	–	–	4	–	–	149	66	32	4	60	OMS	–	7	–	–	–	–	–	–	7	7	–	–
49	71	–	–	103	–	–	3	–	–	226	160	92	9	150	RCS	–	12	39	–	–	–	–	–	51	12	39	–
5	10	8	–	36	12	–	5	–	–	76	20	42	4	20	ABPS	80	80	–	–	–	–	–	–	160	160	–	–
–	6	15	–	81	–	53	–	–	37	192	150	57	15	60	GN&C	24	6	–	–	172	10	–	–	212	212	–	–
–	–	–	14	51	–	–	–	–	–	65	21	–	–	21	COMM	–	–	–	–	26	–	–	–	26	26	–	–
–	4	–	45	38	–	–	–	–	–	87	63	63	27	50	EPD	–	10	–	9	–	–	–	–	19	19	–	–
27	7	–	–	14	–	8	8	–	–	64	40	12	14	40	Hydraulic pwr	17	32	–	–	–	6	–	–	55	55	–	–
37	40	–	–	–	5	–	12	–	4	98	70	32	12	40	ECLSS	–	–	–	–	–	–	–	–	–	–	–	–
–	–	–	–	–	–	–	–	–	16	16	–	–	2	–	Flt crew supt	–	18	5	–	–	–	4	6	33	30	3	–
–	–	–	6	4	–	–	–	–	–	10	10	3	–	10	Instrumentation	–	25	–	12	3	–	–	–	40	40	–	–
17	19	–	9	72	6	–	4	–	–	127	63	67	6	60	Elect power gen	23	50	18	–	–	–	–	–	91	73	18	–
20	16	–	3	36	4	–	4	–	–	83	45	20	4	40	Mech power gen	–	12	6	–	–	–	–	–	18	12	6	–
–	–	–	–	600	–	–	–	–	–	600	–	–	–	–	D&C	–	–	–	–	–	–	–	–	–	–	–	–
262	505	44	77	1164	45	88	40	18	57	2300	730	457	100	571	Orbiter totals	378	1419	354	21	223	16	202	6	2619	2267	352	57
2	8	–	10	–	–	–	–	–	–	20	–	–	–	–	SRM totals	–	18	12	–	–	–	–	–	30	–	–	–
4	16	–	–	19	–	–	–	–	–	39	20	4	2	20	ETS totals	2	112	30	–	10	–	–	–	154	–	30	–
268	529	44	87	1183	45	88	40	18	57	2359	750	461	102	591	Grand totals	380	1549	396	21	233	16	202	6	2803	2267	382	57

*All data available for recording/telemetry and ground checkout **Data used for engineering analysis

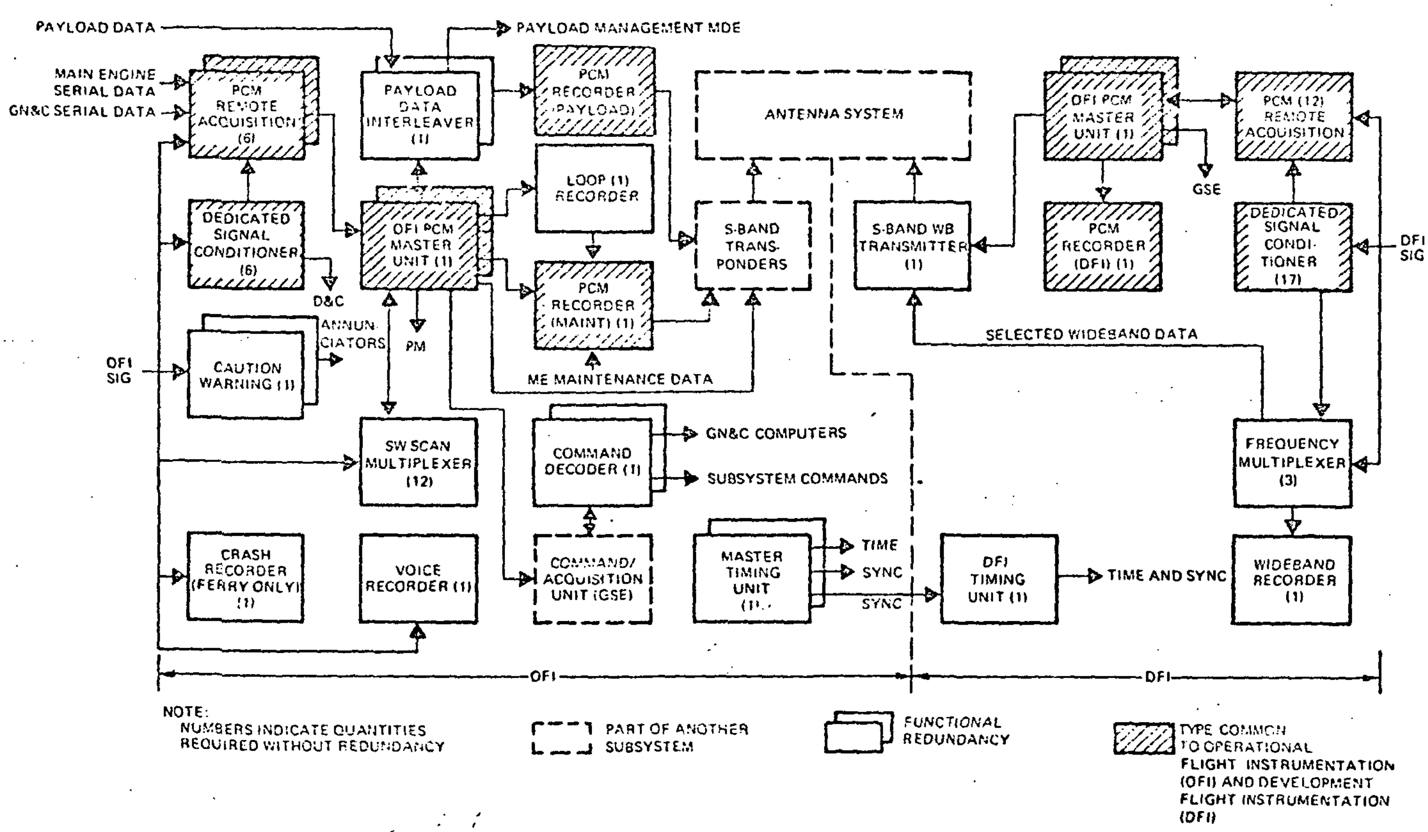


Figure 4.11-1 *Development and Operational Instrumentation Block Diagram*

166 is outside of crew training requirements. The voice recorder is required. The system should provide near real world simulation interfacing with all active voice or T/M (simulated) lines.

4.11.2 Sensors and Signal Conditioning

166 The types of signal sensors are indicated in Table 4.11-1 Table 4.11-2 indicates the sample rate for PCM inputs. The data rate (maximum) is 256 kbs and is programmable in 4 formats. Inputs to the signal conditioning units is 0-5 volt and 0-20 millivolt full scale. Dedicated spacecraft instruments use dual 0-5 volt inputs. Total error for instrumentation is distributed as follows:

Sensors	3.0%
Signal Conditioners	1.0%
Noise	0.6%
PCM	0.4%

4.11.3 Ground Support Equipment PCM Links

66 There is no requirement for simulation of preflight GSE activity occurring prior to the crew boarding the vehicle. GSE maintenance/monitoring must be complete prior to that time. By the same ground rule, there is no requirement for preventive maintenance simulation for main engine performance evaluation.

4.11.4 Caution and Warning System

66 A Instrumentation sensors are provided for 200 major critical flight parameters. Parallel systems are provided to display the problem to the crew and to provide input data to the GNC computer where required.

10/20/72

Operational Instr					Development Flight Instr			
No. Chan.	Rate S/S	Bits	Bits/ Second	Data Type	No. Chan.	Rate S/S	Bits	Bits/ Second
435	1	8	3,480	Analog	1,910	1	8	15,280
500	10	8	40,000	Analog	140	10	8	11,200
—	—	—	—	Analog	3	50	8	1,200
2	100	8	1,600	Analog	75	100	8	60,000
—	—	—	—	Discrete	35	100	1	3,500
1,120	10	1	11,200	Discrete	344	10	1	3,440
6	10	80	4,800	Ser. Dig. (ME)	—	—	—	—
3	25	80	6,000	Ser. Dig. (GN&C)	—	—	—	—
1	100	256	25,600	Ser. Dig. (Payload)	—	—	—	—
1	10	24	240	MET	1	10	24	240
1	200	32	6,400	Sync & ID	1	200	32	6,400
1	10	32	320	Perf. Mont. Sync.	—	—	—	—
Total used			99,640		Total used			97,760
Spare capacity			28,360		Spare capacity			30,240
Total capacity			128,000		Total capacity			128,000
S/S—samples/second MET—Mission elapsed time								

TABLE 4.11-2 OFI and DFI Rates

A Redundant caution and warning indicator display light units are provided in the orbiter crew station. Klaxon horns, intercom speakers, and headset tone signals provide audio cues for alert. The audio tones are coded by frequency and pulse duration for easy recognition by the crew.

A The tone signals used from the Skylab Mission will also be used for the shuttle. A siren signal will indicate Fire Emergency. A 1000 cps tone modulated by a square wave of 1.25 cps will indicate Warning, and a continuous 1000 cps tone will indicate Caution conditions. A rapid loss of pressure (Emergency) will be represented by an interrupted buzzer signal.
B Stall (Emergency) will be represented by an intermittent horn while gear down and locked (coupled to a low power level) will be represented by a continuous horn. A single pulse will represent a crew alert for incoming crypto message.

A Other crew alert tones will be present when crossing the Tacan Outer Marker
B (400 Hz modulated once every two seconds), Middle Marker (1300 Hz with alternate dots and dashes), and the Boundary Marker (3000 Hz dots @ two per second).

A Redundant sensors provide input parameters to two C/W units. Each unit is equipped with redundant power supplies. Caution, Warning, Emergency, and Crew Alerts provide inputs to set the Master Alarm units. Master Alarm is visually indicated on a telelight switch. Depress the switch with an alarm on will silence the alarm. Reoccurrence of the condition will re-trigger the alarm circuit.

A The C/W system provides a means of disabling malfunctioning sensors through the use of inhibit switches. In the Inhibit position, the sensor input is not fed into the C/W system.

A. A Memory feature provides the crew with the capability to recall an alert signal that triggered the C/W circuit. Memory may be cleared by depressing the CLEAR switch.

4.11.5 Rationale

A. The Caution and Warning System will be similar in logic and performance to the Skylab C&W System. Refer to Saturn Workshop Systems Handbook, DC-5 July 11, 1972.

B. It is assummed that the C/W tones used for shuttle will be very similar to existing commercial aircraft warnings.

4.11.6 References

166 Pages 3-111- 3-113

4.12 Environmental Control and Life Support Subsystem

166 The environmental control and lift support subsystem (ECLSS) provides atmospheric revitalization, life support, and thermal control (Figure 4.12-1) with subsystem assemblies as shown in Figure 4.12-2. The atmospheric revitalization subsystem furnishes a shirt-sleeve environment for the four-man crew for a seven-day mission and a 96-hour contingency by controlling CO_2, humidity, odor, pressure, oxygen/nitrogen cabin atmosphere, and cabin temperature. Six additional crewmen can be accommodated for a short duration. Expendable capacity is supplied for 42 man-days. The mission can be extended up to 30 days by adding expendables in the payload. The life support subsystems provide for food and waste management, fire control, and extravehicular/intravehicular activities. The thermal control subsystem affords active thermal control for avionics and mechanical equipment, dissipates the heat from the crew compartment, and provides for water management. The fluid and energy interchange functions performed by the ECLSS and interfacing vehicle systems are shown in Figure 4.12-3.

4.12.1 Atmospheric Revitalization

4.12.1.1 Pressure Control

166 A two-gas atmosphere, oxygen/nitrogen at a nominal pressure of 14.7 psia, is used to reduce fire hazard potential and to provide compatibility with payload module experiments and proposed space station environments. The oxygen stored in a supercritical state, is acquired from the electrical power generation reactant storage system. The nitrogen is stored at 3,000 psi in two carbon filament storage tanks. Normal gas usage and losses as a result of venting and seal leakage are shown in Table 4.12-1. The subsystem includes provisions for a 10-minute emergency

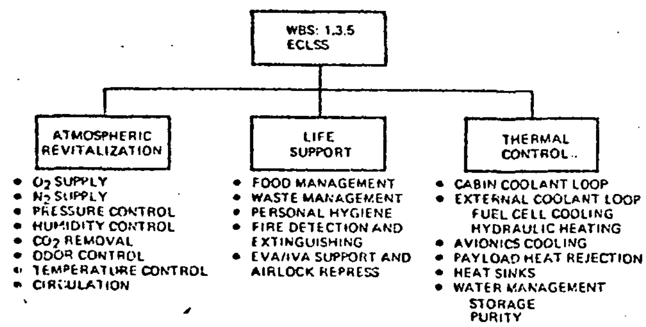


FIGURE 4.12-1

ALL FUNCTIONS PROVIDED BY SYSTEM PROPOSED

10/20/72

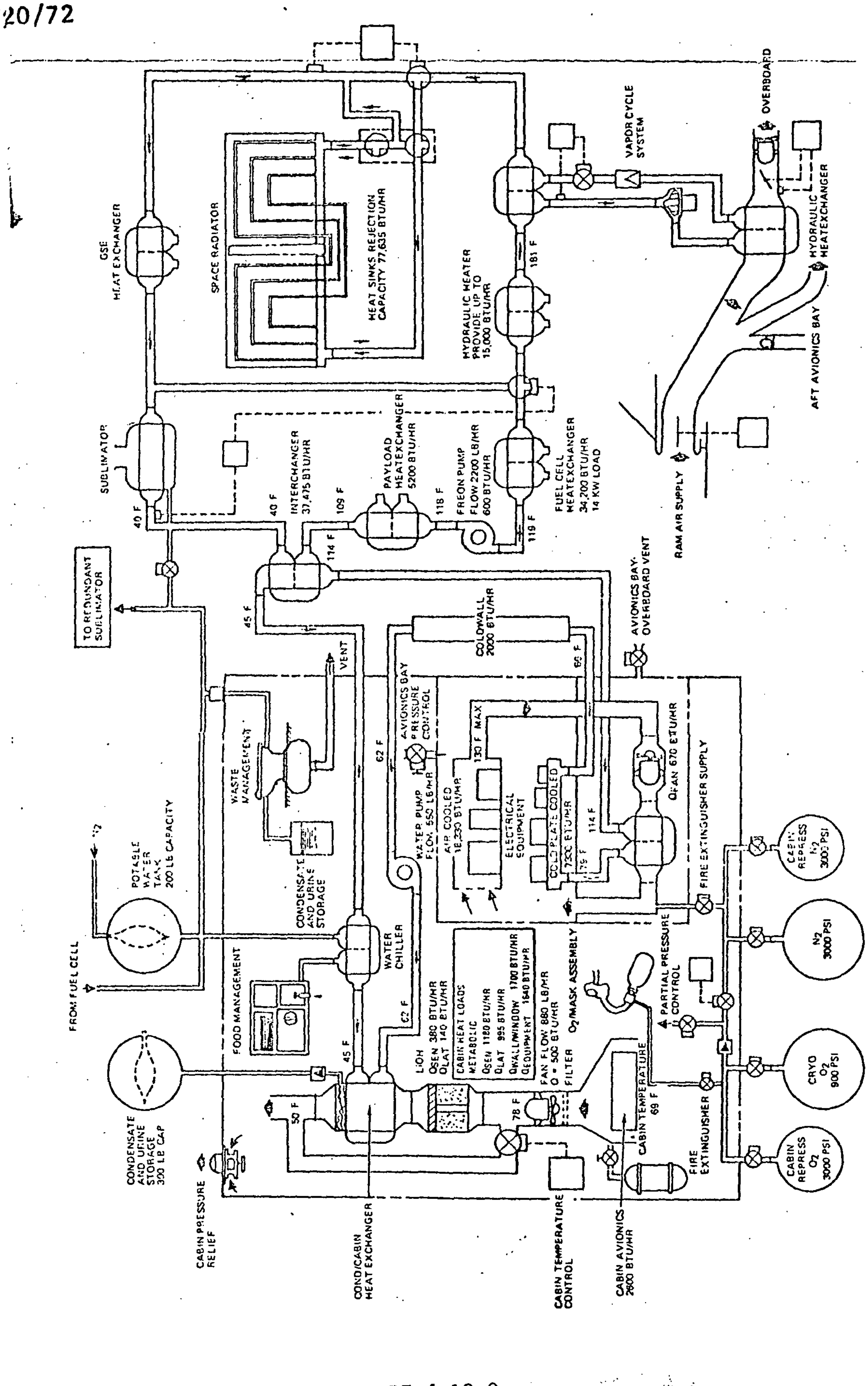


FIGURE 4.12-2

ECLSS

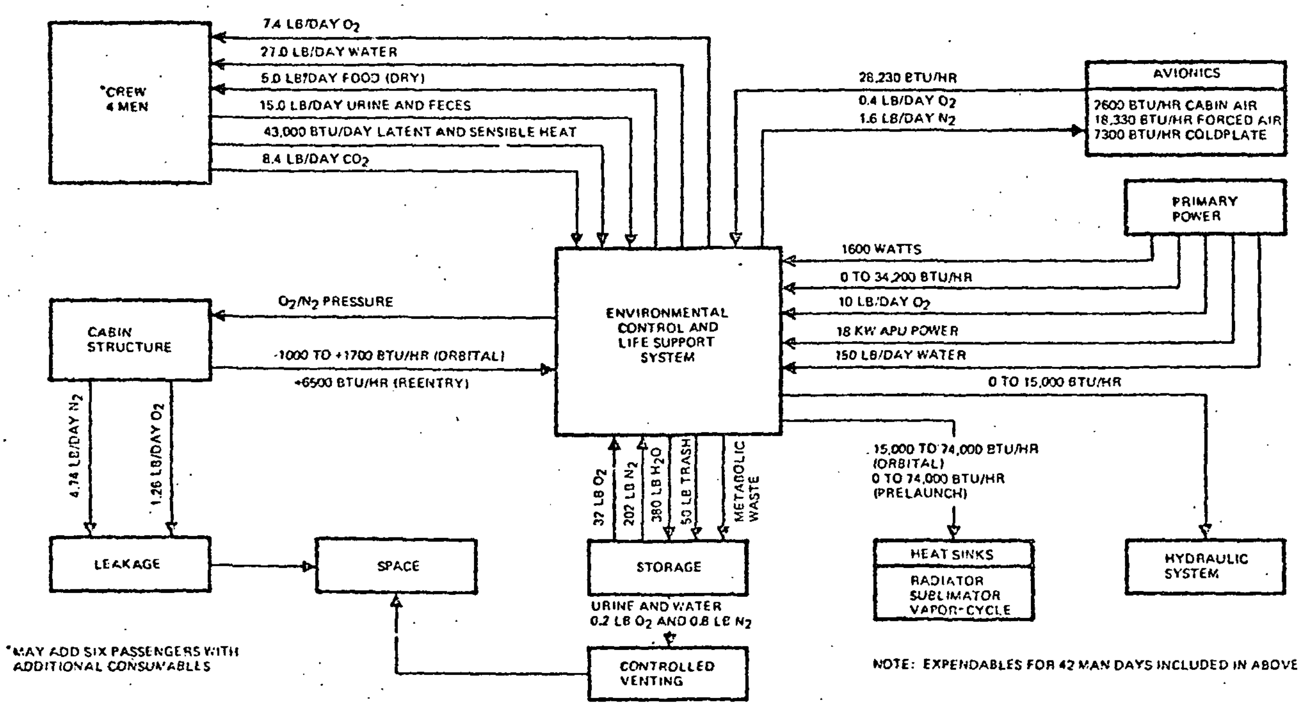


FIGURE 4.12-3

BALANCED AND INTEGRATED SYSTEM

Description	Gas Loss (Pounds Per Day)		
	Oxygen	Nitrogen	Total
Waste management port venting	0.21	0.79	1.0
Avionics bay controlled venting	0.42	1.58	2.0
Cabin leakage	1.26	4.74	6.0
Metabolic	8.0		8.0
	Gas Required (Pounds)		
Mission total (including reserves and cabin repressurization)	142.0	202.0	344.0

TABLE 4.12-1

NORMAL GAS USAGE AND GAS LOSSES

166 high oxygen gas flow of 55 pounds per hour. An alarm to the Caution and Warning system is initiated after a time delay if the oxygen flow exceeds three pounds per hour. Additional quantities of oxygen and nitrogen, stored at 3,000 psia, are connected to the primary system for one cabin repressurization at a flow of 150 pounds per hour, offering the four-man crew a 96-hour contingency, or seven airlock operations.

166 Gas flow into the cabin is controlled by the cabin pressure regulator, which sets the N_2 flow based upon the cabin O_2 partial pressure of 3.0 to 3.2 psia. The partial pressure control opens when the O_2 partial pressure is 3.2 psia or above, permitting only nitrogen to flow into the cabin. Nitrogen will continue to flow until a total cabin pressure of 14.7 psia is reached. When the O_2 partial pressure drops to 3.0 psia, the partial pressure control closes to permit only O_2 to flow into the cabin.

166 Cabin overboard pressure relief valves are located in the cabin area and in the avionics bays. A differential pressure relief valve is provided between the cabin and each avionics bay to maintain the avionics bay pressure at approximately 0.4 psi below cabin pressure. A total of two pounds per day for three avionics bays is bled overboard to maintain bay pressure below cabin pressure. During times of airlock repressurization or emergency fire extinguishing operation in the avionics bays, the pressure in the bays may be as much as 0.6 psi higher than in the crew cabin. Gas flow into the crew cabin is prevented by check valves in the differential pressure regulator and the sealed avionics bay door design.

166 Portable face masks and emergency oxygen assemblies are designed to furnish a ten-minute supply of oxygen to each crew member and passenger. Provisions are also available to connect these assemblies to the oxygen system.

4.12.1.2 Cabin Humidity, CO_2, Odor and Temperature Control

166 A condensing heat exchanger for humidity control and cabin temperature, lithium hydroxide for controlling CO_2 level, and activated charcoal for controlling air odors are used because of their proven performance in the CSM environmental control system. The system is comprised of three two-speed fans, two lithium hydroxide (LiOH) canisters, a condensing heat exchanger, and an air bypass valve. Air volume through the system is governed by the cabin temperature control range of 65° to 80° F, selectable to $\pm 2^{\circ}$ F. The fans can operate at maximum flow for high heat load flight conditions or at reduced flow for station keeping operations. Figure 4.12-2 shows system performance for the four-man design condition. With 10 men and maximum heat load, the cabin temperature will approach 75° F. A payload heat exchanger is included in the cargo bay with a capacity of 5,200 Btu/hr. Additional heat dissipation from operation of the mission specialists station of 1,000 Btu/hr is absorbed by the cabin atmosphere heat exchanger plus 2,000 Btu/hr from equipment in the avionics bay. The LiOH canisters are installed in two parallel flow paths. Each canister will be installed in the system for 24 hours, based on a seven-day, four-man mission. The canisters will be replaced more frequently when 10 people are carried. Each canister also contains activated charcoal for trace contaminate control. Low toxicity outgassing materials in the cabin design are used to reduce contaminant problems. The condensing heat exchanger removes the cabin sensible and latent heat load. Water is removed from the heat exchanger by water separators and trasnferred to the waste water system for storage. A replaceable, disposable filter is

166 provided at the inlet to the fans for particulate control. A coldwall configuration has coolant flow passages attached to the cabin wall to control the wall temperature to prevent condensation in orbit and excessive cabin temperature during and after entry. All surfaces normally in contact with the crew will not exceed 113^{o} F.

4.12.2 Life Support

4.12.2.1 Food Management

166 The food management system consists of a galley area offering a food preparation center for food and equipment storage, hot and cold water dispensers, and waste storage. The GFE food concept assumed includes the use of protective canisters for storing food serving cans, dehydratables, and drink packages. Foods are categorized as thermostabilized, rehydratables, wafers, and beverages. Some foods are ready to eat, and others require chilling or heating in the galley oven. Rehydratables are prepared by injecting hot or cold water and mixing. A freezer can be installed in the galley for extended missions by moving stored food to other areas of the vehicle.

166 Simulation of the food management system outputs/inputs such as electrical loads, water usage, heat absorbtion/rejection is required.

4.12.2.2 Waste Management

166 The waste management subsystem accumulates solid waste and collects, transfers, and stores liquid wastes. Urine and urinal rinse are collected, separated from air, and stored for return to earth. The commode uses a slinger and vacuum drive concept for feces storage. The bactericide for bacteria control in the urine is stored in a bladder tank. Air flow

leaving the urinal and commode passes through bacteria filters and charcoal canisters before reentering the cabin. This process provides odor and bacteria control to the entire vehicle.

166 Simulation of the electrical loads, heat rejection/absorbtion, and water usage is required.

4.12.2.3 Personal Hygiene

166 For the seven-day mission, personal hygiene provisions will be a GFE kit stored in the waste management compartment. Space is available above the waste management system for installation of a hand wash basin and a more extensive personal hygiene system for extended missions.

166 Simulation of the electrical loads, heat rejection/absorbtion, and water usage is required.

4.12.2.4 Fire Detection and Extinguishing

166 The fire control subsystem is located in the avionics bay and flight deck. The detection system consists of a light source, a gas filter interferometer, and a detection and localization logic similar to the Skylab system. Several gases can be monitored at once, and it can localize incipient fires or toxic elements. This system simultaneously serves as a fire detector and as a contaminant detector. It detects a problem before it represents a hazard to the crew.

166 Different types of fire extinguishing systems are provided for the three avionics bays and for the crew compartment. The avionics bays are isolated and differential pressure regulators normally maintain the bays 0.4 psig below cabin pressure. If a fire occurs in the avionics bay, a high N_2 purge flow is introduced to reduce the O_2 partial pressure below 1.0 psia, extinguishing the fire in less than one minute. Operation of avionics

166 bay fans will effectively mix all the gases to give the minimum reaction time. The high gas flow into the bay will be vented overboard by the individual bay vent valve. The vent valve will operate between 14.7 to 15.0 psia.

166 The crew compartment will use four hand-operated portable, foam fire extinguishers. Each extinguisher is identical to that used in the CSM.

4.12.2.5 Extravehicular/Intravehicular Service and Recharge Station and Airlock

166 Two portable life support systems (PLSS) service stations are provided for two-man operations. Each station has provisions for water (for recharging the suit sublimator cooling system), oxygen and battery recharge, and oxygen open-loop prebreathing. To recharge the PLSS, a pump boosts the 900-psia oxygen supply to the 1500-psia PLSS oxygen storage tanks. The airlock will normally be repressurized from the cabin gas in less than five minutes by opening the airlock pressurization valve. Expendables are supplied for seven airlock repressurizations and PLSS recharges.

4.12.3 Thermal Control

4.12.3.1 Coolant Loops

166 Figure 4-12.2 presents the coolant loops and design performance data. Water is used to cool the cabin because it is nontoxic. Freon 21 cools the unpressurized areas because it has a low freezing point and viscosity. Thermal integration is accomplished in the Freon loop through interfaces with the environmental control subsystem, fuel cells, and

F-398-8-A

hydraulic systems. Up to 15,000 Btu/hr of Freon system heat transfers to the hydraulic fluid to maintain its temperature. Electrical heaters provide thermal control of other passive systems in space. Refer to Figure 4.12-3.

166 Thermal control of the three avionics compartments (located in the crew compartment) is accomplished by pinfin coldplates and by air cooling systems of one of two parallel fans operating with a heat exchanger in each bay. The CSM-developed coldplates will be improved in durability by increasing the faceplate material thickness to reduce potential damage during normal maintenance procedures.

4.12.3.2 Heat Sinks

166 Space radiators are the primary heat sink and can reject the maximum heat load without attitude constraints during all space operations. The water sublimator is the normal heat sink for boost and before and during reentry to 100,000 feet altitude, with system thermal lag absorbing heat between that altitude and the cruise condition. During atmospheric cruise (below 30,000 feet), vapor-compression cycle refrigeration systems reject the heat to ram air-cooled heat exchangers. Heat is rejected during ground operations through GSE heat exchangers or the vapor-cycled systems. The radiator is a two-sided, deployable radiator consisting of eight modular panels stowable in the cargo bay and enclosing the cargo space as a door. The radiator is in turn provided with a thermal protection cover or outer door during ascent, reentry, and recover conditions. The panels are constructed from aluminum, and the top panels are thermally isolated from the bottom panels. Figure 4.12-4 shows one side open,

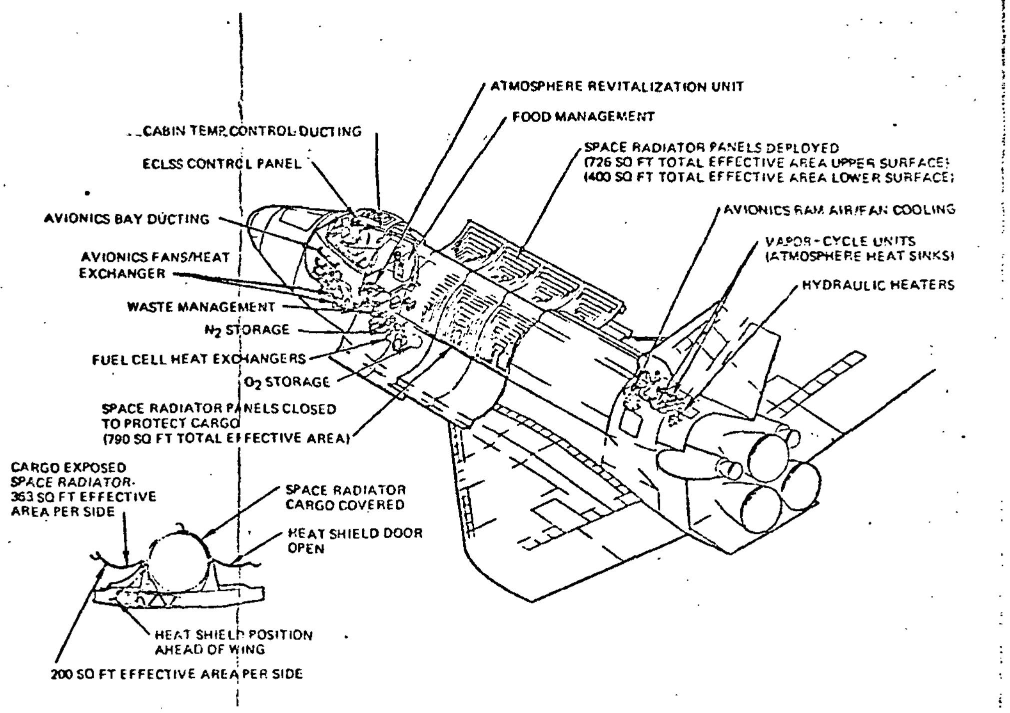


FIGURE 4.12-4

RADIATOR CONFIGURATION

exposing the cargo bay, and the other closed, protecting the cargo. In the open position the radiator has 726 square feet of effective area on the bottom side. The radiator is most efficient when in the deployed position, for heat can be rejected from both sides without supplemental cooling for all orbital mission phases and without attitude constraints.

166 When the radiator is in the stowed position, the effective radiator area is 790 square feet and supplemental cooling will be required, using excess fuel cell water in the sublimator for the worst-case mission heat load and radiator attitude. The radiator coating has a solar absorptivity of 0.2 and an infrared emissivity of 0.92.

4.12.3.3 Water Management

166 The water management system stores, distributes, and disposes of potable water and collects and stores waste water. Potable water is stored in two tanks, each having a capacity of 100 pounds. It is supplied to the galley at temperatures of 150^{o} and 45^{o} f. When the potable storage tanks are fully charged, the system pressure will rise to 20 psi above cabin pressure. With continued fuel cell flow, the tanks will become full and normal disposal is accomplished by the water sublimator. For emergency, the water will be dumped overboard through two heated nozzles. Bacteria are controlled by processing potable water through silver ion generators. In the event the water from one of the fuel cells should become contaminated, the water can be dumped directly overboard to bypass the potable system. Waste water condensate from the humidity control heat exchanger is stored in three tanks, each of 100 pounds capacity and normal operating pressure of 14.7 psia.

4.12.4 Fault Detection Management

166 The fault detection subsystem allows continuous monitoring of critical paths of the subsystems during flight and minimizes ground checkout procedures. The instrumentation will permit determination of redundant paths in flight and status of each line replaceable unit within the subsystems during ground checkout. Controls will isolate units or groups of units when an out-of-tolerance condition is detected. Notification of out-of-tolerance conditions will activate Caution and Warning system alarms and displays.

4.12.5 Ground and Launch Operations

166 Postlanding cooling of the avionics equipment and cabin will be supplied by the ECLSS. System heat will be transferred to the GSE heat exchanger, which will be activated by connecting the ground coolant lines and electric power to the service connections. With ground power available to the vehicle, the vapor-cycle system will serve as a standby cooling system if the GSE coolant system is not operating or if it is not available. Prelaunch cooling will be provided by the GSE system through the upper swing arm to T-25. Subsequent to T-25 the vapor-cycle system will provide system cooling.

4.12.6 Cabin Noise Level

166 Special provisions have been incorporated in the design of the cabin systems to control noise level. The avionics bay doors and the noise-producing avionics equipment, such as the inverters and fans, are sound insulated. The cabin fan and water pump assemblies have been remotely

166 located under the lower compartment floor away from living areas. The pump and fan incorporate local sound insulation and the duct system design contains sound absorbing baffles. Audio cues for these devices are not required.

4.12.7 Corrosion and Contamination Control

166 A program for corrosion and particulate contamination control will be employed for all ECLSS fluid systems. Stainless steel tubing and components will be used in the water system loop and the potable and waste water system to reduce corrosion. The Freon system will be made of aluminum.

4.12.8 Rationale

Not required

4.12.9 References

166 Pages 3-135 to 3-142

4.13 Payload Accommodation System

4.13.1 Structural/Mechanical Interfaces

4.13.1.1 General Structural Attachment

The payload attachment mechanism Figure 4.13.1.1-1 consists of easily moved attachment fittings that can be placed at 14 locations along the length of the bay. This design provides felxibility for payload changes between missions. Alignment of the payloads within 0.5 degrees of the orbiter's reference system is maintained by this system.

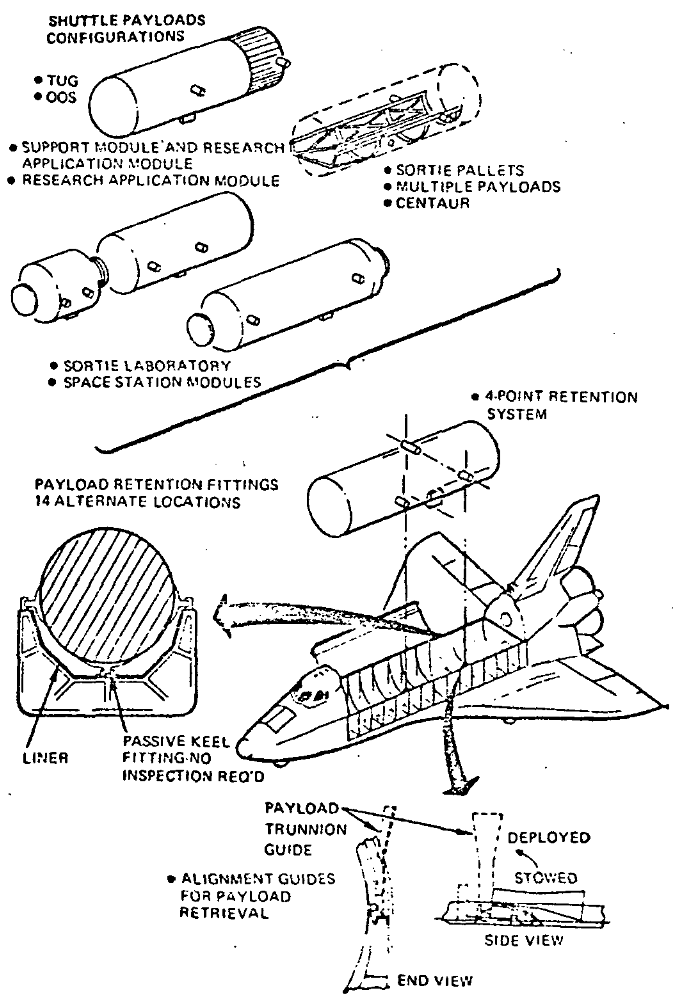


FIGURE 4.13.1.1-1 PAYLOAD RETENTION

4.13.1.2 Payload Deployment/Retrieval

166 Deployment and retrieval of payloads is provided by the general purpose remote manipulator system (RMS) illustrated in FIGURE 4.13.1.2-1. The system is adaptable for various payloads to perform multiple payload deployment, retrieval on a single orbit mission, and the docking/retrieval of light mass payloads.

A payload is retrieved in three basic steps: (1) transmission of commands for stabilization, orientation for manipulator attachment, retracting solar arryas, antenna, etc.; (2) manipulator engagement, translation, and securing in the payload bay; and (3) connection of payload utilities, (e.g., caution/warning, power, data, and fluid/gas venting when required)..

These utilities are connected without EVA by the standard interface connections provided at the docking port and the payload bay access hatch.

For multiple payloads having propulsive stages, actuation of interface connections remotely controlled from the payload handling station will be provided by kits.

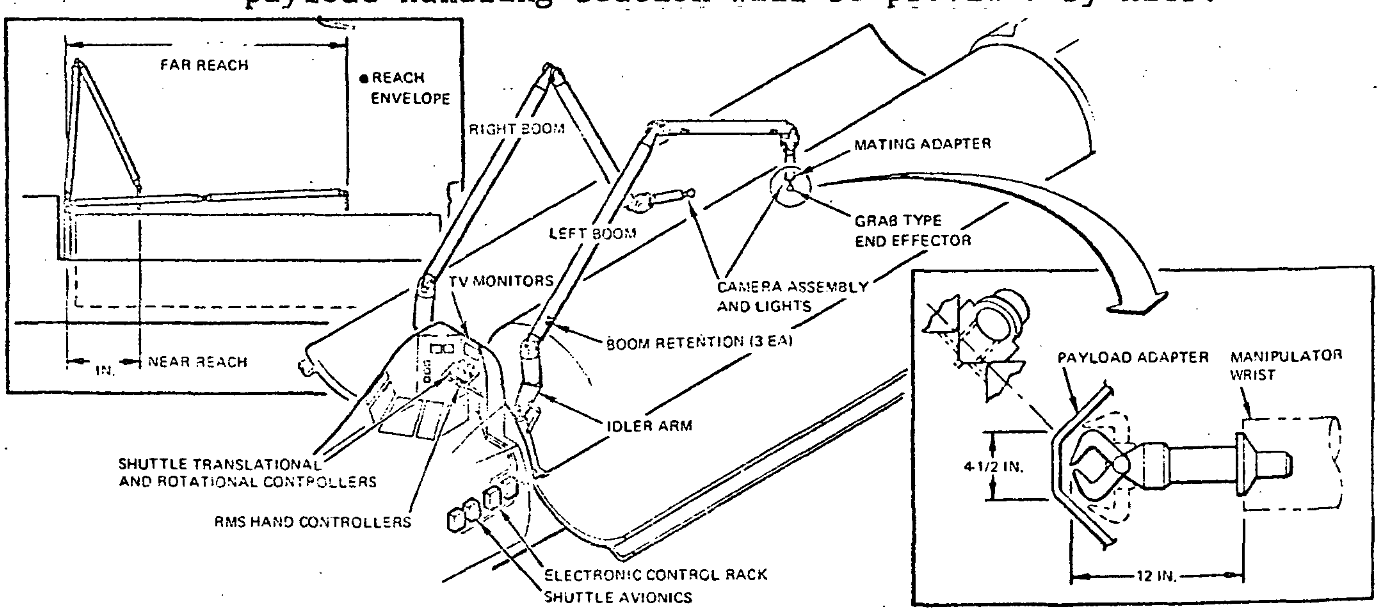


FIGURE 4.13.1.2-1
Manipulator Arms Adaptable to Multiple Payload Deployment and Retrieval

398-8-A

Shuttle orbital payload deployment and retrieval will be
1, 2, 3 accomplished by two remotely controlled manipulator arms. The two identical
50 foot long arms each possess shoulder, elbow, and wrist joints. The
shoulder, located at the attachment point of the arm to the orbiter,
possesses two rotational joints (pitch and yaw). The elbow, connected
4 to the shoulder by a 23½ foot long rigid tubular beam, possesses two
rotational joints (roll and yaw). The wrist, connected to the elbow by
a 23½ foot long beam "forearm," possesses three rotational joints (yaw,
pitch, and roll). The terminal device is connected to the wrist by a
3 foot long beam. The standard terminal device will be a hand-type
5 grasping mechanism, but special devices may be attached for particular
6 mission. Each arm will weigh about 1000 pounds. Each arm is attached
1 to the fuselage near the forward bulkhead of the payload bay. During
7 launch and entry, the arms are stowed along the top of the payload bay.
8 Each arm is extended, and attached at seven points to one of the payload
1 doors, near the mating line between the doors. Each arm is attached to
a different door. When stowed, each arm occupies a cylindrical envelope.
50 feet in length by 8 inches in diameter. The attachments must be
released before opening the payload doors, and, the arms are ordinarily
locked in them shortly after closing the doors. After the payload doors
1 are opened, an arm deployment mechanism is actuated which raises the
shoulders out of the payload bay and opens a 20 foot separation distance
between them. The deployment mechanism must be retracted in order to

2, 9, 10, 11 close the doors. Electric motors and gear reducers located at each of the seven rotational joints, plus the terminal device, provide torque
1 to move the arm and terminal device. The selected motors have a maximum shaft power of 28 watts. Each joint uses two motors. If one motor
1 fails, the joint can still be moved with full speed and torque capability. Each rotational joint has a brake, which is locked up whenever power to that joint is off (not powered up, or power failure). Each rotational joint also contains a potentiometer to measure angular position and a tachometer to measure angular rate.

Control of the manipulator arm may be either automatic or
13 manual. The on-board computer will be able to execute certain basic
14 arm maneuvers itself. Manual control will be effected through a force-reflecting system. To provide both the coarse control helpful for positioning the arm in the approximate attitude desired, and the fine
13 control necessary for delicate maneuvers, a variable-gain system was chosen. For coarse control, angles transcribed by the manipulator arm and the controller will be equal. For fine control, angles transcribed by the controller will be 18 times as large as those transcribed by the manipulator arm. Interface between the controller and the manipulator arm will take place through the on-board computer, which will calculate the necessary transformations.

A checkout system is provided to verify the condition of
8 arm motors, tachometers, and potentiometers, before power is applied to the system. Upon selecting the desired device to be evaluated, a predetermined voltage is applied to it. The actual voltage accepted and returned is then available for display. Ordinarily, no motion will

result, since brakes are locked when the system is powered down. In case of certain failures, the checkout system may be used to move a joint independently. A switch will be provided to release the brake on a joint, thereby moving the joint when the checkout system applies electrical power.

Sensory inputs to the crewman controlling the manipulator
arm are provided by the force-reflecting controller, a window through
15, 8 which the operator can see into the payload bay, and three TV monitors.
1 Four TV cameras are mounted in the payload bay - the operator chooses
which three he wishes to monitor. Two floodlights are attached to each
camera. One camera is mounted near the end of each arm (after wrist
16 yaw and pitch joints, before wrist roll). Another TV camera is mounted
at the base between the two manipulator arms. This camera can be used
to automatically track the terminal device. Iris and focus control is
17 provided for all cameras, as is a power switch. Both the fore and aft
payload bay cameras can be zoomed as well. Both the aft and forward
(when in manual mode - not automatic terminal device track) can be panned
and tilted by the operator. Brightness, contrast, and test controls are
provided on the crew station monitors. Resolution of at least 300 scan
lines is provided by the TV system.

In case of certain arm failures which would prevent closing
1, 2 of the payload doors, an explosive device will separate the failed
manipulator arm from the vehicle, allowing the orbiter to be flown away
from the derelict arm.

1, 2 Efficient manipulator arm design requires certain limitations
6, 12 on arm dynamics. Rotational joint design requires limitations on angular

travel at each joint. Each joint is also torque limited to prevent damage to the manipulator arm. Travel limits and torque capability (based on 10 pounds maximum tip force) are:

		Angular Travel	Torque (ft-lb)
Shoulder	Pitch	$\pm 200^{\circ}$	500
	yaw	$\pm 130^{\circ}$	500
Elbow	roll	$\pm 200^{\circ}$	350
	yaw	$\pm 155^{\circ}$	350
Wrist	yaw	$\pm 120^{\circ}$	150
	pitch	$\pm 120^{\circ}$	150
	roll	$\pm 200^{\circ}$	65

Travel limits are enforced by mechanical stops and/or microswitches. Joints and the control system are designed so that
6, 19 maximum tip position error is ±2 inches and maximum tip velocity error is ±.05 ft/sec. (excluding structural deflection under load). This requires an angular position accuracy of $\pm.113^{\circ}$ and an angular rate accuracy of $\pm.033 \frac{deg}{sec}$ at each joint. This places requirements on the precision and frequency of arm control inputs recieved from the on-board
20 computer. Angular position information will be exchanged between the computer and the arms in 13 bits, every 20 to 40 milliseconds.

Structural deflection at maximum (10 pounds) tip force will not exceed 1 inch.

The arm electric torque motors, controller torque motors,
6, 21 22 and TV camera operate off 28 vdc. Floodlights will operate off 110 vac. The largest electrical load is expected during payload unload and deploy. An average power demand of 1.008 KW is anticipated for a period of ten minutes during this process.

4.13.1.2.1 Rationale for Assumptions

Not applicable.

4.13.1.2.2 Data References

1. 22 pages VIII-1, VIII-2, VIII-7
2. 76 page 4-371
3. 7 page B.3-40; 13 pages 4-32, 4-33
4. 22 pages VIII-41, VIII-43
5. 22 pages VIII-6, VIII-89
6. 22 pages X-2, X-3, X-4, X-5
7. 40 page 177
8. 22 pages VIII-9B, VIII-9C
9. 40 page 179
10. 22 page VIII-21
11. 22 page VIII-67
12. 22 page VIII-59
13. 22 pages VIII-10, VIII-11
14. 7 page B.3-77
15. 22 pages VIII-89, VIII-90, VIII-91
16. 22 page VII-11
17. 22 pages VIII-95, VIII-96
18. 22 pages VIII-106 through VIII-110
19. 22 page VIII-5
20. 22 pages VII-24, through VII-28
21. 16 page 4-373
22. 22 page VIII-65
23. 166 pages 3-156

4.13.1.3 Payload Control/Display Panels

166 Payload services such as electrical power, fluid and gases filling, venting, and draining, etc., are provided by replacing blank panels with payload-unique panel kits. These panels will be located adjacent to orbiter service panels of a similar nature to aid mission preparations and operations.

4.13.2 Propulsive Fluid System Interfaces

166 For propulsive stages using LH_2 and LO_2 propellants, a fill, drain, dump and vent system interfacing with the MPS will furnish payload fuel requirements. Payload kits provide the interface connections from the payloads to the orbiter's plumbing.

Propulsive payloads that utilize storage propellants will have interface panels and connecting lines for propellant dump but are independent of the orbiter propulsion system lines.

4.13.3 Electrical/Instrumentation Interfaces

166 Standard connectors are provided for all hardwired electrical interfaces for payload power, monitoring, communications, navigational data, and caution/warning located in the payload bay, docking port, at the payload handling stations and the mission specialist station (MSS).

Payload electrical loads of 50 kwh of electrical energy from the orbiter's EPS is sufficient for the majority of payloads; Therefore, mission-to-mission changes to the base power system will not normally be required. For payloads requiring more energy, provisions have been incorporated to allow additional reactants and tankage to be installed in the payload box.

166 The baseline orbiter's time phased average power profile for a typical mission given in the EPS loading includes the specified payload requirement of 1000 watts average (1.5 kw peak) during all mission phases except the orbital operations phase where excess capability exists. This power is supplied to the payload bay and to the docking port in the form of regulated dc.

4.13.4 Payload Avionics Signal Interface

166 The baseline configuration accommodated the majority of payload and has sufficient flexibility that between-flight changes will be required only for infrequent special missions. This flexibility is provided by simple design extensions of the basic avionics subsystems, and simplifies crew procedures, flight preparations, and payload adaptation.

166 The orbiter communication system and the payload signal interface that is common to all payloads is shown in Figure 4.13.4-1. Except for the safety of flight signals (critical displays and safing discretes) which are routed to the forward flight station and the MSS, the individual selection, routing, and switching of payload signals is accomplished at the payload control panel of the MSS. Payload data communication flexibility is provided by multiple recorders, multiple transmission channels, general purpose alpha-numeric displays, and alterable memory computers. The junction and switches shown are repeated for simultaneous accommodation of up to five payloads. Digital commands from ground stations are encoded, and a payload data interleaver combines payload data for inclusion with the normal 128 kilobits per second operational flight instrumentation OFI (Mode 1) transmission. When orbiter OFI is not transmitting (Mode 2), up to 256 kilobits per second payload digital data can be accommodated.

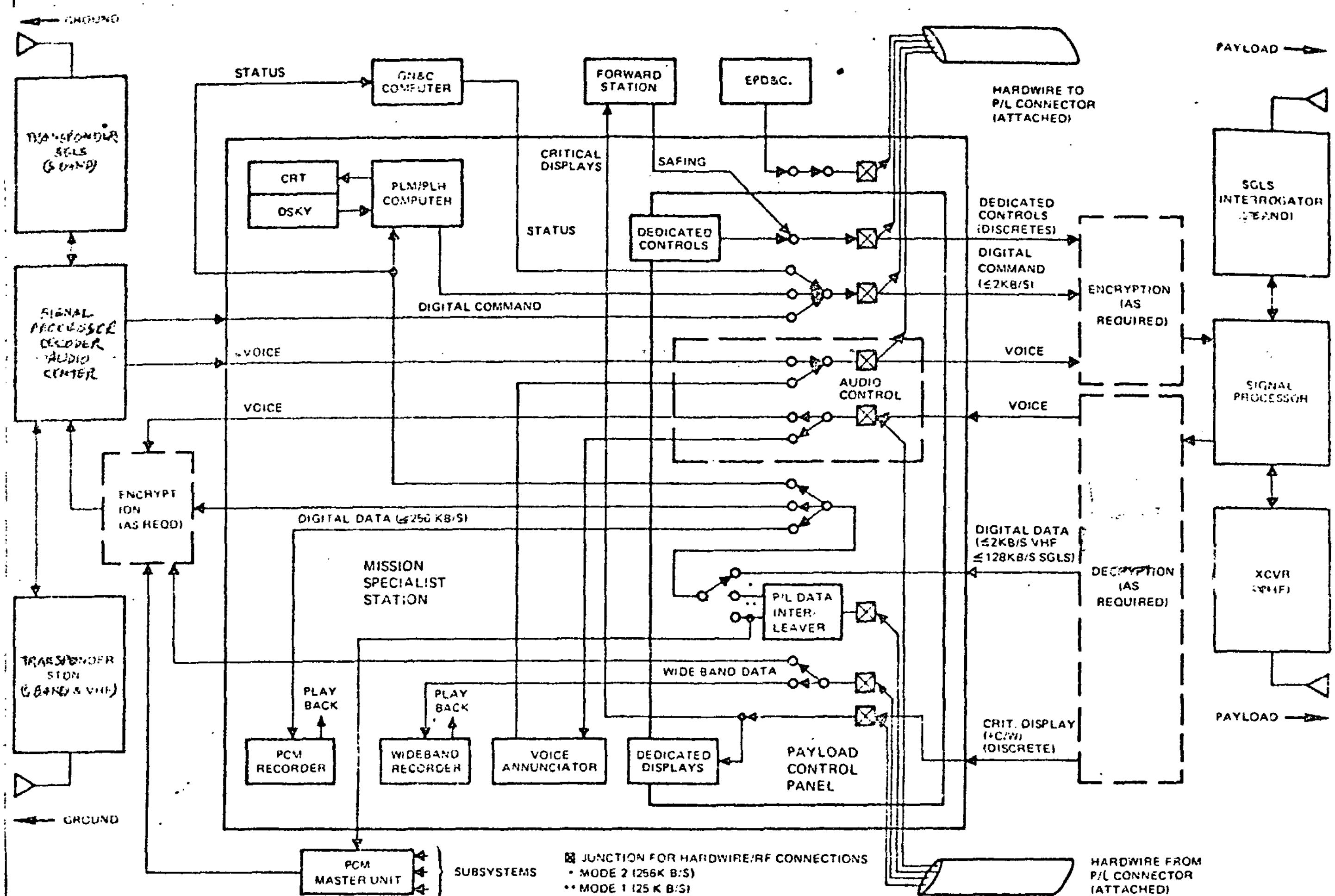


FIGURE 4.13.4-1 Payload Interface Signal Flow

166 The MSS provides for managing orbiter/payload interfaces and supporting operation of active payloads. The console houses most of the payload signal support equipment. The MSS caution and warning system annunciates payload critical malfunctions and is interconnected with the flight deck performance monitor during periods when the payload computer is not operating or the MSS is not manned.

The computational support system is required for status determination, control, limited onboard checkout, initialization, data display, and gross verification of proper data transmission for all payloads, and to support the manipulator arm control. The read-only tape memory enables the MDE to operate in either the payload signal processing or manipulator control mode, and provides for payload variation and program growth needs. Either of the payload data

Area	Payload Bay Environment
Acoustics	145-db interior overall sound pressure level
Pressure	Air vents control pressure to differential ± 2.0 psi
Humidity	Nitrogen purge by GSE – dew point – 65°F
Contamination:	
Ground	Payload bay liner, GSE purge
On-orbit:	
H_2O	Hold for 2 days and dump*
Waste	Hold for 7 days
Propellant	RCS thrusters do not impinge on payload bay
Material outgassing	Material selection and design criteria per NASA guidelines (OMSF Handbook 8060.1)

*Ice particles decay time due to drag and sublimation ≈ 3 hr

Table 4.13.5-1 Payload Bay Environment

Mission Phase	Low-Temperature Limit	Baseline Design	High-Temperature Limit	Baseline Design
Prelaunch	+40°F	GSE purge gas conditions bay to 75°F ±10°F for extreme 95% cold atmospheric temperatures (i.e., KSC: 44°F; WTR: 39°F)	+120°F	GSE purge gas conditions bay to 75°F ±10°F for extreme 95% hot atmospheric temperatures (i.e., KSC: 89°F; WTR: 88°F).
Launch	+40°F	Adiabatic payload bay wall will remain near prelaunch temperature	+150°F	Entry designs TPS; during launch, bay wall temperature rises a maximum of 18°F (i.e., 103°F maximum)
On-orbit (doors closed)	-100°F	Insulation and radiators maintain temperature above minimum.	+150°F	Extensive multidimensional analysis shows maximum temperatures less than 120°F
Entry and post landing	-100°F	Same as on-orbit minimum temperature	+200°F	Payload bay insulation designed to meet entry temperature requirements. Repressurization air cooled to less than 200°F by air vent heat sinks. Post-landing ground purge connection incorporated.

Table 4.13.5-2 Payload Bay Thermal Environment

166 recorders can be controlled at the payload control panel to play back data for ground transmission, subject to the bandwidth limitations of the ground RF link. In addition to the installed equipment which can accommodate most payloads, the MSS incorporates space and modular installation provisions for additional unique displays and controls provided with special payloads.

166 A three-way conference voice is available, as are dedicated digital data channels and a time-shared (with orbiter TV) wide band ground downlink. Also, payload access to orbiter antennas is provided for ground checkout of transmission functions.

4.13.5 Payload Environment Control

166 The orbiter incorporates a payload heat exchanger in the ECLSS Freon loop to provide a minimum cooling capability to the payload of 5200 Btu/hr. This exchanger is located near the payload bay service panel for convenient access for hookup during payload changes.

166 Provisions within the orbiter ECLSS distribution system permit the installation of payload-supplied fans and ducting for atmosphere revitalization of pressurized payload modules.

166 The characteristics of the environmental conditions in the payload bay are presented in TABLE 4.13.5-1. Thermal control provisions are presented in TABLE 4.13.5-2. In addition, the air vents provided for pressure control use heat sinks to cool incoming air during re-entry to 200°F.

4.13.5.1 Rationale

Not required

4.13.5.2 Reference

166 Pages 3-155 to 3-161

4.13.6 Payload Bay Door Mechanisms

The door mechanism rotates axially while maintaining contour control between door segments. Each co-linked segment is indenendently hinged to the fuselage longeron at three places and contains three latches on the door centerline edge. Latches also are located at the forward and aft cargo bay bulkheads. The latches first grasp and pull the door edges together against thermal distortions and the peripheral door-to-fuselage seals holding the door shut. To avoid large latch reaches, each latch actuates upon contact with its mating pin via a proximity sensor signal. The latches have a zip-fastener action, initiating at the point of first contact; i.e., each latch will pull the next adjacent latch into contact. This will continue even if one latch on each segment is failed or tripped.

Each door segment is supported by either three idler hinges or two idler hinges and one powered hinge. There are four powered hinges on each side of the orbiter. The four powered hinges are driven by a common torque shaft with redundant electric motors at each end of the shaft. Intermediate stopping, provided through sequencing proximity switches, provides manipulator positioning and deployment and retrieval. The powered hinges are driven by

4.13.6 Payload Bay Door Mechanisms

The door mechanism rotates axially while maintaining contour control between door segments. Each co-linked segment is independently hinged to the fuselage longeron at three places and contains three latches on the door centerline edge. Latches also are located at the forward and aft cargo bay bulkheads. The latches first grasp and pull the door edges together against thermal distortions and the peripheral door-to-fuselage seals holding the door shut. To avoid large latch reaches, each latch actuates upon contact with its mating pin via a proximity sensor signal. The latches have a zip-fastener action, initiating at the point of first contact; i.e., each latch will pull the next adjacent latch into contact. This will continue even if one latch on each segment is failed or tripped.

Each door segment is supported by either three idler hinges or two idler hinges and one powered hinge. There are four powered hinges on each side of the orbiter. The four powered hinges are driven by a common torque shaft with redundant electric motors at each end of the shaft. Intermediate stopping, provided through sequencing proximity switches, provides manipulator positioning and deployment and retrieval. The powered hinges are driven by

Ref.
Key

reversible jackscrew linear actuators. The reversible jackscrews are used so horizontal payload loading can be accomplished by simply disconnecting the gear boxes from the drive torque shaft, allowing the remaining system connected to the door to free-wheel as the doors are opened through use of GSE. Reliability is designed into the system through the use of independent drive systems at each end of the drive torque shaft. A drive system is automatically declutched from the torque shaft in the event of failures resulting in its loss.

4.13.6.1 Rationale

Not Required

4.13.6.2 Reference

166 Page 3-23

Ref.
Key

4.13.7 Rendezvous and Docking Sensors

4.13.7.1 Requirements

22 The potentiometers and tachometers mounted on the manipulator joints can also be utilized as rendezvous and docking sensors. Normally, they provide angular position and velocity of each joint to the on-board computer to close the force-feedback control loop. The on-board computer can calculate the exact position and velocity of the terminal device at any time with this information. After payload capture by the manipulator arm, knowing the position and velocity of the terminal device fixes the position and velocity of the target vehicle.

4.13.7.2 Rationale for Assumptions

Not applicable.

4.13.7.3 References

-22 pp. II-4, VIII-5, VIII-6

4.13.8 Payload Dedicated Recorders

A. 166 Two recorders are provided for payload wideband data and PCM data. The dedicated recorders and interface equipment will be installed as required as specialized units at the mission specialist station. Additional recorders may be transcribed or replaced during servicing of payloads. On board reduction of these records is not required; however, the recordings may be mounted and transmitted to the ground via the orbiter communication system.

4.13.8.1 Rationale

There may be specialized recorders on the payload.

Ref.
Key

4.13.8.2 References

166 Page 3-159

4.13.9 Payload Handling Station

The payload handling station is equipped with five closed circuit television cameras (CCTV) and two monitors. In addition, the handler will have visual contact. Two CCTV cameras are located on the ends of the manipulator arms which insure a close image of the final mating target. Two TV cameras are mounted in the payload bay to provide remote viewing of the payload attachment, release, and stowage operations. The fifth TV camera is used to help the operator align the payload during manipulator controlled docking operations.

4.13.9.1 Rationale

Not required.

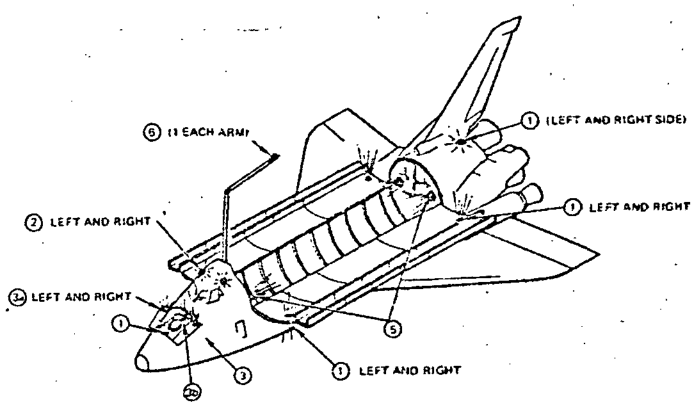
4.13.9.2 References

166 Page 3-159 and 3-160

Ref.
Key

4.13.10 Payload/Bay Lighting

166 The exterior lighting system fulfills the requirements for payload visual acquisition and tracking, determination of gross attitude, relative attitude and alignment, and gross range and range rate during terminal rendezvous, docking, deployment, and retrieval operations. Types of lamps and their locations are illustrated in FIGURE 4.13.10-1.



APPLICATION	QUANTITY	TYPE
1 ATTITUDE AND RUNNING LIGHTS	13	LOW VOLTAGE AC, TUNGSTEN-HALOGEN LAMPS.
2 ACQUISITION, TRACKING, ANTI COLLISION LIGHTS	2	XENON FLASHING – (MODIFIED APOLLO)
3 DOCKING LIGHTS – SPOTLIGHT (a)	2	LOW VOLTAGE AC, TUNGSTEN-HALOGEN LAMPS, 15K BCP
– FLOODLIGHT (b)	1	LOW VOLTAGE AC, TUNGSTEN-HALOGEN LAMP
4 EVA LIGHTS*	2 (OR AS REQMTS DICTATE)	LOW VOLTAGE AC, TUNGSTEN-HALOGEN LAMPS, 100 WATTS
5 PAYLOAD BAY FLOODLIGHTS	4	75 WATT FLUORESCENT
6 MANIPULATOR ARMS SPOTLIGHTS	2	LOW VOLTAGE AC, TUNGSTEN-HALOGEN, 100 WATTS
7 LANDING LIGHTS*	2	700K BCP QUARTZ, SEALED BEAM.

* NOT SHOWN

FIGURE 4.13.10-1

ILLUMINATION FOR ALL SPACE

LIGHTING CONDITIONS

REF. KEY

4.1 6 Miscellaneous Systems

This section includes those subsystems which are not described elsewhere in this document.

4.1 6.1 Purge and Vent System

166 The orbiter requires a controlled venting and purging throughout the vehicle because of hazardous fluids and gases, thermal heat dissipation, and air frame delta pressure limitation.

166 The large quantities of on-board hazardous fluids and combustible gases dictate the use of an inert gas dilutant for purging airframe cavities. On the launch pad, prior to fueling, the vehicle undergoes an air purge furnished by GSE. When the vehicle is loaded, the airframe is then purged with GN_2. Internal to the vehicle, the compartments are isolated to avoid gas mixing and are provided with independent non-propulsive vent ports.

166 The vent ports are sized for maximum flow rate during the boost phase of flight. A two-position mechanically controlled non-propulsive vent port having lanyard-type pull out plugs will be used to give the two stage orifice control required.

166 Two feeders (top fuselage and bottom fuselage) are provided from the aft located GN_2 tanks for the purge system. This system will be manually controlled by the crew.

4.1 6.1.1 Rationale for Assumptions

Not required

4.1 6.1.2 References

166 Pages 3-22

REF. KEY

4.16.2 Landing/Braking System

167 The orbiter vehicle is equipped with a conventional tricycle landing gear system. Each set of gears is a dual wheel having hydraulic retraction and hydraulic/free-fall extension. The two main gears and nose gear have the same material construction and components as conventional commercial aircraft. Each wheel has a folding drag brace, a gear activation system, up/down locks, and position indication system. The nose gear is equipped with a combination shimmy damper and steering system. Nose wheel limits are $\pm 30°$ either side of center. Turning radius of the vehicle is 116 feet. The main gear has an anti-skid brake system with locked-wheel protection at touchdown. Brakes and nose gear steering are electrically control-hydraulic activators. Accumulators in the hydraulic system provide stored energy for towing and braking.

167 Emergency fail-safe free-fall capability is provided without hydraulic power.

167 The nose gear uses two 32x8.8 type VII tires at 200psi pressure. The main gear uses four 44.5x16 type 21 tires at 220 psi pressure. For ferry missions the main gear tire pressure will be increased to 250 psi.

167 The braking system and the drogue parachute provide the capability to stop the vehicle within 6,000 feet on a dry runway and 10,000 feet on a wet runway.

167 The brake assembly is a carbon-on-carbon surface. The brakes are sized to provide five normal stops or to stop an orbiter during aborted takeoff without overheating.

REF. KEY

167 The drogue chute assembly is basically a B52 main chute and an Apollo pilot chute and mortar assembly.

167 For ferry flight, the static tire deflection is 32% of full travel. Maximum tire loading occurs during the horizontal takeoff roll when the elevon deflection causes nose gear liftoff. Main gear deflection at this condition is 50% of maximum travel.

167 For cross-wind landing, the gear is designed for a maximum ground wind speed of 35 knots.

167 During in-orbit flight, the landing gear system and the drogue chute system will be thermally protected by passive insulation and electric heater blankets. The design limit of the gear system is -60°F to 275°F.

167 The nose gear is equipped with ground tow attachments. Simulation of the tow system is not required.

4.16.3 Glide Brake System

167 The orbiter vehicle employs an integral split rudder design which provides aerodynamic dra forces. The design is manually controlled and may be adjusted for the amount of braking force required.

167 The glide brake (or speed brake) system is normally deployed at 250 knots airspeed at approximately 41,000 ft. During the landing glide and approach, the split rudder glide brake is deployed and modulated to control airspeed and glide range.

167 The nominal glide brake setting is 40°. This setting may be varied to increase range/speed by closing the brake or to decrease range/speed by opening the brake further.

REF. KEY

167 A 100% glide brake setting on the control gives a 70° brake deployment - a 30% setting gives an angle of 20°, and a 40% setting gives an angle of 30°.

4.16.3.1 References

167 Page 4-5 and 4-6

4.16.3.2 Rationale for Assumptions

Not required.

REF. KEY.

4.1 6.4 Ejection Seat Mechanism

166 Orbiters 1 and 2 will be equipped with two ejection seats for the Horizontal Flight Test phase and the first six manned orbital flight tests.

166 The ejection seat will operate at dynamic pressures from zero to a maximum of 650 psf. Existing ejection seats which have zero altitude/zero velocity performance capability will be used.

166 The ejection sequence will separate the canopy hatch and remove it from the ejection path. The crew member will automatically be positioned for safe ejection. The seat will then be ejected by a rocket-catapult blast through the canopy hatch.

166 The ejection seat mechanism will be removed from the operational vehicle.

4.1 6.4.1 Rationale

Not required.

4.1 6.4.2 References

166 Pages 2-2, 4-20, 7-11, 7-5, 7-3.

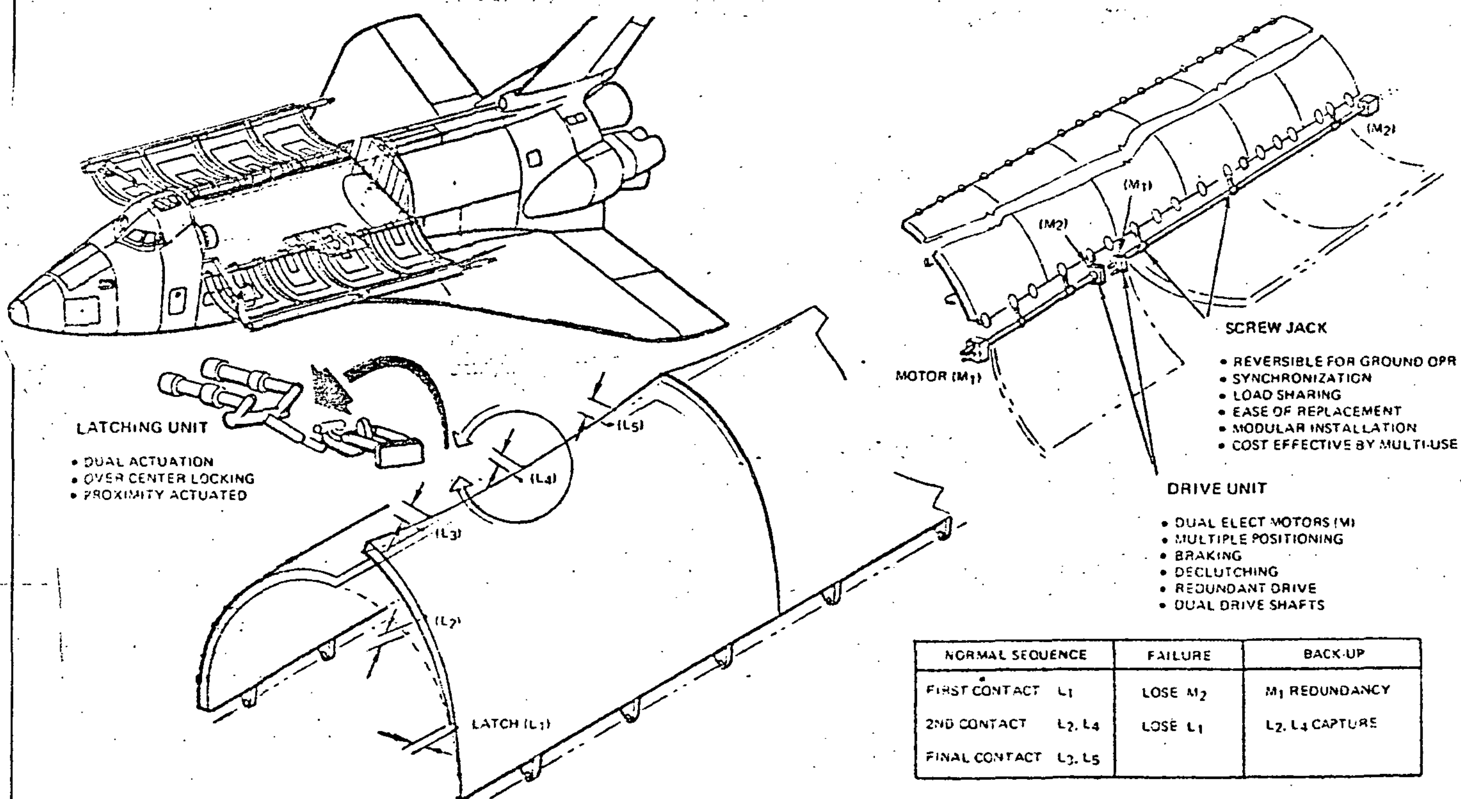
Ref.
Key

4.16.5 Docking Mechanism

The orbiter docking mechanism employs a neuter docking concept. Figure 4.16.5-1 schematically illustrates this concept. It presents the orbiter assembly and the corresponding docking element assembly. During the mission, the neuter active assembly is installed with its extendible tunnel retracted into the air lock. The tunnel extension provides the clearance for the docking operation. All docking elements will provide a passive assembly that interfaces with the orbiter docking assemblies.

The seals and latches are designed to accommodate relatively large tolerances accumulated from manufacturing, thermal, and pressurization distortions. The arrangement provides the capability for inspection and unscheduled maintenance and interface connector engagement in a shirtsleeve atmosphere after a docking has been accomplished. The total neuter assembly is contained within an 80-inch-diameter clear passage. Attenuation of docking forces is obtained by using a 10-inch stroke hydraulic shock unit. The standard utilities interface connectors are located around the 40-inch diameter clear opening. This arrangement provides the flexibility for replacement and for mission configuration changes. Visual alignment aids and displays assist the crew in aligning the docking element within the accuracy required for mating.

The passive docking assembly with an outward opening hatch for the baseline orbiter configuration provides the capability to attach a payload to the docking port by utilizing the payload handling manipulator or to dock with other orbiting elements or an orbiter vehicle. This tunnel assembly provides a minimum 36-inch clearance between the docking element and the orbiter and eliminates need of a loose docking adapter. Separation and jettison provisions provide positive separation of the docking assembly for emergency or contingency conditions.



NORMAL SEQUENCE		FAILURE	BACK-UP
FIRST CONTACT	L_1	LOSE M_2	M_1 REDUNDANCY
2ND CONTACT	L_2, L_4	LOSE L_1	L_2, L_4 CAPTURE
FINAL CONTACT	L_3, L_5		

Figure 4.16.5-1
PAYLOAD DOOR ACTUATION AND LATCHING

4.17 On-Board Computer System

The Shuttle Vehicle on-board Computer System is functionally divided into four semi-autonomous federated systems: the GN&C, the Performance Monitor, the Payload system, and the Main Engine Controllers.

Key features of the on-board computer system are:

166
- Computation tasks grouped and allocated for manageability, separation of high development activity, isolation of high traffic data from flight critical functions and ease of integration.
- Display, keyboard and non GN&C computation functions mechanized in a single type, small computer augmented by tape mass memory, enables low development and equipment costs to satisfy redundancy requirements.
- No direct exchange of data between computers performing redundant functions (multiple cross switching but no cross strapping), low data rate and non-interrupt transfer of data, memory protection.

166 A block diagram of the on-board computer system is shown in Figure 4.17-1.

166

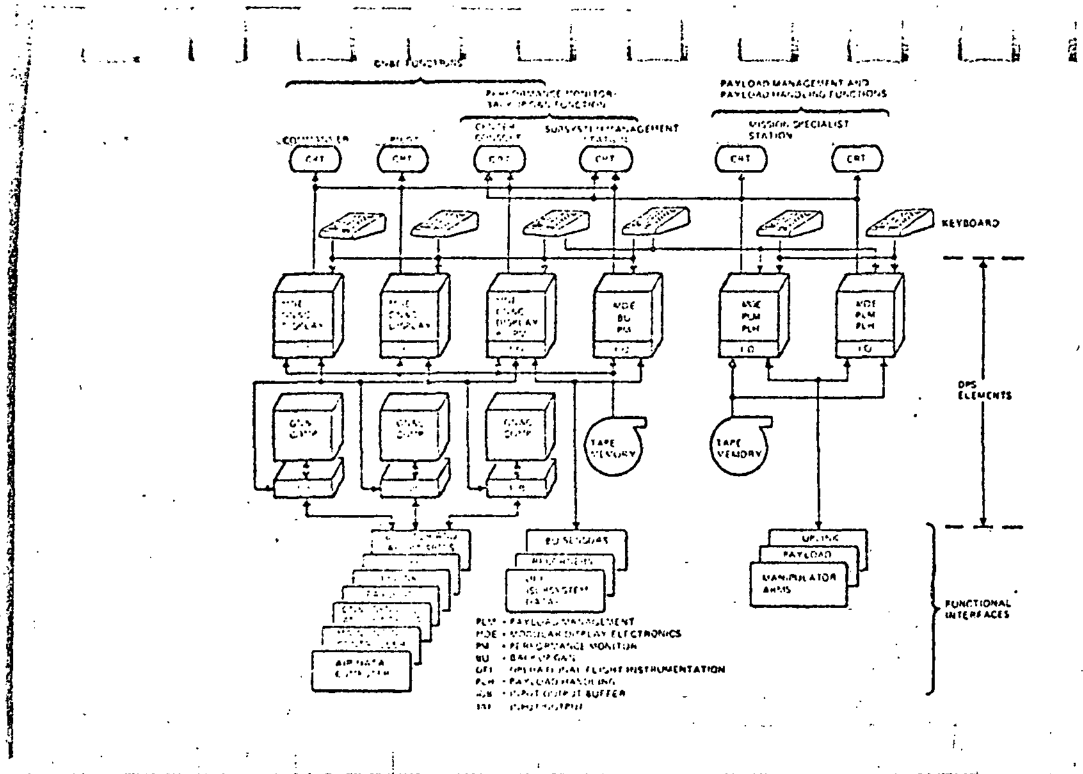


Figure 4.17-1. Shuttle Vehicle On-Board Computer System and Interfaces

166 The input/output (I/O) interface between the computers and the other vehicle subsystems is implemented using a modular design concept to provide the flexibility to accommodate changing requirements and permit early computer design. The modular display electronics I/O, computing, and display subassemblies are combined into a single unit, which permits a simple design and is feasible because the identified I/O channel requirements are primarily multiplexed, serial digital. The I/O to the GN&C computers is significantly more complex, and subject to change; therefore, these functions have been grouped into a separate unit identified as the GN&C I/O buffer. The I/O buffer provides the functions of signal conversion, multiplexing, and transfer of data to and from the computer memory. The transfer functions are accomplished by using a direct memory access (DMA) channel which operates independently of the CPU, except for initialization. Modifications to the I/O buffer capabilities are accomplished by exchange of available standard modules and/or where necessary exchanging custom I/O modules. The

61 & 42 transmission of commands and data to and from the Main Engine Controllers will be via a dedicated serial interface

F-398-8-A

with DMA interface to controller computers.

41 Guidance, Navigation, and Control (GN&C) System

The primary GN&C system consists of three 64K 32-bit word computers dedicated to the solution and status of the Orbiter guidance, navigation, and control functions. The computers contain identical programs to allow a triple redundant computer system mode of operation.

Data will be routed to each computer to allow program vote, average, or compare on input data. Output data will be voted, or compared, at the actuators rather than at the computer.

41 & 166 The Performance Monitor System Modular Display Electronics (PM MDE's) will contain backup G&N programs so that a "get me home" capability is available for all flight phases in the event of a generic software error or other critical system failures. Separate dedicated sensors and electronics are used with this system.

41 Display data for crew observation of the GN&C system performance and status is provided by the GN&C Modular Display Electronics (MDE) processors which are connected to CRT displays and keyboards.

166 The CRT's can operate in tabular alphanumeric, or graphic mode and contain an interactive selection feature for rapid indexing and data selection. Major display mode selection and manual data entry are accomplished via an associated keyboard. Switching is provided to permit several modes of CRT/MDE interconnection for flexibility of access to different display generation programs which may be accommodated among the MDE's.

Performance Monitor (PM) System

41 The PM system consists of two MDE processors dedicated to the monitor and display of non-GN & C systems status or to the solution and display of the backup guidance and navigation function. One of the PM MDE processors capabilities is obtained by time sharing the center console GN & C MDE, between the GN & C display functions, the PM function, and the backup G & N functions. Reload of the MDE memory for a change of functions will be provided by a tape read-only memory under control of the crew.

41 Payload (PL) System

The PL system consists of two MDE processors dedicated to the status, control, checkout, initialization, and display of payload data. The PL computers which contain identical programs will both be active but only one will be in control. The PL system will not be required to process onboard experiment data which will be either recorded onboard or transmitted to the ground for processing. Data transmission and recording will not require the PL computer participation. Memory growth capability for the PL computer system is provided by a dedicated read-only tape memory unit.

42 Main Engine Controller

The controller is a single integral electronics package mounted directly on each of the three main engines. The controller contains electronics for interfacing with the engine and the Vehicle with double and triple redundancy. The controller also contains two independent 16 bit HDC-601 Digital Computers, each of which performs the computations necessary for full-authority closed-loop control of the engine thrust and mixture ratio in response to commands from the Vehicle and data from the engine sensors.

Normally, Computer No. 1 is in control and Computer No. 2 is in operational standby. In the event of a failure, control is automatically transferred to Computer No. 2 without impairing engine operation.

4.18 GN&C On-Board Computer & Interface Systems

4.18.1 GN&C General Functional Description

The GN&C subsystem (Figure 4.18.1-1) provides (1) automatic and manual control capability for all mission phases except docking, which is manual only; (2) guidance commands that drive control loops and provide steering displays for the crew; and (3) inertial navigation updated by star and horizon sensors for autonomous orbital flight and by RF navigation aids for rendezvous, approach, and landing. Three independent redundant strings and a backup provide FO/FS redundancy. The equipment is divided into three subsystems: the aerodynamic stability augmentation (ASAS); the primary GN&C; and the backup GN&C. Sensor requirements are found in table 4.18.1-2. Section 4.9 discusses overall GN&C subsystem functional requirements. The various system control modes are summarized in Table 4.18.1-1.

The basic aerodynamic stability of the orbiter is augmented by using the ASAS, an F-14 type conventional system that employs body mounted rate gyros and accelerometers. Gain scheduling is provided by inputs from a DC-10 type digital air data computer and deployable probes. Side stick rotation controllers, rudder pedals, and trim controls allow manual control, and the GN&C computer provides commands to the aerodynamic stability augmentation subsystem for automatic flight control functions, such as automatic landing.

TE | THE SINGER COMPANY SIMULATION PRODUCTS DIVISION BINGHAMTON, NEW YORK | PAGE NO.
/. | REP. NO.

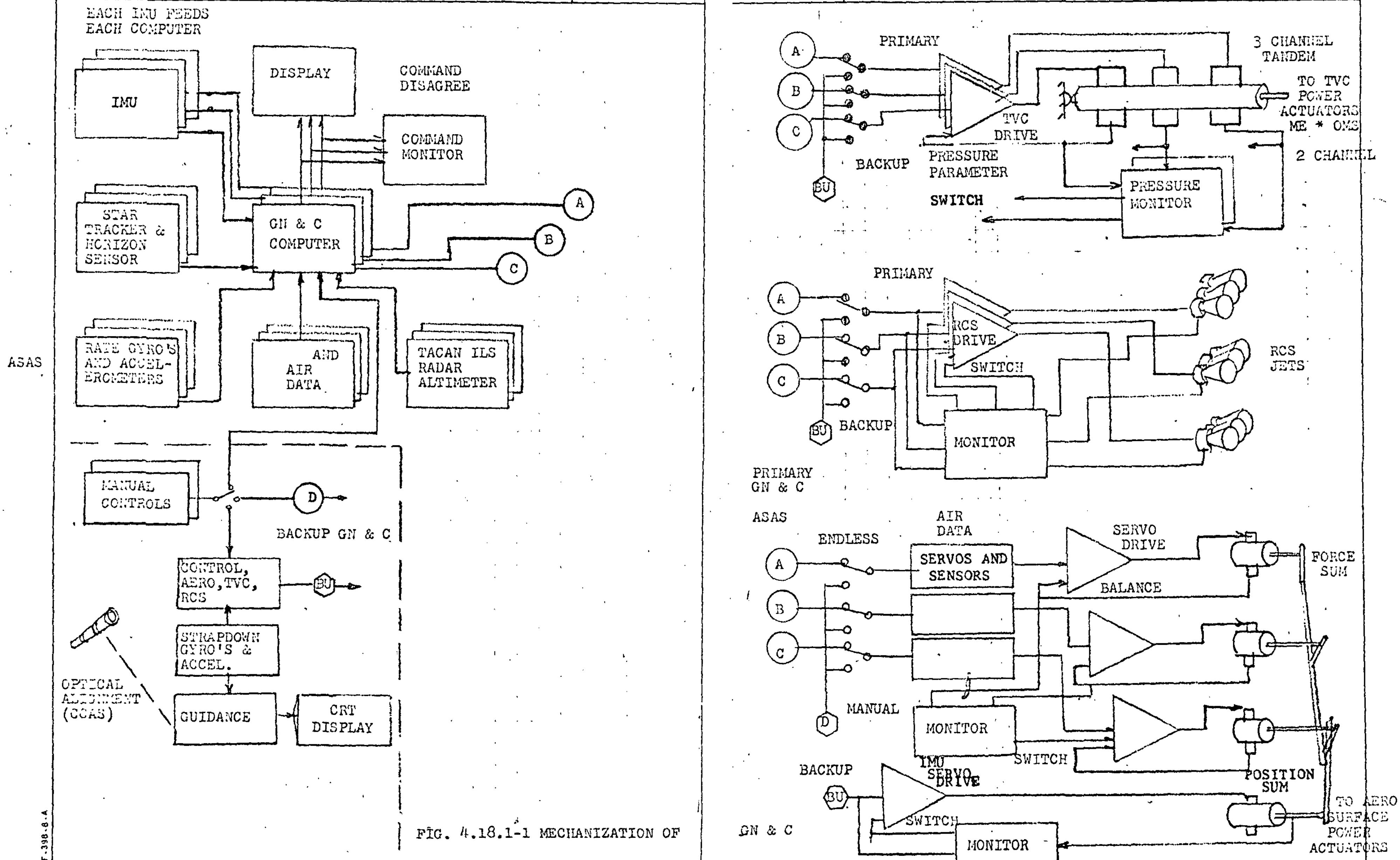


FIG. 4.18.1-1 MECHANIZATION OF GN & C

F.398.8.A

Mode	Flight Phase	Characteristic
Command	.Boost/Insertion (IVC) Orbital (OMS,TVC,RCS), Entry, Aero,Landing	.Crew initiates guidance and control modes and monitors control automatic through GN&C computer.
Control Stick Steering	.Orbital (OMS TVC or RCS), Entry, Aero,Landing	.Manual control and guidance displays through GN&C computer . Rate command, attitude hold, and RCS minimum impulse: RCS translation-acceleration command
Manual	.Aerodynamic	.Manual control through ASAS Rate command and damp
Manual (backup)	.Boost/insertion(IVC) Orbital (OMS TVC or RCS), Entry,Aero, Landing	.Backup sensors and computer, Rate command and damp .Direct RCS jet

Table 4.18-1-1 Control Modes

The primary GN&C provides guidance, navigation, and control for all flight phases with both automatic (command) and manual (control stick steering) modes. The flight control loops used with the main engine and orbital maneuver subsystem (OMS) thrust vector controls, and the reaction control subsystem (RCS) are closed through the GN&C computer; aerosurface control uses the computer and the ASAS. An additional manual mode for aerodynamic flight control bypasses the computer and uses only the ASAS. Rate gyros and accelerometers provide sensing for damping and load relief. Attitude information is obtained from the inertial measurement unit (IMU). Air data is provided by vehicle nose pressure ports at high altitudes, and by redundant probes deployed at lower altitudes.

The backup GN&C subsystem provides a safe return capability for all flight phases. It is separate from the primary subsystem and uses dedicated sensors and electronics. Backup flight control is manual; rotation controller and rudder pedals inputs are used. The change from tip pods to body-mounted RCS jets eliminates severe control cross-coupling and allows direct RCS control on-orbit as additional backup. Steering information for ascent, entry, and terminal area energy management is provided through visual interpretation of the backup G&N data on a cockpit cathode-ray tube (CRT). Backup system alignment is accomplished by an optical sighting device, much like the CSM crewman's optical alignment sight.

Currently, a gimbaled IMU provides the navigation reference with horizon and star sensors for autonomous alignment and state vector update. During active rendezvous, TACAN is used for range much as it is used in military air-to-air refueling operations. Range rate is derived from the range data in the GN&C computer. A star sensor acquires a target light to provide bearing. Rendezvous with a passive target employs ground tracking of the target combined with orbiter on-board navigation and, when required, range, range rate, and angle data from rendezvous sensors (considered part of the payload). TACAN update is used to assure approach and landing path intercept above 10,000 feet altitude.

Automatic landing is accomplished via a computed flight path generated in the GN&C computer using the inertial navigation system for reference with continuous updates from TACAN and instrument landing system (ILS). Radar altimeter updates are used near touchdown. The two segment (15-and3-degree) glide slope approach requires separate ground ILS transmitters for each segment. Rollout control similarly uses the GN&C computer and inertial navigation system with continuous ILS localizer updates.

For all critical mission phases where automatic switching (i.e., allowable recovery times exceed pilot capability) is required, the three independent strings are operated in parallel with the backup string available. (Table 4.18.1-3). The required automatic detection and switching are accomplished through comparisons with a predetermined threshold.

Component	Characteristic	Equipment Capability (1a)	Budget Budget (1a)
IMU			
Accelerometer	Bias (g)	25	100
	Scale Factor (%)	0.0025	0.0150
	Input-Output axis misalign (sec)	10	40
Gyro	Scale Factor (%)	0.05	0.10
	draft (/hr)	0.03	0.05
	Mass unbalance input axis (/hr/g)	0.1	0.1
	Mass unbalance spin axis (/hr/g)	0.03	0.10
	(/hr/g)	0.0003	0.100
Gimbal	Readout (two-speed) (sec)	25	72
Horizon sensor	Angie (min, limited by horizon definition	6	9
Star sensor	On-Axis random and bias (min)	0.5	0.6
Backup optical signt	instrument, alignment, limit cycle (min)	11	12
IMU Horizon sensor/star sensor	Alignment (min)	1.0	1.2
Backup Sensor gyro	Scale factor (%)	0.05	0.0065
	Bias (/hr)	0.25	0.50
Accelerometer	Scale factor (%)	0.005	0.10
	Bias (g)	400	500
RF NAVAIDS			
Radar altimeter	Accuracy at touchdown ( )	1	1.5
TACAN	Range accuracy-specification value ( )	300	600
	Bearing accuracy-specification value ( )	0.5	1.0
ILS	Azimuth, localizer (deg)	0.03	Minimum
	Elevation glide slope (deg)	0.005	Category 11

Table 4.18.1-2 Sensor Performance Requirements

F-398-8-A

First Failure		Second Failure		
Element	Detection and Correction Sequence	Element	Detection and Correction Sequence	GN&C Status
IMU	A	IMU	E	Safe
		Navigation and Computer	B	Operational
		Drivers, servos, sensors	C	Operational
		Backup Channel	D	Safe
Navigation aids computer	B	IMU	A	Operational
		Navigation aids, computer	F	Safe
		Drivers, servos, sensors	G	Safe
		Backup channel	D	Safe
Primary Drivers Servos & Sensors	C	IMU	A	Operational
		Navigation aids, computer	F	Safe
		Drivers, servos, sensors	G	Safe
		Backup channel	D	Safe
Backup	D	IMU	A	Safe
		Navigation aids, computer	B	Safe
		Drivers, servos, sensors	C	Safe

Detection and correction sequence

A. Detection: Software compare in GN&C computers
Correction: Software midvalue select in GN&C computers
B. Detection: Hardware compare in command monitor and in pressure driver monitor
Correction: Manually disengage failed channel, or pressure driver monitors automatically disengage failed channel
C. Detection: Hardware compare in pressure driver monitors
Correction: Fail soft by force fight-automatically disengage failed channel
D. Detection: Servo monitor (aero) CRT backup guidance BITE
Correction: None required for space since normally disengaged auto disengage for aero
E. Detection: Software compare in GN&C computers
Correction: Manual switch to backup automatic (software freeze on last inertial disposition discompare); if isolated by BITE, option to revert to primary

Table 4.18.1-3 GN&C Failure Recovery Sequences (continued on next sheet)

Table 4.18.1-3 GN&C Failure Recovery Sequences (continued from previous sheet)

F. Detection: Hardware compare in command monitor and in pressure driver monitor
   Correction: Manual switchover to backup GN&C or pressure driver monitors automatically switch to backup GN&C

G. Detection: Hardware compare in pressure driver monitor
   Correction: Fail soft by force fight and automatically engage backup channel, if isolated by BITE, option to revert to primary channel

Note: The detection and switching techniques used cover the different combinations of failures. The major ones are described above.

BITE - built-in test equipment

4.18.2 GN&C System Elements & Interfaces

This section provides detailed descriptions of the least replaceable units (LRU's) included in the GN&C system.

The size and weight of the individual LRU's are included in Table 4.18.2-1. This table does not include the various displays, controls and navigational aids which interface directly with the primary GN&C equipment. Table 4.18.2-2 contains a tabulation of the individual LRU power requirements by mission phase. Individual detailed LRU descriptions follow.

LRU	Qty.	Weight Lb/LRU	Volume Ft^3/LRU	Total Wt.-Lbs.	Total Vol-Ft^3
Air Data Package	3	25	0.52	75	1.56
Attack Angle Transducers	3	10	0.15	30	0.45
IMU & Power Supply	3+3	47	0.80	141	2.40
Star Tracker and Control Unit	3+3	10	0.18	30	0.54
Horizon Sensor (1) & H/S Heads (2)	3+6	22	0.20	66	0.60
Rate Sensor	9	1.5	0.02	13.5	0.18
Accelerometer	6	0.8	0.01	4.8	0.06
GN&C Computer	3	40	1.03	120	3.09
Prog I/O Processor	3	35	1.03	105	3.09
Memory Unit (0)	3	30	0.70	90	2.10
TVC Driver Unit	3	8	0.12	24	0.36
MTVC Electronics	1	6	0.10	6	0.10
Aero Control Elect.	3	13	0.29	39	0.87
Speed Brake Driver	1	6.4	0.10	6.4	0.10
APS Logic/Driver Unit	2	19.6	0.48	30.2	0.96
Totals, GN&C LRUs	55			789.9 lbs.	16.46 Ft^3

Table 4.18.2-1 System Weight and Volume of Orbiter GN&C LRUs

F-398-8-A

Table 4.18.2-2

GN&C Power Requirements by Mission Phase - Orbiter

GN&C Equipment	Qty/ System	Unit Power (DC/AC)	System Power (DC/AC)	Duty Factor In				Average Power in Watts			
				Ascent	Orbit	Entry	Aero	Ascent	Orbit	Entry	Aero
Air Data Package	3	0/89	0/267	0	0	1	1	-	-	0/267	0/267
Attack Angle Transducer	3	0/10	0/30	0	0	1	1	-	-	0/30	0/30
IMU & Power Supply	3	50/70	150/210	1	1	1	1	150/210	150/210	150/210	150/210
Star Tracker Equipment	3	15/6	45/18	0	0.1	0	0	-	5/2	-	-
Horizon Sensor Equipment	3	22/0	66/0	0	0.1	0	0	-	7/0	-	-
Rate Sensor	9	0/2	0/18	1	0	1	1	0/18	-	0/18	0/18
Accelerometer	6	0/0.5	0/3	0	0	1	1	-	-	0/3	0/3
GN&C Computer Equipment	3	150/0	315/0	1	1	1	1	315/0	315/0	315/0	315/0
MTVC Electronics	1	12/8	12/8	1	0	0	0	12/8	-	-	-
Aero Control Elect.	3	42/29	126/87	0	0	1	1	-	-	126/87	126/87
APS Logic/Driver Unit	2	2/2	4/4	1	1	1	0	4/4	4/4	4/4	-
Speed Brake Driver Unit	1	5/10	5/10	0	0	1	1	-	-	5/10	5/10
					Totals			511/271	481/216	600/629	596/625

F-398-8-A

4.18.2.1 Air Data Equipment

The Air Data Equipment for shuttle has been selected for maximum use of existing equipment and permits a phased development program.

For the Mark I Orbiter, the horizontal flight control system requires only the computation of dynamic pressure for gain scheduling of the stability augmentation system; displayed parameters include altitude, altitude rate, mach number, and airspeed. These parameters are computed by the Air Data Package from input probe pressure signals corresponding to static and total pressures.

The selection for Air Data Package is a modified version of the Honeywell Digital Air Data System designed for the DC-10. Utilizing a computer (7 CPU cards, 1 memory card), this device converts pressure signals to digital format, computes flight parameters, and produces both analog and parallel digital output to displays and to the aerodynamic control electronics.

Updating the Air Data System for the Mark II vehicle requires the addition of an Angle of Attack Transducer package, which produces digital outputs permitting computation of angle of attack by the GN&C computer. This device consists of two differential pressure-to-electrical signal transducers and utilizes module building blocks in converting to the required digital outputs.

Three each of the Air Data and Angle Transducer units are required in each system, for single-string redundancy in a spatially diverse vehicle layout.

4.18.2.2 Rate and Accelerometer Sensors

Sensors for both the orbiter and booster vehicle GN&C systems include both body rate gyros and accelerometers for control loop stabilization.

The hardware elements for both rate and acceleration sensors are based on existing equipment in current production by Honeywell for the Grumman F-14 flight control system. Available configurations include both dual and triple redundant single-axis packages utilizing heaterless GNAT gyros, and triple accelerometer packages. In applying the single-string system criterion of spatial diversity to the redundant gyros and accelerometers, subassemblies and circuits of the F-14 units will be handled as building blocks in repackaged sensors of four types: roll, yaw, and pitch body rate packages, and a 2-axis accelerometer package; each LRU includes the inertial sensor, plus loop electronics and power supply. Separating rate sensors by axis is assumed to be required to permit locating the sensors at different airframe body stations because of vehicle bending mode variation by axis.

4.18.2.3 IMU

Each IMU consists of two LRU's; the platform and the power supply. Both are shown in Figure 4.18.2-1.

The platform has four gimbals with appropriate synchros, resolvers and torque motors for each gimbal. The angular sequence starting with the inner gimbal is pitch, roll, and yaw with the fourth gimbal providing redundant roll. The stable element (inner member) contains two 2DOF gyros with SRA's directed along the pitch and yaw gimbal axes, respectively. One gyro controls the roll and yaw platform gimbals while the other controls the pitch gimbal axis. The platform baseline is typified by a Kearfott KT-70.

The accelerometers, also mounted on the stable member, consist of a two axis accelerometer, measuring accelerations in the X and Z axis and a single axis accelerometer measuring Y axis accelerations. This definition applies when the gimbal angles are driven to zero in all axis with respect to the vehicle body axis system. The coordinate system X, Y, Z, defined in the conventional sense, corresponds to the roll, pitch, and yaw axes of the vehicle. The gimbal torquing electronics and the accelerometer rebalance electronics are located in the platform assembly.

4.18.2.4 Star Tracker System

The star tracker is a strapped down optical sensor using electronic gimbaling to determine star positions within the eight degree diameter field-of-view (FOV). Usage of the tracker is extended to include acquiring and tracking a space station light beacon. The acquisition mode, results in a scan of the entire FOV after which the brightness object is selected. The tracker then enters a tracking mode in which the selected object is scanned over a very small FOV on the order of 16 arc-minutes. The position of the object is measured in two axes with respect to the boresight of the tracker.

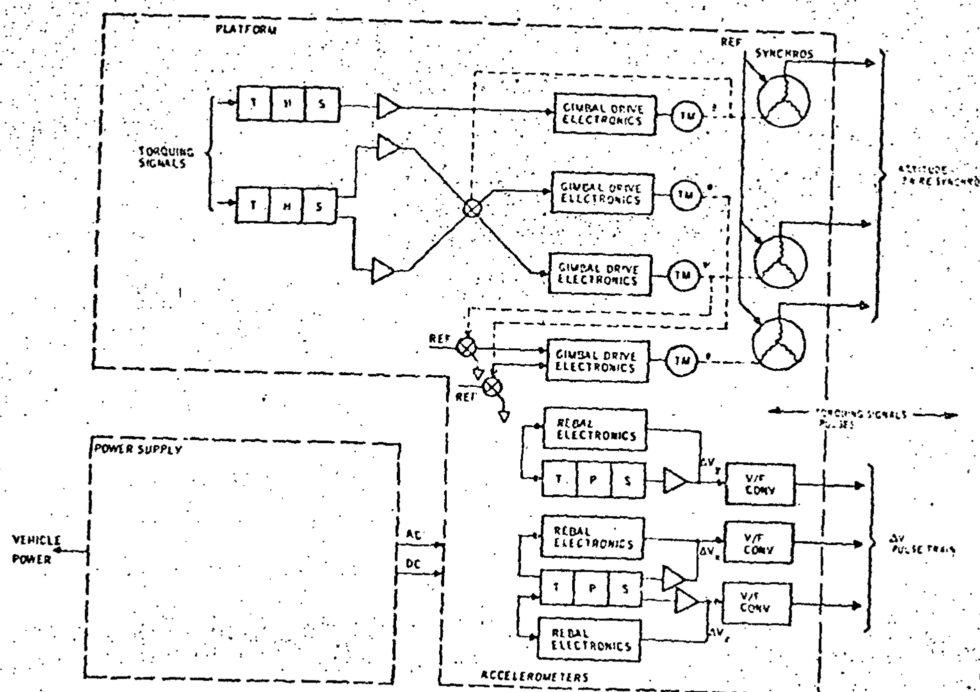


Figure 4.18.2-1 IMU Block Diagram

The star tracker(baseline is typified by the ITT Dual Mode Star Tracker) consists of an optical lens system, photosensor and electronic circuitry as shown in Figure 4.18.2-2. The lens gathers and brings to focus the radiant energy from the source at the photo cathode of the multiplier phototube. The photo cathode surface forms an electron image of the focused light source. An accelerating voltage applied between the photo cathode and a limiting aperture causes electrons from a particular area of the photo cathode (instantaneous photo cathode area) to pass through the aperture. This then defines an instantaneous FOV of a small region in space. A multiplier section behind the aperture amplifies the signal.

Deflection coils placed around the image section of the phototube provide a means of deflecting the electron image. A search sweep generator and a track sweep generator provide deflection signals to the coil to cause the electron image to sweep across the aperture during the acquisition and track modes, respectively. The search sweep generator is used ~~to~~ scan the entire tracker FOV whereas the track sweep generator scans a small preselected area.

The video amplifier and star selection circuits amplify the star presence signal, and set a reference level corresponding to the largest signal encountered in the FOV. On receipt of a tracking signal from mode control, the tracker scans the FOV and stops when the brightest star enters the instantaneous FOV. Once this happens, the tracker enters the tracking mode. The tracking loop circuits develop an analog error signal which is used to control the deflection-soil signals so that the fine scan is centered on the star image. The output error amplifiers sample the deflection coil signal and provide output error signals to the computer.

The power supply requires both plus and minus 27 vdc input power.

The star tracker system is comprised of 3 LRU's:

Star Tracker/Shuttler

Sun/Shield/Sun Sensor/Shutter Assembly

Control Unit

4.18.2.5 Horizon Sensor

The horizon sensor, used operationally during navigation is of the conical scan variety. The sensor approach uses two horizon sensing heads interfacing with a signal processing box. A block diagram of the sensing system is shown in Figure 4.18.2-3.

Each head contains an optics section with a motor driven rotating mirror. A bolometer/amplifier provides a signal indicating the level of radiance received through the rotating FOV. A marked change in radiance levels indicates a horizon crossing. Under normal operation, two horizon sensor crossings per scan are received by each head. A motor packoff provides a reference timing signal which locates the center of each scan pattern.

Within the processor box, pitch information from each head is derived from the horizon crossing signals and the reference pulse.

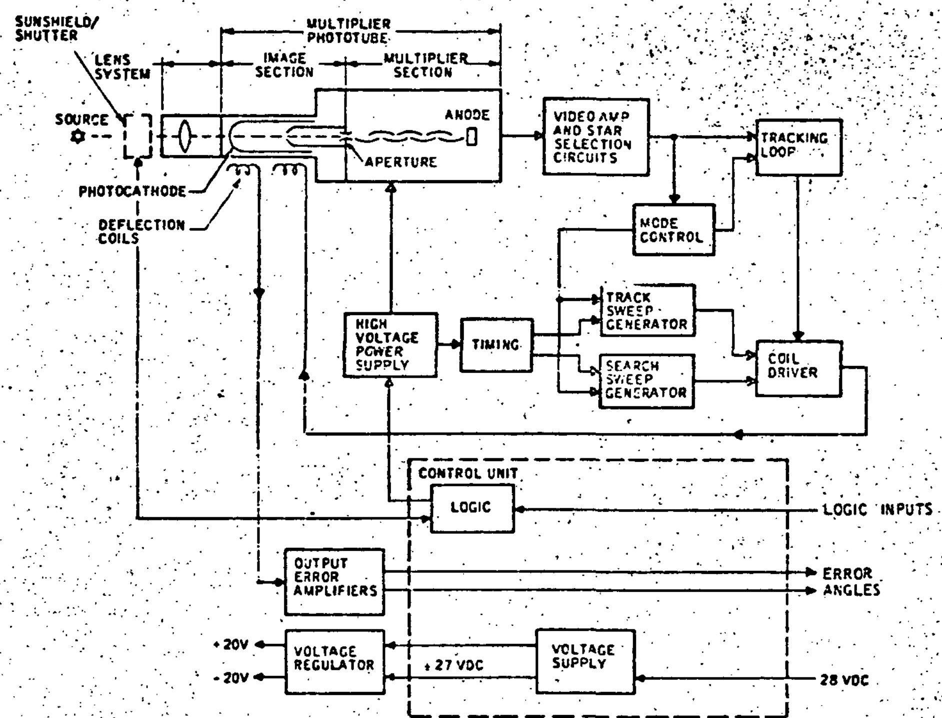


Figure 4.18.2-2 Star Tracker Block Diagram

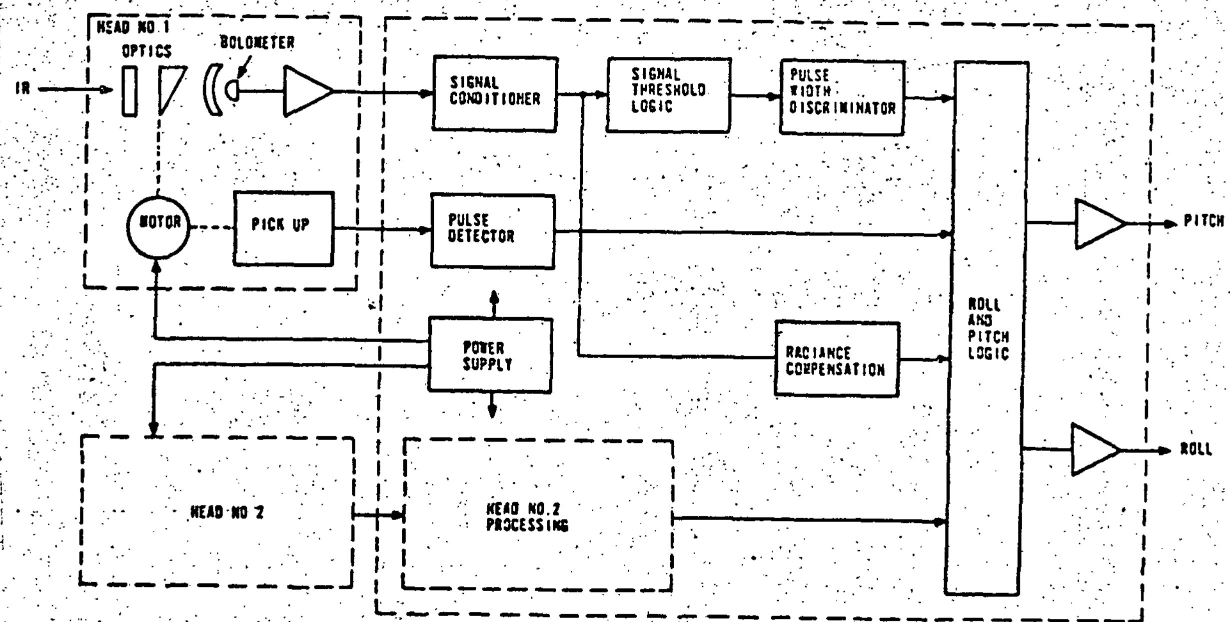


Figure 4.18.2-3 Horizon Sensor Block Diagram

The pitch information from each head is averaged to obtain a pitch angle output signal. A roll output signal is obtained by comparing the period of "Earth crossing" pulse from each head.

Radiance compensation circuitry is provided to suppress the effects of horizon anomalies on the output performance of the horizon sensor.

The horizon sensor, typified by the Barnes 13-156 Conical Scan Horizon Sensor System, provides a highly accurate, long life, local vertical reference over a wide range of altitudes (80 - 6000 nautical miles). The optics are designed to view the CO_2 special band (15 microns) which is the optimum spectral band for minimizing horizon variations. The output error angle signals are linear over a ± 5 degree region saturating at 10 degrees. A sun presence signal, detected in the short wavelength spectral band, inhibits the signal output thus preventing corruption of the navigation processing with erroneous updates.

4.18.2.6 Analog SAS

The selection of an analog SAS shown in Figure 4.18.2-4 as opposed to a digital SAS was based on the fact that there are several flight proven and operational fail operative and fail safe analog flight control systems in use today. All aircraft currently flying employ this method of control. There is no production digital stability augmentation other than Apollo. Therefore, analog is tentatively selected as the conventional off-the-shelf approach in order to minimize risk and be available for horizontal flight.

Analog aerodynamic flight control electronics are packaged in three identical 3-axis LRU's in adherence to the spatial diversity concept of redundant hardware independence. In each, the pitch, roll and yaw control channels receive control commands from the GN&C computer and provide crossfed commands to the elevon control surface actuators, and in yaw channel from transducers in pedals. Capability also exists in the roll and pitch channels to provide surface actuation signals in response to analog manual input commands from transducers on the center control stick. Vehicle body rate and lateral and normal acceleration feedback signals are introduced for axis stabilization and a dynamic pressure signal from the air data system is used in each axis for gain scheduling.

In addition to the control electronics and actuator servo amplifiers, each LRU includes an independent set of middle-select comparators and the required power supplies. In packaging of the aero control electronics, extensive use will be made of existing aircraft flight control hardware and packaging techniques. Plug-in circuit boards will be modified from C-5 and F-14 hardware; wire-wrap interconnecting base plate and fabricated aluminum enclosure are adapted from the existing F-15 aircraft flight control electronics hardware.

4.18.2.7 <u>TVC Electronics</u>

Hardware for thrust vector control is packaged in four LRUs" 3 identical TVC gimbal servo driver units and one manual TVC electronics box.

Each of the TVC gimbal servo driver units contains 8 servo

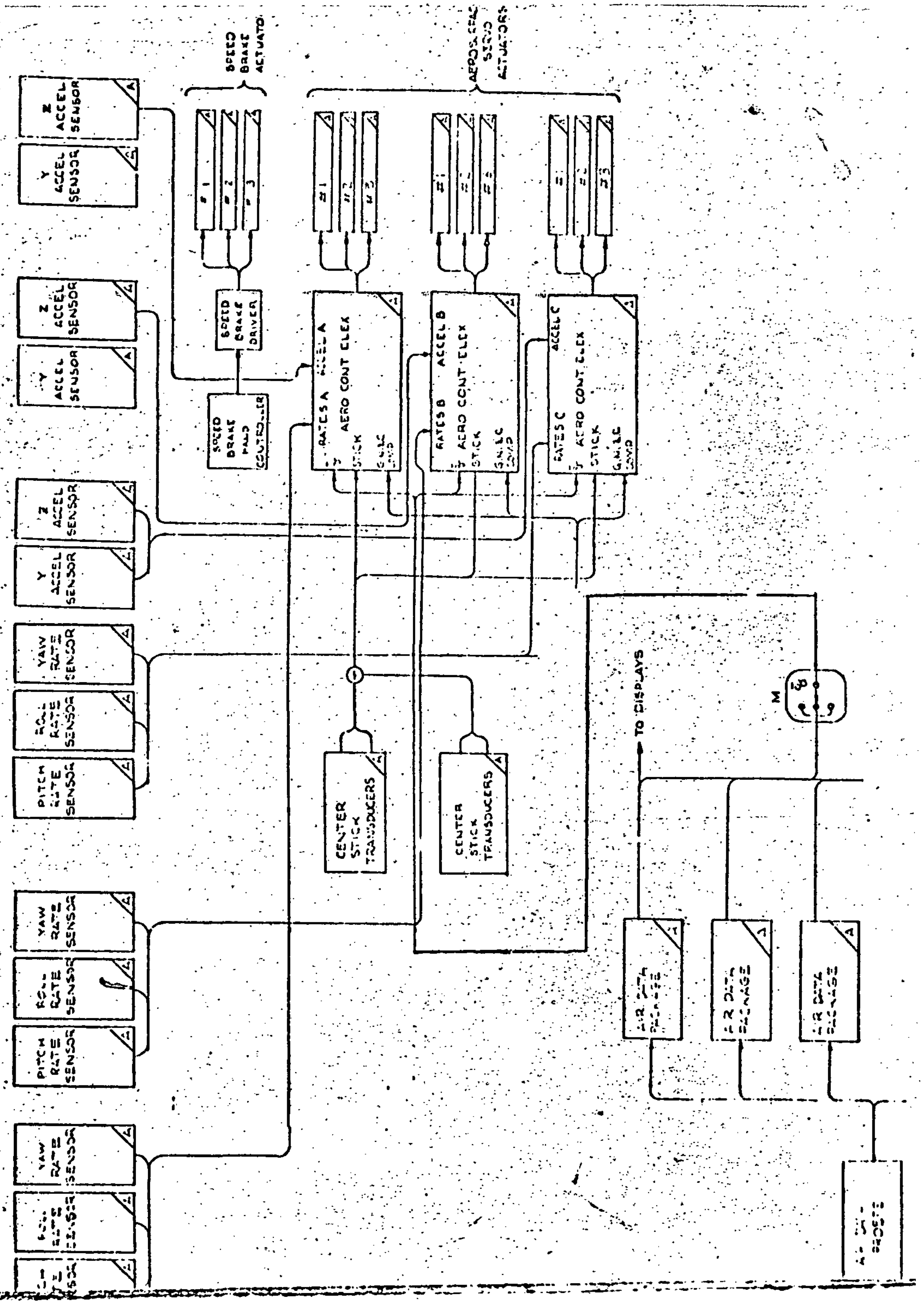


Figure 4.18.2-4 Analog SAS

amplifier circuits, middle select level detectors for redundancy management, and bias supplies, and represents a single layer of redundant drivers for the 8 engine gimbal actuators. Gimbal command inputs are accepted from either the GN&C computer in automatic control mode, or upon manual switchover to manual TVC, from the MTVC electronics in a backup mode.

The MTVC electronics unit accepts 3-axis commands from the two rotation hand controllers and body rate signals. Stick and rate inputs are amplified, and summed in an integrating amplifier; the resultant command signal outputs are fed to the appropriate gimbal servo drivers in each of the three TVC driver LRU's. All of the TVC electronics hardware is new design, with circuits based on Apollo BG286 MTVC and BG288 servo amp background experience. Packaging will utilize plug-in circuit boards, wire wrap interconnecting matrix, and non-hermetic aircraft-grade enclosure design of fabricated aluminum sheet, with thermal design based on convection cooling.

4.18.2.8 APS Logic/Driver Unit

Control of the ACPS thruster valve solenoids and of OMS engine ignition is handled by two APS Logic/Driver Units. Based on the existing Apollo Reaction Jet/Engine on-off control, this device accepts ACPS thrust commands from either the GN&C computer or from the Translation Hand Controller; it contains the logic necessary to select the appropriate thruster in response to the commanded rotation or translation maneuver, and provides the switching circuits necessary for controlling current to the thruster valve solenoids. This device also includes the necessary transient suppression and bias supply circuits, and circuits for OMS engine ignition and cutoff control.

The unit will utilize the BG287 chassis as well as the basic existing welded matrix and cordwood module configuration. Growth space is adequate to accommodate revisions necessary to tailor the existing jet select logic to shuttle ACPS system requirements, add one 2-jet driver module to the existing 16-jet control capability, and modify the SPS engine ignition timing and logic to meet shuttle OMS needs.

4.18.2.9 GN&C Computer Interface

The GN&C computational facilities consist of three, single thread, dedicated data acquisition strings and computers. At the present time, no provisions are being made to permit the transfer of data between computers. Figure 4.18.2-5 presents a simplified block diagram of the selected concept.

During the quiescent on-orbit periods, where the consequences of a hard failure are minimal, only one GN&C string is used. In Table 4.18 are the key features of the redundancy configuration. The failure recovery scheme is shown in FIGURE 4.18.1-1 and TABLE 4.18.1-3.

Table 4.18.2-2

Redundancy Management Key Features

- Redundant strings interconnected only at IMU output and at the output force servos and jet drivers to maintain design simplicity. IMU interconnection prevents divergence in guidance computations and allows detection of slow degradation failures. Servo and jet driver interconnection (via monitor) required for "fail-soft."
- RCS engines divided into 3 groups electronically, each with dedicated GN&C computer and capable of doing all required maneuvers.
- Hydraulic pressure monitors in the main engine TVC and aerosurface actuators automatically disengage a failed channel for first failure and switch to backup for second. OMS TVC is not time-critical and does not require a monitor.
- When the backup system is engaged, there are no significant transients requiring initial corrective pilot inputs. Backup is rate command system that damps out failure transients automatically with rotational controller centered.
- Steering display command monitor detects guidance loop discrepancies for display; no automatic action is initiated by this monitor.
- Built-in test and self-test furnish LRU status to crew but are not used to automatically manage redundancy.
- After a second failure, if the failed string can be isolated using BITE, control may be restored to the remaining good string at crew option.

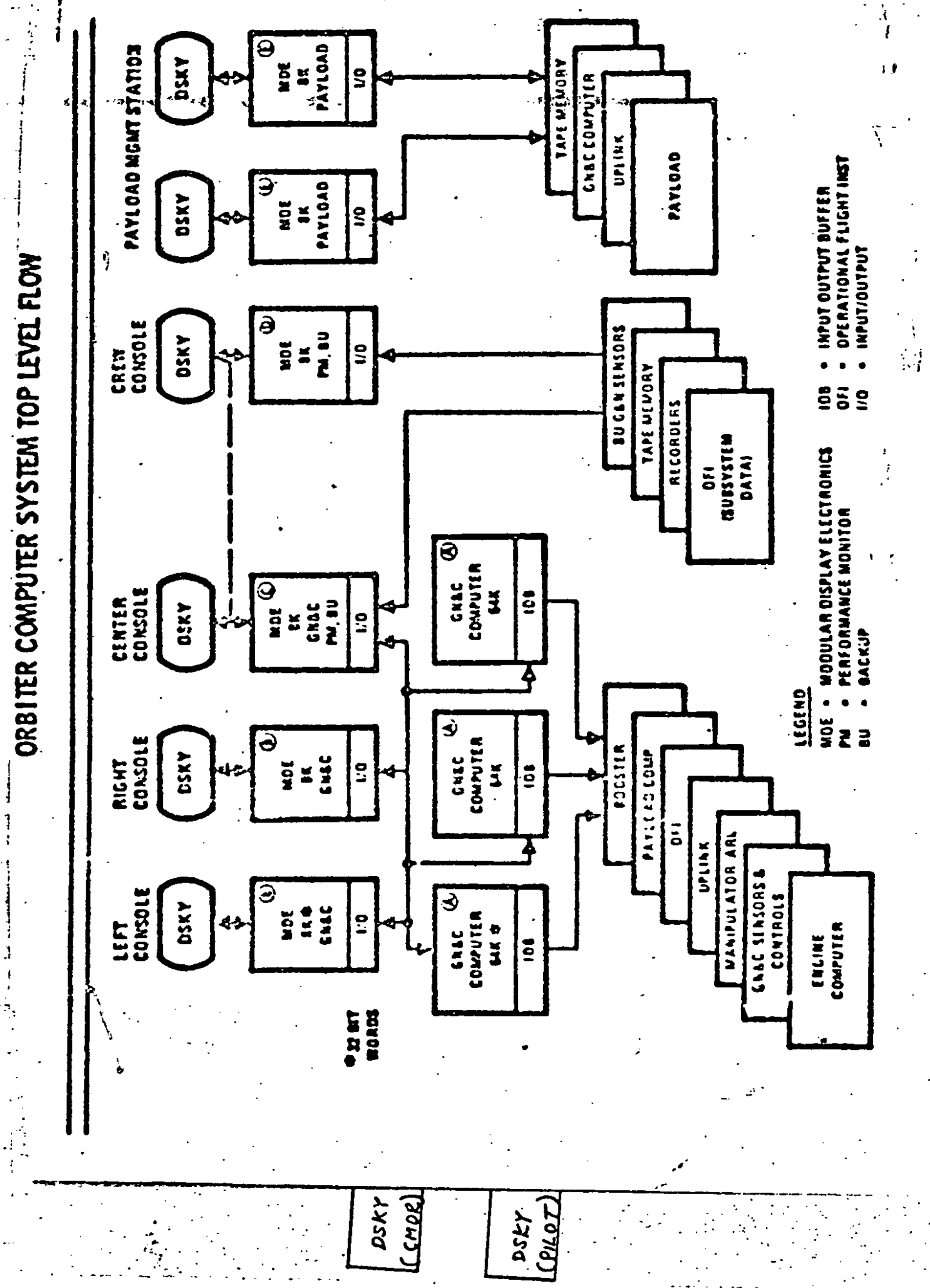


Figure 4.18.2-5

4.18.2.9.1 GN&C I/O Buffer

The input/output I/O interface between the computers and the other vehicle subsystems is implemented using a modular design concept to provide the flexibility to accommodate changing requirements and permit early computer design. The modular display electronics I/O, computing, and display subassemblies are combined into a single unit, which permits a simple design and is feasible because the identified I/O channel requirements are primarily multiplexed, serial digital. The I/O to the GN&C computers is significantly more complex, and subject to change; therefore, these functions have been grouped into a separate unit identified as the GN&C I/O buffer. The I/O buffer provides the functions of signal conversion, multiplexing, and transfer of data to and from the computer memory. The transfer functions are accomplished by using a direct memory access channel which operates independently of the CPU, except for initialization. Modifications to the I/O buffer capabilities are accomplished by exchange of available standard modules and/or where necessary exchanging custom I/O modules. The feasibility of this approach has been verified by detailed I/O interface studies summarized in Table 4.18.2-3.

Table 4.18.2-3

Input/Output Buffer Subsystem Interface Summary

Interfacing Subsystem	Signal Type	Type of Interface	Interface Device	Number	Rate	Typical Signal Names	Use
GN&C	Discrete	Hardwire	Standard	291	25 s/s	Status and control	GN&C performs monitor and equipment control
	AC analog	Hardwire	Custom	9	1 and 25 s/s	Gimbal angles	Guidance and flight control computations
	Pulse	Hardwire	Custom	23	1 and 25 s/s	Accelerometers, rate, gyros	Guidance and flight control computations
	DC analog	Hardwire	Standard	47	1 and 25 s/s	Star angles, vertical errors, air data, commands	Navigation and guidance computations, flight control
	Serial	Hardwire	Semi custom	5	25 s/s	Air data	Flight control computations
		MUX	Semi custom	1	25 s/s	Main engine	Main engine control and monitor
Booster	Discrete	Hardwire	Standard	3	25 s/s	Status and control	Booster status and discrete control
Communication	Discrete	Hardwire	Standard	45	1 and 25 s/s	Status, control, antenna switching commands	Status, equipment control and switching
	Serial	MUX	Semi custom	2	2 Kb/s	Up data	Ground data load
	Parallel	Hardwire	Custom	6	1 s/s	Range, bearing	Aero-navigation guidance; orbital rendezvous
	DC analog	Hardwire	Standard	43	25 s/s	Altitude, glide slope	Autoland
D&C	Serial	MUX	Semi custom	10	4.8 Kb/s	Attitude/translation commands	Flight control
OFI	Discrete	Hardwire	Standard	4	1 s/s	Status and control	Status, equipment control
	Serial	MUX	Semi custom	1	2 Kb/s	Computed data	Telemetry
	Serial	Hardwire	Semi custom	1	1 s/s	Time	Mission/event sequencing, navigation update
Computer/computer	Serial	MUX	Semi custom	12	256 Kb/s	Intercomputer data transfer	Indirect transfer of data between computers for reinitialization, support information, data exchange, and GN&C display data (CRT and annunciator)

s/s—sample per second Kb/s—kilobits per second; MUX—multiplexed

The GN&C I/O Buffer consists of two standard remote multiplexers and input buffer, and an output decoder. Data collected via the multiplexer/buffer will be, with very limited exceptions, already conditioned to a specified DC level. The decoder will be custom designed to meet subsystem interface requirements.

After signal conditioning has been performed, data from the various subsystems is distributed to one of the two multiplexers.

The input data is multiplexed and routed to the buffer units of the computers. Each multiplexer accepts: (1) 96 discretes, (2) 1 serial data channel, (3) 80 high level analogs, and (4) 50 differential analog signals. The multiplexer's sample rate is under internal control; its output is 8-bit plus parity serial data.

The buffer units accept input data from the multiplexer at the bit rate programmed by the multiplexer programmable read only memory. The buffer stores the input data in an internal memory. Each multiplexer input has a separate starting address. Two multiplexer channels are stored simultaneously (16 bits).

The buffers used in conjunction with the G&N computers receive data from the various subsystems via the multiplexers described above. Using input data and the stored program, the computer system performs the required G&N functions. Computer outputs are obtained either under direct software control or under decoder control via direct memory address through the parallel channel. Nominal decoder output capacity is as follows: (1) 15 serial signals, (2) 100 analog signals, and (3) 200 discretes.

4.18.2.9.2 Remote Multiplexing Unit (RMU)

The selected multiplexer is typified by an off-the-shelf Teledyne Remote Multiplexer Unit (RMU) Model DS-704 with minor modifications. The RMU is the device which interfaces the user subsystems to the GN&C computers, via the appropriate Buffer Unit. It consists of a time division multiplexer, A/D converter, programmable read only memory, and the necessary control/timing logic and modulator circuitry required to output multiplexed subsystem data into a PCM format for transmission to the Buffer Units. The modification required is the addition of a serial digital input channel(s).

The following is a description of the functional capabilities of the RMU:

Inputs:

. High level (00 to +5 VDC) analog: 80 single ended channels

. Low/High Level Analog: 50 differential channels

. Discretes: 96 points

. Serial Digital: Total input capability is limited to 142 equivalent 8 bit words, i.e., Analog Input = 1-8 bit word, 8 discrete inputs = 1-8 bit word. The number of serial digital 8 bit words inputted to the RMU will subtract from the existing analog/discrete capability, i.e., - 1 serial digital word block consisting of 142-8 bit words would delete any analog/discrete input capability for that RMU.

. Format Select: Two lines specifying one of four formats to be read from the programmable read only memory.

. Power: +28 ±10% VDC @ 400 ma.

Outputs:

. Data Output: 9 bit (8 bit + parity) words being transmitted at approximately 30 Kbps with Bi-phase - L Modulation.

Functional Operation The RMU will operate according to the mode stored in the programmable read only memory and selected by the Format Select lines. Figure 4.18.2-6 presents a simplified functional block diagram of the RMU. One of four modes can be externally selected which specify: output bit rate, input channel selection, output modulation, output word length, and gain selection for the differential analog input channels. The RMU will then scan the appropriate input channels, perform the required A/D conversion for formatting, add the necessary parity, modulate the resulting serial data stream, and output this data stream for transmission to the appropriate Buffer Unit.

Functional Configuration Two functional RMU's are provided within one case. These RMU's may be used redundantly by external crossstrapping or to expand the total number of input channels.

Physical Characteristics (for Dual RMU's)

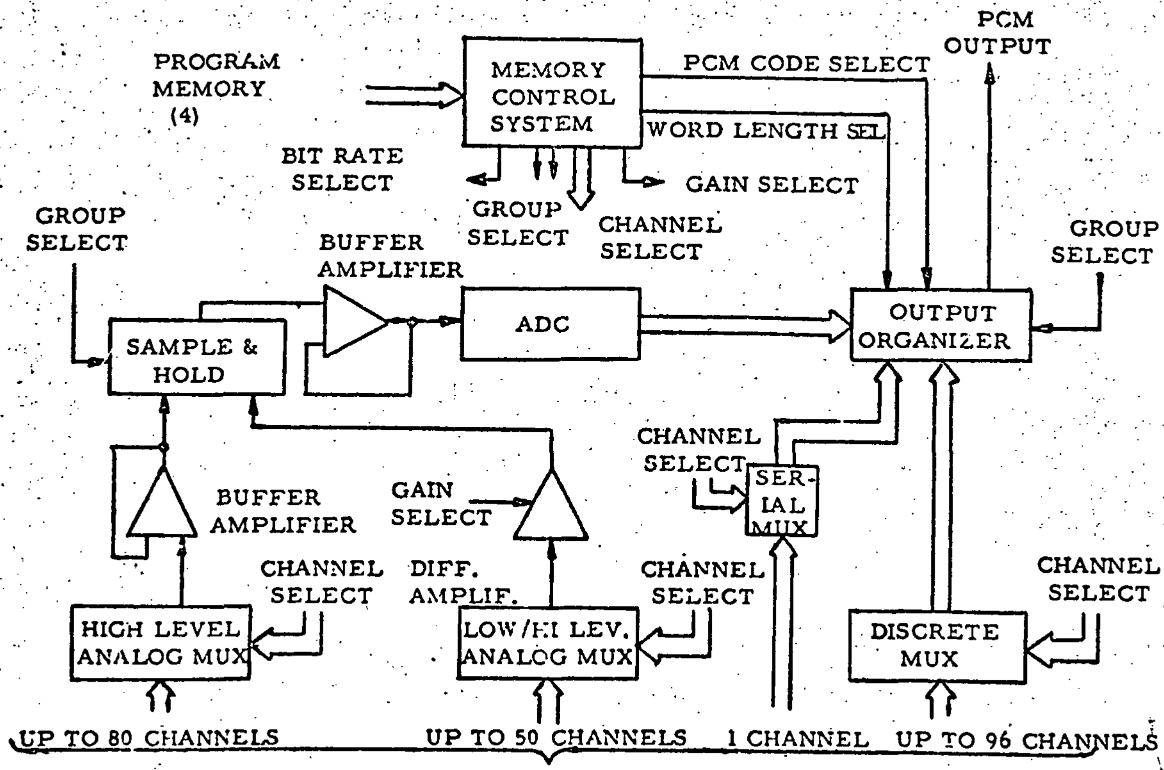
Power:	20 Watts (both units ON)
Weight:	10 lbs.
Dimensions:	8.5 H x 4.5 W x 3.5 L (Preliminary est.)
Volume:	0.08 cu. ft.

4.18.2.9.3 Input Buffer

The input buffer contains all of the circuitry and storage necessary to accept, decommutate, and store the inputs from four multiplexers which are outputting serial data streams. In the baseline system only two will be used. The multiplexers are free running and

Figure 2.18.2-6

- SIMPLIFIED BLOCK DIAGRAM, REMOTE MULTIPLEXER (TELEDYNE DS-704)



operating asynchronously. As each pair of words of data are received, from a multiplexer channel, the data will be transferred to a predetermined location of core storage in the buffer unit, or the computer.

In addition to the four PCM input channels, provisions will be made to custom interface, on an exception basis, those specific inputs where significant cost savings may accrue. Typical candidates for consideration are data inputs from TACAN and Up-Data Link equipments.

The input buffer for the GN&C computers will have a buffer-computer interface compatible with the computer parallel Direct Memory Access (DMA) channel. The computers will initiate data fetches from the buffer under control of software. By use of an I/O call, a starting buffer address and word count will be transferred and then sequential fetches of up to 24 locations will begin. Transfers from the buffer core to main memory will take place via DMA.

A simplified block diagram of the input buffer is shown in Figure 4.18.2-7.

Physical Characteristics

Weight	24 lbs.
Dimensions:	4.1 H x 10.1 W x 15.5D
Volume:	0.25
Power:	100 watts

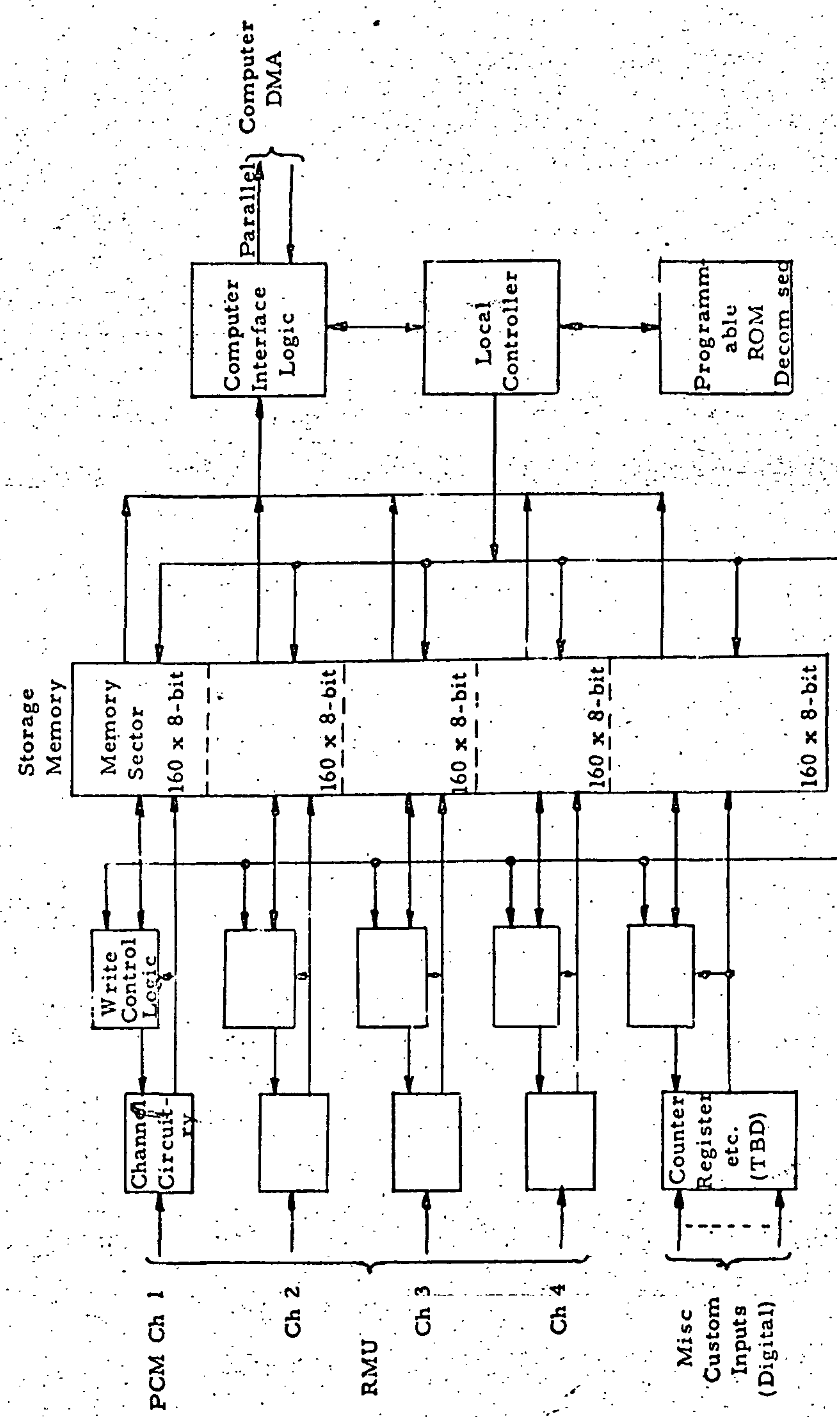


Figure 4.18.2-7 Block Diagram Input Buffer

4.18.2.9.4 Output Decoder

The output decoder provides a means of distributing and/or formatting the G&N computer output. The output decoder has the following capabilities.

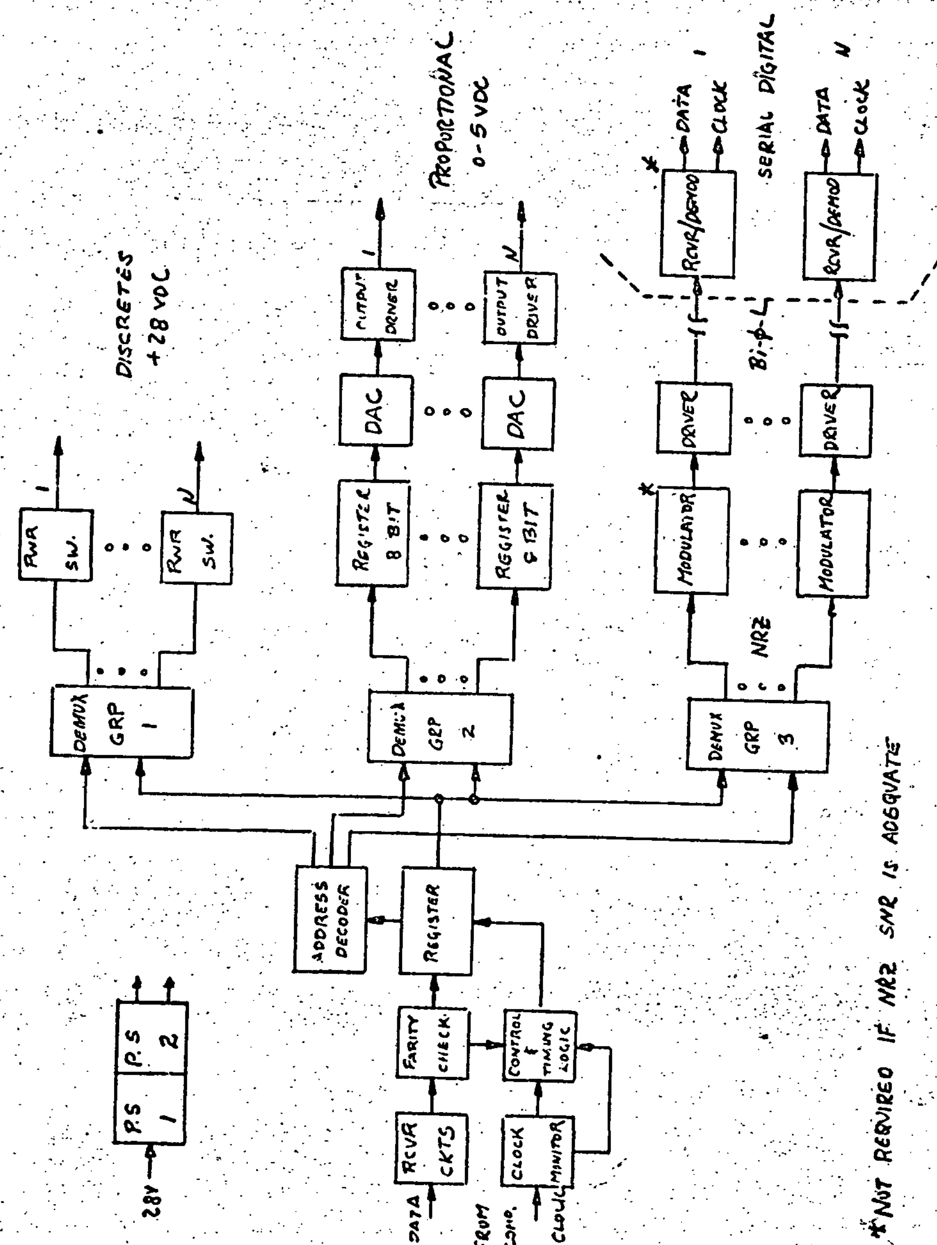
- 16 independent serial channels - each serial channel is loaded via a direct I/O instruction
- 128 discretes (0 - 5 ^{v}) - four fixed main memory locations are fetched via the DMA at selected intervals and their contents are used to update the discrete outputs.
- 64 DC analogs (0 - 5^{v} @ 8 bit accuracy) - 16 fixed main memory locations (4 outputs per location) are fetched via the DMA at selected intervals and their contents are used to update the digital to analog converters.
- Gyro Pulse Torque Commands to gyro - a three-channel gyro pulse torquer converts command signals from the computer to precise binary torque pulses which may be used to drive the gyro packages. Each pulse delivered to the gyro represents an increment of angle at the gyro input axis.
- 48 bit register to telemetry - one and one half fixed memory locations are fetched via the DMA at software controlled intervals and their contents are used to reload the 48 bit register. The register is made available for periodic sampling by the telemetry data acquistion system.

The Output Decoder interfaces with the parallel DMA channel of the computer. The DMA facilities of the computer are shared with the Input Buffer.

Figure 4.18.2-8 depicts a simplified block diagram of typical functional paths which may be provided by the Output Decoder.

Physical Characteristics

Weight:	27 lbs.
Dimensions:	4.1 H x 10.1 W x 15.5 D
Volume:	0.25
Power:	80 watts



*NOT REQUIRED IF NRZ SNR IS ADEQUATE

Figure 4.18.2-8 DECODER UNIT (COMMAND GROUPS)

4.18.3 GN&C Computer System

The GN&C system consists of three 64K 32-bit word computers dedicated to the solution and status of the Orbiter guidance, navigation, and control functions. The computers contain identical programs to allow a triple redundant computer system mode of operation. The computer clocks will be phase locked to the master timing unit to provide an inherent computer synchronization capability.

The GN&C program also provides navigation and guidance data to the Booster control computer and provides the control program for the manipulator arms.

Although the final choice for the GN&C computer has not yet been made, there are at present two candidates for the GN&C computer for the Orbiter. One is the Model AP-1 (designated AP-101 for the Space Shuttle) manufactured by IBM Corporation. The other candidate is the SKC-2000 manufactured by the Kearfott Division of The Singer Company.

The Model AP-1 Computer

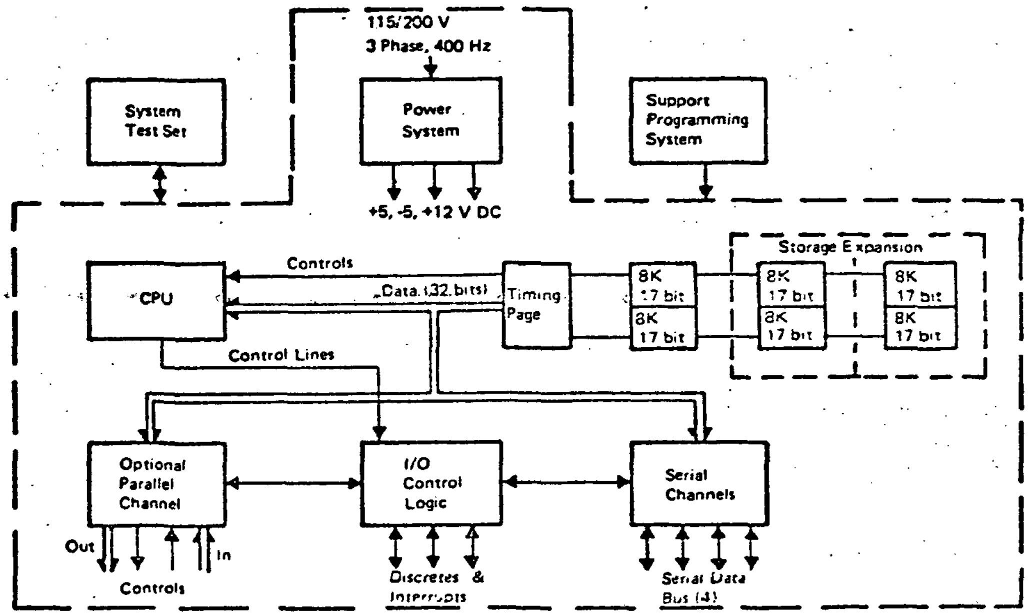
The IBM System/4Pi, Model AP-1 computer is a medium-sized, high-speed, general-purpose computer for real-time space and military control use. A standardized parallel channel and AGE interfaces permit the AP-1 to be integrated into an optimum system design with other Advanced System/4Pi computers. A summary of AP-1 characteristics follows.

The AP-1 has a full parallel 32-bit word machine organization. Data transfers to and from main storage and the I/O are handled on a fullword basis as well. The architecture provides addressing of each 16-bit storage halfword, thus allowing both 16-bit and 32-bit operands to be processed. This, together with the short and long format instructions (16 and 32 bits), significantly increases the overall storage efficiency of the AP-1 computer. It further expands the applicability of the AP-1 to a wide range of problems, since most uses require only 16-bit precision in the operands. Navigation and similar high precision computations are effectively addressed using 32-bit operands. The short format instruction, 16 bits, provides a similar advantage, because more than 70% of the instructions in a typical application program may be of the short format.

AP-1 Functional Characteristics

Machine Organization	
Type	General purpose, stored program, parallel
Organization	Binary, fixed point, fractional
Data Flow	32-bit parallel
Data Word Length	32 bits
Instruction Word Length	16 and 32 bits
Average Instruction Execution Rate	450,000 operations/second to 550,000
General Registers	Two selectable sets of eight 32-bit hardware registers
Main Storage	
Type	Random access, nonvolatile, destructive readout core
Capacity	16,384; 32,768; or 49,152 18-bit halfwords (including 1 parity bit and 1 store-protect (optional) bit)
Modularity	8,192 18-bit halfwords per pluggable page
I/O	
Serial Channels	Four channels; transformer coupled, two-way party line, 15 devices per channel; 1-MHz bit rate; 45,000 16-bit halfwords per second per channel
Parallel Channel (option)	One channel; multiple devices; 16 bits transfer; 150,000 to 750,000 halfwords/second
Interrupts	16 internal and external interrupts; 9 levels of priority; maskable
Discretes	12 DC-coupled inputs; 9 DC-coupled and 3 relay-controlled outputs
Logic Circuits	
Class	Monolithic integrated
Type	TTL and MSI
Package	Flatpacks; 14, 16, and 24 leads
Power System	
Primary Power	115/200 V AC, 3 phase, 400 Hz OR 28 VDC
Power	250 W (32 K halfword main storage)
Features	Overvoltage and overcurrent protection; transient protection; power sequencing
Physical	
Volume	0.87 ft^3
Weight	40 lb (32 K halfword main storage)
Environment	Meets or exceeds MIL-E-5400, Class 2X

A block diagram of the AP-1 computer is shown below.



4.18.3-1 *AP-1 Functional Block Diagram*

CPU Characteristics

Type	General-purpose, stored program, simplex – 4 SETS OF GENERAL REGISTERS		
Number system	Binary, fixed-point, two's complement, fractional		
Operation	Full parallel		
Data word length	32 bit fullword, including sign		
Instruction word length	16 or 32 bits		
Instruction set	42 short (16 bits) and 41 long (32 bits) AP-101 ≈ 117 INSTR.		
Typical execution times		Storage to Register (μs)	Register to Register (μs)
	Add	2.0 1.6	1.2
	Multiply	6.6 5.4	6.0
	Divide	10.6 8.6 AP-101	10.0
Average instruction execution rate	Over 450,000 instructions per second using typical instruction mix (68% add or equivalent, 18% branch, 5% multiply, 1% divide, 8% miscellaneous) (AP-101 550,000)		
General registers	Two selectable sets of eight 32-bit hardware registers; in each set, all are usable as accumulators, seven are usable as index registers, and four are usable as base registers		

Instructions

The five basic instruction formats shown below allow most operations to be coded in either a halfword or fullword instruction. The halfword format instructions not only provide increased throughput, by means of the instruction-look-ahead feature of the AP-1, but significantly improve the storage efficiency of the computer. Improved storage efficiency means decreased storage capacity requirements for operational programs, since typical applications have consistently demonstrated that the short format (16-bit) instructions account for over 70% of the instructions used. The AP-1 Assembler program automatically optimizes the object program by selecting the short format instruction whenever feasible.

OP	R1	1 1 1 0 0	R2		RR (Register to Register)

OP	R1	DISP	B2		SRS (Short Register to Storage)

OP	R1	1 1 1 1	b b	B2	ADDRESS SPECIFICATIONS	LRX (Long Indexable Register to Storage)

OP	OPX	DISP	B2		SSS (Short Storage to Storage)

OP	OPX	DISP	B2	MASK	LSS (Long Storage to Storage)

OP and OPX	Operation code and operation code extension
R	General register designation
DISP	Displacement field
B	Base register designation
bb	Addressing mode

The AP-1 general-purpose computer features an efficient instruction set, resulting in low main storage overhead and high throughput. A comprehensive set of halfword and full-word instructions are provided for arithmetic, logical, bit manipulating, data formatting, and input/output operations. Extended mnemonics for the Branch-on-Condition instruction aid in simplifying programmer source coding.

AP-1 Instruction Set

Instruction Type	Basic Instruction	Typical Execution Time (SRS) (μs)	Format RR	SRS	LRX	SSS	LSS
Arithmetic	Add	2.00 (1.6)	X	X	X		
	Compare	2.00	X	X	X		
	Divide	10.75 (8.6)	X	X	X		
	Load	2.00	X	X	X		
	Modify Storage	3.12					X
	Multiply	6.75 (5.4)	X	X	X		
	Store	2.00		X	X		
	Subtract	2.00 AP-101	X	X	X		
	Tally Down	3.12 examples				X	
Logical	AND	2.00	X	X	X		
	Exclusive OR	2.00	X	X	X		
	OR	2.00	X	X	X		
	OR Bits	3.12					X
	OR Halfword	3.12				X	
	Test Bits	2.75					X
	Test Halfword	2.75				X	
	Zero Bits	3.12					X
	Zero Halfword	3.12				X	
Branch	Branch and Link	1.37	X	X	X		
	Branch on Condition	1.37	X	X	X		
	Branch on Count	1.62	X	X	X		
	Branch on Overflow & Carry	1.37	X	X	X		
Shift	Normalize and Count	1.32 + n/4	X				
	Shift Left Single & Double Logical	1.87 + n/4		X			
	Shift Right Single & Double Arithmetic & Logical	1.87 + n/4		X			
Input/Output	In	1.87			X		
	Load AGE	2.00		X	X		
	Out	2.12			X		
System Instructions	Load Multiple	10.37		X	X		
	Load PSW	2.25		X	X		
	Load System Mask	2.25	X	X	X		
	Store Multiple	10.12		X	X		

Organization

The organization of the AP-1 computer offers very versatile addressing. The addressing technique accommodates 65,536 halfword addresses in the main storage. The generation of Storage Operand Effective Address is a function of the instruction format; and offers short or extended displacement addressing, direct and indirect indexed address, and immediate operands.

The AP-1 computer contains two selectable sets of eight 32-bit general registers, which may be used as accumulators for fractional binary arithmetic; for logical and shifting operations; or as base registers, index registers, or temporary storage.

Interrupt handling normally requires a put-away routine and a storage area to save the general registers whenever an interrupt servicing routine is entered; then the registers must be restored at the end of the interrupt routine. Assigning one set of the general registers to interrupt (executive) routines and the other set to the problem programs gives an improvement in throughput and storage efficiency. Alternatively, in a multiprogramming environment, assignment of the general register sets among the problem programs may produce even greater throughput.

Main Storage

Main Storage Features

Type	Ferrite magnetic core, nonvolatile, random access, destructive readout
Capacity	Choice of 16,384; 32,768; or 49,152 18-bit halfwords, including 1 parity bit and 1 store protect bit, for the optional store protect feature, within the existing structure. A slightly larger structure will accommodate 65,536 halfwords
Modularity	8192 18-bit words
Cycle time	1 μs write or read/restore → 900 ns AP-101
Access time	450 ns
Electronics	Monolithic
Special features	Power transient protection, separate sense winding, 2-1/2D Organization, temperature compensation

A militarized, 1-μs core storage array organized in a 2-1/2D addressing scheme forms the basis for the AP-1 main storage, which is implemented primarily with monolithic circuits on two unique page types: an 8 K X 18-bit storage page and a timing page. The storage page contains 8/13 mil,* wide-temperature-range, ferrite cores with self-contained, temperature-compensated voltage regulators, address drivers, sense amplifiers, and a storage data register. A single timing page can handle up to eight storage pages, providing 8 K X 18-bit modularity up to a maximum of 64 K halfwords of storage. No adjustments are required to provide complete interchangeability of the pluggable modules.

Input/Output

High-speed internal operations, coupled with a versatile serial and/or parallel input/output, provide for excellent throughput. The four-channel multiplex serial I/O services up to 15 peripheral devices per channel at a data rate of 1 megabit per second. The data-handling capacity of each transformer-coupled, party-line, serial data bus may be up to 45,000 halfwords per second, depending on the amount of data blocking.

I/O Characteristics

Serial channel	Direct memory access; four multiplexed I/O channels; up to 15 devices per channel
Data Interface	Serial, 1 MHz, transformer-coupled (70-ft lines)
Maximum data transfer rate	45,000 halfwords per second per channel 180,000 halfwords per second for 4 channels
Parallel channel (Option)	Direct memory access, externally controlled, 750,000 halfwords per second, maximum Buffered I/O, externally controlled, 480,000 halfwords per second, maximum Direct I/O, program controlled, 240,000 halfwords per second, maximum
Data interface	16 bits plus address and control, single-ended TTL interface
Maximum data transfer rate	240,000 to 750,000 data words per second
Features	Channel-to-channel interface Externally specified channel-control-word chaining on buffered I/O
Discrete inputs (typical)	12, DC coupled
Discrete outputs (typical)	12: 3 relay controlled; 9 DC coupled
Program loadable/readable counters	Two 6-second counters with 100-μs resolution; one 3-second counter with 50-μs resolution
Interrupts (typical)	12 internal and 4 external; 9 levels of priority with program mask control

Transformer coupling and the sinusoidal Manchester phase-coded signals provide data transmission reliability not possible with direct-coupled DC lines. The AP-1 transmission lines can withstand a short or an open without affecting data transmission. This is particularly desirable in a party-line I/O system. An unterminated line design allows removing units without disrupting operation of the line.

Each serial channel is initialized separately by a program instruction. Facilities exist to permit block transfer of data words and to provide for chaining blocks of data. Data and control words are transferred to and from the CPU general registers or main storage. Peripherals respond only when a select code commands a response.

An optional multiplex interface module (MIM) can provide a compatible interface between an I/O device and the serial channel. One version of the MIM (shown in the serial I/O application diagram) features a back-up channel for improved I/O reliability, parity checking and generation, peripheral unit selection logic, parallel-to-serial data formatting, and automatic gain control to assure uniform data amplitude during transmission.

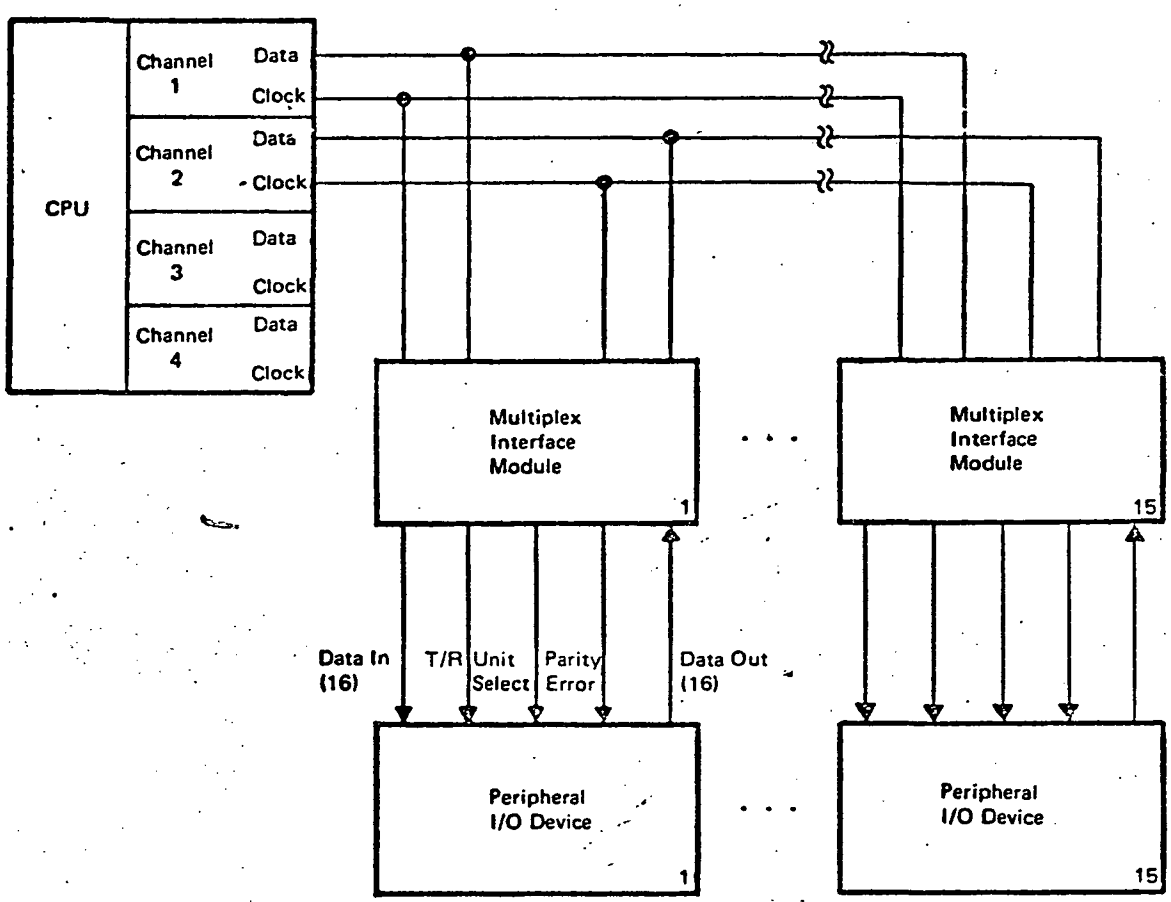


Figure 4.18.3-2

Typical serial channel application shown for channel 1 primary and channel 2 backup. Four channels are available.

Designed for inclusion in each peripheral I/O device, the MIM connects the serial bus interface to a parallel TTL interface, and includes the control lines necessary to service most I/O devices.

The I/O capacity of the AP-1 may be increased by adding an optional high-speed parallel channel. This parallel channel may be used with, or in place of the serial channel in applications requiring support of such high-speed I/O devices as the IBM System/4 Pi Drum Mass Memory.

Input and output data busses provide transfer of data unidirectionally to the peripheral devices. The parallel channel interface has been standardized to assure I/O compatibility with other computers of the Advanced System/4 Pi family, such as the SP-1.

The parallel channel can support four devices, each of which may be an I/O unit, such as a drum or magnetic tape unit, or a control unit. Each control unit may interface with as many as 127 I/O devices, giving the AP-1 excellent and versatile I/O support potential.

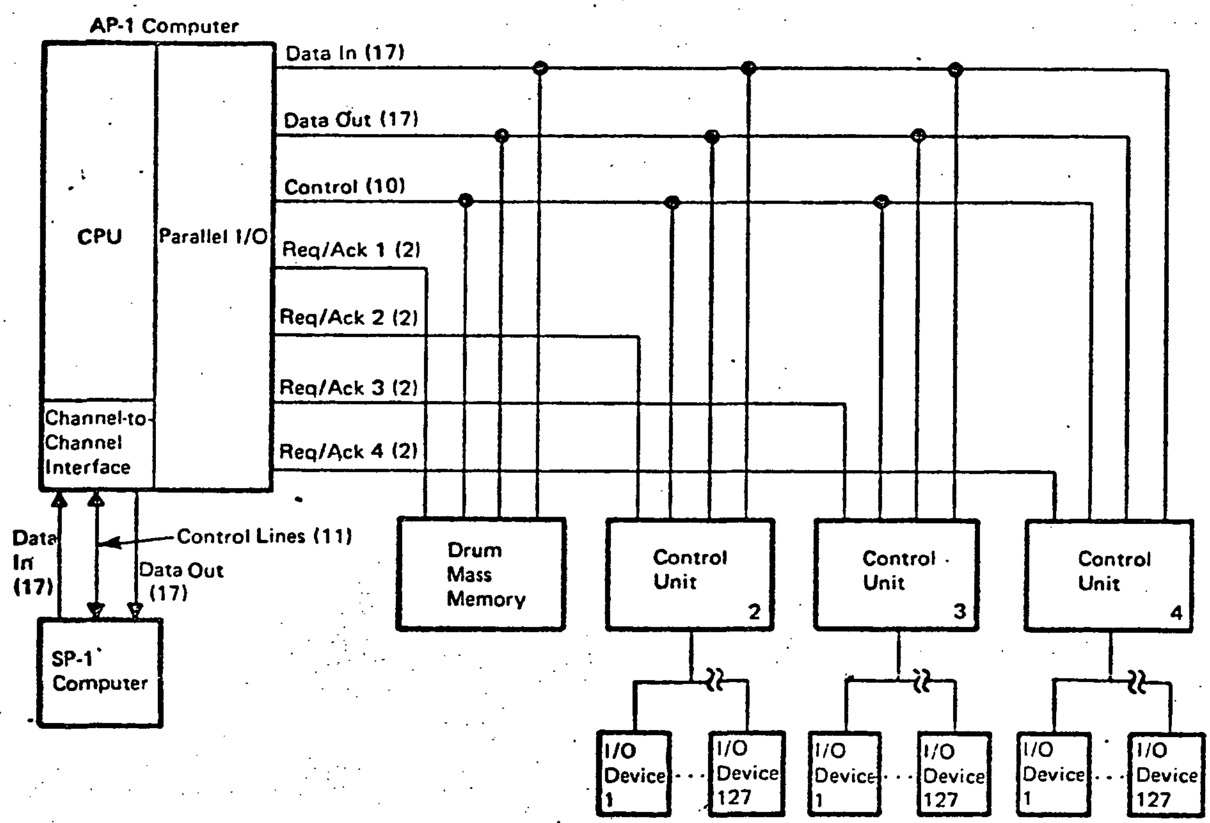


Figure 4.18.3-3

Typical parallel channel application, showing direct device attachment (drum) and control units with with maximum number of peripherals attached.

Three I/O modes are provided for transfer of data and command functions between the AP-1 computer and peripheral equipment:

1) Direct Memory Access (DMA) I/O – The I/O device supplies a 16-bit main storage address, and 16 bits of data are written into or read from the storage location. DMA I/O is device initiated. A CPU lockout feature significantly increases the data rate to a maximum of 750,000 halfwords per second.

2) Buffered I/O – The I/O device provides an "address tag", used to access a channel control word (CCW) containing count and address; a single word or a block of data words are written into or read from main storage. Buffered I/O is device initiated. A maximum data rate of 480,000 halfwords per second may be reached when the CPU is locked out.

3) Direct I/O – Under control of the CPU instruction stream, a command is sent to an I/O device, and one word (16-bits) of data is written into or read from the device. Direct I/O is program initiated. The data rate is a function of the operation following the transfer and may be as high as 240,000 halfwords per second.

In addition to high speed, the AP-1 parallel channel provides several features for expanded performance. Most significant is the channel-to-channel interface, which permits the AP-1 to transfer data to or from another member of the advanced System/4 Pi family, such as the SP-1 or another AP-1. The AP-1 channel, although faster than the SP-1 channel due to the higher main storage speed, is compatible with the SP-1, thereby permitting communication between the AP-1 and the SP-1.

Being able to externally chain two channel-control words while in the Buffered I/O mode increases the length of the data block that may be transferred. Provisions for an expanded control word for Direct I/O enhance the number of different I/O functions or transfers that may be performed.

A priority interrupt system is implemented with full masking capability for nine levels. Four external interrupts and 12 internal interrupts automatically store the current program status word in an assigned main storage location and begin an interrupt servicing routine. The internal interrupts include arithmetic conditions, time, machine errors, and I/O termination interrupts.

Three program-loadable/readable counters are provided in the I/O logic of the AP-1. One of these is used principally to implement I/O time out monitoring. The other two may be used in any countdown application. All three issue an interrupt when the countdown reaches zero.

The I/O serial and parallel channel interfaces are shown on the following page. Four serial channels are available as standard I/O on the AP-1. Each consists of a clock line and a two-way serial data bus. On the parallel channel each device has unique Service Request and Service Acknowledge lines. Four pairs of these lines are built into the parallel channel. In addition, the I/O implementation includes an interchannel interface that interacts with the data out and/or data in bus of another AP-1 or SP-1 computer.

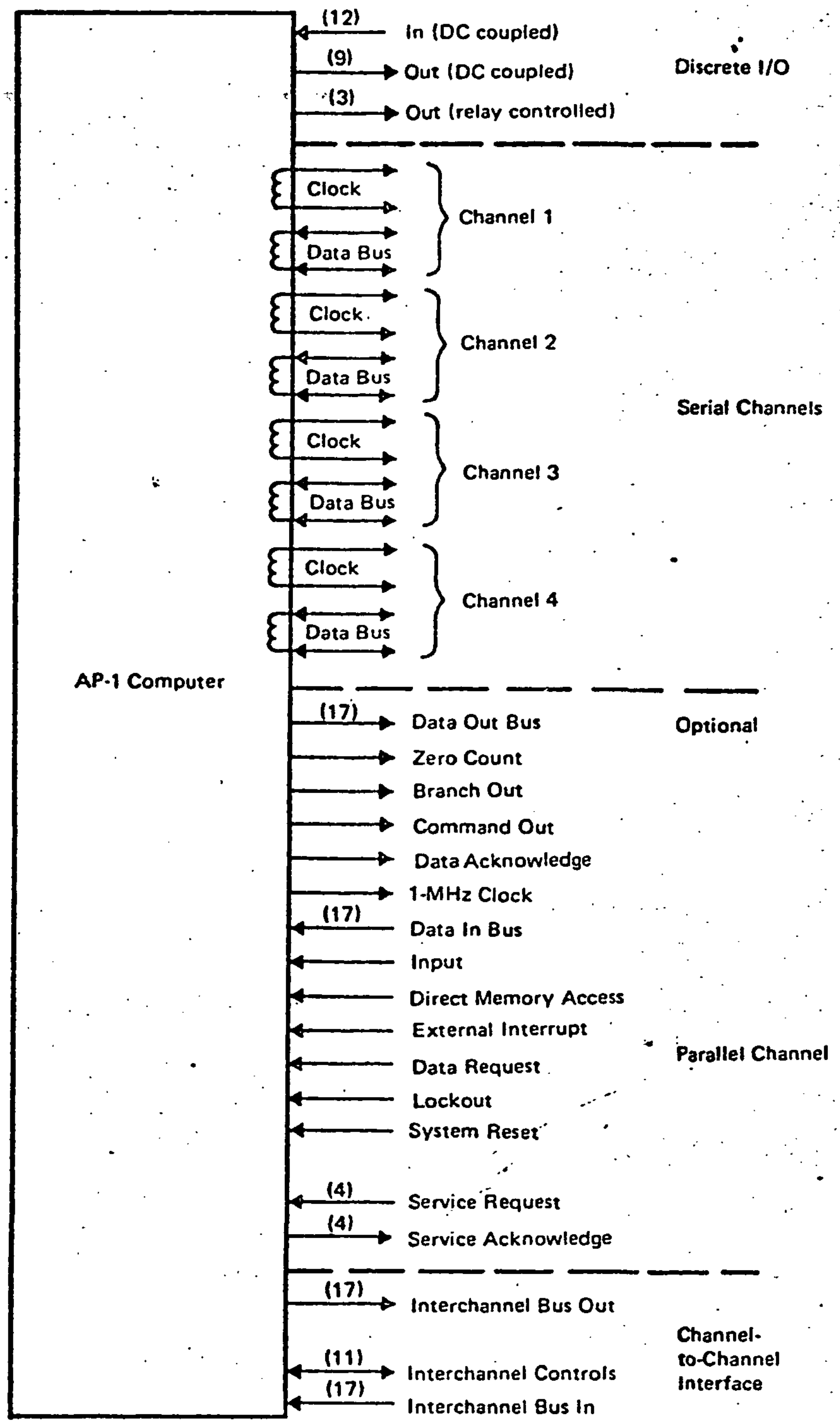


Figure 4.18.3-4 *I/O Interface*

Options

Nine major options are available for extending the capability of the AP-1 to meet specific application requirements. These options are summarized below:

1) Expanded Main Storage — The minimum configuration AP-1 has a 16-K halfword main storage. This may be expanded in 16-K increments to 48 K halfwords by inserting pairs of main storage pages (plus replacing the timing page with a new timing page for a 48-K storage). This modification is field installable.

2) 64-K Main Storage — An expansion to a 64-K halfword main storage may be done without any changes to the architecture and organization of the computer. A slightly larger structure to accommodate the additional main storage pages is the only other requirement in addition to that listed for a 48-K halfword main storage.

3) Main Storage Extension — Additional 64-K halfword modules of external main storage may be attached to the AP-1 computer. A factory modification, this feature permits a memory extension to 256-K halfwords directly addressable by the CPU through storage interface logic (SIL). The interface is implemented on an additional SIL page pluggable module working with the external main storage extension packages.

4) Parallel Channel — A high-speed parallel channel with support for four devices (four pairs of unique request and acknowledge lines) is available. Featuring a channel-to-channel interface and compatibility with other Advanced System/4 Pi computers, this channel provides both general register and direct memory access data transfers. The parallel channel option is field installable. Details for the parallel channel are described in the I/O section of this document.

5) Store Protect Feature — A store protect option, which prevents the accidental overwrite of protected halfwords, is available as a factory modification. The store protect bit is set on the selected halfwords during the memory load through the AGE interface. It is not program resettable, assuring maximum integrity of the protected data. If an attempt is made to write in a protected area, an interrupt will be issued and serviced by the interrupt routine. Each MCM-1 main storage page is populated with the necessary store protect bits.

6) Maintenance Library Program — An optional software feature, the MLP is used to test and debug operational programs on the AP-1 computer. Used in conjunction with an external data adapter and peripheral equipment such as a keyboard, CRT display, and tape drives, the MLP simplifies the task of operational program checkout and maintenance.

7) Floating Point — This option allows floating-point operations for extended range applications. A factory modification, the floating-point feature provides both long and short fraction operands. The floating-point operands consist of a 2's complement, signed integer exponent, and a 2's complement fraction. General registers 4 and 5 serve as the floating-point register first operand during floating-point operations.

AP-101 FORMATS
8/24 + 8/40

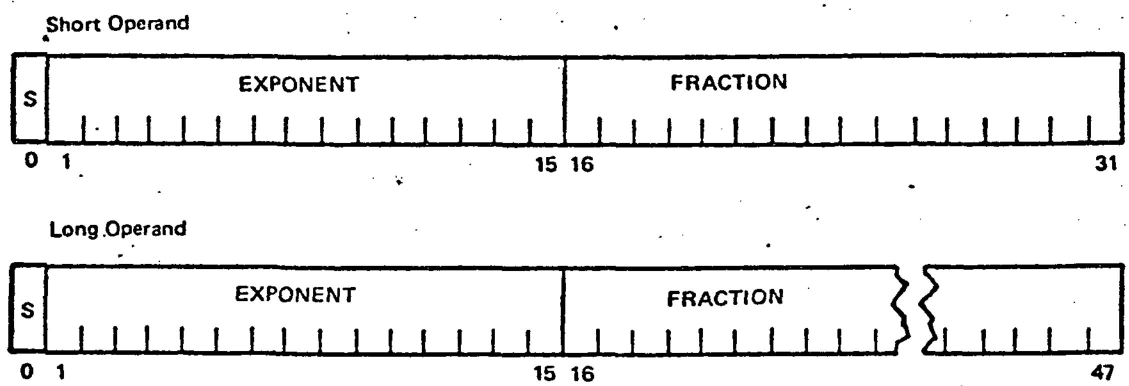
8 BIT EXPONENT
WILL BE HEXADECIMAL

8) Field Programming System (FPS) – The FPS is an AP-1 software option that provides for generating and testing operational software directly on the AP-1 computer using the computer test set, paper tape reader, paper tape punch, and a typewriter.

9) Compiler – A compiler may be provided to extend the AP-1 language translation facilities to include a higher order language (HOL) such as JOVIAL, FORTRAN IV, HAL, ETC. Translation of the HOL by the compiler to AP-1 macro assembler language simplifies implementation of this option.

Floating-Point Arithmetic (Optional)

Floating-point data occupy a fixed-length format, which may be either a fullword short format or a three-halfword long format.

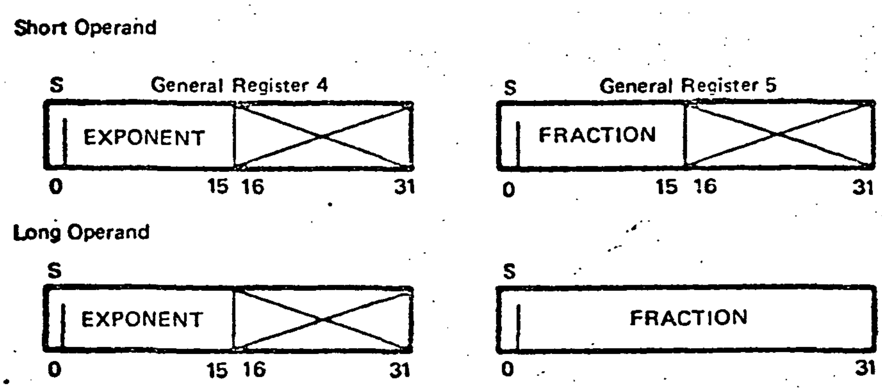


AP-101
8/24

8/40

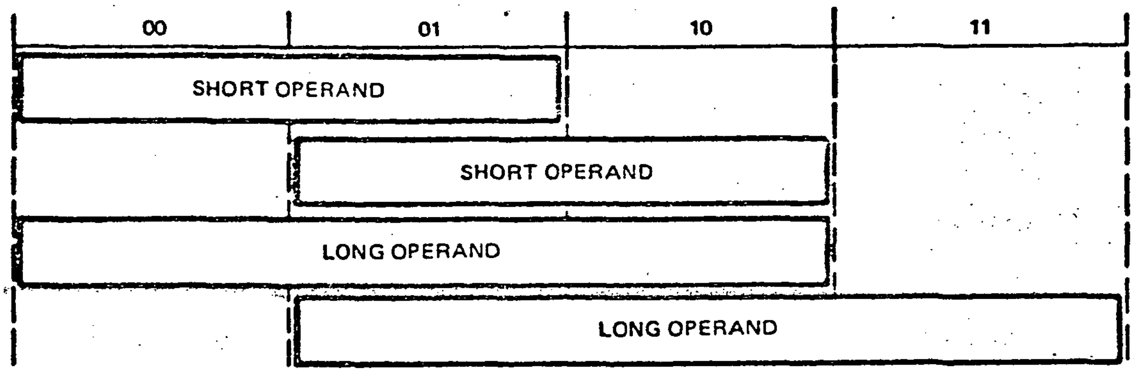
Note: The sign of the normalized fraction is unlike the first binary fraction digit.

The first operand is contained in a pair of general registers. General register 4 contains the exponent of the first operand in bits 0 through 15. Bits 16 through 31 are ignored. General register 5 contains the fraction of the first operand. The fraction of a short floating-point number is contained in bits 0 through 15 of general register 5. The fraction of a long floating-point number is contained in bits 0 through 31 of general register 5.



F-398-8-A

The floating-point second operand may start at any halfword address. The figure below illustrates floating-point data placement in main storage.



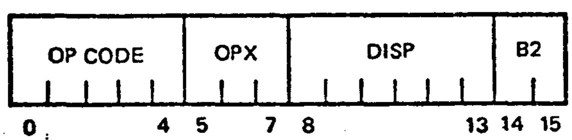
Floating-Point Data Placement in Main Storage

Floating-Point Exceptions

The operand incompatibility interrupt, exponent overflow interrupt, and the floating-point divide interrupt are the three floating-point exceptions. A bit in the program status word (PSW) may be used to mask and hold these pending until the PSW bit is reset to zero. The exception is then honored by a PSW swap. The interrupt code in the "old" PSW identifies the particular exception.

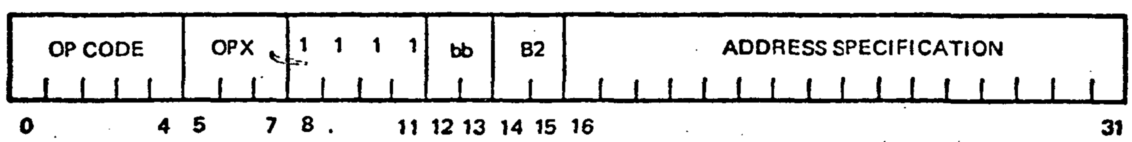
The instruction formats are:

Short Format



OP and OPX	Operation code and operation code extension
R	General register designation
DISP	Displacement field
B	Base register designation
bb	Addressing mode

Long Format



F-398-8-A

These formats are identical in form and addressing modules to the SRS and LRX instruction formats.

Floating-Point Features

		AP-101
Data	Exponent: 16 bits	8 BITS - HEX
	Fraction: Short 16 bits	24 BITS
	Long 32 bits	40 BITS
	Range: 10^{-9864} to 10^{+9864} (approx)	
Instruction Length	Short: 16 bits	
	Long: 32 bits	
Repertoire	Add; Add Long	
	Subtract; Subtract Long	
	Multiply; Multiply Long	
	Divide; Divide Long	
	Load; Load Long	
	Store; Store Long	
	Convert to Floating Point	
	Convert to Fixed Point	
		AP-101
Average Execution Times	Add 4.00 μs →	2.6 μSEC.
	Multiply 9.50 μs	
	Divide 13.50 μs	

Field Programming System (Optional)

The AP-1 FPS provides the programmer with the facilities to

1) Generate and edit source input tapes

2) Assemble the source program modules from the generated tapes

3) Link the program modules into an executable program

4) Control program checkout through entry and display of data during program execution

The AP-1 FPS consists of the following program components:

1) Tape Generation and Edit (TGE) program

2) Assembler

3) Linkage editor

4) Program verification package.

The TGE module gives the problem programmer a tool for placing the source program module on punched tape, modifying that tape module before assembly, and listing the contents of the source tape.

The FPS assembler provides programmers with a convenient means of writing operational programs to be run on the AP-1 computer. The assembler accepts as input a symbolic source program module written in the assembler language, and produces as output an object program module. The module is ready for subsequent processing by the FPS linkage editor. The assembler language is upward compatible with AP-1 macro assembler, which operates on the IBM System/360. Programs written in the FPS assembler language may be assembled on either the System/360 or the AP-1.

The FPS linkage editor, like the System/360 AP-1 linkage editor, accepts as input independently produced object modules generated by the assembler. The linkage editor links these modules together into a single AP-1 executable program.

The program verification package (PVP) helps the user check out his program by providing dynamic control over the program while it is being executed on the AP-1. The user may control program debug by entering commands to the PVP directly through the typewriter or indirectly through a user-written test information program.

Compiler (Optional)

A compiler can be provided to extend the AP-1 language translation facilities to include a higher order language. The complier translates programs written in HOL JOVIAL to the AP-1 macro-assembler language.

The selective use of a HOL for military computer applications has been demonstrated to be cost- and schedule-effective. In addition, HOL programming addresses the military need for software commonality and maintainability.

With the compiler, the AP-1 application programmer has at his disposal three languages to express his problem:

1) JOVIAL

2) Macro

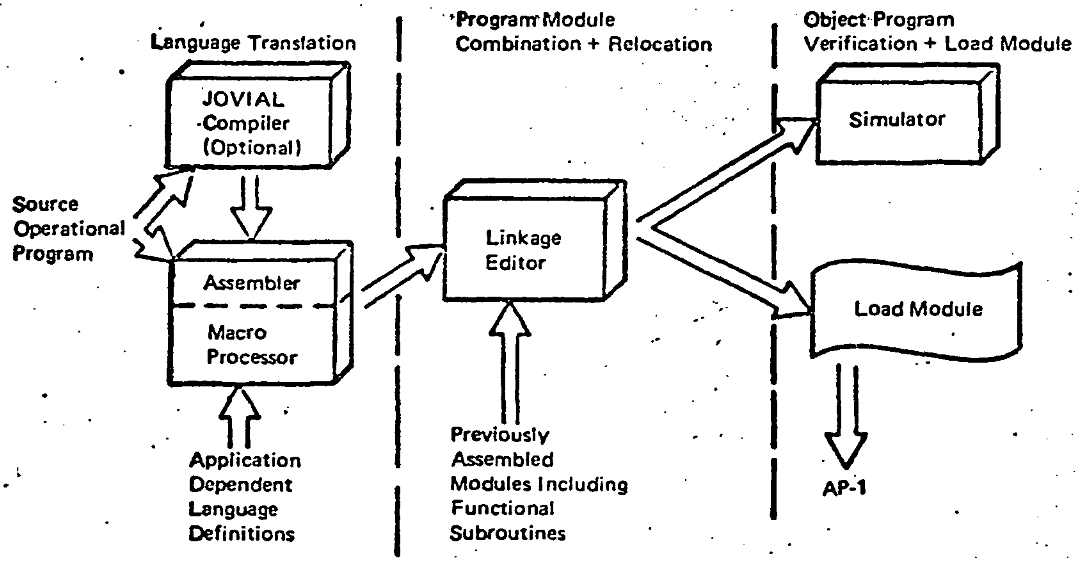
3) Assembler.

To augment this capability, the subroutine library may be called by any one of the above languages. The programmer may intermix the use of these languages in any combination to achieve an optimum program for the AP-1 application.

For example, the HOL is well suited for the major and primary mission tasks. The macro language is ideal for defining unique application functions, including input/output, and the assembler language best handles time-critical and bit-handling operations.

AP-1 Computer Support Software

The AP-1 Support Programming System (SPS) provides the application programmer with a flexible set of aids designed to effectively reduce the time required to develop application programs to support program modification and maintenance. Continuing a field-proven concept, the AP-1 SPS, operating on System/360 under OS control, gives the user an assembler, a linkage editor, a simulator, a subroutine library, and an optional compiler. The program generation process is shown in the following illustration. An optional Field Programming System (FPS) permits generating and testing programs directly on the AP-1 computer.



AP-1 Support Software Program Generation Process

Macro-Assembler

The AP-1 assembler provides the programmer with a symbolic source language that produces storage-efficient, high-throughput programs using the effective instruction set defined by the computer's architecture. An excellent macro capability combined with mathematical subroutines significantly reduces the repetitive coding required of the application programmer.

Identical in most respects to the widespread, well-known System/360 assembler language, the AP-1 assembler requires minimum retraining of experienced programmers and provides early on-line release of new trainees, thus significantly reducing training costs. High-quality user and programmer manuals assist the programmers in the effective use of the AP-1 assembler language. Programming rules and techniques, such as modular programming (C-sects, D-sects, etc.), syntax, macro generation, symbols and labels, source instructions, comments and combinations, and subroutine linkage, are in most cases identical to System/360 operations.

Linkage Editor

The AP-1 linkage editor accepts as input independently produced object modules generated by the AP-1 assembler. The linkage editor combines and relocates these modules, resolves module linkages, and generates an executable load module suitable for AP-1 execution or simulation. This load module is available on paper and magnetic tapes for loading the AP-1 memory. Corrections can be made to the object modules at linkage edit time. The AP-1 linkage editor also produces a symbolic listing of the relocated load module complete with operation mnemonics, labels, and absolute addresses.

Simulator

The AP-1 simulator dynamically analyzes the executable load module produced by the AP-1 linkage editor for programming errors and unusual conditions. The user has complete control of portions of code to be simulated and of stopping conditions. In addition, the user can choose several types of debug outputs, including full or partial dumps; instruction-by-instruction traces, complete with location symbols; and location snaps. He may, if he desires, devise his own format for simulation control inputs through entries that are available to attach simulated i/O devices.

The AP-1 simulator program consists of a series of subroutines. To form an executable simulation capability, these subroutines are combined with a user-written control program. This control program reads the control input and passes control information to the simulator through subroutine calls.

Subroutine calls are made through a screening interface, which detects syntax errors, if any, in the call; then passes information to the appropriate subroutine.

The actual simulation is done interpretatively, with an execute table of interpretative instructions being built and changed dynamically during simulation. Corrections to the load module to be simulated can be made at any time during the simulation.

Functional Subroutine Library

The AP-1 functional subroutines consist of the elementary trigonometric, vector, and conversion routines. The linkage editor selectively includes these in AP-1 application programs as needed.

AP-1 Support Software

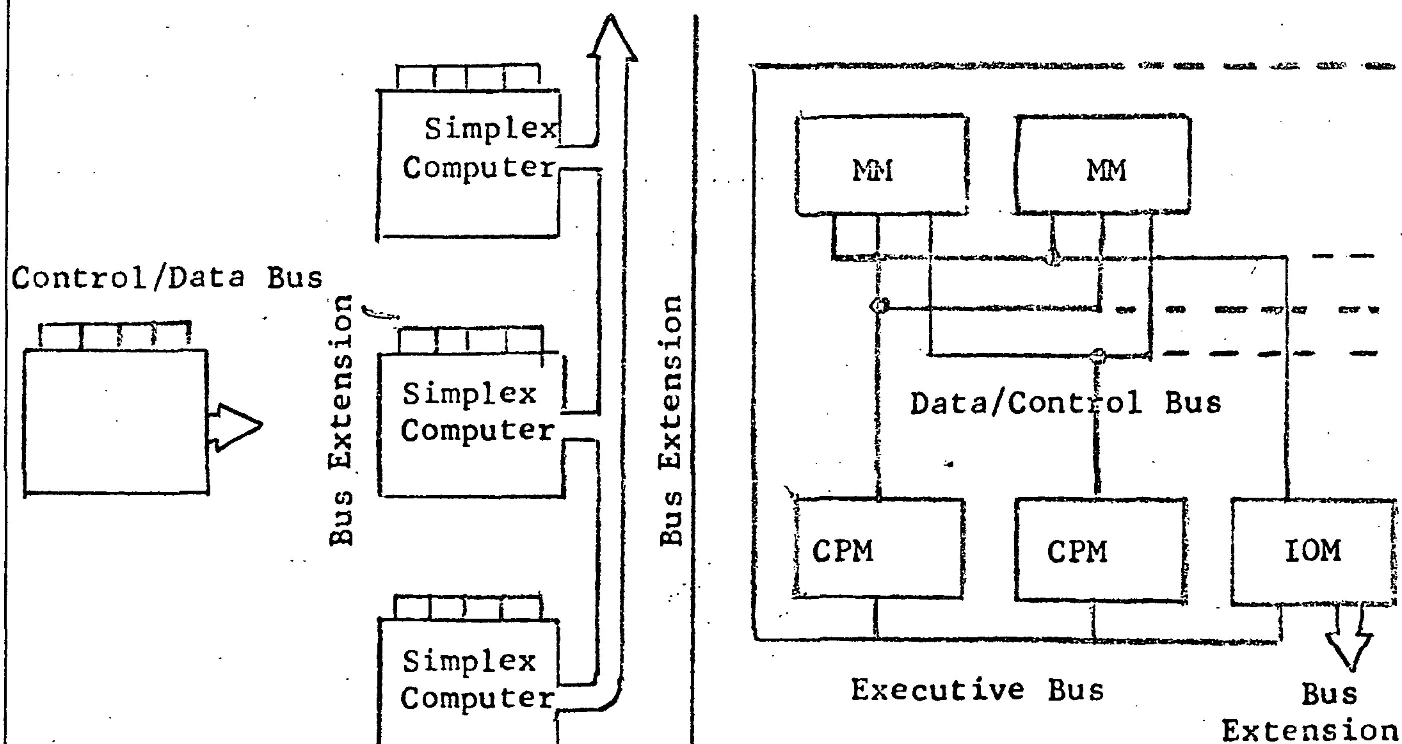
Assembler	Efficient symbolic language Modular programming and testing Assembles relocatable programs Selects instruction length for the AP-1 machine instructions Provides syntax error detection and identification Allows macro-processing and conditional assembly Produces listed output
Linkage Editor	Combines and relocates program modules assembled at separate times Resolves program linkages Creates input for the Simulator Creates core image object programs for loading the AP-1 Computer
Simulator	AP-1 program analysis Dynamic simulation facilitated through a user-written control program User access to simulated computer object program data (with absolute and symbolic reference) Object program correction Program debugging options (dump, snap, trace) Provision for simulating input/output and interrupt initiation and response
Subroutine Library	Square root Root sum square Sine/cosine Arc tangent Arc sine Arc cosine Log_e (X) Exponential e^x Exponentiation a^x Matrix operations Vector cross product Vector dot product Rectangular-to-polar coordinate conversion Polar-to-rectangular coordinate conversion BCD-to-binary conversion Binary-to-BCD conversion
Maintenance Library Program (MLP) (Optional)	Operational program debug (on the AP-1) Keyboard, printer, tape, CRT, I/O control Program load and verify Provides core dumps Memory modify

The SKC-2000 Computer

A general purpose, high performance digital computer, the SKC-2000 bases its modular design upon a single data and control bus interconnecting all modules with a standard interface to provide true modularity and maximum flexibility. Modular flexibility is further augmented by asynchronous module operation, a complete spectrum of input output capabilities, and contemporary mechanical design utilizing sandwich construction which is easily expandable.

SKC-2000 modules' strict compliance with asynchronous modularity concepts permits module mixing or matching, and growth by adding modules or through replacement with modules using newer technologies.

SKC-2000 modules can be used, for example, to form a simplex central computer, a multi-computer, or a multi-processor installation with equal ease. Modules for a simplex computer typically include a central processor module, one or more memory modules, a power supply module, and an input/output module. Because individual modules are relatively small, they are packaged most efficiently as part of the larger SKC-2000 computer.

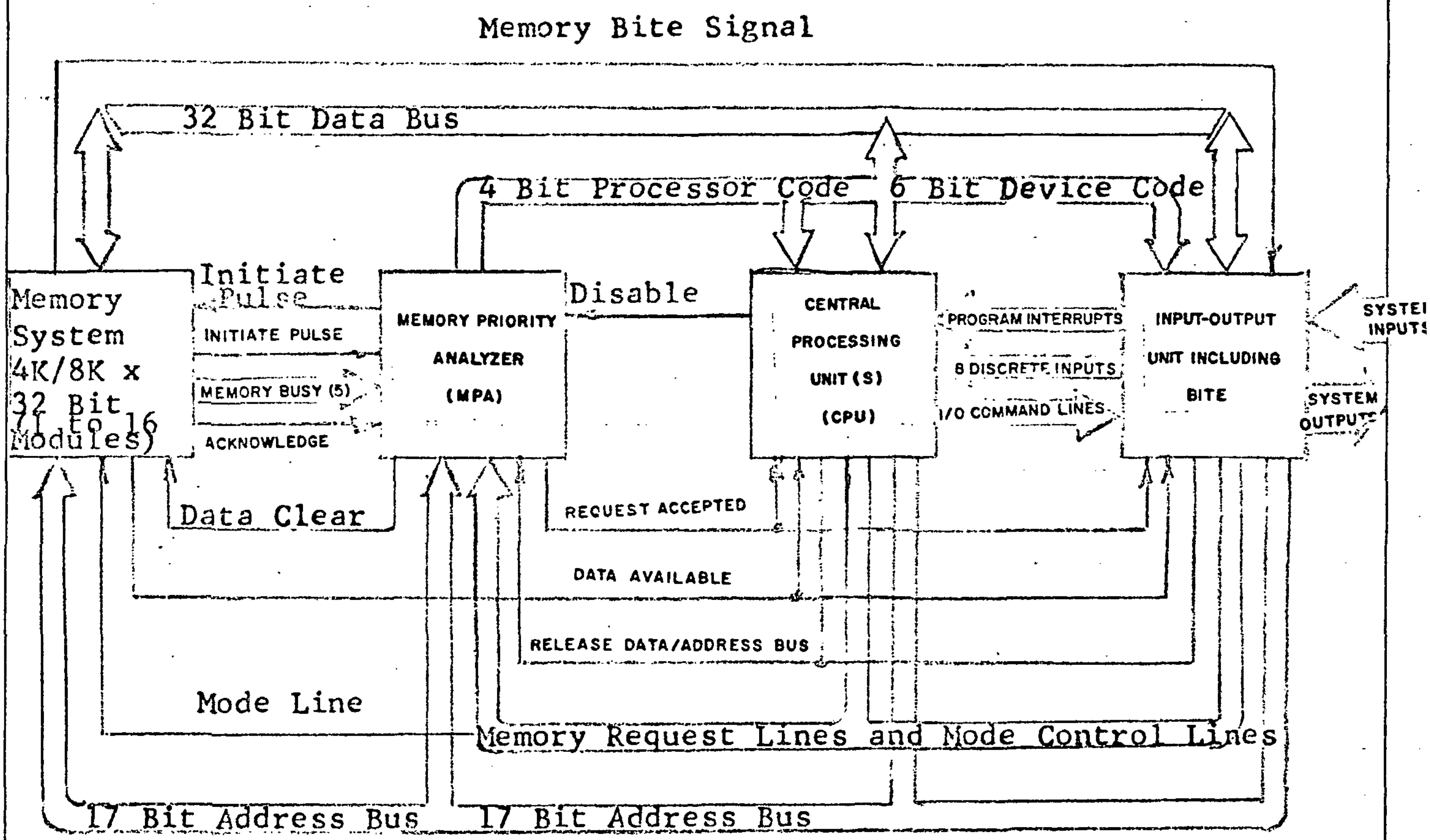


Simplex Computer with Two Memory Modules

Multi-Computer Configuration

Multi-Processor Configuration

Inter-module communication is accomplished via a bus consisting of 32 data lines, 17 memory address lines, 6 processor/device code lines, and several control lines. This bus, the bus priority analyzer, and asynchronous module operation allows direct memory access by processors or peripheral devices. It also permits processor-to-processor communication on a priority basis with minimum loss of time which would otherwise be wasted to provide synchronization.



SKC-2000 Basic Computer Bus Interconnection

SKC-2000 computer through-put is outstanding, and varies with the choice of memory module. A simplex computer performs 320,000 add type operations per second using the 1.9 μsec memory module, 470,000 operations per second with the LSI scratch pad and 530,000 operations per second when a solid state ROM and an LSI scratch pad memory is employed. Multi-computer and multi-processor configurations typically provide more than 1,000,000 operations per second with core memories. High density packaging gives the SKC-2000 computer outstanding through-put-to-volume and through-put-to-weight indices.

CENTRAL PROCESSOR MODULE

Parallel organization and floating point arithmetic in the central processor module provide outstanding arithmetic capabilities. The processor operates upon 32 bit operands and has double precision add and subtract instructions for those computations requiring exceptionally high accuracy. Simultaneous indexing capability on two levels facilitates matrix operations, and the 88 index registers located in memory assure sufficient capacity for large computation loads.

SKC-2000 FEATURES

Type	General purpose, parallel, asynchronous module, high speed data/control bus
Military Specifications	MIL-E-5400 Class 2, MIL-E-16400

PARALLEL CENTRAL PROCESSOR MODULES

Number Systems	Binary, floating point and two's complement fixed point		
Data Words, Floating Point	24 bit mantissa, 8 bit exponents		
Data Words, Fixed Point	32 bits including sign		
Instruction Words	16 bits short and 32 bits long		
Instructions	99 total long and short		
Address Modes	direct, indirect, relative, immediate		
Indexing 1st Level Registers	7 } 4 groups for a total of 88		
Indexing 2nd Level Registers	15 }		
Execution Times, Average for:	1.9 μsec Memory	1.9 μsec Memory with LSI	ROM Program Store & LSI Scratchpad
Add Fixed Point, μsec	3.125	2.125	1.875
Multiply Fixed Point, μsec	6.625	5.875	5.5
Divide Fixed Point, μsec	10.75	10.0	9.625
Add Floating Point, μsec	4.0 + NT	3.25 + NT	2.875 + NT
Multiply Floating Point, μsec	7.0	6.25	5.875
Divide Floating Point, μsec	9.5	8.75	8.325
Δ Time Indexing, μsec	2.0	1.0	1.0
Normalization Time (NT)	.25 μsec/shift		
Memory Words Directly Addressable	131,072		
Electronic Technology	MSI/TTL		
Power	75 Watts		
Voltage	+5 volts ±5%		
Weight	3 pounds including structure		
Options	Fast shift matrix, ½ word arithmetic		

MEMORY MODULES

Standard Type	20 MIL DRO lithium ferrite core, coincident current 3 wire, 3D
Optional Types	Plated wire, 1.2 μsec 2½ D core, solid state LSI scratchpad, solid state ROM
Module Size	4K, 8K words by 32 bits
Access Time, LSI	0.25 μsec
Access Time, 1.9 μsec Core	<1.00 μsec
Protected Program Storage	2 AGE and/or program controlled subsets
Contents Protection	Over-voltage; under-voltage; over-temperature; erroneous input signals
Special Techniques	Selection currents controlled by single thermal sensor; strobe timing slaved to selection current amplitude
Electronic Technology	MSI/TTL, Hybrid
Power, Long Term Average	85 watts typical, 1.9 μsec cycle core
Voltages	±5 volts ±5%; ±12.5 volts ±2%
Weight	6 pounds including structure

INPUT/OUTPUT CAPABILITY

Program Interrupts	16 priority levels — expandable (7 preassigned including power failure, BITE, Control Console)
Direct Memory Access I/O	16 priority levels (2 preassigned)
Data Bus	32 bit parallel — 4 MHz
Data Flow, 1.9 μsec Memory	500,000 words/sec
Data Flow LSI Memory	1,000,000 words/sec
Programmed I/O	To/From Memory; To/From CPU, command or data designation, with or without acknowledge
Channel Device Codes	61 directly addressable; 14 are useable for DMA
Discretes (Input Switch Type)	8 directly interrogated and branched upon

PHYSICAL FEATURES

Packaging	*½ ATR cross section, variable length*
Cooling	Forced air or cold plate
Circuit Cards	X-Y
Baseline SKC-2000	Parallel CPU, 8K x 32 bits 1.9 μsec cycle memory, 3 card I/O and power post-regulators
Size	15.33 in. long × 7.50 in. wide × 4.88 in. high
Power	245 watts including regulator loss
Weight	19.7 pounds

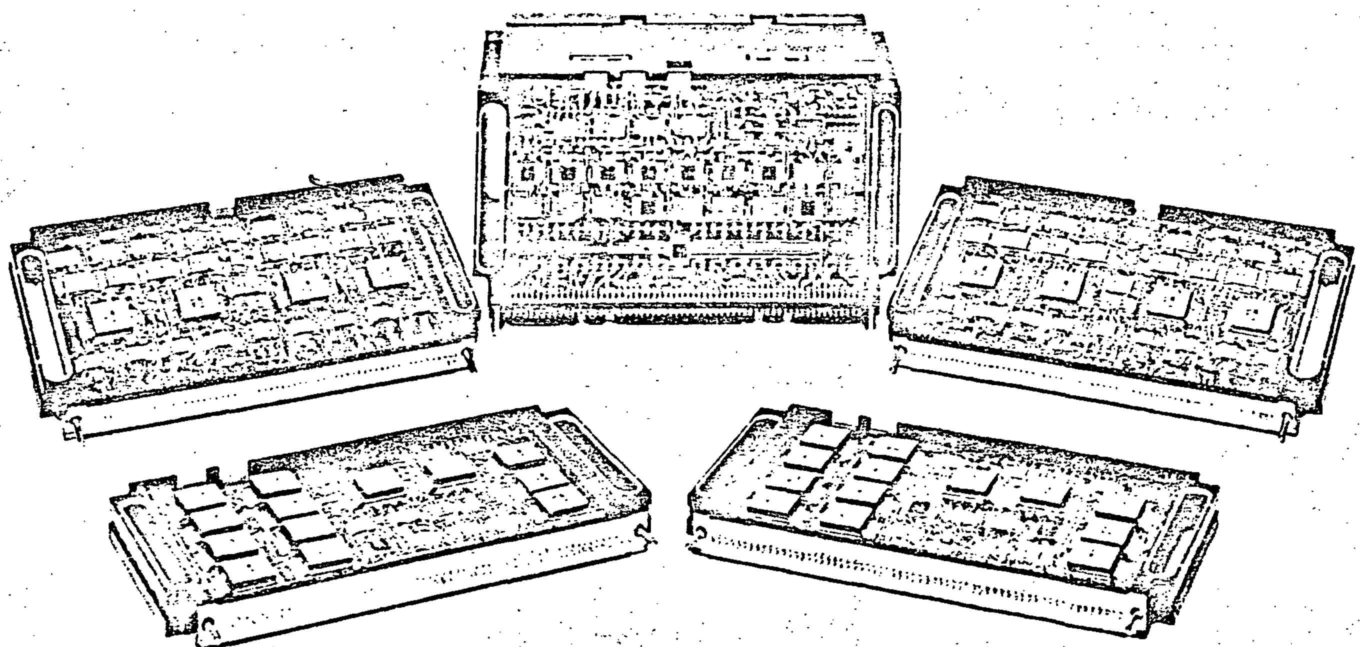
The order set of 99 was derived by detailed consideration of aerospace computer programming requirements and is a balanced set providing an optimum combination for computation and programming ease. Inclusion of many big-machine features facilitates transition to the SKC-2000 by the programmer who is familiar with large commercial computers.

Instructions having short and long formats, 16 and 32 bits respectively, enhance efficient memory usage. Experience shows that 80% to 90% of the instructions will be in short format form. Typically, instructions use the greater part of memory. Hence, packing two instructions per memory word results either in appreciable reduction in memory size or a greatly increased number of instructions for a given memory size. An 8192-word memory module provides approximately 10,000 instructions and 2,000 words of scratch pad, making it the equivalent of a 12,000-word memory.

Addressing modes include: (1) direct, for operand location at the address specified by the instruction; (2) indirect, for operand location at the address specified, at the address specified by the instruction (up to 16 levels); (3) relative, for operand location at the effective instruction address plus first and second level index registers; and (4) immediate, for operand location in the address portion of the instruction. Relative addressing may be combined with either direct or indirect addressing.

MEMORY MODULES

Designed as independent, asynchronous modules of 4K or 8K words by 32 bits, standard ferrite core memory modules have 1.9 microseconds cycle time. Asynchronous modules operate independently of the 4 MHz computer clock and use an initiate pulse from the requesting processor or device to start the memory cycle and issue a "data available" signal at the cycle's completion. Thus, asynchronous operation allows use of any memory cycle time while maintaining full compatibility with other modules.



Optional memory modules include LSI read only, LSI read/write, plated wire, and 1.2 μsec 2½ D core.

Standard core memories use MSI, TTL, and hybrid circuits to provide high reliability and small size. A novel circuit using a single precision alloy resistor mounted on the core stack to sense temperature and to control X and Y currents for both read and write, precisely slaves read and write currents to each other.

Another unique circuit slaves strobe timing to selection current amplitude and rise time, assuring interrogation at the output signal's optimum point.

Memory modules provide program storage protection against inadvertent write cycles, the protected portion being adjustable by program control or during manufacture to fit system requirements. Another design feature is the inclusion of memory address latches within each module.

Protection against over-voltage, under-voltage, over-temperature and erroneous input signals is built into the memory module. Resulting signals force the computer into a shut-down routine and are transmitted to the system's built-in test equipment (BITE).

INPUT/OUTPUT MODULE

Extensive input/output capabilities of the SKC-2000 facilitate design of an application-oriented I/O module and provide direct communication with external units. I/O capabilities of the computer include:

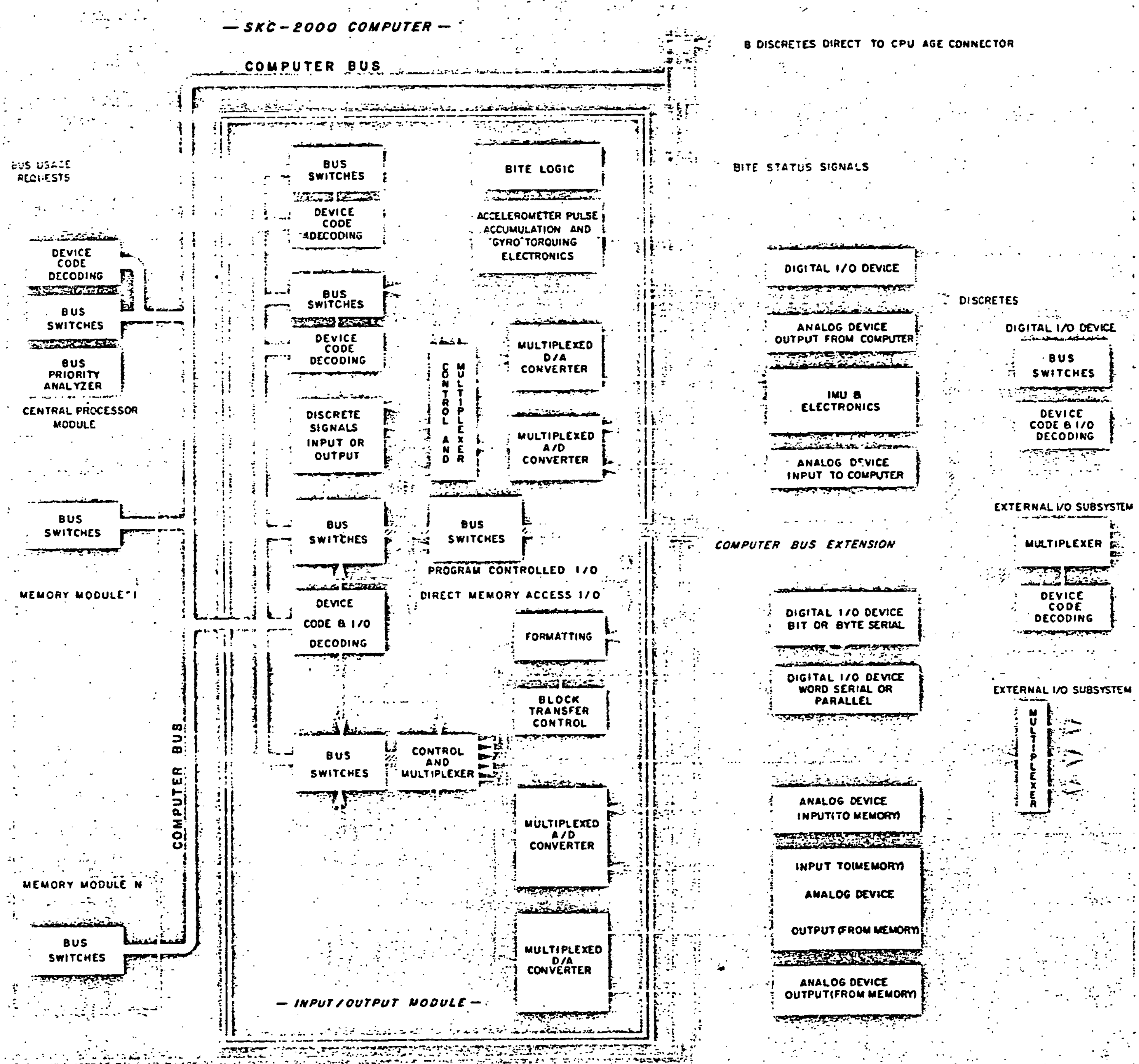
1. Data Transfer
   a) Program Controlled; "A" Register, Memory
   b) Direct Memory Access
2. Priority Interrupts
3. Discretes (switches)

Program-controlled I/O utilizes programmed instructions to transfer information to and from the computer "A" register or to and from the computer memory. These instructions may request an acknowledgement and may designate the information being transmitted either as data or as a command. All combinations of input/output, acknowledgement, and data/command give 16 types of program-controlled transfers for each of 61 directly addressable devices. Program-controlled I/O instructions are long or short instruction formats for global and non-global addressing with indirect addressing applicable to global addressing.

Direct memory access (DMA) by other computers or devices allows high speed data transfer to and from SKC-2000 memory modules independent of central processor module operation. Access to the memory is accomplished via the bus system allocated to various users on a priority basis. Use of DMA facilities permits an external device to insert or extract words almost arbitrarily from the computer memory, bypassing all program control logic. DMA transfer is particularly well suited to devices transferring large amounts of data in block form such as tapes, drums and disks, for information exchange between computers, or for sensors generating or requiring large amounts of data.

Program interrupts are signals having preassigned priority which divert the program to assigned locations in memory. Jump commands in each of these locations direct the computer to the interrupt subprogram. The program counter value at the time of the interrupt is stored for use when returning to the main program. Of the 16 interrupts provided in the basic computer, 7 are preassigned, leaving 9 for other system uses. Unlimited interrupt expandability may be provided in the input/output section.

8 discrete (switch) inputs which are directly accessible to the program via the "Jump on Switch i" order are accepted by the SKC-2000. Additional discretes may be handled by conventionally programmed I/O techniques.



TYPICAL INPUT/OUTPUT CAPABILITIES

Typical I/O channel capabilities include serial channels with I/O bit rates up to 4 megabits/second. Channel control can be by direct program control, by an I/O processor operating independent of the program, or by an I/O processor which operates independently but is directed by software. Similarly, parallel channels operating at up to 16 megabits per second are available.

A block transfer channel for communication with standard peripherals in byte format is also available. This channel is started under program control by setting a register with the starting address and number of words to be transferred. The transfer is then automatically controlled by the block transfer channel which stops when the correct number of words have been transferred.

Analog to digital and digital to analog converters built into the SKC-2000 digital computer's input/output module are compact, high accuracy, high reliability circuit board-mounted units demonstrating high-speed conversion compatible with high speed computers. Self-checking capabilities assure operational precision.

BITE SUBMODULE

A comprehensive combination of software and hardware built in test equipment provides efficient self-test features for error detection and program protection and is integral to computer design. The BITE submodule is located in the I/O module.

In operation, the combination —

- determines computer performance
- protects hardware from damage
- protects memory against information loss
- isolates faults for maintenance and traceability
- processes and traces errors

Software is provided to check program storage (memory sum), check the computer's central processor, end around check the I/O channels etc., and to process the BITE priority interrupt.

BITE priority interrupt is based on the following signals:

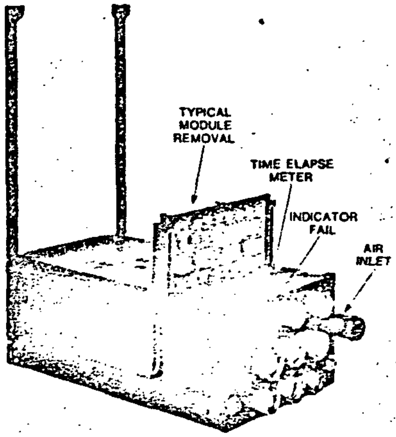
SIGNAL	FUNCTION
Memory Power Fail	Out-of-specification memory input power levels detected
BITE Memory One	Protected storage alteration attempted by program
Memory Over-Temperature	Memory temperature exceeds specified value
Bus Priority Time Out	User failed to generate memory release signals
Indirect Addressing Fail	Indirect addressing exceeded 16 levels
Time-Out Alarm	Time-out counter improperly reset by CPM
T_1 Time-Out Alarm	CPM reset time exceeded time allotted
Master Clock Pulse Fail	Master clock failure detected
Primary Power Fail Interrupt	120 μsec warning signal to CPM
4 Spare Signals	Available per application requirements

POWER SUPPLY MODULE

Power supply modules for the computer system vary depending upon the type of memory modules chosen and input/output module design. The power supply typically operates from 115 volts 400 Hz or from 28 volts DC prime power. Each power supply module is equipped with a high efficiency switching regulator designed in accordance with MIL-STD-704A, and has over-voltage, over-load, and short circuit protection. In addition to providing line conditioning and RFI attenuation, continuous fault monitoring (BITE) operates to assure orderly shutdown in the event of out-of-tolerance conditions. Internal energy storage is sufficient to complete the current instruction and store the contents of status and CPU registers. This orderly shutdown is followed by deactivation of the memory circuits.

PACKAGING

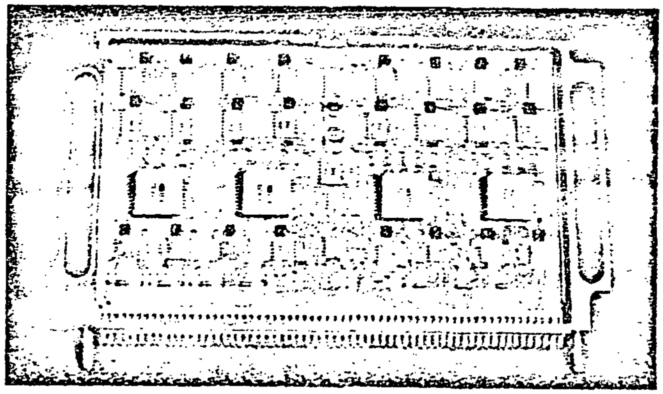
SKC-2000 packaging employs a stacked and clamped card mounting technique in which the cards form an integral "box-like" structure for the overall assembly. Four longitudinal members (straps) are used to clamp the cards between a front connector panel and rear support panel. Lower straps form a subchassis by providing the structure for retaining the female connector plate motherboard assemblies and for card keying and guiding. Card removal is effected by loosening four front bolts and lifting the two upper straps.



Designed to be hard-mounted to eliminate the need for vibration isolators, the computer has been subjected to extensive vibration tests without any failures. A thorough dynamic analysis shows that all stress levels are well below fatigue limits.

SKC-2000 computers are designed to be easily maintained, having conveniently located and readily accessible test points which contribute to a short mean time to repair. The design includes separate external AGE and test connectors and accessible test points on individual cards.

Circuit card construction used throughout incorporates several unique features which provide efficient methods of flatpack interconnection and heat removal while reducing electromagnetic interference. A glass-filled epoxy double-sided laminate 0.008 inch thick provides electrical insulation and serves as a base for the conductors. These are routed in the X direction on the outer surface and the Y direction on the inner surface. Interconnection between X and Y conductors is accomplished by plated-through holes, a type of construction which achieves a significant increase in reliability and producibility over multilayer boards.



SKC-2000 COMPUTER PROGRAMMING AND SOFTWARE

The SKC-2000 computer is significantly easier to program than previous military computers. Features that facilitate programming are:

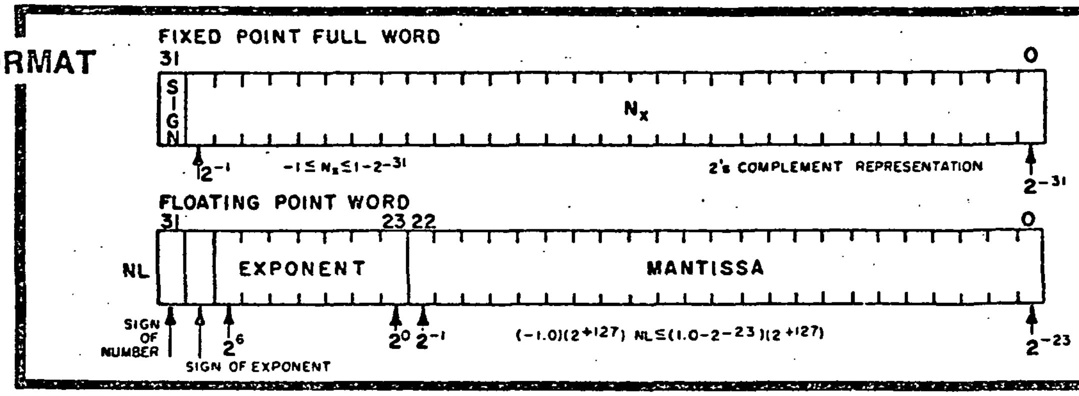
- Floating point arithmetic eliminates scaling
- 88 index registers eliminate memory boundaries
- Dual level index registers simplify reentrant subroutines
- Incremental addressing in branch instructions for relocatable subroutines
- Trap instructions provide for special instructions
- Large instruction repertoire for minimizing computation time

In addition to hardware designed for the programmer, a powerful software package is available to complement machine language programming. It includes an assembler, a loader, an arithmetic simulator, an interpretive simulator, utility routines, and diagnostic routines.

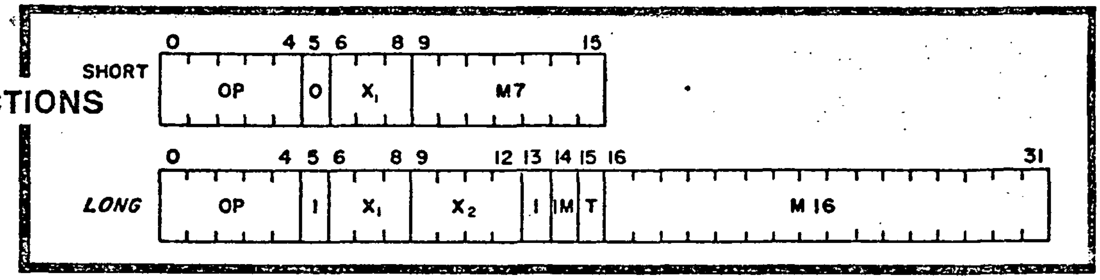
The assembler/loader is designed to allow the user great flexibility and utility in his coding. Features include relocatability, reentrancy, addressing via index register modification, availability of common (blank or labelled) and globally defined symbols. An assembly language, FOCAP, has been developed and documented for the assembler/loader.

SKC-2000 simulators operate on a host machine via the host machine's assembler. They simulate execution of a user program written for the SKC-2000 as if the user program were actually running on the SKC-2000. The arithmetic simulator reproduces the arithmetic of the SKC-2000 in the manner most expeditious in the host machine. An interpretive simulator accepts the binary pattern used in the SKC-2000 and interprets each instruction, performing each operation exactly as in the SKC-2000. The arithmetic simulator runs much faster and is useful for much of the program checkout. The interpretive simulator is a very desirable tool where real-time and I/O problems do not lend themselves to hands-on debugging.

DATA FORMAT



BASIC INSTRUCTIONS



OP	PRIMARY OPERATION CODE (5 BITS)
LONG/SHORT OPTION	BIT 5 IS 0 FOR SHORT BIT 5 IS 1 FOR LONG
X_1	1ST LEVEL INDEX REGISTER SELECTION
X_2	2ND LEVEL INDEX REGISTER SELECTION
IM	IMMEDIATE ADDRESSING
I	INDIRECT ADDRESSING
T	TRAP BIT
M7, M16	MEMORY ADDRESS FIELD
E	EFFECTIVE ADDRESS $= M + (X_1) + (X_2)$

MNEMONIC	INSTRUCTION	EXECUTION TIME (μSEC) SHORT FORMAT CORE	SHORT FORMAT LSI	LONG FORMAT CORE	LONG FORMAT LSI
LDA	LOAD THE A REGISTER	3.125	1.625	4.0	2.25
LDB	LOAD THE B REGISTER	3.125	1.625	4.0	2.25
LDX	LOAD THE X REGISTER	3.125	1.625	4.0	2.25
ADU	ADD UPPER FIX POINT	3.125	1.625	4.0	2.25
ADL	ADD LOWER FIX POINT	3.125	1.625	4.0	2.25
SBU	SUBTRACT UPPER FIX POINT	3.125	1.625	4.0	2.25
SBL	SUBTRACT LOWER FIX POINT	3.125	1.625	4.0	2.25
ADF	ADD FLOATING POINT	3.5 + SC*	2.75	4.125 + SC*	3.50
SBF	SUBTRACT FLOATING POINT	3.5 + SC*	2.75	4.125 + SC*	3.50
MUL	MULTIPLY FIX POINT	6.75	5.325	7.5	6.25
DVD	DIVIDE FIX POINT	10.75	9.5	11.5	10.25
MLF	MULTIPLY FLOATING POINT	7.0	5.75	7.75	6.50
DVF	DIVIDE FLOATING POINT	9.5	8.25	10.25	9.00
SAM	SKIP ON A REGISTER MASKED	3.5	2.5	1.25	3.00
LOR	LOGICAL OR	3.125	1.625	4.0	2.25
AND	LOGICAL AND	3.125	1.625	4.0	2.25
EXO	EXCLUSIVE OR	3.125	1.625	4.0	2.25
STA	STORE THE A REGISTER	3.125	1.625	4.0	2.25
STB	STORE THE B REGISTER	3.125	1.625	4.0	2.25
STX	STORE THE X REGISTER	3.125	1.625	4.0	2.25
RTA	RETURN ADDRESS (UNCONDITIONAL JUMP)	3.5	2.25	4.125	3.00
AFD	ADD DOUBLE PRECISION FLOATING POINT	3.75 + SC*	3.25	4.375 + SC*	3.75
SFB	SUBTRACT DOUBLE PRECISION FLOATING POINT	3.75 + SC*	3.25	4.375 + SC*	3.75
LAE	LOAD THE A REGISTER WITH E	1.625		2.25	
LDS	LOAD STATUS REGISTER			4.0	2.25
STS	STORE STATUS REGISTER			4.0	2.25
LDI	LOAD INTERRUPT REGISTER			4.0	2.25
STI	STORE INTERRUPT REGISTER			4.0	2.25

TIMES ARE BASED UPON 1.9 μSEC CORE AND LSI MEMORY. *NORMALIZATION TIME 0.25 μSEC/SHIFT

F-398-8-A

SHIFT INSTRUCTIONS

0–4	5	6–9	10–11	11–15
OP	02	X_1	04	N

02 AND 04 SECONDARY OPERATIONS CODES
S NUMBER OF PLACES TO BE SHIFTED
$S = (X_1) + N$

MNEMONIC	INSTRUCTION	EXECUTION TIME (μSEC) SHORT FORMAT
SRA	SHIFT A RIGHT ALGEGRAICALLY	1.75 + N/4 WHERE N IS THE NUMBER OF BITS SHIFTED
SLA	SHIFT A LEFT LOGICALLY	
SRL	SHIFT A, B RIGHT ALGEBRAICALLY	
SLL	SHIFT A, B LEFT LOGICALLY	
SRC	SHIFT A RIGHT CIRCULARLY	
SCR	SHIFT A, B RIGHT CIRCULARLY	
SCL	SHIFT A, B LEFT CIRCULARLY	
SRG	SHIFT A, B RIGHT LOGICALLY	

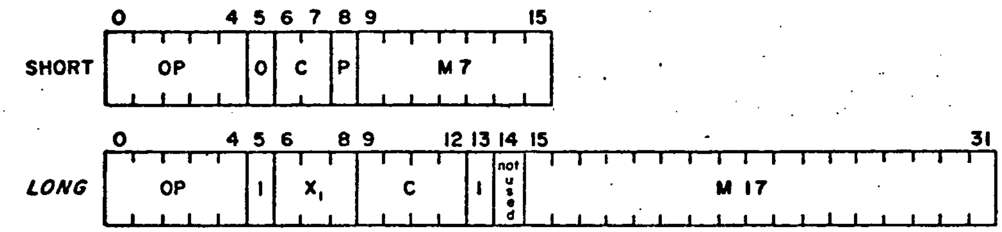
INDEX-REGISTER COMPARE AND INCREMENT INSTRUCTIONS

0–4	5	6–8	9–12	13	14	15	16–31
OP		X_1	X_2	I	IM	T	M 16

T TEST CRITERION/ADD-SUBTRACT
E EFFECTIVE ADDRESS = M16 OR (M16)

MNEMONIC	INSTRUCTION	EXECUTION TIME (μSEC) LONG FORMAT	
		CORE	LSI
ICN	TEST SELECTED INDEX-REGISTER FOR NOT-EQUAL, AND SKIP	4.00	2.75
ICL	TEST SELECTED INDEX-REGISTER FOR LESS-THAN, AND SKIP	4.00	2.75
IMP	MODIFY INDEX-REGISTER BY POSITIVE INCREMENT	4.00	2.75
IMN	MODIFY INDEX-REGISTER BY NEGATIVE INCREMENT	4.00	2.75

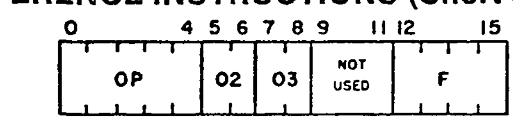
JUMP INSTRUCTIONS



C	JUMP CRITERIA
BIT 5	0 = SHORT; 1 = LONG
P	ADD OR SUBTRACT

MNEMONIC	INSTRUCTION	EXECUTION TIME (μSEC) SHORT FORMAT	LONG FORMAT CORE	LONG FORMAT LSI
JRU	JUMP RELATIVE UNCONDITIONAL	1.63		
JRL	JUMP RELATIVE IF C(A) < 0	1.63		
JRG	JUMP RELATIVE IF C(A) > 0	1.63		
JRN	JUMP RELATIVE IF C(A) = 0	1.63		
JGF	JUMP ON PROGRAM FLAG		2.25	
JAL	JUMP IF C(A) < 0		2.25	
JAG	JUMP IF C(A) > 0		2.25	
JAN	JUMP IF C(A) = 0		2.25	
JGWi	JUMP ON SWITCH i		2.25	
JGSi	JUMP ON STATUS BIT i		2.25	
JGU	JUMP UNCONDITIONALLY		2.25	
JGI	JUMP UNCONDITIONALLY TO SUBROUTINE		6.00	4.25
TRPi	JUMP THROUGH TRAP INTERRUPT (4 AVAILABLE)		6.00	

NON-MEMORY REFERENCE INSTRUCTIONS (Short Instruction)

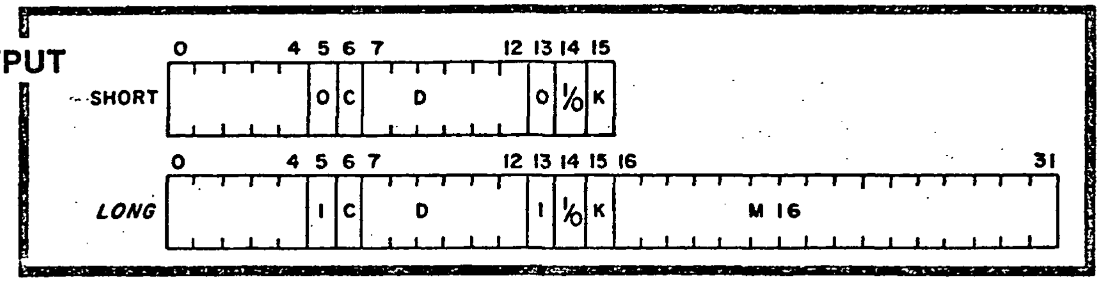


02, 03	SECONDARY OPERATION CODES
F	PROGRAM FLAGS/INTERRUPT CONTROL SELECTION

MNEMONIC	INSTRUCTION	INSTRUCTION TIME (μSEC) SHORT FORMAT
NOP	NO OPERATION	1.63
HLT	HALT	1.63
SET	SET SELECTED PROGRAM FLAGS	1.63
RST	RESET SELECTED PROGRAM FLAGS	1.63
EPI	ENABLE PROGRAM INTERRUPTS	1.63
DPI	DISABLE PROGRAM INTERRUPTS	1.63
DMI	DISABLE MEMORY INTERRUPTS	1.63
EMI	ENABLE MEMORY INTERRUPTS	1.63
CFX	CONVERT FLOATING POINT TO FIXED POINT	2.25 + NT*
CXF	CONVERT FIXED POINT TO FLOATING POINT	4.25 + NT*
EAB	EXCHANGE A & B REGISTERS	1.63

*NORMALIZATION TIME 0.25 μSEC/SHIFT

INPUT-OUTPUT



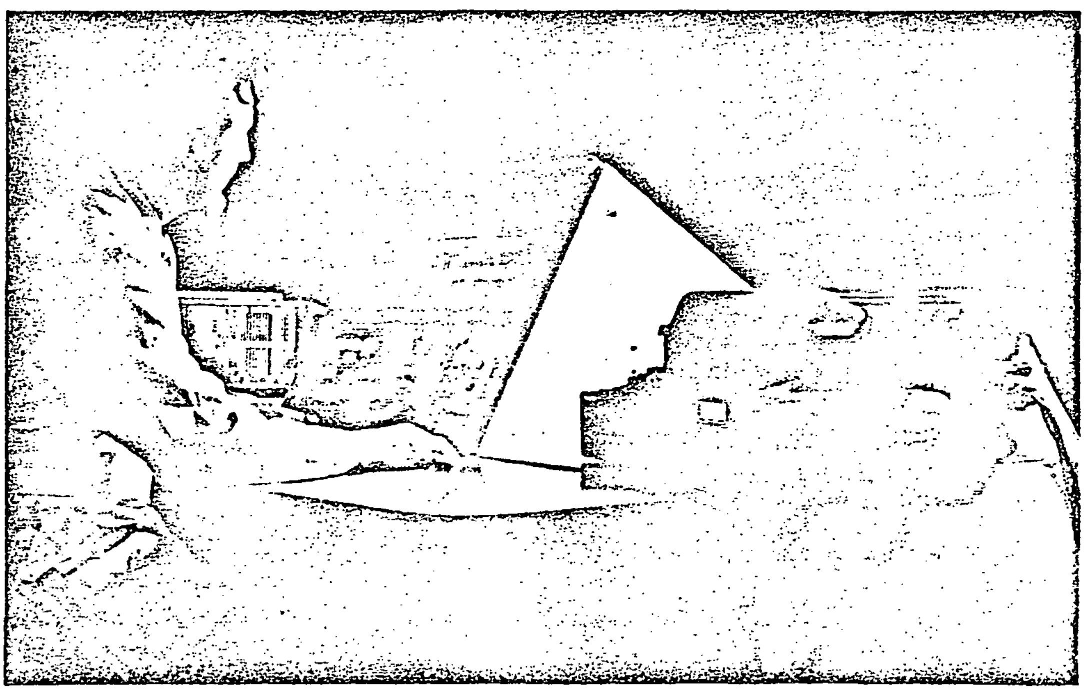
I/O	INPUT/OUTPUT
BIT 5	LONG = 1, SHORT = 0
D	6 BIT DEVICE CODE
C	COMMAND BIT
K	ACKNOWLEDGE

MNEMONIC	INSTRUCTIONS	INSTRUCTION TIME (μSEC)	
		SHORT FORMAT	LONG FORMAT
CIM	CONDITION, I/O TO MEMORY		4**
DIM	DATA, I/O TO MEMORY		4**
CIP	CONDITION, I/O TO CPU (A REG)	2.38 AVG.	
DIP	DATA, I/O TO CPU (A REG)	2.38 AVG.	
COM	CONDITION, MEMORY TO I/O		4**
DOM	DATA, MEMORY TO I/O		4**
COP	CONDITION, CPU (A REG) TO I/O	2.38 AVG.	
DOP	DATA, CPU (A REG) TO I/O	2.38 AVG.	

**PLUS ACKNOWLEDGE DELAY

NOTES:

1. *Times are average, based upon use of the 2.0 μs memory initiation pulse and include memory access. Individual times vary dependent on machine operational state.*
2. *Reductions in execution times are available through memory overlap operation.*
3. *Each indirect operation adds one memory cycle time to the execution time.*
4. *One level indexing does not increase execution time. Two levels add 0.5 μsec.*



4.18.4 Orbiter GN&C Operational Flight Computer Program

The operational flight program for the Shuttle on-board GN&C computer consists of an executive and a set of "mission" modules (see Figure 4.18.4-1 which can be assembled to build up an operational program to accommodate all Shuttle baseline missions. The computer program for the GN&C computer will be devoted to the solution, display and performance monitoring of the Guidance, Navigation and Control problems of the Shuttle Orbiter. Peripheral support such as displays and performance monitoring of non-GN&C orbiter subsystems will be provided by dedicated MDE processors which are divorced from the GN&C processors.

The GN&C Operational Flight computer program description covers the operational software functions for the orbiter from prelaunch through landing. The GN&C software functions include Guidance, Navigation, Control, Attitude reference, Executive, and in conjunction with the MDE computer, Display Processing and status monitoring. The defined functions within the above functions for all mission phases are:

- Guidance

  Prelaunch Targeting

  Atmospheric guidance during boost

  Orbit injection guidance

  Prethrust targeting for rendezvous and deorbit

  Cross-product steering

  Entry guidance

  Approach guidance

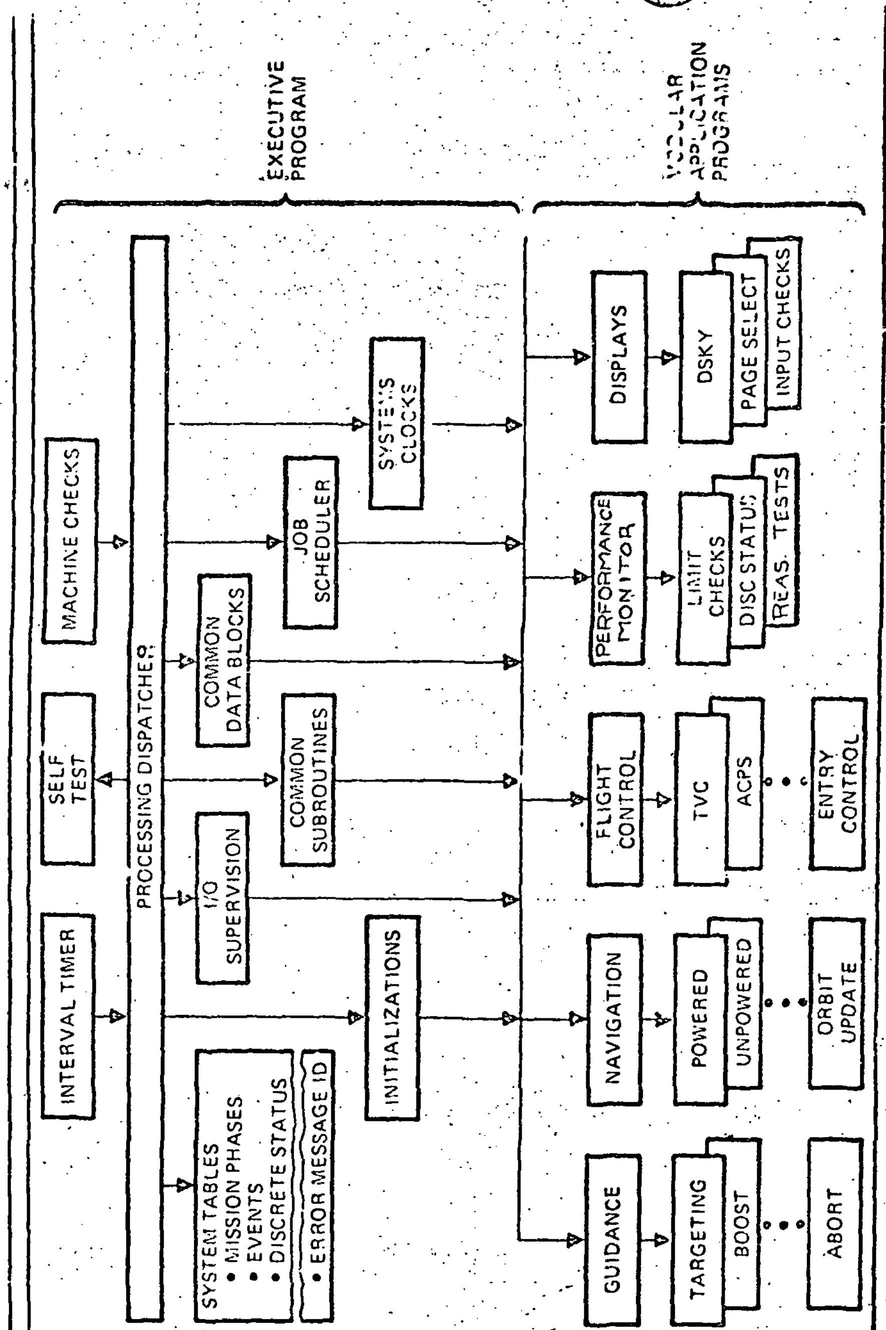


Figure 4.18.4-1 GN&C Flight Software Program Structure

- Navigation

  Prelaunch alignment

  Powered and unpowered navigation

  Earth relative navigation

  On-orbit updating (horizon sensor)

  Target navigation

  Entry navigation

  Atmospheric navigation (TACAN Updating)

- Attitude Reference

  Direction cosine and Euler angle computation

  Platform updates (Star Tracker)

  Gyro compassing during prelaunch

- Flight Control

  Main engine thrust vector control

  ACPS control

  Thrust on-off control

  Entry program control

- Executive

  Task control

  I/O control

  Common routines

  Tables

- Display Processing
  - Display I/O
  - Tables
- Status Monitoring
  - Maintain system status
  - Computer self test
  - Command/response tests
  - Limit checks
  - Reasonableness tests

In addition, the software functions include sequencing and interfacing with the MDE computer and systems other than GN&C.

The overall control of the operational program is provided by the executive sub-program. The Executive sub-program provides for the flow of information between programs and the external environment and provides for standardization of interfaces in the presence of a variety of modules and for management of the program resources to assure the response required by the system. In addition, the executive program will provide an internal environment which will permit application programs to be constructed and executed independently of one another and allow a modular program structure which emphasizes operational flexibility, ease of modification, and capability for growth.

All computer program instructions will reside in protected memory locations while all computer variable data will reside in unprotected memory locations. Protected memory is defined as read-only memory while unprotected memory is defined as read-write memory. The protected and unprotected memory locations are predetermined during the program development process and will not change during a mission time span. All unused memory locations will be assigned as protected memory. If any program attempts to write into protected memory,the computer will enter a fail state.

Individual program will be responsible for all internal program calculation scaling and arithmetic manipulations in order to attain sufficient accuracy as required for the Shuttle Orbiter configuration for all mission phases. Extended precision arithmetic functions will be used as required. The fixed point scaling must be sufficient to accommodate the maximum expected values and maintain the required parameter resolution.

Software Elements

The following paragraphs describe design concepts for the onboard Shuttle GN&C Flight Program.

a. On-board Executive System. This section describes the onboard executive control system.

1. Basic Structure. The executive structure will be based on a timer interrupt, fixed schedule, time slice mode of operation. In effect, this type of executive does not require, nor allow, instantaneous

response to external interrupts.

The timer interrupt interval will be under program control and will provide the initialization and reference for the basic executive time structure. When the executive detects a timer interrupt, it will process its schedule tables to determine which set of program jobs are to be executed during the next program interval.

All program intervals will be approximately equal in execution time. This will be accomplished by predetermining the time required for the longest path in each job and shuffling jobs to balance the time intervals. Each time interval will contain portions of various rate jobs as required to satisfy the time and rate constraints.

The program intervals occur 25 times per second, based on the maximum system sample rate. Each interval contains all 25 per second jobs, every other interval contains 25 and 12.5 per second jobs, every fourth interval contains 25 and 6.25 per second data, etc.

2. Job Schedule. The Job Schedule function manages the sequence in which programs are executed. System considerations determine the priority of the routines driven by the schedule function. The job schedule function is designed to respond quickly to execution requests on a predetermined priority basis.

All external interrupts are treated by the schedule as discrete inputs (ON-OFF). The scheduler will mask the interrupt and sample the interrupt status on a periodic basis. An interrupt detected as ON will result in the scheduler modifying its scheduled jobs or setting a flag to be used by one of the sub-programs.

Internal interrupts such as the Internal Timer or Machine Error interrupts will be handled as true interrupts, as follows:

(a) Internal Timer Interrupt - This interrupt will occur when a pre-set timer counts down. The program will set the time to count down in 40 milliseconds (based on accepted maximum sample rate of 25 per second). This interrupt will cause the program to exit its present scheduled job and return to the basic scheduler entry point. The program structure will be designed so that this interrupt will always occur during a low priority (background) job or during an idle state.

(b) Machine Error Interrupt - This interrupt indicates that the computer circuitry has detected a problem in the computer hardware. This interrupt will force the program into an error detection self test mode. Verification of the machine error will cause the computer to be shut down. One of the redundant computer systems will take over control of the orbiter vehicle by manual or automatic means.

3. Input/Output Supervision. The input/output supervision function controls all the operations associated with the input/output buffer device. Effectively, the input/output supervisor will input data at interval n to be used in calculations in interval n + 1. The output data at interval n is for data calculated during interval n-1. For time critical functions, the input-calculation-output process will be optimized to satisfy the required time constraints.

The input and output data is normally a function of mission phase, sequencing, control panel selection, job scheduling, etc.; as such, the data set is variable and is a function of the jobs schedule by the executive.

The input/output data will be transmitted to or received from the input/output buffer utilizing predetermined unprotected main memory locations.

4. Systems Services. Systems services provides the function of system and applications program interface with the operator of the system. This function will sample all input control media such as control panel switch positions at a predetermined and constant rate, nominally 6 samples per second. If the panel switch inputs are in a changed state for two or more successive samples, the services facility will either change the job schedule to satisfy the inputs or set a flag for usage by the subsystems program.

5. Systems Clocks. This facility will provide the capability of maintaining unique clock facilities for system usage. The clock facility will be utilized for display functions, sequencing, or subsystem usage. System clock accuracy and update rate will be consistemt with accuracy constraints on the onboard system. The clock facilities will include Time From Launch, GMT, Launch Local Time, Time to Go to Event, and Time in Mission Phase. Maximum clock update rate for a minimum number of time bases will be 25 per second. Nominal clock update rate will be 1 per second. The basic time information for

calibrating the Software clocks shall be provided by the Master Timing Unit or the Display Keyboard.

6. Common Subroutines. This service facility will contain all common subroutines to be used by the various subprograms. Included in the common subroutines are the standard mathematical functions to determine sine, cosine, tangent, square root, root mean square, matrix inversion, etc. In addition to the standard set of subroutines, this facility will contain other routines that are used by more than one subprogram. The subprograms will branch to the appropriate subroutine as necessary; exit from the subroutine will be to the using subprogram.

7. Common Data Blocks. The Executive program will maintain common data blocks of unprotected memory. The Common Data Blocks will be available for usage by all programs, subprograms or routines. Inclusion of data in the Common Data Blocks implies that data is used across program interfaces. Data that is unique to individual programs such as scratch pad memory, constants, etc., will be maintained in individual local program data blocks which are not accessible to other programs.

8. Self Test. The Self Test function will be executed on a minor cycle basis to determine and verify the operational state of the GN&C computer. This function will be designed to execute a maximum number of computer program instructions and to compare results of mathematical manipulations against a predetermined result. Failure of the Self Test function in a particular computer will result in a shutdown of that computer.

9. <u>Systems Tables</u>. The executive program structure will be designed to maximize the usage of System Tables in conjunction with simple subroutines for execution of the System Table data. The System Table concept simplifies the programming, checkout and validation of the GN&C Flight Program in addition to providing a relatively simple means of incorporating program changes and updates.

b. <u>Guidance, Navigation, and Control Programs</u>. The GN&C software consists of the operational software functions from prelaunch through landing. The GN&C software functions include Guidance, Navigation, Control, attitude reference, Executive, Display processing, and status monitoring. These functions are required for both the orbiter and the booster of the flyback (LOX/RP) configuration.

However, a pressure fed booster which is controlled by the orbiter and which is towed to the launch site after a free fall return would not require any software functions: the orbiter GN&C computer would be augmented to provide the pressure fed booster control during the mated boost phase.

The LOX/RP booster configuration contains its own GN&C computer with the same functions as the orbiter during prelaunch, mated boost, and separation. A period of booster coast and retunr to atmospheric flight requires computer Guidance, Navigation and ACPS control. These functions are performed by the computer until the pilot initiates manual control.

Table 4.18.4-1 summarizes the orbiter GN&C program functions by mission phase.

Table 4.18.4-1 Orbiter Mission Phase Requirements Summary

PHASE	GN&C FUNCTIONS
Prelaunch	Targeting Update Powered Navigation Prelaunch Nav Update Direction Cosines Computations Prelaunch Gyro Compassing
Mated Boost	Powered Navigation Earth Relative Navigation Direction Cosines Computations Open Loop Guidance Closed Loop Guidance Abort Guidance
Separation/Orbit Insertion	Powered Navigation Earth Relative Navigation Separation Guidance Abort Guidance Injection Guidance Separation Control Thrust Vector Control Thrust On-Off Throttle Control Direction Cosine Computations Euler to Body Transforms
Rendezvous/Unpowered	Orbital Navigation Direction Cosine Computations Update Platform Attitude Euler to Body Transform ACPS Control Thrust On-Off Control Rendezvous Targeting Prethrust Maneuver Targeting Targeting Conic Routines
Rndezvous/Powered	Powered Navigation Direction Cosine Computations Euler to Body Transform Thrust On-Off Control Cross Product Steering
Docking	Unpowered/powered Navigation Manipulator Arms Target Relative Navigation Direction Cosine Computations Euler to Body Transform ACPS Control

Table 4.18.4-1 Orbiter Mission Phase Requirements Summary (Cont.)

PHASE	GN&C FUNCTIONS
Deorbit Maneuver Sequence (Unpowered)	Unpowered Navigation Earth Relative Navigation Direction Cosine Computations Euler to Body Transform Attitude Update Deorbit Targeting Target Conic Routines Prethrust Maneuver ACPS Control
Deorbit Maneuver Sequence (Powered)	Powered Navigation Earth Relative Navigation Direction Cosine Computations Euler to Body Transform Thrust Vector Control Thrust On-Off Control Cross Product Steering
Entry	Powered Navigation Earth Relative Navigation Direction Cosine Computations Body Axis Error Computations Attitude Control
Aerodynamic	Approach Guidance Earth Relative Navigation Atmospheric Nav/Aids Updates Aero Flight Control Attitude Reference

Baseline GN&C computer sizing estimates are based on well-understood requirements from previous programs, are therefore relatively accurate, and are not expected to grow beyond the 100-percent reserve provided. If added functions during the definition phase cause growth beyond the reserve, available access channels will accommodate mass memory. The performance monitor (PM) and payload (PL) functions are not as well defined, and the use of tape mass memory allows expansion and the sharing of functions to minimize the number of computers. Tape memory was selected for the modular display electronics functions, which are not time critical, based on availability and low power, weight, and cost. The payload data processing requirement of 10,000 32-bit words is satisfied by the use of a modular display electronics unit with an equivalent of 8000 32-bit words in the main memory and the tape mass memory. (Preliminary analysis indicates a 4000-word resident memory augmented with incremental 2000pword tape loads is adequate.)

Hardware floaring point arithmetic is implemented in the GN&C computer to reduce the problem of scaling errors and provide the lowest program cost for the primarily arithmetic GN&C software. Hardware floating point is not provided in the modular display electonics; the added cost to provide this capability for the primarily data handling and logic functions of the modular display electronics, which do not involve a large amount of scaling, is not justified by offsetting software savings.

Table 4.18.4-2 illustrates the assignment of programs and requirements to the functionally dedicated GN&C computers and MDE's and identifies the memory size, speed estimates, and capabilities. These values were derived by analyses of vehicle requirements, modeling, and coding of selected functions. The results of this process compared within 5 percent with equivalent programs from Apollo RTCC and Saturn IU.

Table 4.18.4-2

Program Functional Requirements and Memory Estimates

GN&C			PM and Backup GN&C MDE's			PLM and PLH MDE's		
Function	Size and (Speed) GN&C	Size and (Speed) MDE	Function	Size and (Speed) PM	Size and (Speed) BU GN&C	Function	Size and (Speed) PLM	Size and (Speed) PLH
Executive Data pool, subroutines, input/out, sequencing, scheduling, tables	3950 (13.8)	350 (0.5)	Executive Subroutines, input/output, data pool, tables	400 (0.5)	400 (5.0)	Executive Subroutines, input/output, data pool, tables	900 (2.5)	900 (2.5)
System status Redundancy management, GN&C, system performance monitor, ground checkout	2200 (25.5)		System status Non-GN&C system performance monitor, annunciator lights, self-check	900 (20.0)		System status Performance monitor, analog, discrete, serial limit checks	*2000	1400
Flight crew displays GN&C system status, procedures	450 (0.4)	3600 (5.6)	Crew station displays Input/output conversions, formats, tables, display services, time share GN&C, PM, and backup GN&C	3900 (5.6)		Crew displays Input/output conversions, formats, tables, display services	500 (5.6)	500 (5.6)
Navigation State vector, attitude reference	5850 (2.1)		Backup GN&C Steer to displays, get-home guidance and navigation		2950 (33.1)	Payload checkout Minimal on-board checkout, no experiment processing	*2000 (3.0)	
Guidance Steering commands, engine control, abort	7500 (39.5)					PLH control program Closed-loop control of attitude, rate, and position		1000 (36.3)
Targeting	4200 (19.3)					PLH constraint tables Deployment and retrieval trajectories, payload dependent tables		1000
Flight control TVC, RCS, digital outer loop, boost blending	2950 (20.1)					Program loader Tape memory load on request	150	150
Program services Uplink, PCM, preflight, system align and calibration, interfaces (booster, payload, main engine controller, GSE interface)	2700 (5.0)							
Unmanned orbiter Sequencing, calculation for autonomous flight	500 (2.5)		Program loader Tape memory load on crew request	150	150			
Program loader		100						
Peak memory loading	30300 (128.2)	4050 (6.1)		5350 (26.1)	3500 (38.1)		3550 (13.1)	4950 (44.9)
Machine capability main memory	65536 (380.0)	8192 (296.0)		8192 (296.0)			8192 (296.0)	
Tape memory		256 K		256 K			256 K	

Note: Sizing is in terms of equivalent 32-bit words. Most GN&C computer instructions are 16 bits.
MDE computers are all 16 bits. Floating point data are 32 or 48 bits and other data are 16 or 32 bits. Speed in KADS.
MDE—modular display electronics PLM—payload management PM—performance monitor PLH—payload handling

*Mutually exclusive, tape memory load

4.18.5 Rationale

Not applicable.

4.18.6 Assumptions

None.

4.18.7 Data References

232	MSC-03329	Space Shuttle Alternate Avionics System Study and Phase B Extension Final Report dated 12 November 1971 Pgs. 2.3-29 to 2.3-46 2.10-1 th 2.10-48
166	SD72-SH-50-s	Technical Proposal for Space Shuttle Program Volume III dated 12 May 1972 pgs. 3-95 to 3-122
41	SD72-SH-0023	Space Shuttle Phase B Avionics Final Report dated 8 March 1972 Pgs. 70-73
65	SKC2000	Aerospace Digital Computer SKC-2000 dated Nov. 30, 1970
187	AP101/SP1	IBM OBC Candidates for Shuttle