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LMSC-A995931
15 Nov 1971
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ERRATA
FOR
ALTERNATE CONCEPTS STUDY
EXTENSION

FINAL REPORT

LOCKHEED MISSILES & SPACE COMPANY

6 December 1971

NASA Manned Spacecraft Center
Houston, Texas 77058

Attention: J/86 - Distribution Operations

Subject: Contract NAS8-26362 - Supplemental Agreement No. 6
Alternate Concepts Study Extension Final Report

Reference: (A) Lockheed letter to W.C. Farries (NASA/MSC), dated 3 December 1971 (LMSC-A997281)

(B) Lockheed letter to J.W. Harden (NASA/MSFC), dated 29 November 1971 (LMSC-A997275)

Enclosure: (1) Sixty (60) copies of Part 4 of Volume II of Final Report (LMSC-A995931)

(2) Errata for Alternate Concepts Study Extension Final Report, dated 15 November 1971 (LMSC-A995931)

(3) Distribution List

Gentlemen:

Enclosure (1) is provided to complete the submittal of the subject Final Report in accordance with DR MA-03 referenced in Attachment 1, Data Procurement, to Exhibit "D", Statement of Work.

All but the abovementioned part of the Final Report was forwarded by Reference (B). Changes to such material are set forth in Enclosure (2).

This transmittal, together with the distribution effected by Reference (A) and Enclosure (3), completes the distribution requirements for the subject Final Report.

LOCKHEED MISSILES & SPACE COMPANY, INC.
SPACE SYSTEMS DIVISION

R. Noel Mendoza

R. Noel Mendoza
Chief Contract Administrator
Manned Space Programs

RNM:Jo

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LMSC-A995931
15 Nov 1971
ACS-201

ERRATA
FOR
ALTERNATE CONCEPTS STUDY EXTENSION

FINAL REPORT

Volume II
Concept Analysis and Definition,
Part 4, Avionics

I

ERRATA FOR ALTERNATE CONCEPTS STUDY EXTENSION FINAL REPORT

1. Pages 1-5 and 1-6, remove and substitute T-punched pages. Mark I Avionics Key Characteristics becomes Fig. 1-3
2. Page 1-8, third paragraph, first line, change to read "Software requirements were determined for development test, ..."
3. Page 1-11, Table 1-1, delete "(OFICM)" and substitute "(COFIRM)"
4. Page 2.1-6, line 11, delete "Mark I" and substitute "Mark II"
5. Page 2.2-3, paragraph 2.2.1, second line, delete "load" and substitute "level"
6. Page 2.2-7, second paragraph, lines 9 and 12, delete "manual" and substitute "mechanical"
7. Page 2.2-11, fourth paragraph, delete "load relief" and substitute "load limiting"
8. Pages 2.2-11 and 2.2-12, remove and insert revised T-punched page
9. Page 2.2-13, change first two lines to read "Speed brake control is "fly-by-wire" to the servoactuators, with mechanical linkage from actuators to brake surfaces. Each actuator has series electrohydraulic valves,"
10. Page 2.2-15, second paragraph, lines 4 and 5, change to read "launch the orbiter provides guidance to the composite orbiter-booster vehicle and, subsequent to reentry below Mach 2 speeds, the spacecraft GN&C system provides"
11. Page 2.2-23, Table 2.2-2, Candidate CDC Alpha-1, Column 10, delete "XDS Signal 7" and substitute "XDS Sigma 7"; Column 1, delete "Candidate IBM 4X CP" and substitute "IBM 4π CP"; Candidate Univac 1832, Column 10, delete "Solo Leader" and substitute "Solo Loader"
12. Page 2.2-25, Table 2.2-3, Parameter Star Magnitude last 3 columns, delete dashes and add +1.0 (Bendix), +2.0 (Litton), and +1.8 (Kollsman)

13. Pages 2.2-27 and 2.2-28, delete and insert revised T-punched pages

14. Page 2.2-39, Fig. 2.2-18, first block, add "S-3A above "Audio Panel"

15. Page 2.2-57, second paragraph, line 3, delete "flight dock" and substitute "flight deck"

16. Page 2.2-97, fifth paragraph, line 4, change to read
"data-gathering subsystem from the LMSC Space Experimental Scientific Program (SESP); ....."

17. Page 2.2-98, Table 2.2-34, delete "P-50" and substitute "SESP" in 5 places

18. Page 2.2-101, Fig. 2.2-51, Analog Mux S/S, delete "P-50" and substitute "SESP"

19. Page 2.2-129, paragraph 2.2.3, second bullet listing, change to read
   - Flexible (CRT/KEYSET) man interface and DMS highspeed, digital access to all LRUs without resort to a complex data bus offers straightforward economical methods of addressing shuttle-unique and combined aircraft-spacecraft instrumentation/checkout/control problems."

20. Page C-3, Fig. C-2, after "Signal Acquisition Remotes" add "(SAR)"

21. Page C-6, Table C-2, delete "[your list]" after MADAR

22. Pages D-11, D-12, remove and insert revised T-punched pages

VOLUME I, EXECUTIVE SUMMARY

None.

VOLUME II, PART 1, O4OA SYSTEM

1. Page 3-12, subsection 3.4.2, line 3: Change "(GE F100/F12A3)" to read "(GE F101/F12A3)"
2. Page 3-13, caption for Fig. 3-9: Change "F101/AB" to read "GE F101/F12A3"
3. Page 3-13, caption for Fig. 3-9: Change "F101/A3" to read "GE F101/F12A3"
4. Page 3-14, caption for Fig. 3-10: Change "F101/A3" to read "GE F101/F12A3"
5. Page 4-26, subsection 4.1.4, paragraph 3: Change to read "A structural modification kit is added to the orbiter payload bay to provide for the installation of jet engines for the orbital mission. This kit, which minimizes the scar weight, includes structural support for the pylons, doors in the payload bay upper shell structure to permit pod deployment, and the deployment mechanism which is shown in Fig. 4-16. The engine bay doors are designed to return to the closed position with the pods deployed. For ferry operation, a fixed dual engine pod may be substituted.
6. Page 4-27, caption for Fig. 4-14a: Change to read "Orbital Mission Installations"
7. Page 4-28, caption for Fig. 4-15: Change to read "Orbital Mission Engine Pod Arrangement"
8. Page 4-28, caption for Fig. 4-16: Delete "for Ferry Kit"

VOLUME II, PART 2, ONE-AND-ONE-HALF STAGE SYSTEM

1. Pages 2-1, 2-2: Remove and substitute attached T-punched pages. A new configuration replaces old Fig. 2-1
2. Page 3-5, line 1: Change "was sized" to read "was not sized"

VOLUME II, PART 3, SRM BOOSTERS

1. Page 2-19, Fig. 2-9: Delete dashed line in upper lefthand graph
2. Page 3-6, Fig. 3-3: Change legends on all vertical axes to read "Recurring Costs (10^6 \$)"
3. Page 3-10, Table 3-2, next to last line: Change "3 Percent" to "4 Percent"

VOLUME III, COST ANALYSIS

1. Page 1-2, Fig. 1-1: Deletes callout "11B" and arrow at extreme upper right of graph

15 Nov 1971

LMSC-A995931
ACS-201

Final Report

ALTERNATE CONCEPTS STUDY EXTENSION

Volume II

PART 4: AVIONICS

Contract NAS 8-26362

Prepared for George C. Marshall Space Flight Center By
Manned Space Programs, Space Systems Division

LOCKHEED MISSILES & SPACE COMPANY
SUNNYVALE CALIFORNIA

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FOREWORD

This is the final report of a four-month extension of the Phase A Study of Alternate Space Shuttle Concepts (NAS 8-26362) by the Lockheed Missiles & Space Company (LMSC) for the National Aeronautics and Space Administration, George C. Marshall Space Flight Center (MSFC). This study extension, which began on 1 July 1971, was to study two-and-one-half stage, stage-and-one-half, and SRM interim booster systems for the purpose of establishing feasibility, performance, costs, and schedules for these system concepts.

The final report consists of three volumes (6 books) as follows:

Volume I - Executive Summary
Volume II - Concept Analysis and Definition
- Part 1 - O4OA System
- Part 2 - One-and-One-Half Stage System
- Part 3 - SRM Booster
- Part 4 - Avionics

Volume III - Cost Analysis

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ILLUSTRATIONS

ILLUSTRATIONS (CONT'D)

TABLES

Section 1

SUMMARY

Section 1
SUMMARY

INTRODUCTION

The objectives of the Alternate Avionics System Study were to evaluate Avionics System alternatives and conceive an overall vehicle/ground system that (1) significantly reduces total program cost and peak annual funding and (2) reduces the technological risk. Three alternate systems were evaluated, and a baseline Avionics System which meets the study objectives was selected. The major portion of this study report is devoted to this recommended baseline system; the alternates, which were evaluated but not selected, are summarized in the appendixes.

The study scope included the avionics subsystems onboard the 040A Orbiter; the avionics subsystems onboard the recoverable pressure-fed LOX/propane ballistic booster; the electronics ground support equipment for direct support of flight avionics; and avionics-related ground support for maintenance, launch, and mission operations. A 40 to 50 percent cost-growth allowance over the Mark I Orbiter avionics nonrecurring costs was used as a constraint in defining the Mark II Orbiter Avionics System.

This study approach emphasized a requirement for a functional analysis to establish the basis for a minimum, safe, flyable system. Alternate systems were defined for the Mark I Orbiter after an initial estimate of major cost and risk factors. An extensive compilation of developed available equipment was prepared for use in mechanizing the alternate systems. Each alternate was defined in detail, and on the basis of tradeoffs of overall costs, risks, and system capability, the baseline system was selected. Also, the projected Mark II Orbiter Avionics System configuration was considered in the choice of the Mark I Avionics System baseline in order to simplify the transition from Mark I to Mark II.

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Development costs, technological risks, and schedule risks were minimized by extensive selection of equipment already developed and/or demonstrated. Equipment modifications were identified and costed as were qualification tests for those equipment not environmentally protected and not previously qualified for the operational environment. In a few cases, required equipment are presently being developed for other applications in the 1972-1976 time period. Some equipment, such as thrust vector control drive electronics or Attitude Control Propulsion System (ACPS) drive electronics, must be designed and developed for the specific application but will use proven techniques and hardware elements.

RECOMMENDED BASELINE SYSTEM

The recommended Mark I Orbiter Avionics System baseline concept is illustrated in Figs. 1-1 and 1-2. All equipment required for safe return is dedicated and hardwired – including displays and controls for crew information and control. The data management subsystem (DMS) plus the programmable displays and integrated control panels are overlaid on the basic "safe system" to provide access to all subsystems and to permit crew access to all information available to the data management computer. Onboard checkout, fault-isolation, inflight performance monitoring, and redundancy management are but some of the onboard capabilities provided by the DMS which reduce dependence upon ground support. Manual override of the DMS and programmable displays is available to the crew in addition to hardwired dedicated equipment controls which may operate independently of the DMS subsystem. The data management computer, interfaces, programmable displays, and integrated control panels are S-3A aircraft program-developed equipment that are presently being demonstrated as an integrated system.

This baseline Avionics System configuration provides flexibility for growth to an expanded Mark II capability without a major change of system configuration. The functions assigned to the DMS computer may be increased to include subsystem operational computations with either backup or primary responsibility as desired. Onboard mission planning, targeting, and onboard mission operations support will increase vehicle autonomy and reduce operations cost, thereby reducing total program cost.

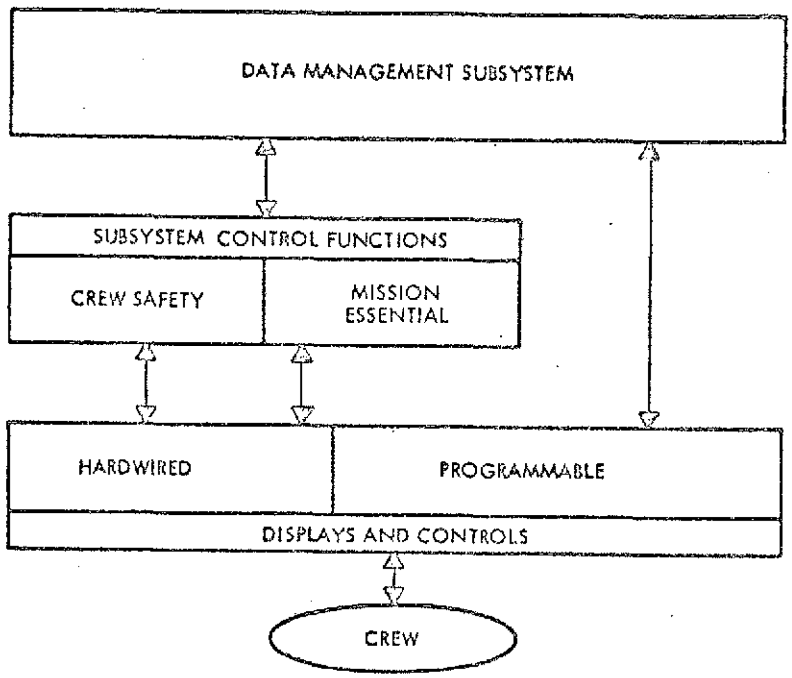

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Fig. 1-1 Mark I Orbiter Avionics System Concept

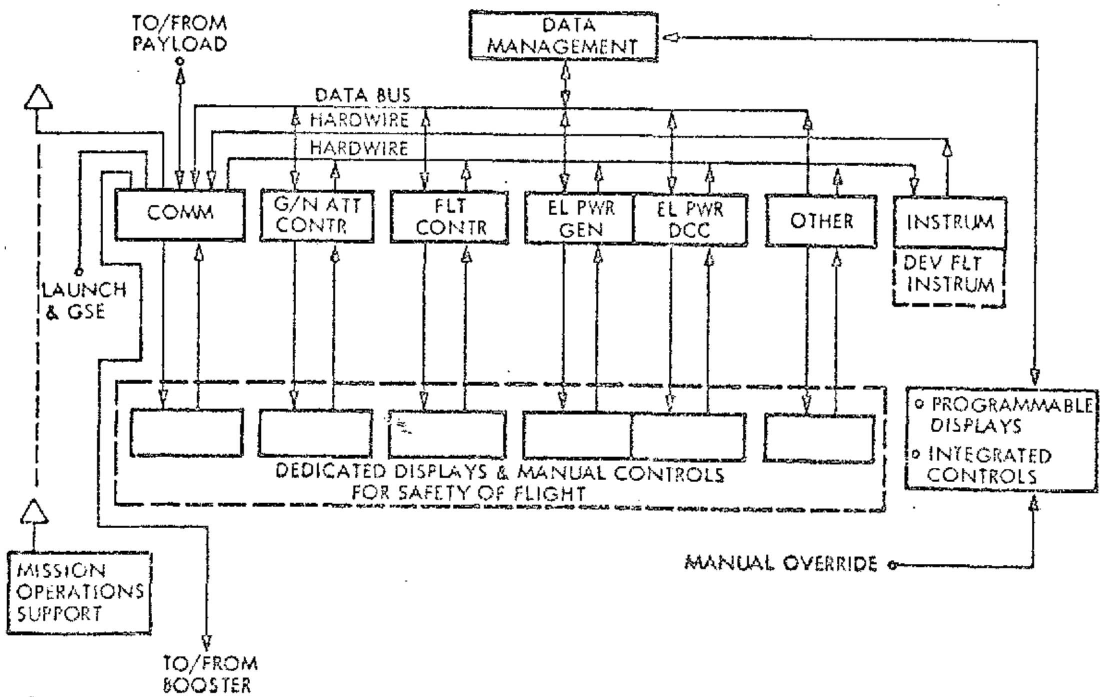

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Fig. 1-2 Mark I Orbiter Avionics Baseline Recommendation

Some of the key characteristics of the baseline Avionics System are presented in Fig. 1-3. The Guidance, Navigation, and Control (GN&C) subsystem provides automatic and manual operating modes with manual override of automatic functions. The dedicated guidance/navigation computer is digital and is identical to one-half of the dual computer (S-3A Univac 1832) used in the DMS. Spacecraft orientation and translation control is affected via this computer. The flight control system employs L-1011 aircraft-type dedicated analog computers for primary flight control and for autopilot/autoland capability. Stability augmentation is provided, and fly-by-wire was selected in preference to mechanical control cables.

The Controls and Displays (C&D) subsystem is configured to reduce crew workload through display programming by providing data specific to mission phase only, eliminating random caution and warning (C&W) annunciators, and having less random instrument scan. Task allocations for horizontal flight test indicate the need for a third crewman to monitor subsystems, aid in checkout, and provide expanded engineering flight data to the pilot and copilot. For vertical test flights and orbital operations, a two-man crew can perform all defined manual tasks using the integrated control and

GUIDANCE, NAVIGATION, ATTITUDE CONTROL
- DEDICATED COMPUTER
- MANUAL OVERRIDE OF AUTOMATIC
- AUTO/MANUAL TVC AND RCS CONTROL
- RCS ENTRY CONTROL TO MACH 2
- PHASED ADDITION OF SENSOR SUIT

FLIGHT CONTROL SYSTEM
- DEDICATED ANALOG COMPUTERS
- FLY-BY-WIRE
- STABILITY AUGMENTATION SYSTEM
- AUTOPILOT/AUTOLAND
- QUAD-REDUNDANT AEROSURFACE DRIVE/CONTROL
- FLIGHT DATA DISPLAYS HARDWIRE

COMMUNICATIONS & TRACKING
- APOLLO BLOCK II FOR SPACE
- S-3A TYPE FOR ATC VOICE
- ALL NAVIG AIDS OFF-THE-SHELF

DISPLAYS & CONTROLS
- DUAL PILOT/COPILOT STATIONS
- SAFETY OF FLIGHT: DEDICATED, HARDWIRED
- PROGRAMMED DISPLAYS
  - REDUCED CREW WORKLOAD
  - INCREASED CREW CAPABILITY
  - REDUCED PANEL AREA REQUIREMENTS

ELECTRICAL POWER
- PRIMARY POWER
  - MINI-TECHNOL 2000 HR FUEL CELL DEVELOPMENT
  - AC GENERATOR SYSTEM
- SAFETY-OF-FLIGHT HARDWIRE INSTRUM AND CONTROL
- SOME DMS CONTROL ACCESS

INSTRUMENTATION
- FLIGHT CRITICAL: HARDWIRED
- NON CRITICAL: DMS-CONTROLLED (MULTIPLEXED)
- DEVELOPMENT FLIGHT INSTRUMENTATION OVERLAID

DATA MANAGEMENT
- NON-FLIGHT-CRITICAL
- MANUAL OVERRIDE
- ACCESS TO ALL SUBSYSTEMS
- ON-BOARD C/O; R/M AIDS
- FLEXIBILITY FOR MARK II GROWTH
- SOFTWARE TEST BED FOR MARK II

D0627B

Fig. 1-3 Baseline Mark I Avionics Key Characteristics display configuration. As in horizontal flight test, the vehicle is flyable from either seat and critical controls and displays are duplicated at pilot/copilot main instrument panels. Backup minimum flight-instruction displays are not CRT dependent.

Electrical power generation, not strictly part of "avionics" but considered within the scope of this study, presents the only area in which a new development is required. The electrical power subsystem (EPS) configuration for the orbital mission includes (1) three H_2-O_2 fuel cells, rated at 8 kW continuous power, which provide FO/FS 28 Vdc power for distribution to users; and (2) a centralized static 3-phase inverter system, which provides 115 Vac power. Three 200/115 Vac generators, driven by chemical-dynamic auxiliary power units (APUs), provide peak power for ascent phase and atmospheric flight operation of the propulsion systems. These generators also power transformer-rectifier (T-R) units for horizontal development flight tests. Installation of the fuel cells and their cryogenic storage system is phased for the first orbital development flight test. The fuel cell is a new low-cost development providing a 2000-hr lifetime for Mark I that is increased to 5000 hr for Mark II. The cryogenic tankage for AAP is used with development completion for Mark I and minimum change for Mark II. Power is distributed over a two bus system. It is hardwired to the crew stations for manual control and override of automatic control of the EPS. Automatic control is provided by sensors, circuit breakers, equipment controllers, and the DMS which interfaces with the EPS through subsystem interface units (SIUs).

The Data Management Subsystem performs the major functions of checkout, fault isolation, and redundancy management as follows. During prelaunch activities, orbiter avionics is automatically checked out and fault isolated to the major replaceable unit by the data management computer. Orbiter nonavionics and all booster systems are checked out and fault isolated by automatic GSE and manual inspection. Redundancy management is manually initiated, except for time-critical items and unit internal redundancy, which are automatic. Orbiter in-flight checkout and fault isolation for safety of flight items is by dedicated built-in test with caution and warning annunciators and operational displays. Orbiter avionics checkout and fault isolation is automatic; nonavionic checkout and fault isolation is accomplished by a combination of operational displays with semiautomatic crew instructions. The checkout fault isolation and redundancy management functions of the DMS and vehicle systems are periodically validated via MCC until programmatic confidence is established to support complete autonomy. Redundancy management is the same as during prelaunch. For accomplishing between flight maintenance on vehicle systems, use is made of checkout, fault isolation, and redundancy management prelaunch capability, supplemented by discrete GSE units for periodic test and calibration plus special software routines in the data management computer.

The recommended Mark I Orbiter Avionics System Baseline equipment block diagram (See figure at end of this section) illustrates the major functional flow among subsystems and equipment and identifies the program source for each equipment. The extensive use of developed equipment from aircraft and spacecraft programs reduces technological risk and reduces program costs.

Software requirements were determined from development test, for orbiter vehicle functions, and for direct support functions. The development test software includes programs for integrated test, equipment simulation, and test data reduction plus utility programs, and requires 270K words. The orbiter vehicle software includes 34K words for guidance navigation and control plus 232K words for data management. The latter consists of 127K words for system test programs (OBCOFI) and 105K words for operational programs including common control and services (33K), system management aids (27K), and subsystem operations support (45K). Direct support software includes launch checkout, preflight data insertion, MCC system library, simulation, system generation, and data reduction for a total of 254K words. The baseline system software requirement is thus 790K words.

Since the baseline equipment configuration employs the S-3A computer and data management techniques, the software also would be based on S-3A developed software and would use or would modify, to the extent appropriate, the already existing S-3A programs. Of course, additional software development will be required.

ALTERNATE AVIONICS SYSTEMS

Alternate avionics systems configured and evaluated but not selected are illustrated in Figs. 1-4 and 1-5. System Alternate A provides separate aircraft and spacecraft subsystems which are dedicated and hardwired. Controls and displays for aircraft and spacecraft functions are provided at completely separate stations. The lack of onboard checkout and fault isolation capability means that extensive mission operations support from ground facilities is required. System Alternate B combines aircraft and spacecraft dedicated and hardwired subsystems into one set, eliminating overlapping functional equipment. Manual controls and displays are combined at pilot and copilot stations. An onboard checkout and fault isolation system, incorporating a data bus for equipment test access, provides status, caution, and warning information to the crew. Dependence on mission operations support from the ground is reduced. System Alternates A and B, when compared with the recommended baseline, are lower in cost for onboard avionics, higher in total cost (flight avionics plus ground support), have considerably less capability, and do not have the flexibility for easy growth to Mark II capability.

MARK II AVIONICS

The baseline Mark I Orbiter Avionics System configuration can be applied directly to the Mark II Orbiter requirements, thereby minimizing changes and associated costs, and providing flexibility of choice to the program insofar as the time of affecting a partial or complete transition. Increased onboard capability, improved performance, and improved equipment characterize the Mark II. Projected avionics changes to achieve the Mark II system are listed in Table 1-1. Reduced turnaround time of two weeks requires more extensive onboard checkout and fault isolation capability. The addition of area navigation with a real-time programmable CRT display will permit observation of orbiter position relative to a computed track and will reduce crew workload for repetitive types of navigational tasks. The horizon sensor and orbital altimeter may be deleted if the Precision Ranging System (onboard Mark I for rendezvous) is applied to orbital navigation.

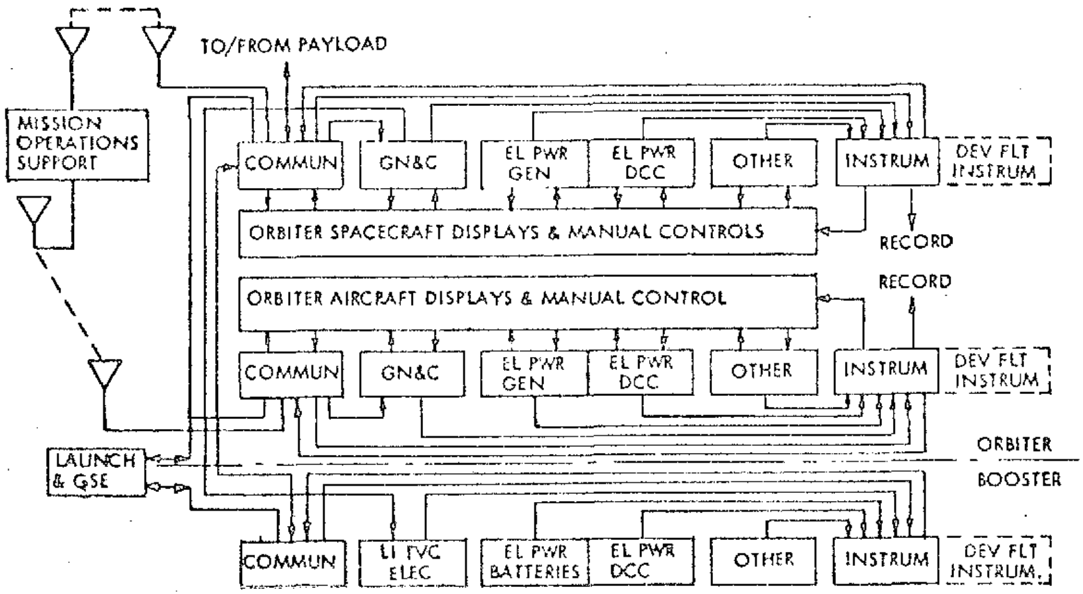

Fig. 1-4 Alternate A Avionics System

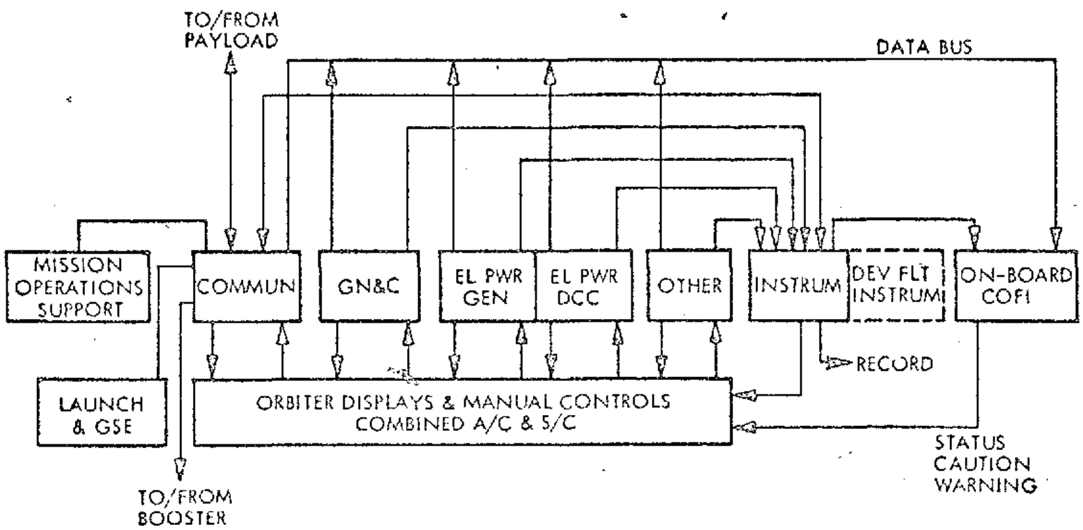

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Fig. 1-5 Alternate B Avionics System

Table 1-1

MARK II AVIONICS CHANGES

Item	Changes
Subsystems	
Guidance, Navigation, and Control (GN&C)	Improve performance and quality of equipment; accuracy improvements reduce ACPS, ΔV propellant use and reduces reentry dispersions.
Control and Display	Add area navigation/autoland CRT display.
Communications & Tracking	Improve performance and quality of equipment.
Instrumentation	Improve performance and quality of equipment.
Electrical Power	Provide 5000 hour life fuel cell. Improve performance and quality of equipment.
Data Management, Checkout, Fault Isolation & Redundancy Management (OFICM)	Increase onboard COFIRM for nearly complete autonomy for both avionics and non-avionics.
Software	Performance of functional operations through software instead of hardware could significantly increase mission flexibility and decrease change reaction time. Greater reliance on software in flight controls and COFIRM will require advanced management techniques.
Orbiter	
Mark I Equipment Deletions	Horizon sensor and orbit altimeter.
Tracking Satellite	Used to augment navigation.
On-Board Navigation, Data Management and COFIRM Improvements	Minimize dependence on ground control and remote stations.
Safety/Reliability	Improved quality of equipment will increase probability of mission success and enhance safety. More autonomous fault isolation and redundancy mangement will reduce crew workload and decrease corrective action time.

Other changes to the Mark II avionics subsystems are primarily for improvement of performance and quality and the ability to withstand the space environments. These improvements will reduce propellant loading requirements by providing greater navigational accuracies and will reduce operational costs since fewer failures requiring removal and replacement of equipment are expected.

BOOSTER AVIONICS

The interim recoverable pressure-fed ballistic LOX/propane booster avionics requirements were estimated on the basis of previous experience, since only sketchy information was available to describe the booster or its subsystems. Those functions that could be performed by the orbiter without overly complicating the interface with the booster were not mechanized in the booster to avoid duplication of equipment and development costs. Thus, the guidance and control computations for composite vehicle ascent are performed in the orbiter. An equipment block diagram for the booster avionics is presented in Fig. 1-6. For this booster, which is unmanned and is not guided or actively controlled after staging, orbiter/booster avionics commonality is virtually nonexistent.

COST

The baseline Mark I Orbiter Avionics System total cost for a phased program was determined to be $323.3 million with a maximum peak annual cost of $78 million. The estimated total cost for Mark II is $202.6 million, for a total program cost of $525.9 million.

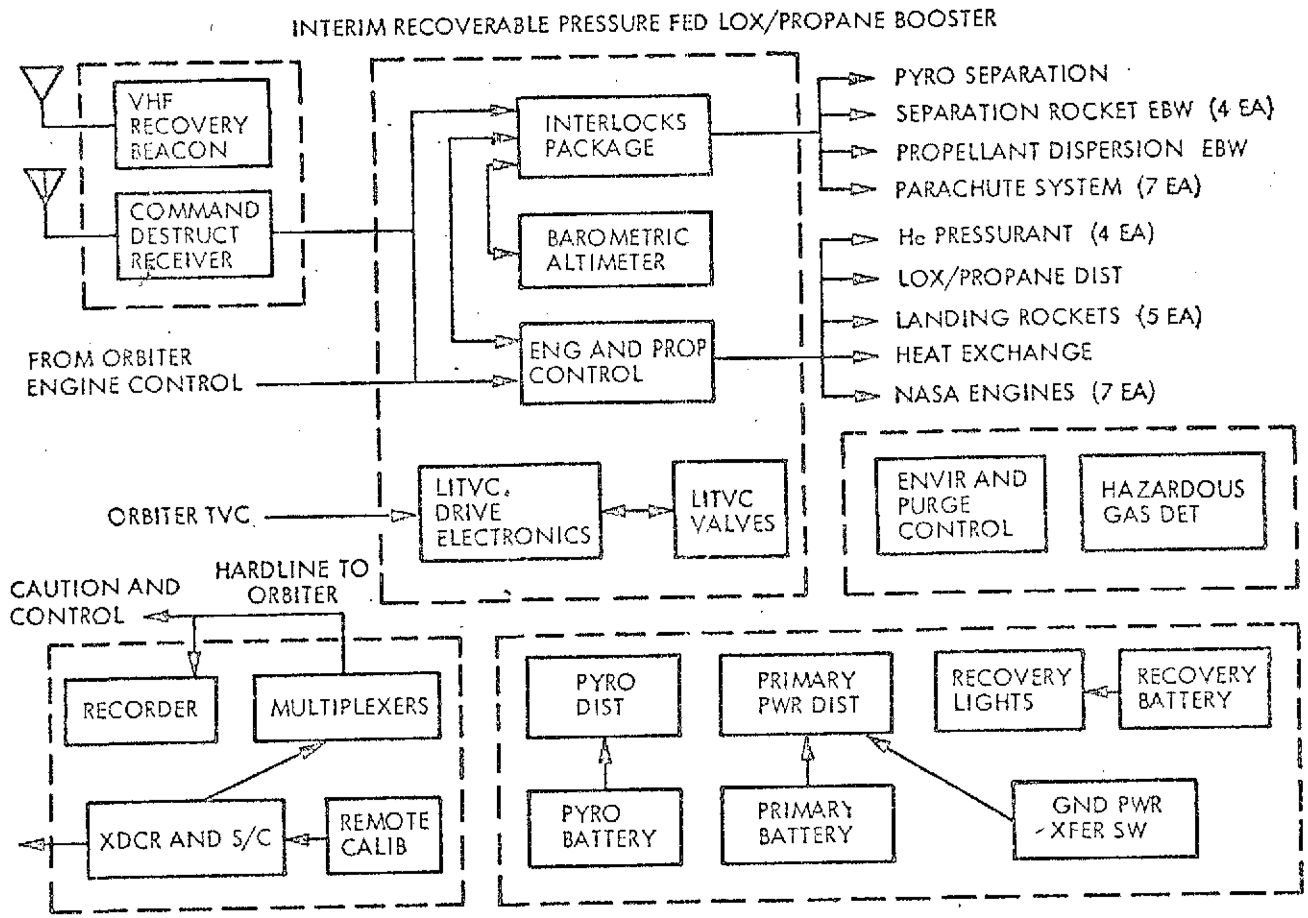

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Fig. 1-6 Booster Avionics Selected Point Design

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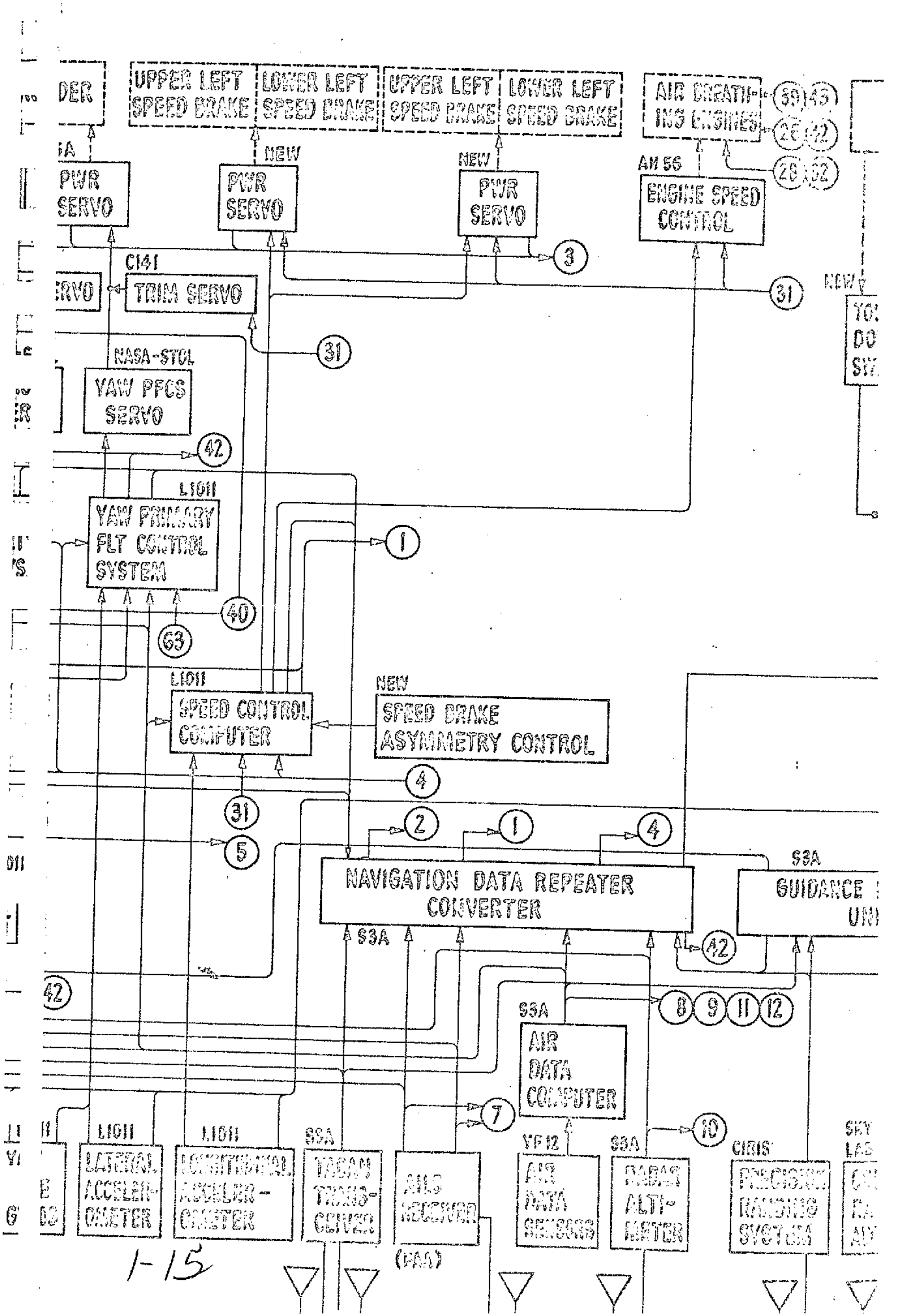
CONCLUSIONS

The recommended baseline Avionics System significantly reduces total program cost and peak annual funding, and reduces technological risk. These principal study objectives were achieved by extensive application of developed, proven equipment from aircraft and spacecraft programs. Additional study is recommended to define in more detail all subsystems of the baseline system. In particular, the application of S-3A Avionics System techniques, equipment, and software plus modifications and redesign of other equipment for compatibility with the S-3A checkout, fault isolation, and inflight performance monitoring should be investigated. Interfaces among subsystems and equipment should be more precisely defined. Safety/reliability studies should be performed to verify adequacy of selected redundancy levels. A thorough packaging and installation study must be performed to determine impact on the crew station, on the environmental control system, and on access for maintenance. The preparation of an avionics system management plan for design, development, test, and integration is of primary importance.


UPPER LEFT SPEED BRAKE
LOWER LEFT SPEED BRAKE
UPPER LEFT SPEED BRAKE
LOWER LEFT SPEED BRAKE
AIR BREATHING ENGINES
PWR SERVO
NEW
PWR SERVO
NEW
PWR SERVO
AM 56
ENGINE SPEED CONTROL
C141
TRIM SERVO
NASA-STOL
YAW PFCS SERVO
L1011
YAW PRIMARY FLT CONTROL SYSTEM
L1011
SPEED CONTROL COMPUTER
NEW
SPEED BRAKE ASYMMETRY CONTROL
NAVIGATION DATA REPEATER CONVERTER
S3A
GUIDANCE
S3A
AIR DATA COMPUTER
L1011
LATERAL ACCELEROMETER
L1011
LONGITUDINAL ACCELEROMETER
S3A
TACAN TRANSCEIVER
AILS RECEIVER
(FAA)
YF12
AIR DATA SENSORS
S3A
RADAR ALTIMETER
CIRIS
PRECISION RANGING SYSTEM

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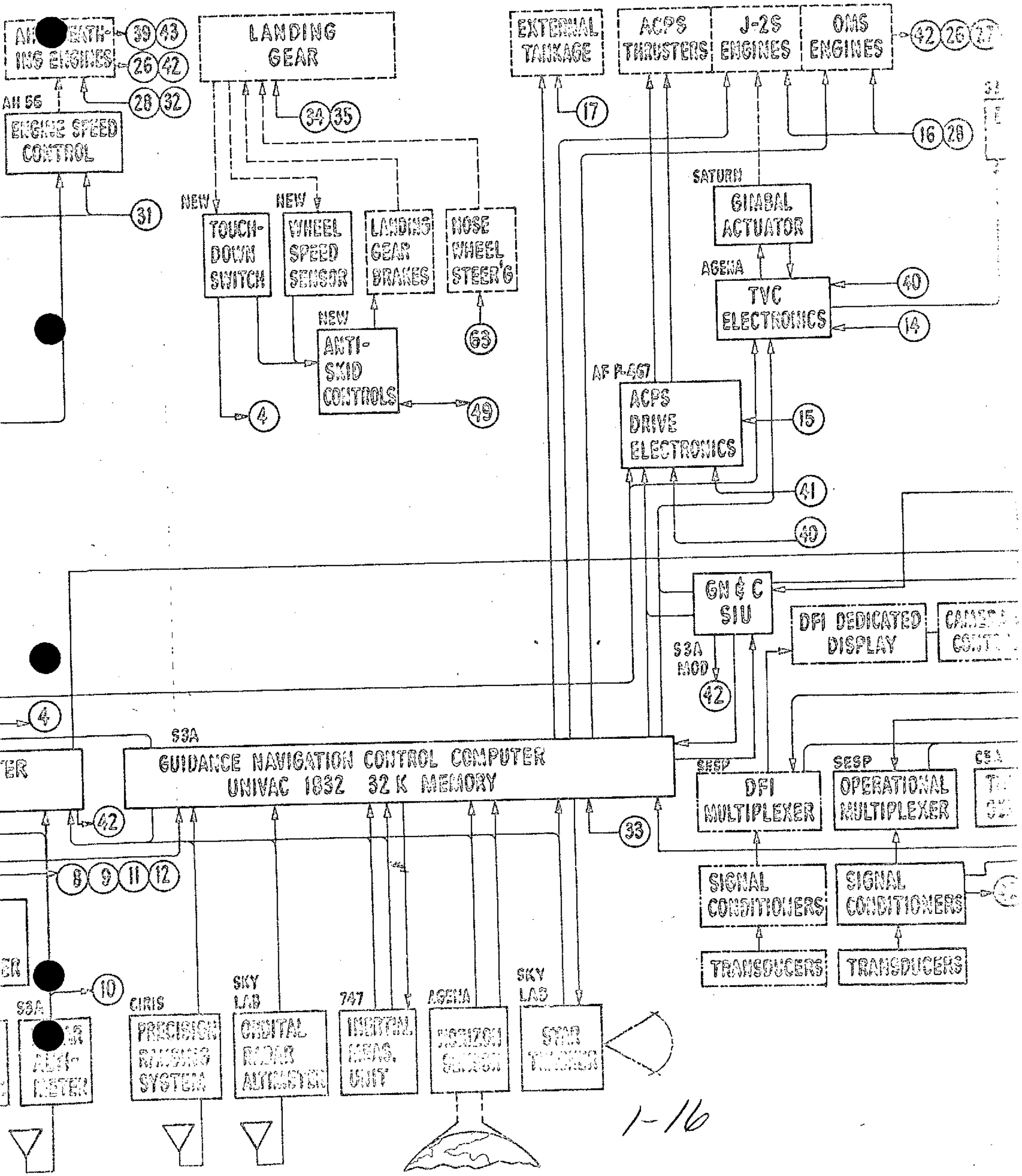

LANDING GEAR
EXTERNAL TANKAGE
ACPS THRUSTERS
J-2S ENGINES
OMS ENGINES
AH 56
ENGINE SPEED CONTROL
NEW
TOUCH-DOWN SWITCH
WHEEL SPEED SENSOR
LANDING GEAR BRAKES
NOSE WHEEL STEER'G
ANTI-SKID CONTROLS
SATURN
GIMBAL ACTUATOR
AGENA
TVC ELECTRONICS
AF P-467
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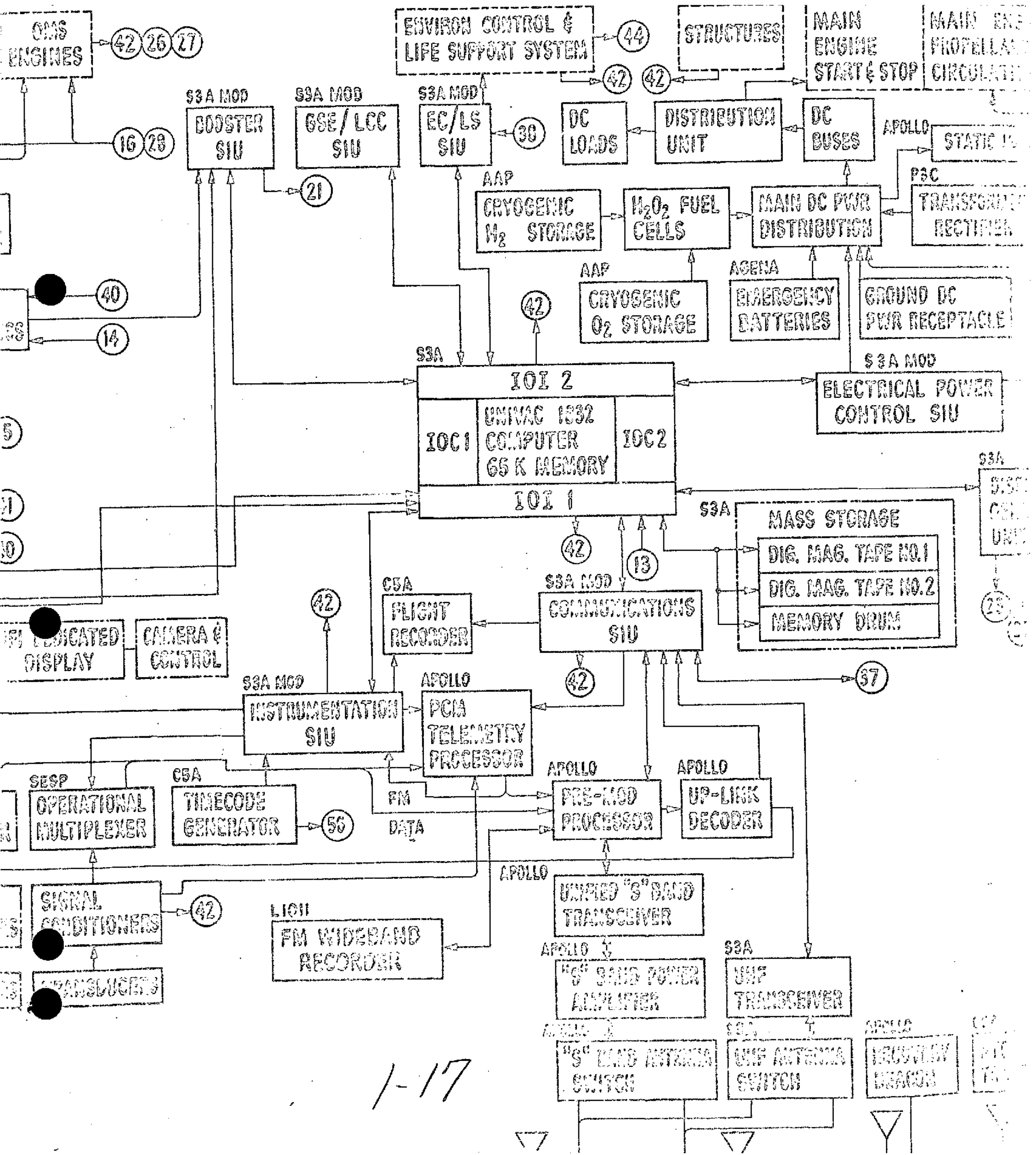

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BOOSTER SIU
S3A MOD
GSE/LCC SIU
S3A MOD
EC/LS SIU
DC LOADS
DISTRIBUTION UNIT
DC BUSES
APOLLO
AAP
CRYOGENIC H_2 STORAGE
H_2O_2 FUEL CELLS
MAIN DC PWR DISTRIBUTION
PSC
AAP
CRYOGENIC O_2 STORAGE
AGENA
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GROUND DC PWR RECEPTACLE
S3A MOD
ELECTRICAL POWER CONTROL SIU
S3A
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IOC 1
UNIVAC 1832 COMPUTER 65 K MEMORY
IOC 2
IOI 1
S3A
MASS STORAGE
DIG. MAG. TAPE NO.1
DIG. MAG. TAPE NO.2
MEMORY DRUM
C5A
FLIGHT RECORDER
S3A MOD
COMMUNICATIONS SIU
DEDICATED DISPLAY
CAMERA & CONTROL
S3A MOD
INSTRUMENTATION SIU
APOLLO
PCM TELEMETRY PROCESSOR
APOLLO
PRE-MOD PROCESSOR
APOLLO
UP-LINK DECODER
SESP
OPERATIONAL MULTIPLEXER
C5A
TIMECODE GENERATOR
FM
DATA
APOLLO
UNIFIED "S" BAND TRANSCEIVER
SIGNAL CONDITIONERS
TRANSDUCERS
LIOII
FM WIDEBAND RECORDER
APOLLO
"S" BAND POWER AMPLIFIER
S3A
UHF TRANSCEIVER
"S" BAND ANTENNA SWITCH
UHF ANTENNA SWITCH
APOLLO
RECOVERY BEACON
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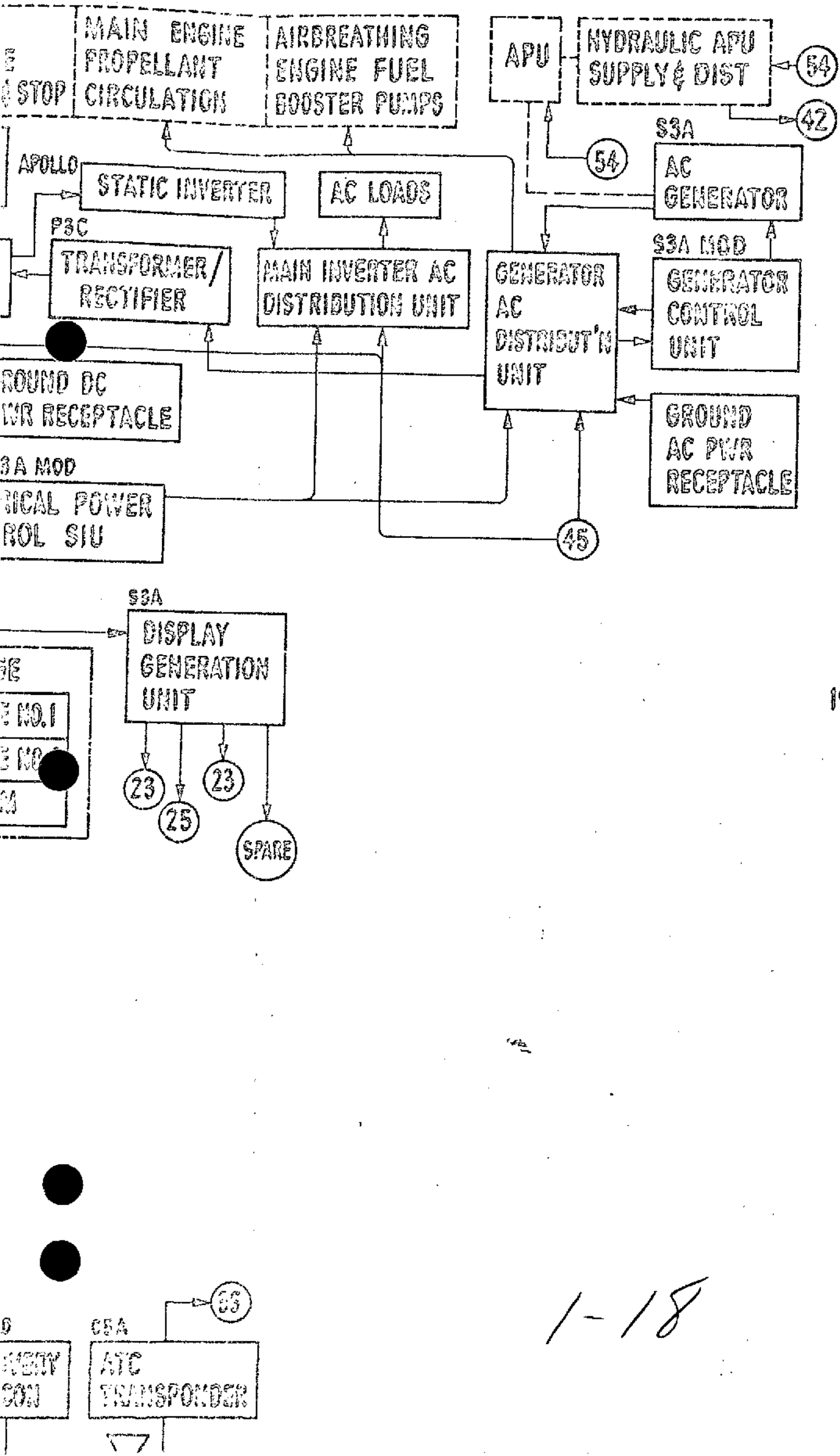
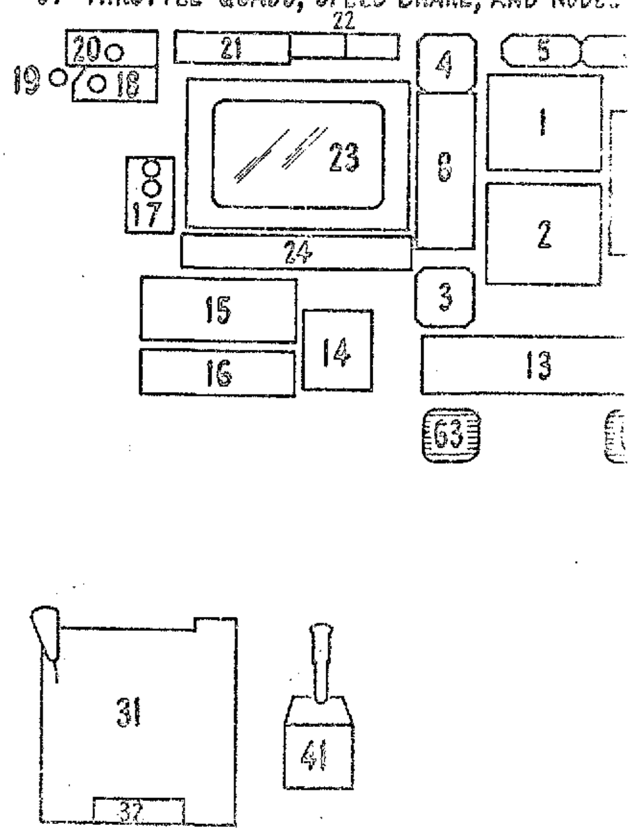


1 FLIGHT ATTITUDE INDICATOR - S3A
2 HORIZONTAL SITUATION INDICATOR - S3A
3 AERO SURFACE INDICATOR - NEW
4 AFCS MODES
5 AFCS WARNING
6 INSTR WARNING
7 AUTOPILOT / LAND
(4–7: L1011)
8 METER - AIRSPEED / MACH / α - C5A
9 METER - ALTITUDE / VERTICAL SPEED - C5A
10 ALTIMETER - S3A
11 TRUE AIRSPEED INDICATOR - S3A
12 ALTITUDE INDICATOR - S3A
13 MULTI-PURPOSE KEYBOARD - S3A
14 ENGINE GIMBAL OVERRIDE - APOLLO
15 ACPS CONTROL OVERRIDE - NEW
16 MAIN / OMS OVERRIDE - NEW
17 TANK JETTISON OVERRIDE - NEW
18 ABORT - NEW
19 INSTR BRIGHTNESS CONTROL - NEW
20 CAUTION AND WARNING TEST PANEL - NEW
21 BOOSTER STATUS PANEL - NEW
22 MASTER SYSTEM CAUTION AND WARNING - NE
23 MULTI-FUNCTION CRT (FLIGHT MGMT) - S3
24 FLIGHT MODE INDICATOR - NEW
25 MULTI-FUNCTION CRT (SUBSYSTEMS) - S3
26 ENGINE / PROPULSION DISPLAYS
27 ENGINE / PROPULSION DISPLAYS
28 MODE SELECT (MAIN / OMS / ABES
29 MODE SELECT (ACPS / APU)
(26–29: NEW)
30 AREA NAV - GROWTH (MARK II) - L1011
31 THROTTLE QUADS, SPEED BRAKE, AND RUDD



1-18

3A

- C5A

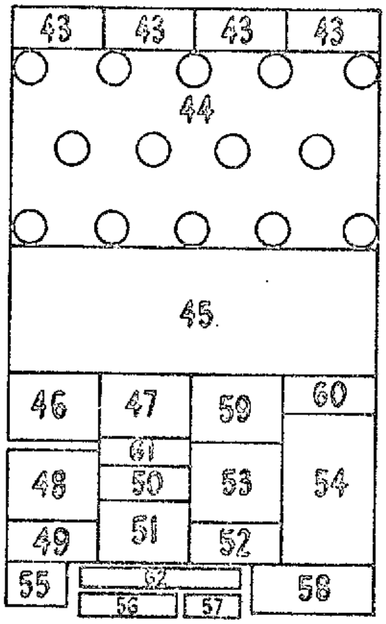
L-NEW

NING - NEW
T) - S3A

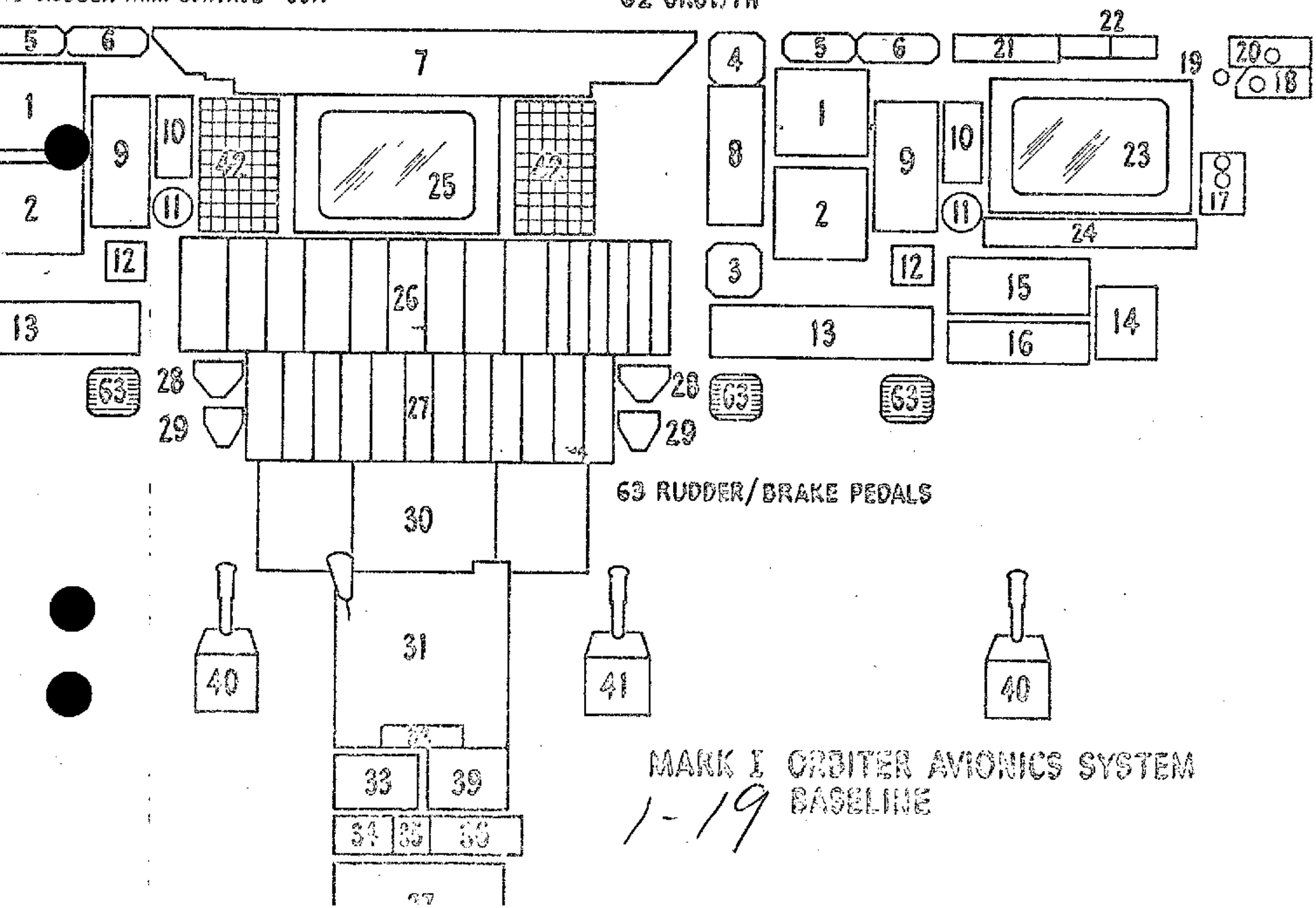
3) - S3A

NEW

L1011
RUDDER TRIM CONTROL - C5A



32 ARES CONTROL - C5A
33 G&N MODE SELECT PANEL - NEW
34 LANDING GEAR CONTROLS - C5A
35 EMER LANDING GEAR EXTENSION CONTROLS - C5A
36 ATC PANEL - C5A
37 COMMUNICATIONS PANEL - S3A
38 EC/LS PANEL - NEW
39 ENGINE START - C5A
40 ATTITUDE HAND CONTROLLER } APOLLO - MOD
41 TRANSLATION CONTROLLER
42 SUBSYSTEM C&W INDICATORS NEW
43 ENGINE FIRE CONTROL PANELS - L1011
44 EC/LS GAS SUPPLY OVERRIDE VALVES - NEW
45 ELECT. PWR GENERATION AND DIST - NEW
46 ELEVON DISABLE } NEW
47 RUDDER DISABLE
48 SAS, PITCH, ATS, AND TRIM EMER CONTROLS - L1011
49 ANTISKID CONTROLS } L1011
50 SENSOR HEAT CONTROLS
51 PFCS MON., RUDDER AND ELEVON EMER CONTROLS - L1011
52 RUDDER LIMITER } L1011
53 APU ENGINE START
54 APU ENGINE CONTROLS - NEW
55 CABIN LIGHTS - L1011
56 MISSION TIMER } APOLLO CM
57 EVENT TIMER
58 EXTERIOR LIGHTS - L1011
59 GROWTH
60 GROWTH
61 GROWTH
62 GROWTH



63 RUDDER/BRAKE PEDALS

MARK I ORBITER AVIONICS SYSTEM BASELINE

1-19

Section 2
SYSTEM CHARACTERISTICS

2.1 REQUIREMENTS AND GROUNDRULES

The requirements and groundrules for design of the Model LS 200-11 stage-and-one-half system are nearly identical to those for the system using the Mark II O4OA Orbiter, except for configuration-peculiar requirements. The only exceptions to applicable requirements reported in Part 1, Subsection 2.1 (for the Mark II O4OA) are:

a. OMS propellants are to be H_2/O_2 rather than storables.

b. OMS tankage is not to be limited to 1000 ft/sec capability if the delta-body volume and tankage arrangement allows more.

c. The crossrange of 1100 nm is to apply with 40,000 lb payload and no ABES aboard. (TPS is not to be increased to handle higher wing loading with ABES on the resupply mission; rather, a less-severe trajectory with reduced crossrange is to be used.)

d. Avionics development is to follow the same sequence as is appropriate for the Mark I - Mark II progression, rather than meeting Mark II capability initially. Retrofit of Mark II type equipment in the operational vehicles is to be implemented as it becomes available.

The reason for use of the H_2/O_2 system for the OMS is that this approach is most cost-effective for stage-and-one-half. No major development is required, since the RL-10 engine is used, and the relatively high on-orbit weight of a stage-and-one-half orbiter would cause a significant penalty with use of less efficient propellants. Also, the delta-body configuration allows for sufficient volume to store the less dense H_2/O_2 propellants.

2.2 SYSTEM CONFIGURATION

Figure 2-1 depicts the launch configuration of the Model LS 200-11 vehicle. The system has been designed by modification of Model LS 200-10 reported in detail in LMSC-A989142. The driving purpose for modification has been cost reduction, and the net result of a

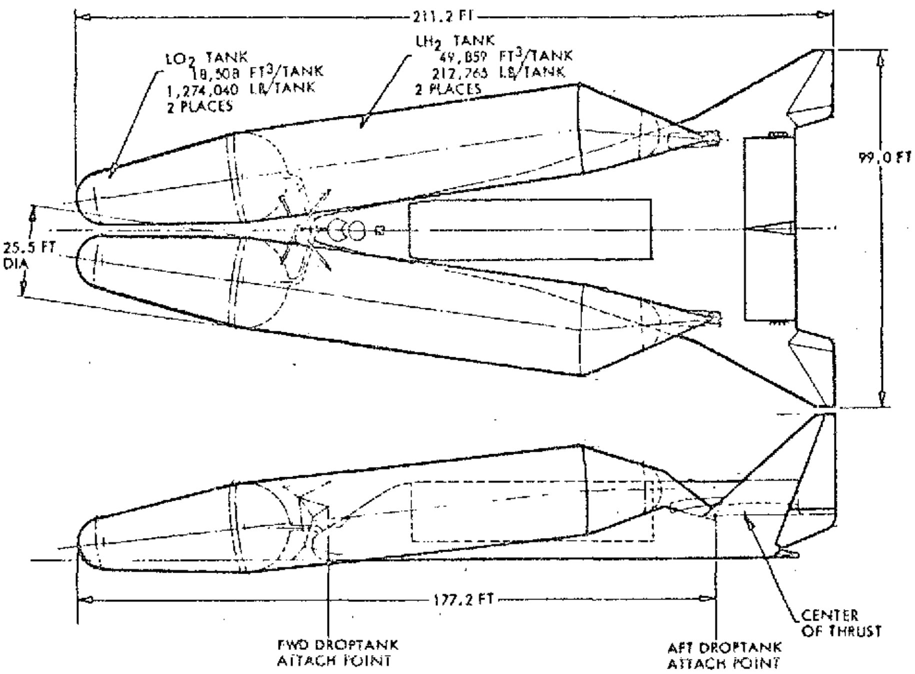


Fig. 2-1 Launch Configuration of LS 200-11 Vehicle

number of changes has been weight increases as summarized in Subsection 1.2. (See Table 1-1.)

The decrease in costs is the net result of many fairly small reductions, no one of which dominates. Of the ten changes which contribute most to cost reduction, five increase system weight, four decrease it, and the tenth causes no change in weight. The following five changes result in system weight increases:

A1. Low-cost external tank design employing weld-bonding where feasible, rather than fusion-welding, using a single bulkhead between the O_2 and H_2 tanks, and replacing the titanium thrust cone with a maraging steel design.

A2. Using an all-aluminum primary structure in the orbiter rather than titanium in some areas.

A3. Eliminating titanium panels in the thermal protection system by using insulation bonded to the primary structure throughout.

A4. Using storable propellants for the attitude control propulsion system and the auxiliary power unit rather than H_2/O_2 systems.

A5. Using acionics system designs employing available equipment.

Section 2

ANALYSIS AND DISCUSSION

2.1 INTRODUCTION

2.1.1 Objectives and Scope

The objectives of this study were to evaluate avionics system alternatives and to conceive an overall vehicle/ground system that significantly reduces total program cost, peak annual funding, and technological risk.

The study scope was established by the primary study guidelines and the contractor tasks as defined by the statement of work for Modification No. 10, Contract NAS8-26362. The scope was modified and clarified by the NASA direction given to LMSC at the midterm review on 7 October 1971. The resultant study scope is delineated below:

- The orbiter vehicle is the O4OA, and both Mark I and Mark II orbiters are to be considered. Orbiter nonavionics subsystems which interface with the avionics system are therefore as identified for the O4OA.
- The booster is the interim recoverable pressure-fed ballistic LOX/propane booster. The extent to which avionics could be defined for this vehicle was limited by the amount of vehicle and subsystem information available during the time span of this study.
- The avionics system for the orbiter and booster includes the following where required: (1) data processing; (2) displays and manual controls; (3) development, test, and onboard software for the avionics system; (4) guidance, navigation, and flight controls including actuators; (5) electronics ground support equipment; (6) operational and development flight instrumentation; (7) electrical power distribution, conditioning, and control; (8) signal and electrical power wiring and shielding; (9) communications and navigational aids; and (10) power generation.
- The supporting ground system includes avionics-related checkout/launch operations support, mission operations support, and maintenance/refurbishment support. Specifically excluded are facilities, facility equipment, and facility personnel other than those required for the above support.
- Potential orbiter/space station interfaces are outside the scope of this study.
- As defined by NASA, the orbiter/payload interface is to consider a very minimal payload health status information display capability in the orbiter, minimum orbiter avionics for deployment and retrieval of payload, and electrical power from orbiter for payload as follows: 3 kW average, 6 kW peak nominally, but 500 watts average and 800 watts peak during orbiter peak loads.

- The mission model and program schedule are as defined in NASA Technical Directive 3004. The phased program expendable booster schedule is considered to apply to the interim recoverable pressure-fed ballistic booster.
- Rendezvous and manual docking are considered to satisfy the mission on-orbit functional requirement for Mark I.
- In defining the Mark II avionics system, a 40 to 50 percent cost growth over the Mark I avionics costs was allowable. (It was assumed that nonrecurring cost was the basis for comparison.)

2.1.2 Approach

The basic study approach is depicted in the simplified flow diagram of Fig. 2.1-1. The system functional requirements were identified for a representative Mark I mission (100 nm polar) which included rendezvous and manual docking. The mission was divided into phases, and the functional requirements for each on-board subsystem were identified for each mission phase. In addition, the functions within each mission phase were categorized as to their criticality for crew safety and for mission success. Subsequently, types of equipment required to perform the various individual functions

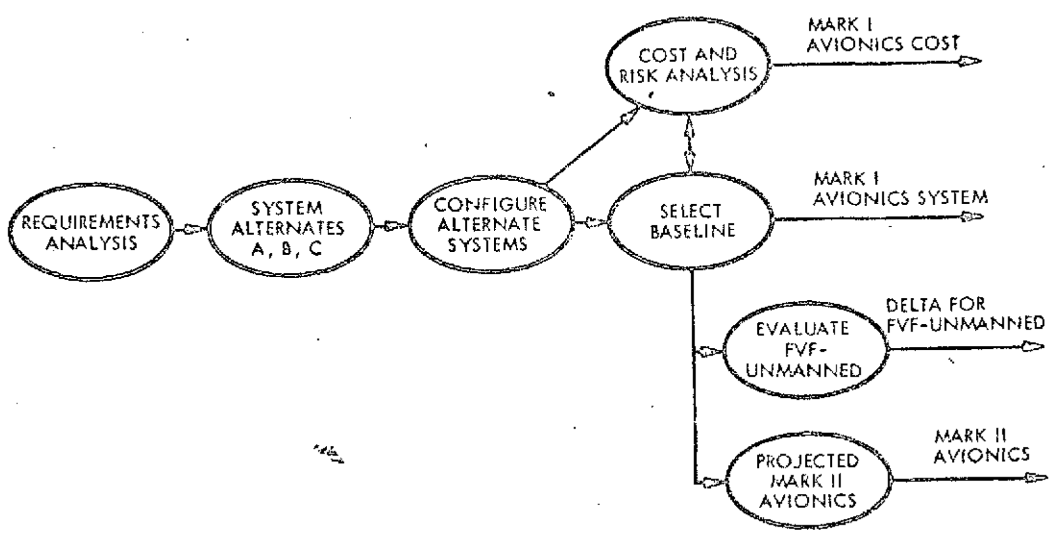


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Fig. 2.1-1 Alternate Avionics Systems Study Flow were identified, and those required to effect safe return of the crew were designated versus functions and mission phases. This analysis provided the basis for a minimum-redundancy fail-safe configuration of on-board equipment. The same technique was used for horizontal flight functions, mission phases, and equipments.

In order to select meaningful alternate system configuration candidates (in addition to the "separate aircraft/separate spacecraft functions" configuration described in the statement of work) which would focus attention on major differences of on-board functional capability versus required ground support, on major risk factors and on major cost factors, the individual subsystem functions were examined to determine which, in fact, might be performed off the vehicle. The result was that only functions associated with checkout and fault isolation, data management, and position updating were in this category. It also was recognized that on-board displays and manual controls plus crew station arrangements have significant impact on overall system effectiveness. Three alternate systems were considered adequate to span the study problem and to highlight major choices available to the system designer. The tradeoff of on-board capability versus the extent of ground support was estimated to encompass the major risk and cost factors.

The first system, Alternate A, consisted of separate sets of aircraft and spacecraft subsystems which are functionally dedicated and are hardwired. Separate aircraft and spacecraft crew stations are provided. The vehicle system is heavily dependent on mission support from the ground, since no on-board checkout and fault isolation system is provided, and no data management system processes data to assist the crew in decision making or in performing routine functions.

The second system, Alternate B, contains one set of functionally dedicated and hard-wired subsystems. Displays and manual controls for aircraft and spacecraft are combined at crew stations, and aircraft and spacecraft displays may be intermingled on any one panel. A passive monitoring on-board checkout and fault isolation system employing a data bus for test access to hardwired subsystems provides status, caution, and warning indications to the crew. Safety-of-flight items are still hardwired to panel annunciators. Dependence on ground support for launch is the same as for Alternate A, but dependence on mission support from the ground is reduced.

The third system, Alternate C, retains the feature of functionally dedicated and hardwired subsystem equipments required for safety of flight but significantly increases on-board system capability by adding a data management computer with communication links to each subsystem or major equipment and to programmable displays and integrated control panels which are also added in this configuration. The crew has access to information available to the data management computer and thus can more effectively make on-board decisions and be relieved of performing routine tasks. The crew can override the data management system, reverting to the dedicated hardwired subsystems (and displays) for safety-of-flight functions. Improved on-board checkout and fault isolation reduces the dependence on mission support from ground stations. This configuration provides flexibility for growth to eventual Mark I capability. The points in time at which increased capabilities (such as mission planning) are incorporated on-board are options available to the Shuttle Program.

In order to configure the alternate systems by utilizing developed, proven equipments, a significant effort was made to compile information on applicable equipment from both aircraft and spacecraft programs. Programs researched included S-3A, L-1011, C-5A, C-141, Jetstar, NASA/STOL, Agena, Apollo, AAP, and Gemini. The minimum redundancy level for fail-safe, identified in the requirements analysis, was increased on the basis of equipment reliability data, if available; if not available for specific equipment, a judgment was made on the basis of similarity to known equipment or on the basis of comparable complexity. This is recognized as an area requiring better definition based on more complete data and extensive reliability analyses. For each alternate system, equipment effectivity was identified for first horizontal flight, first vertical flight manned (and unmanned), and for Mark II. Excluding displays and controls, on-board checkout, and data management, the equipment for the three alternate systems was not significantly different.

Costs were compiled for the three alternate systems and an estimate of reduced ground support costs for Alternate B and Alternate C (versus Alternate A) was developed. The Mark I baseline system was then selected on the basis of cost, capability, and flexibility for growth to the Mark II configuration. Part C of this document is a brief Cost Summary for the Baseline System. A more complete costing analysis is given in Volume III, Cost Analysis.

The delta impact of performing the first vertical flight unmanned was determined on the basis that, manned or unmanned, the first vertical flight vehicle would have the same complement of Mark I avionics on board. The additional requirements unique to the unmanned vehicle were identified and additional hardware, software, horizontal flight tests, and support in the form of simulation, chase plane modification, and training of ground controllers were estimated. Cost and programmatic impacts were then identified.

The Mark II Orbiter avionics system projection was confined within a nonrecurring cost growth allowance of 40 to 50 percent over the Mark I nonrecurring costs. Within each subsystem area, desired improvements in performance and quality of equipment were identified and cost estimates were prepared. Similarly, increased software requirements corresponding primarily to an expanded role for the data management system were estimated. A determination was then made that the desired additions and improvements were within the allowable cost growth allowance. No general allocation of funds was made for redesign of equipment to reduce weight and size or for redesign to space environments.

The interim recoverable pressure-fed ballistic LOX/propane booster avionics requirements were estimated on the basis of previous experience, since only sketchy information was available to describe the booster or its subsystems. Those functions which could be performed by the orbiter without overly complicating the orbiter interface booster were not mechanized in the booster so that duplication of equipment and development costs could be avoided.

2.2 MARK I AVIONICS SYSTEM BASELINE

The overall vehicle/ground system addressed in this study consists of orbiter avionics, booster avionics, and avionics-related ground support for maintenance, for launch, and for mission operations. The avionics system for the interim recoverable pressure-fed ballistic LOX/propane booster is not a significant factor in determining the Mark I orbiter avionics configuration, since orbiter/booster avionics commonality is minimal. The booster avionics system is therefore treated separately (in Par. 2.6) of this document.

The baseline system description in the following text includes orbiter subsystems, electronics ground support equipment, and software for development, test, and onboard functions. Ground support for maintenance, launch, and mission operations is treated within the above framework.

The baseline system concept is illustrated in Figs. 2.2-1 and 2.2-2. Safety-of-flight and mission essential functions are performed by dedicated, hardwired subsystems, including displays and controls for crew participation and control.

A tremendous increase in onboard capability and a consequent reduction in dependence on ground support is provided by incorporating a data management subsystem (DMS) plus programmable displays and integrated control panels. The DMS accesses all subsystems, provides the crew with information to assist decision-making, and assists the crew by performing many routine functions. Manual override of the DMS and programmable displays reverts the system to hardwired, dedicated equipment. This configuration permits an evolutionary growth of onboard capability within the Mark I time frame and the flexibility for growth to Mark II capability without a major change of system configuration. Figure 2.3-3 illustrates the planned growth in functional capability of the baseline Data Management subsystem.

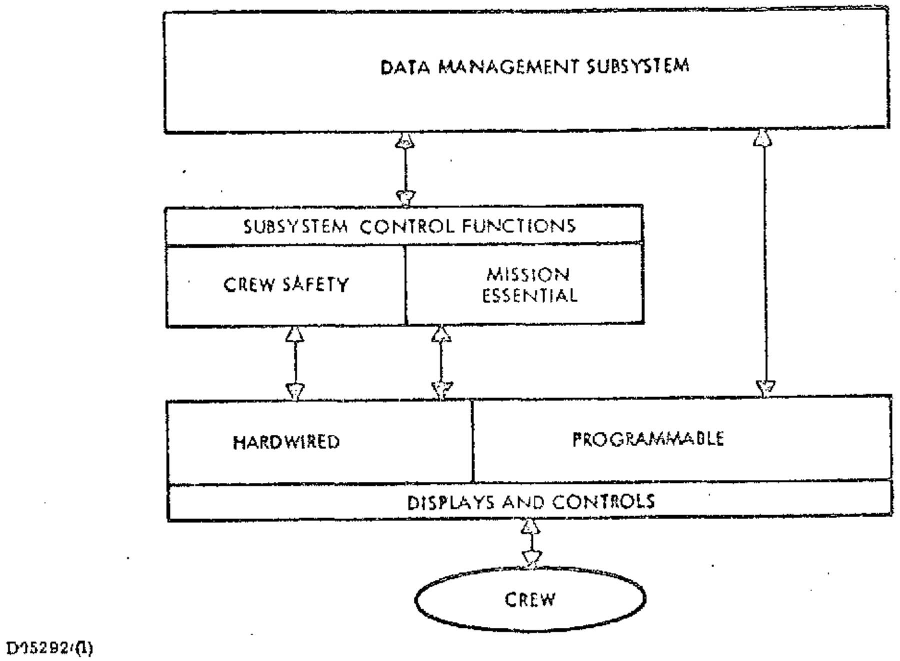

Fig. 2.2-1 Mark I Orbiter Avionics System Concept

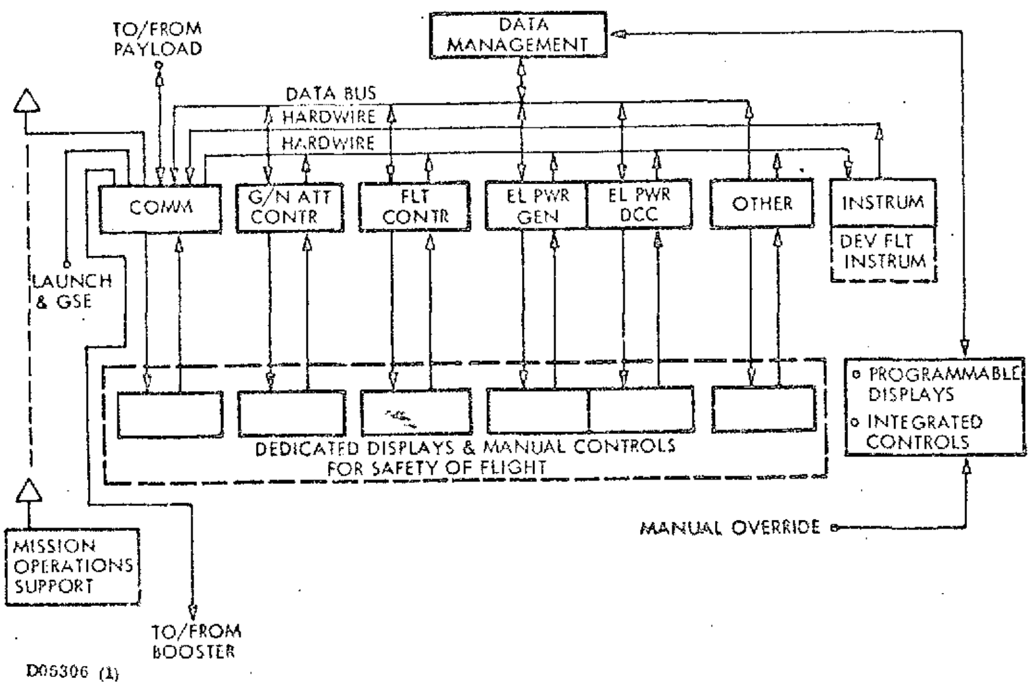

Fig. 2.2-2 Mark I Orbiter Avionics Baseline Recommendation

FUNCTIONS	MARK I			MARK II
	HFT	VFT	OP'L	
ONBOARD CO/FI AND DATA EXTRACTION	O	O	O	O
INSTRUMENTATION AND ELECTRICAL POWER CONTROL	O	O	O	O
ABORT AIDS	O	O	O	O
GN&C COMPUTATIONS		O	O	O
ONBOARD CO/FI/RM		O	O	O
SYSTEM MANAGEMENT AIDS		O	O	O
AVIONICS CONFIGURATION CONTROL			O	O
CONSUMABLES MANAGEMENT			O	O
RENDEZVOUS COMPUTATION			O	O
PAYLOAD MANAGEMENT				O
A/C AND S/C FLIGHT CONTROL				O
NONAVIONICS CONFIGURATION CONTROL				O
MISSION PLANNING				O

D08314

Fig. 2.2-3 Data Management Functions Effectivity

2.2.1 Mark I Orbiter Avionics Subsystems

The functional requirements analysis (Appendix A) was basic to the definition of equipment types for each subsystem and the minimum load of redundancy was required for crew safety. An extensive search for aircraft and spacecraft programs was made to compile information on developed, available equipment of the types identified in the functional analysis. Appendix D summarizes the baseline system equipment and some of its pertinent characteristics. Figure 2.2-4 highlights some of the key characteristics of the Mark I Orbiter avionics subsystems.

GUIDANCE, NAVIGATION, ATTITUDE CONTROL
- DEDICATED COMPUTER
- MANUAL OVERRIDE OF AUTOMATIC
- AUTO/MANUAL TVC AND RCS CONTROL
- RCS ENTRY CONTROL TO MACH 2
- PHASED ADDITION OF SENSOR SUIT

FLIGHT CONTROL SYSTEM
- DEDICATED ANALOG COMPUTERS
- FLY-BY-WIRE
- STABILITY AUGMENTATION SYSTEM
- AUTOPILOT/AUTOLAND
- QUAD-REDUNDANT AEROSURFACE DRIVE/CONTROL
- FLIGHT DATA DISPLAYS HARDWIRE

COMMUNICATIONS & TRACKING
- APOLLO BLOCK II FOR SPACE
- S-3A TYPE FOR ATC VOICE
- ALL NAVIG AIDS OFF-THE-SHELF

DISPLAYS & CONTROLS
- DUAL PILOT/COPILOT STATIONS
- SAFETY OF FLIGHT: DEDICATED, HARDWIRED
- PROGRAMMED DISPLAYS
  - REDUCED CREW WORKLOAD
  - INCREASED CREW CAPABILITY
  - REDUCED PANEL AREA REQUIREMENTS

ELECTRICAL POWER
- PRIMARY POWER
  - MINI-TECHNOL 2000 HR FUEL CELL DEVELOPMENT
  - AC GENERATOR SYSTEM
- SAFETY-OF-FLIGHT HARDWIRE INSTRUM AND CONTROL
- SOME DMS CONTROL ACCESS

INSTRUMENTATION
- FLIGHT CRITICAL: HARDWIRED
- NON CRITICAL: DMS-CONTROLLED (MULTIPLEXED)
- DEVELOPMENT FLIGHT INSTRUMENTATION OVERLAID

DATA MANAGEMENT
- NON-FLIGHT-CRITICAL
- MANUAL OVERRIDE
- ACCESS TO ALL SUBSYSTEMS
- ON-BOARD COFI; R/M AIDS
- FLEXIBILITY FOR MARK II GROWTH
- SOFTWARE TEST BED FOR MARK II

D06279

Fig. 2.2-4 Baseline Mark I Avionics Key Characteristics

2.2.1.1 Mark I Orbiter Guidance, Navigation, and Control (GN&C). The Mark I orbiter GN&C consists of a spacecraft section for operation during phases from launch-through-reentry and atmospheric flight to Mach 2 and in an aircraft section for orbiter operations at velocities below Mach 2. In selecting the baseline configuration, two previously established key program objectives were major drivers - minimization of: (1) technology risk, and (2) program cost and peak funding. These Mark I requirements can be best implemented by selecting equipment and software that are now in production or that will be amply flight-proven by mid 1973, i.e., in time for use on the orbiter vehicle. Figure 2.2-5 summarizes methods for meeting these cost and risk objectives.

MINIMIZE TECHNOLOGY RISK

- ALL MAJOR COMPONENTS ARE (OR WILL BE BY 1973) WELL DEMONSTRATED, IN PRODUCTION HARDWARE.
- NEW COMPONENTS (INTERFACE EQUIPMENT) WILL USE EXISTING CIRCUITS AND EQUIPMENT FROM CURRENT PROGRAMS SUCH AS APOLLO AND AGENA
- SOFTWARE TECHNIQUES, GENERATION, AND VALIDATION WILL USE TECHNOLOGY DEVELOPED AND FLOWN ON AGENA G&N AND OTHER PROGRAMS

MINIMIZE COST

- NEW DEVELOPMENT MINIMIZED
- CHANGES TO EXISTING EQUIPMENT ONLY WHEN ABSOLUTELY MANDATORY
- SELECTION OF COMPARABLE EQUIPMENT BASED ON DEMONSTRATED LOW COST
- USE OF MULTIPURPOSE EQUIPMENT TO REDUCE QUANTITY, MAINTAINING REDUNDANCY
- PEAK FUNDING REDUCED BY DELAYING NEW DEVELOPMENT

D05847(I)

Fig. 2.2-5 Meeting GN&C Objectives

Exceptions to GN&C technical requirements* evolved where conflicts appeared with cost and technical risk requirements and where NASA specifically requested technical deviation. Specific deviations affecting the configuration are:**

- Mark II System and Orbiter. No specific redundancy requirement. Contractor to determine and recommend desired level.
- Mark I System and Orbiter. Polar orbit payload (25K lb desired; 10K min). Turnaround time relaxed to one month.
- Booster. Reusable LOX/RP F-1 booster. (This requirement subsequently deleted at mid-term review and changed to ballistic interim water recoverable booster.)***

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* Ref NASA MSC-04075, Rev. B, "Functional and Performance Requirements Specification, Space Shuttle Avionics, Orbiter", dated 10 May 1971

** Ref. Technical Directive No. 3003

*** Mid-term Review, 7 October 1971, Rye Canyon, Calif.

Additionally, the automatic docking requirements were deleted at mid-term review, and LMSC was notified that the reference to space stations no longer applied. Key technical requirements for configuration definition and methods of implementation are shown in Fig. 2.2-6.

Under the minimum cost and technology risk ground rule, the first horizontal test flight vehicle (FTV-1)**** will use only aircraft type GN&C equipment. Space flight type GN&C equipment initially will be installed in the first vertical flight test vehicle (FTV-2); the full set of equipment will be installed in the first operational (Mark I) vehicle.

Figure 2.2-7 is an overview of the program schedule for the Mark I and Mark II vehicles. Additional details of required specific equipment and levels of redundancy for test and operational vehicles are discussed in the following sections.

**** Scheduled flight July 1976

REQUIREMENT	IMPLEMENTATION
o LANDING AND HANDLING TO REQUIRE NO MORE SKILLS THAN OPERATIONAL LAND-BASED AIRCRAFT	USE OF SIDE STICK CONTROLLERS, AIRCRAFT TYPE PEDALS, STABILITY AUGMENTATION, CONTROL LAW MANAGEMENT, APPROACH AND LANDING AIDS AND INDICATORS
o AUTONOMOUS NAVIGATION CAPABILITY	USE OF STAR TRACKER, HORIZON SENSOR AND RADAR ALTIMETER TO UPDATE IMU. PRECISION RANGING SYSTEM USED AS BACKUP. COMPUTERIZED TRAJECTORY DETERMINATION AND STEERING
o AUTOMATIC AND MANUAL ORBIT VEHICLE STABILIZATION, CONTROL, AND TRANSLATION	HAND CONTROLLERS FOR TRANSLATION AND STABILIZATION INTERFACE DIRECTLY TO ACPS LOGIC AND ELECTRONICS. OVERRIDES AUTOPILOT ACPS CONTROL WHEN USED
o GUIDANCE AND STEERING TO SHAPE TRAJECTORY TO ENTRY VEHICLE HEATING CONSTRAINTS, PRESCRIBED G LIMITS AND TERMINAL FOOTPRINTS	UPDATE PRIOR TO DEORBIT, CLOSED LOOP CONTROL FROM INERTIAL GUIDANCE DURING REENTRY, VERTICAL CHANNEL UPDATE POST BLACKOUT
o AUTOMATIC GUIDANCE AND NAV CAPABILITY PRIOR TO FINAL APPROACH	TACAN, RADAR ALTIMETER, AUTOPILOT, AIR DATA, AUTOTHROTTLE, INERTIAL NAV BACKUP
o APPROACH AND LANDING NAVIGATION THROUGH GROUND AIDS OR BY INERTIAL UPDATING	SCANNING BEAM ILS, (MICROWAVE) IMU AND RADAR ALTIMETER BACKUP
o AUTOMATIC-CONTROLLED INSTRUMENT LANDING WITH PILOT-CONTROLLED INSTRUMENT LANDING AND PILOT-CONTROLLED VISUAL LANDING AS BACKUP	DUAL DUAL L-1011 AUTOPILOT, AIR DATA, AUTOTHROTTLE, AUTOSPEED-BRAKE CONTROL, MICROWAVE ILS

D05765 (I)

Fig. 2.2-6 Orbiter Spacecraft GN&C Key Requirements Implementation

2.2.1.1.1 Aircraft Flight Controls. The baseline Mark I orbiter flight control system consists of dedicated analog sensors, computers, servo-actuators, command displays, and includes provisions for interfacing with the data management computer. This computer is programmed to assess system status, provide BITE stimuli, and collect and process data for the integrated cockpit CRT displays. Flight control equipment is hardwired, redundant, and operationally independent of the data management computer. The flight control system functions encompass provisions for: manual aero-surface control using side arm controllers and rudder pedals; automatic stability augmentation; automatic surface trim throughout reentry (to aid the ACPS); manual control of speed brakes and airbreathing engine throttles; and autopilot operations for all atmospheric operations from Mach 2 through automatic landing.

Primary Flight Control System (PFCS). Manual aero-surface control and stability augmentation features have been integrated into a common hardware computer grouping. This baseline subsystem is depicted schematically in Fig. 2.2-8. The Primary Flight Control System (PFCS) utilizes pitch, roll, and yaw rates; lateral and normal accelerations; side-arm controller positions; and rudder pedal position signals obtained from sensors located in an environmentally-protected avionics rack area and pilot and copilot manual input controllers. These signals are used to control dual tandem servoactuators serving as inputs to the surface power unit manual control valve manifold. The servoactuator inputs are mechanically added in series with automatic trim actuator inputs to control the surface actuators. The servo loop is closed by a mechanical feedback arm to the manual control valve.

The PFCS is engaged in each of the three axes for operation by depressing a control panel switch; this panel also provides logic and switching to engage the emergency controls in the event of catastrophic multiple PFCS failures. Disengagement of the PFCS is accomplished by either manual or automatic means, and indicated on the cockpit annunicator display panel.

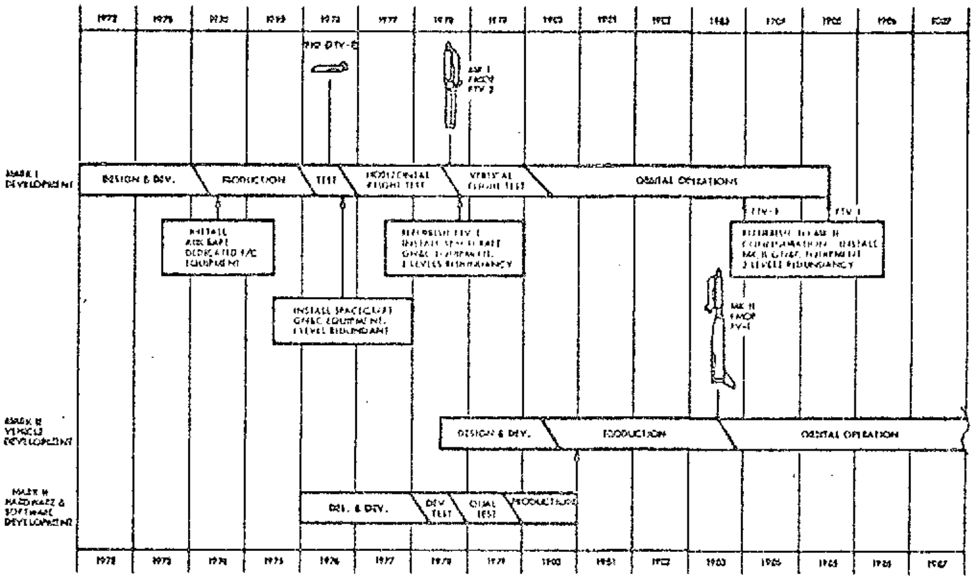

Fig. 2.2-7 GN&C Mark I/Mark II Phase-In

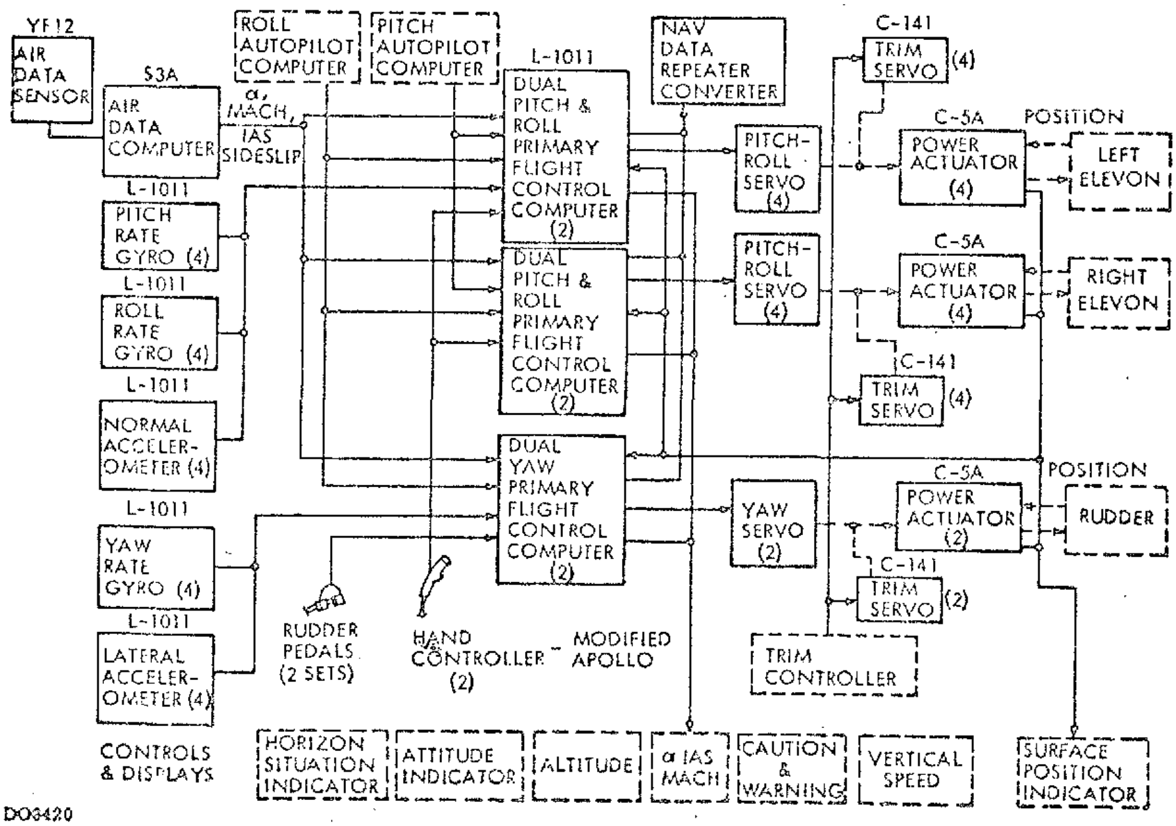

Fig. 2.2-8 Orbiter Aircraft Primary Flight Control System

System mechanization is aimed at fail operational/fail operational/fail-safe operation. For a combination of any two system malfunctions, the system will operate at original gain and authority. Subsequent malfunctions result in unsatisfactory, degraded system performance. Malfunctions result in the system being automatically configured for access by the data management computer during flight time-noncritical portions for isolation of improperly operating channels. During time-critical flight phases, detection of catastrophic PFCS failures automatically places the system in emergency control on the affected axis, provided that the pilot has armed the system on the PFCS control panel.

A single PFCS failure is indicated to the data management computer but does not activate panel warning lights in the cockpit; system status can be addressed by the pilot during any phase of the mission. A second parallel axis fault is indicated by illumination of an indicator strip on the annunciator panel in the cockpit and is addressed to the data management computer. This light notifies the pilot that subsequent faults in the affected axis of control will degrade performance, allowing him to arm system logic for automatically switching to the emergency control mode. Alternatively, the pilot may address the data management computer and perform end-to-end tests to determine remaining system capabilities.

PFCS redundancy management logic is disseminated in the PFCS computers, thus preventing any single fault in the logic or prime power sources from inhibiting the redundancy management function. System checkout is accomplished by the data management computer supplying the necessary stimuli and determining the degree of system readiness. PFCS hardware also is amenable to other design and operational alternatives; however, the present discussion is only concerned with the baseline approach. Hardware selected is basically off-the-shelf L-1011 equipment which is modified in logic and control law implementation for the space shuttle application.

The hardware consists of the following major off-the-shelf components (excluding wiring and installation provisions):

- P/N 672300-1-1, Rate Gyro (Pitch, Roll, Yaw), Four/Axis. Pertinent characteristics include:

  Range - ± 20 deg/sec

  Threshold - 0.01 deg/sec

  Natural Frequency - 20 Hz undamped

  Output - Phase Reversing 400 Hz ac voltage

- P/N 672301-101, Lateral Accelerometer, (4). Pertinent characteristics include:

  Range - ± 2 g

  Resolution - 0.0002 g

  Break Frequency - First Order - 20 Hz

- P/N 672302-101 Normal Accelerometer, (4). Pertinent characteristics are identical to lateral accelerometer

- P/N 672293-101, Computer Unit, (Six: 4 elevon, 2 rudder); Pertinent characteristics include:

  Independent computation channels - two/unit

  Majority voting - quadruplex voting

  Two or three layer printed circuit boards

  Multi-layer side plane board

  Fault isolation monitor annunicators on computer front panel

  GSE test connectors on front of unit

  Isolated channel wiring within the unit

  Growth provisions on each card

- AYN-5, Central Air Data Computer, (2). Pertinent characteristics include:

  Altitude Accuracy - ± (12.5 ft + 1 ft/1000 ft.) to 50,000 ft

  Altitude rate accuracy - ± 4% to ± 4,000 ft/min.

  IAS - ± 5 knots to 500 knots

  Built-in interface with data management computer.

- P/N 697660, PFCS Servo, (Ten: 8 elevon, 2 rudder). Pertinent characteristics include:
  - Dual Independent Servo Loops
  - Dual Independent Hydraulic Systems
  - Break Frequency - As Desired
  - Maximum Rate - As Desired
- P/N L16-81-1, Trim Actuator, (Ten: 8 elevon, 2 rudder). Pertinent characteristics include:
  - Linear Screw Jack Actuator
  - Lightweight
  - 28 VDC Motor

The PFCS provides three-axis stability augmentation functions including rate damping and turn coordination. Although the "SAS Off" vehicle flying qualities at present are incompletely defined and the control laws are in preliminary development stages, enough analysis has been done on the basic (Alternate Concept) study to establish that SAS is required in at least the lateral axis.

The PFCS pilot control loop for elevons and rudder control employ inputs from central air data computers to schedule gain changes and provide load limiting during the high Mach number aerodynamic portions of flight.

Emergency Flight Controls. Emergency means of flying the vehicle after catastrophic failures within the PFCS are provided in the orbiter aircraft flight control system. The concept for the system is simple and highly reliable. It consists of powering trim actuators off isolated battery busses through the standard trim switches on the side arm controller for pitch and roll, and rudder trim pot on the PFCS control panel. During normal operating conditions the trim actuator driving voltage is modulated, according to flight conditions, to schedule trim rates. When the emergency controls are activated by the pilot, this modulator is removed from the circuit and the pilot "flies trim" with the actuators providing maximum trim rates.

The emergency controls can be activated on an axis-by-axis basis by pilot selection on the PFCS control panel. Alternatively, the PFCS system can be armed by the pilot at any time throughout the flight and the emergency controls will be automatically engaged and annunciated to the pilot upon PFCS failure. Rigorous simulation/evaluation of pilot and trim control compatibility must be conducted before system performance capability is verified. Most jet power aircraft today are flown primarily by means of the trim systems. The XV-4B aircraft also was flown in simulation primarily using the "beep" trim system. This is the simplest approach to emergency controls.

Speed Brake and Airbreathing Engine Throttle Control. For powered approach and landing the orbiter speed control system, in conjunction with the GN&C computer, determines the characteristics of the descent trajectory required to achieve a low approach path angle. Speed is controlled through automatic adjustment of the engine throttle. For unpowered flight, velocity reduction must be made before initiation of final glide, soon after the "engine thrust not available" decision point. For this condition, the speed control system automatically operates the speed brakes to control rate of orbiter energy dissipation so that the correct residual energy remains for heading alignment turn, flare, final glide and touchdown. Both the engine throttles and the speed brakes can be operated manually by the crew. The block diagram of the speed brake and engine control system is shown in Fig. 2.2-9.

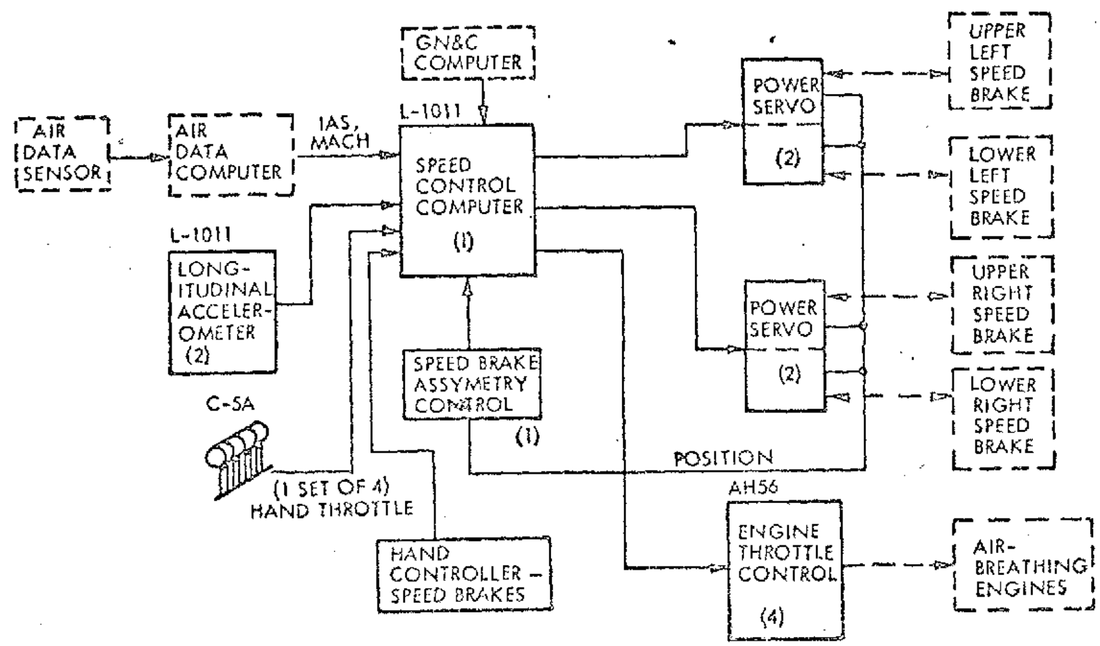

Fig. 2.2-9 Orbiter Aircraft Automatic Speed Control

cable runs from each handle to the upper pair speed brake servos and lower pair of speed brake servos, respectively. Each actuator has series electrohydraulic valves, controlled by a simple asymmetry system patterned after an aircraft spoiler or flap system. However, instead of shutting down the system when asymmetry is detected, the error signal is used to equalize panel deflections. A mechanization of this type allows control functioning in the presence of a fault in the system, which is highly desired for a system upon which the crew depends so heavily during reentry. The electrohydraulic valve inputs afford easy access to speed brake control for automatic approaches made dead-stick. The series input, therefore, is used by the speed control computer during automatic approach and landing for automatic energy management.

The speed brake controls interface with the data management computer to provide status information on system health. The interface is also used to isolate failures to the replaceable unit.

Each engine fuel controller is displaced by a separate rotary servo actuator actuated electrically from an independent throttle level on the pilot and copilot throttle quadrants. The system is basically non-redundant; however, the fact that the MTBF for the system is substantially better than that of the engine, coupled with the fact that the loss of an engine will not cause loss of control, provides the rationale for single-channel operation. Similar controls are used by the AH-56 Cheyenne.

Autopilot Control Modes. The approach being taken toward the use of automatic controls is to take maximum advantage of the proven L-1011 hardware to provide maximum system capability for a relatively small increase in total system cost. Therefore, Mark I Orbiter incorporates an autoland function capable of automatic approach and landing through rollout after the landing guidance beam is captured. The L-1011 system is an integrated autopilot/flight director system, so the pilot plays the role of overall system manager and works in conjunction with the autoland fault detection/correction logic, improving system integrity throughout the approach and landing.

The system consists of dual-dual analog computation channels for pitch and roll steering command generation and dual analog computation channels for speed control. The system interfaces with dual radar altimeters, dual scanning beam ILS receivers, dual control air data computers, the digital guidance computer in the spacecraft GN&C, program flight director and HSI displays, communication and control panels, and the PFCS computers as well as the data management computer. A block diagram of the autopilot is shown in Fig. 2.2-10.

The autopilot/flight director system (APFDS) is fail-operative, since for any single failure its performance is not degraded and it operates with same authority and gains. It is fail-safe in that subsequent faults involving system integrity cause the system to be automatically disengaged. The fault and/or disengaged status is processed by the data management computer and displayed on hardwired annunciators in the cockpit.

The APFDS receives checkout stimuli from the data management computer for system integrity listing prior to deorbit. These stimuli are interchecked with dual switches on the PFCS control panel to prevent inadvertent testing during initial phases of flight.

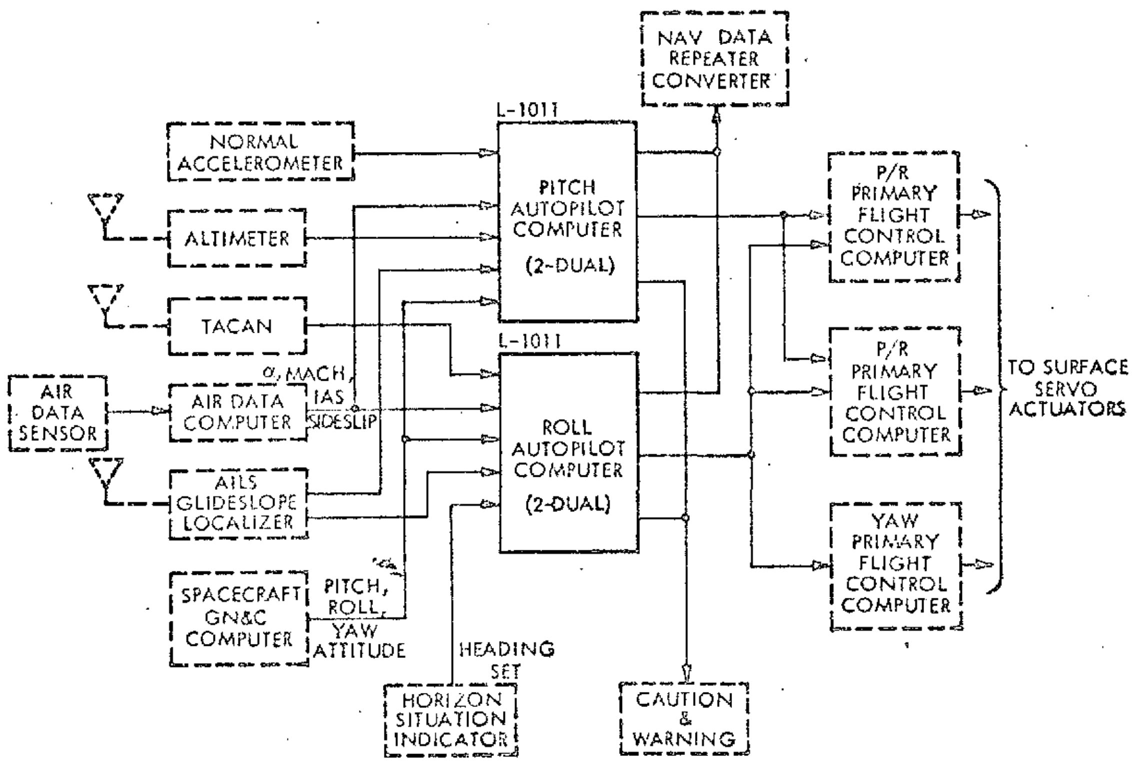

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Fig. 2.2-10 Orbiter Aircraft Autopilot System

In general, the L-1011 hardware will be used intact and unmodified whenever possible to eliminate interface problems; however, S-3A hardware will be used where particular benefit is derived from the built-in data management system interface, since the orbiter DMS is basically the S-3A system. The autoland control laws will have to be modified to account for the steeper approach glide paths, higher speeds, and different flare altitude. The impact of these developments is relatively minor and will not significantly affect overall program costs nor historically demonstrated capability.

2.2.1.1.2 Spacecraft Guidance, Navigation, and Control. The orbiter spacecraft guidance, navigation, and control system navigates, steers, maneuvers, and stabilizes the Space Shuttle from launch through reentry down to a velocity of Mach 2. During launch the orbiter provides guidance to the composite orbiter-booster vehicle and subsequently to reentry below Mach 2 speeds. The spacecraft GN&C system provides inertial navigation backup to the aircraft flight controls and terminal navigation operations.

A "single thread" GN&C system consists of an inertial measurement unit (IMU) to provide inertial acceleration and attitude information; a digital computer to perform the computations required for navigation, guidance, and control; a star tracker to align the IMU in orbit; a horizon sensor and radar altimeter (together with the star tracker) to provide position, altitude, and velocity measurements for navigation; a precision ranging system to provide range and range rate information for rendezvous with a cooperative target; and attitude control propulsion system (ACPS) and thrust vector control (TVC) electronics to provide the control laws and equation solution for steering, maneuvering, and attitude stabilization.

The system interfaces with the aircraft primary flight control system (PFCS), obtaining rate data from its rate gyros; the booster, providing thrust vector control and calculating the booster engine burn ΔV required; data management subsystem, whose digital computer acts as a back-up to the GN&C computer; displays and control subsystem, which displays orbital attitude, velocity, position, and other information for crew use; communication and tracking subsystem, through which the ground control of the vehicle GN&C functions can be achieved; and the vehicle instrumentation and electrical power subsystems (EPS). Figure 2.2-11 is a simplified block diagram of the orbiter spacecraft GN&C system.

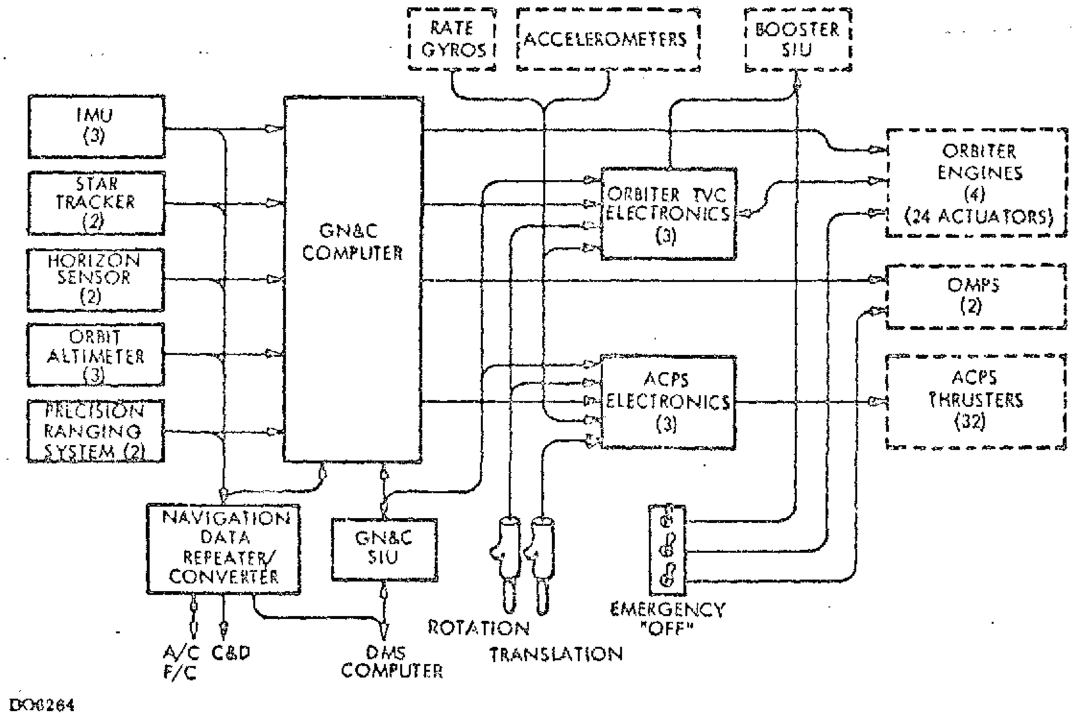

Fig. 2.2-11 Orbiter Spacecraft GN&C System

GN&C Functional Operations. Prior to launch, the inertial navigation platform is aligned using the gravity vector for platform leveling and gyro-compassing to establish azimuth. During or before this period, gyro drift, misalignment, scale factor biases, and accelerometer biases and scale factor errors are measured, and stored in the GN&C computer for calibration purposes. Star tracker, horizon sensor, orbit altimeter and precision ranging system biases, as measured at the equipment suppliers facilities (if determined to be stable), are also included in the program. During this phase, any last-minute changes to mission planning will be loaded into the computer.

During the launch and ascent phase, the orbiter GN&C system will provide the booster with steering commands to keep it on course relative to its programmed trajectory and will provide engine "on" and "off" discretes to keep maximum acceleration below 3 g. The acceleration information can be obtained either from the accelerometers in the IMU or from the longitudinal aircraft flight control accelerometer. Rate information used for control damping is derived from the aircraft flight control rate gyros.

Acceleration during booster engine burn, as detected by the IMU accelerometers, is integrated in the computer and corrected with the IMU calibration data to provide inertial velocity. The velocity information is integrated once more with corrections included for the gravity effects, and the initial conditions added to obtain inertial position. When the required staging velocity is achieved, the GN&C computer sends an engine shutdown discrete to the booster engines and the separation sequence is initiated. After staging, the velocity and position information stored in the computer is used to perform the targeting for orbit injection.

The GN&C computer provides the orbiter engines with an "on" discrete and subsequently controls maximum acceleration to +3.0g in the same manner as it did for the booster. Guidance is provided during orbiter engine burn through the IMU acceleration and attitude data and trajectory calculations. When the computations indicate that the desired altitude and velocity have been achieved by the orbiter, the computer sends an engine "off" discrete.

During coast periods in orbit, the IMU accelerometers will be disabled to prevent noise or biases from contaminating the inertial information stored in the computer. Position, attitude, and velocity update during coast will be performed.

The star tracker, horizon sensor, orbit altimeter, and inertial measurement unit (for attitude angle) will provide the independent measurements required during orbit coast. These measurements are used for the determination of the state variables through the Kalman filter in the digital computer. This function is performed as shown in Fig. 2.2-12.

During the orbiter engine burn period, the estimate of the state vectors will be continuously updated deterministically by integrating acceleration as measured by the IMU and adding the initial conditions to get velocity and integrating once more to get position. Data obtained during ground calibration (gyro misalignment, gyro drifts, accelerometer biases, etc) will be used in the computations to increase the accuracy of the results. The resulting information (estimated position and velocity) will be stored in the computer to be used for initialization when the next orbital update is required.

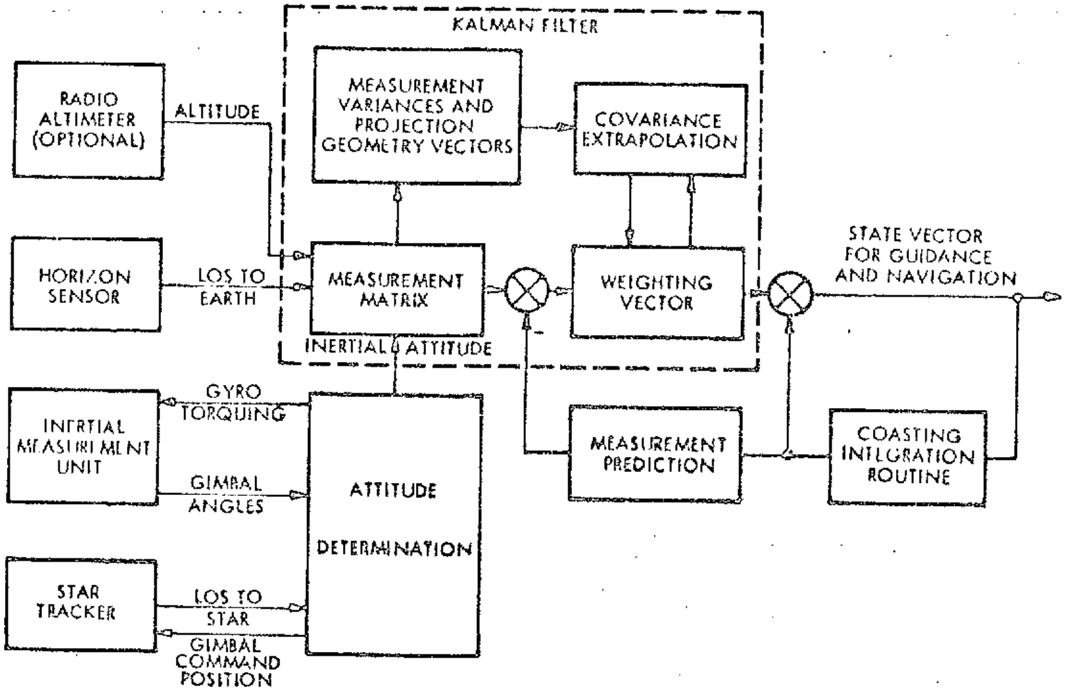

Fig. 2.2-12 On-Orbit and Attitude Determination

For rendezvous with the cooperative target, the precision ranging system is used to obtain navigation information. Through the use of a transmitter/receiver/interrogator on the orbiter and a compatible transponder on the target vehicle, distance to the target and relative velocity are measured. Use of three or more transmitter/receiver antennas on the orbiter, located some 50 feet apart, allows the determination of the line-of-sight (LOS) to the target through phase differences and triangulation.

The range, range rate, and LOS angle data are processed in the GN&C computer to determine the orbital plane and phase change, engine burn attitude, and velocity changes required. With the targeting completed, the applicable commands are given for vehicle orientation and start of engine burn. The subsequent steering is performed by the computer from acceleration and attitude information is obtained from the IMU accelerometers and gimbal readouts, respectively.

When the desired position and velocity, relative to the target, are achieved the GN&C computer turns the orbiter engines off. The crew selects the desired attitude stabilization mode (free inertial, horizon sensor control to local vertical, or target tracking) and prepares for manual control of the orbiter for docking.

Manual control of the orbiter attitude and translation are achieved through the use of a set of controllers, one for rotation and the other for translation, which actuate the ACPS thrusters. The control mode is primarily acceleration for translation and rate for rotation. Controls are provided for both the pilot and co-pilot although a third station (with appropriate controller and displays), with a view looking through the cockpit window in generally the aft direction, may be required if docking is to be done through the payload bay.

Deorbit and reentry operations begin with an update of position, velocity, and attitude. Targeting for desired reentry footprint and retrograde burn initiation point and orbiter orientation are determined in the computer. The sequence is initiated manually and appropriate maneuvers are automatically performed to orient the vehicle in the desired direction. The engines are ignited and attitude is controlled by the computer and the IMU to maintain the desired trajectory. When the computed position and velocity are achieved, the engines are turned off, with attitude control being maintained by sensing angle-of-attach, slip and yaw angles through the IMU accelerometers (with corrections for local gravity affects), and correcting the errors through the ACPS. The IMU gimbal angles will also be used to indicate deviations from the reference trajectory.

When the vehicle exits the "blackout" regime (at about 150,000 feet) vehicle altitude will be updated using the orbit altimeter and the required trajectories to a number of alternate landing sites will be computed. The point in the trajectory where the controls will be handed off to the aerosurfaces will also be calculated. The crew will select the landing site and initiate the automatic control for the terminal phase. Acceleration and attitude data from the IMU will be used to fly the nominal trajectory with information from TACAN, available if needed. When the hand-off point is reached, the aircraft flight control system takes over with some period of overlap before the ACPS system is turned off. The aircraft flight control system is described in par. 2.2.1.1.1.

GN&C Equipment. The study requirement to minimize program costs and technological risk immediately eliminated many system concepts and equipment from consideration, on the basis of early development status. Equipment such as the dodecahedron inertial measurement unit, and concepts such as land mark trackers for orbital navigational update, could not be considered seriously because of production status in the time period 1972 through 1977 when the equipment would be required for the Mark I vehicle. Prior to start of Mark II development, each of these and other areas presently being demonstrated in the laboratory should be re-evaluated to determine if potential technical advantages warrant cost and risks involved.

Further discussion of GN&C equipment follows, in paragraphs a. through f.

a. Inertial Measurement Unit. In the inertial measurement area, the Carousel IV IMU was chosen, primarily because of its low cost ($110K compared to the Honeywell Inertial Sensor cost of $300K) and its demonstrated (in aircraft) capability compared to the Carousel VB or the dodecahedron, neither of which are in production at this time. Two factors remain to be resolved and could have a bearing on the final IMU selection. These are: (1) the possible modification required to the Carousel IV unit to allow operation in the ascent and space environment, and (2) whether the attitude (gimbal) angular readout accuracies of 0.2 deg (0 to 10 deg range), 0.5 deg (10 to 30 deg range) and 1.0 deg (30 to 180 deg range) are adequate for space shuttle use. Both of these concerns would be resolved if the Honeywell Strapdown Inertial Sensor Assembly is used, since the unit is space-qualified and the attitude readout resolution is better than 0.1 minute of arc.

Some of the key characteristics of the Carousel IV IMU and the Honeywell ISA are shown in Table 2.2-1. The Carousel VB, which is to be developed for the Titan IIIC, has performance characteristics similar to the Carousel IV and would cost about the same ($110K) but would use only about one-third the power.

b. Digital Computer. The Univac 1832 general purpose computer was selected as the GN&C computer to perform the ascent and space flight guidance and navigational functions and computations. The Sperry-Rand unit was chosen for this application because

Table 2.2-1

INERTIAL MEASUREMENT UNIT

Parameter	Delco Carousel IV	Honeywell ISA (Used With H448)
Performance Gyro Non-G-Sensitive Drift (1σ) G-Sensitive Drift (1σ) G-Sensitive Drift (1σ) Accelerometer Bias Stability Scale Factor Stability	 0.1°/hr 0.12°/hr/G 0.003°hr/G^2 20×10^{-5} G 350 PPM	 0.1°/hr 0.2°/hr/G 0.05°/hr/G^2 10×10^{-5} G 100 PPM
Power Warmup Operate	 1200 W 400 W	 220 W 125 W
Weight	53 lb	38 lb
Developed (used on which program)	Boeing 747	Airforce Agena P711
When Qualified	–	Mid 1971
Quantity Built/Flown	–	6/1
Cost Modification and Requal Per Unit	 \$1.0 M (LMSC Estimate) \$110 K	 \$0 \$300 K

of the advantages inherent in using the same computer in both the GN&C and data management systems (DMS). The DMS unit which has a dual processor and dual memory stack (65K words) will operate as a backup to the single processor, single-memory stack (32K words) GN&C unit. It will operate in parallel with the GN&C unit during flight, performing simultaneous computations but will be functionally off-line until commanded through the data management system to functionally replace the GN&C computer. The use of the same computer in GN&C and the DMS will reduce the total software generation and validation effort because the identical program will be used in both units.

Other computers investigated during the study were the Delco Magic 362, CDC Alpha-1, CDC 469, and GE CP-24A, IBM 4π units. All of these remain strong candidates for future consideration because of the weight, power, and hardware cost advantages. Table 2.2-2 gives some of the key characteristics of the computers considered.

c. Orbital Initialization and Update Sensors. For alignment of the IMU in orbit, the star tracker was selected because of its accuracy. Other methods which could be used with the equipment already available on the vehicle would be to align the IMU in two axes through the establishment of the local vertical using the horizon sensor and fixing the third axis using the precision ranging system with a ground transponder, or the gyro compassing technique. The latter method takes advantage of the dynamic coupling between the roll and yaw axes inherent in an earth-orbiting vehicle which is kept locally horizontal. The roll rate error is used to correct for out-of-orbital plane yaw errors. The star tracker selected was the Bendix Apollo Telescope Mount (ATM) unit, being developed for the Skylab program. Other units considered were the Litton LTN 300 unit developed for the FB-111 airplane, and the Kollsman KS 199 which was being developed for the Manned Orbiting Laboratory. The reason for the selection of the Bendix unit is that it is being developed for, and will be qualified to operate in, long-term space use. It can be used with little or no change for the shuttle application. The Kollsman unit requires additional development and testing, and the Litton unit might have to be repackaged to withstand the lengthy space operation.

Table 2.2-3 shows a comparison of the three candidate systems.

The method selected for orbital update was to determine inertial position by establishing the line-of-sight (LOS) angles between two or more stars and the local vertical and distance from the center of earth.

The same star tracker required for IMU alignment is used to determine the star LOS, a horizon sensor is used to establish the local vertical, and a radar altimeter is used to determine altitude.

Candidate	Memory Word Length (Bits)	Time Cycle Add Multiply (μ-sec)	No. Of Instructions	
CDC Alpha -1	32	1.0 2.0 7.0	184	Rand Nond Rend Dest (DRC
IBM 4X CP	32	2.5 5.0 18.1	61	DRO
CDC 469	32	1.6 2.4 10.4	44	RA NDR Plat (PW Opti
GECP-24A	24	1.00 3.75 30.5	53	RA NDR PW
Delco Magic 362	16	2.0 4.0 12.0	44	RA DRC Core
Univac 1832	32	0.75 3.00 9.00	131	RA DRC NDR

Table 2.2-2

ORBITER COMPUTER COMPARISON

No. Of Instructions	Memory		Circuit Characteristics	Arithmetic Features	MTBF
	Type	Basic/Max Words			
184	Random Access (RA) Nondestructive Readout (NDRO) Destructive Readout (DRO)	16k/131k Direct Addressing + 0.65k	LSI/IC 115v, 400 Hz 30 Input Power	Fixed or floating Point, Double Precision 2's Complement, Trig Instructions	6,000 hr Predicted
61	DRO Core	8.19k/32k Direct Addressing for 24k	TTL Flatpacs	Fixed Point 2's Complement, Full Length	5,158 hr
44	RA NDRO Plate Wire Memory (PWM) Optional RD Only	4k/64k	MOS-LSI Needs Converter	Fixed Point 2's Complement	8,000 hr Predicted
53	RA NDRO PWM	8k/32k	MOS-TTL	Fixed or Floating Point	12,000 hr Predicted
44	RA DRO Core	8k/65k	MSI-TTL	Fixed Point, Double Precision	5,000 hr Predicted
131	RA DRO NDRO	32k/96k	MSI	Fixed and Floating Point Option, Double Precision	2,000 hr

[illegible]	MTBF	Support Software Availability	Dev/Qual Status	Weight	Power	Remarks
...ating ...ent, ...ctions	6,000 hr Predicted	All Software Support Program Available, Run On CDC 6400, CDC 3300 XDS Signal 7	In Production, Qualified to MIL-E-5400, Class 2	44 lb (32k)	330 W	
...ement, ...th	5,158 hr	Software Available, Run on IBM 360	In Production, Qualified to MIL-E-5400 Class 2	65 lb (32k)	250 W	Liquid Coolant
...ement	8,000 hr Predicted	Assembler Simulator Available, FORTRAN IV Compatible With CDC 6600	3 Prototypes Produced	6.5 lb (32k)	15W	No Active Cooling Required
...Point	12,000 hr Predicted	Assembler Simulator (Regis-Logic), Test and Diagnostic Routine, Utility Routines, Library Routines	One Engineering Model Produced, Qual Test Scheduled for 2nd Quarter 1972	35 lb	35 W	No Active Cooling Required
...ecision	5,000 hr Predicted	Assembler Basic Compiler (IBM 370) Simulator in Development Automatic Check-out for Program Validation	First Prototype Scheduled Dec, 1971	25 lb (32k)	114 W	Air Cooled
[illegible]	2,000 hr	ULTRA Macro Assembler CMS-2 Compiler Language SLIC Librarian and Corrector UTIL Service Routine SOLO Leader	In Production	126 lb (32k)	600 W+	Air Cooled

Table 2.2-3

ORBITER STAR TRACKER COMPARISON

Parameter		Bendix Apollo Telescope Mount (ATM)	Litton LT 300	Kollsman KS 199
Performance				
Accuracy		30 sec	15 sec	15 sec
Field of View		1.0°	2.0°	1.0°
Gimbal Range		$\pm 80^{\circ}$ Outer, $\pm 90^{\circ}$ Inner	$\pm 60^{\circ}$ Outer, $\pm 75^{\circ}$ Inner	$\pm 55^{\circ}$
Star Magnitude		–	–	–
Reliability (MTBF)		5,000 hr	400 hr	12,000 hr
Power		25 W	20 W	23 W
Weight	Tracker	39 lb	62 lb	35 lb
	Electronics	28 lb	28 lb	35 lb
Developed for Which Program		Skylab	FB 111	MOL
Development Status				
When Qualified		Nov 1971	June, 1971	Late 1970
Number Built		6	109	1
Number Flown		0	90-95 Units	0
Cost			LMSC Est.	LMSC Est.
Development		0	$1.0M	$1.0M
Per Unit (Order of 10)		$250K	$200K	$200K

The horizon sensor selected was the Barnes 13-166 conical scan system, which uses a thermistor bolometer to locate the earth horizon along the sensor scan path by differentiating between the earth and space infrared energy levels. This system has been qualified to space environments and a number of systems have flown on earth-orbiting missions exceeding 30 days.

The 13-166 design uses the same basic concepts, the identical electronic circuits, scan motor, and bearings as the Model 13-156, which has flown with great success on over 100 Agena space missions. The 13-166 model can be used for the shuttle without any change in basic design.

Other candidate systems investigated were the Quantic Model IV and the Lockheed Low Altitude Horizon Sensor. These two systems were eliminated from consideration because of their early development status (neither have been qualified), although the attitude accuracy of both systems should be significantly better than the Barnes system.

Table 2.2-4 shows a comparison of some of the key features of the three sensors investigated.

The orbit altimeter selected was the GE 7631 111 G1 system being developed for the Skylab program. In the few weeks of the study, no other source in industry with a system approaching the capability of the GE unit could be found. From discussions with General Electric, apparently all shuttle requirements can be met without modification. Other suppliers contacted were Westinghouse and Teledyne/Ryan. Both of these suppliers had developed test systems for the early Saturn launches; however, considerabl development would be required by both to upgrade these systems.

Other methods of determining altitude were investigated and two of these will be used as backup methods for the shuttle. These are: (1) using the horizon-to-horizon angle measurement taken by the horizon sensor, and (2) using the range and line-of-sight measurements taken with the precision ranging system on the vehicle and a ground transponder.

The reason that the orbit altimeter was chosen over the horizon sensor is that the accuracy of its measurement (over ocean) is within two meters compared to the 1 to 2 miles accuracy of the horizon sensor at an altitude of 100 miles. The reason the altimeter was selected over the precision ranging system (PRS) was that the PRS, capable of measuring altitudes beyond 200 miles, is still in the development stage, although expected to be in production within 5 to 10 years. Also, even for earth orbits below 200 miles, ground transponders within reach of the onboard interrogator would be required during measurement. If transponders are available for other usage, such as for orbital update and landing, the PRS would be a strong candidate to replace the orbit altimeter.

Table 2.2-4

ORBITER HORIZON SENSOR COMPARISON

Parameter	Barnes 13-166	Quantic Mod IV	LMSC Low Alt Horizon Sensor
Performance			
Accuracy	Classified	±0.05 deg	±0.05 deg
Horizon Variation Effects	±0.10 deg	±0.06 deg	±0.06 deg
Altitude Range	80 – 400 nm	80 – 25K nm	80 –400 nm
Acquisition Range	±12 deg	±140 deg (400 nm)	±15 deg
Control Range	±10 deg	±20 deg (300 nm)	±5 deg
Output Type	Analog 1 volt/deg	Digital 14-Bit Serial Analog and 1 volt/deg	Digital 16-Bit Serial
Reliability	0.942 (30 Days)	0.9999 (6 Months)	0.977 (30 Days)
Power	25W	20W	20W
Weight	25 lb	18 lb	20 lb
Dimensions (in.) Tracker	5.1 x 4.8 Dia (2 each)	8-1/2 x 5 x 5 (4 each)	7 x 6 x 5 (4 each)
(LXWXH or LXDIA) Electronics	11.1 x 8.5 x 3.2	10 x 8 x 4	Not required
Developed for Which Program	AF Agena	–	–
Development Status			
When Qualified	Oct 1971	–	–
Number Built	Prototype, Qual and 2 Flight	Partial Prototype	Partial Prototype
Number Flown	0	–	–
Cost			
Development	–	$1.4M	$1.0M
Per Unit (Order of 10)	$100K	$200K	$80K

d. Rendezvous Sensor. For rendezvous navigation, the Cubic precision ranging system (PRS), microwave radar and laser radar were considered. Table 2.2-5 gives some of the characteristics of laser and microwave radar characteristics and Table 2.2-6 lists characteristics of the precision ranging system.

Table 2.2-5
RENDEZVOUS SENSOR CHARACTERISTICS

	Laser (Coop)	Radar (Coop)
Search Angles	$\pm 15^{0}$	$\pm 90^{0} \times 90^{0}$
Range, Max.	75 nm	400 nm
Range Rate, Max.	10,000 n/sec	4900 ft/sec
Accuracy		> 50 nm ±0.1%
Range Rate	±0.5%	< 50 nm ±80 ft
Range Rate	±1 m/sec	±1.0 ft/sec
LOS Angles	±0.02 deg	±0.1 deg
LOS Angles Rate	±0.003 deg/sec	
Shuttle	7.5 in. dia x 17 in. lg. (cyl) and electronic	28 x 8 x 5 in.
Size	6 x 12 x 12 in.	
Weight	24 lb	75 lb
Power Consumption	20 watts	250 watts
Antenna Aperture	Xmtr. 0.5 in. dia Rec. 2-3 in. dia	24 in. dia
Space Station		
Size	4 in. dia	12 x 7.5 x 6 in.
Weight	2 lb	14.5 lb
Power Consumption	N/A	75 watts
Antenna Aperture	N/A	4 in. dia

Table 2.2-6

CUBIC MODEL CR 100-4 PRECISION RANGING SYSTEM (ORBIT MODE)

Error Source	1σ Magnitude
Random Error	
Ranging Error Due to Finite Signal-to-Noise Ratio and Equipment Added Noise	1.0 ft
Phase Shift Over Dynamic Range of Ranging Operations	1.0 ft
Phase Shift With Temperature Over Operating Environment	1.0 ft
Phase Shift of Interrogator Due to Vibration, Shock and G-Loading	Negligible
System Error Due to Craft Dynamics (25,000 ft/sec) and 1000 ft/sec^2	0.2 ft
Multipath Error in Ground-to-Air Range Links	3.0 ft
Digitization Error	0.3 ft
RSS Total	3.5 ft
Bias Error	
Calibration (Equipment)	1.0 ft
Scale Factor	
Stability of Crystal Oscillators	0.1 ppm
Uncertainty in Velocity of Flight	0.5 ppm
Propagation	
N Approximation	50.0 ppm
End Point Correction	10.0 ppm
Electrical Characteristics	
Carrier Frequency	L- or S-Band
Transmitted Power	12 watts
Receiver Sensitivity	
Medium Range and Rate Mode	-119 dbm
Long Range and Rate Mode	-125 dbm
Range Rate Only	-131 dbm
Power Consumption	
Interrogator	120 watts
Transponder	
Standby	7 watts
Transmit	80 watts
MTBF	
Interrogator	2300
Transponder	7000

The laser radar was considered a strong candidate because of its accuracy, low weight, and low power, but was considered a limited use device compared to the multipurpose PRS because of the potential hazard to the eyesight of unprotected crew members on the target vehicle (conventional radar is also of limited use and was discarded for this same reason and because of its higher power requirements).

The system chosen (PRS) is a range and range rate measuring device using phase comparison between the transmitted signal and the signal returned from a transponder for range determination and the doppler effect to determine range rate. Figure 2.2-13 describes the basic technique of determining distance. Three antennas, located at different points on the orbiter, can be used to determine direction to the target through phase differences and triangulation.

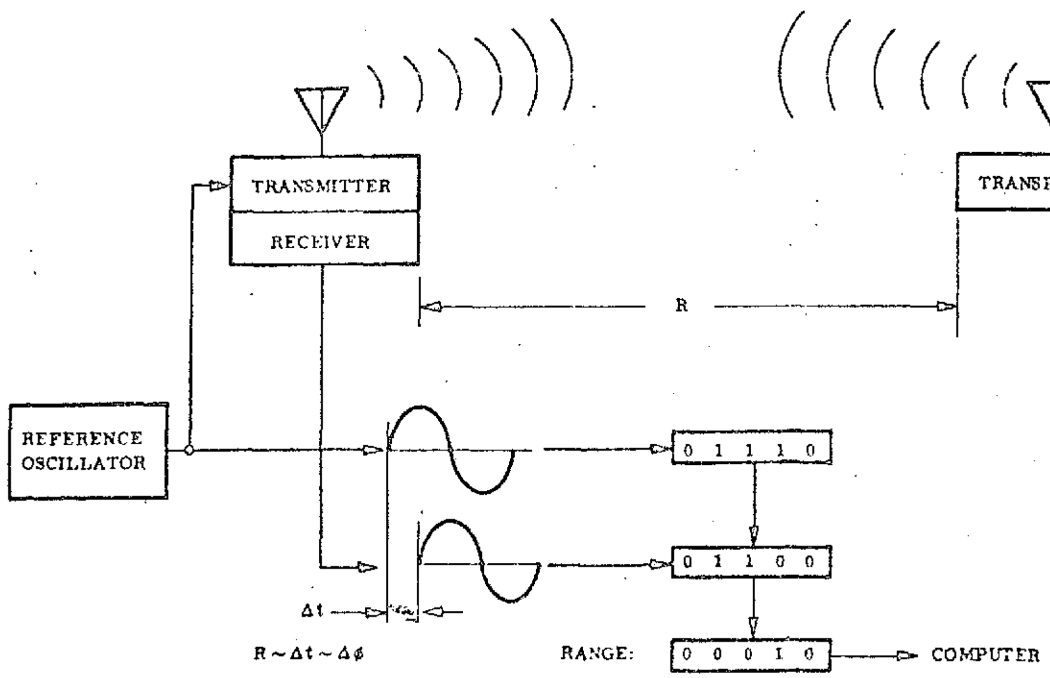

Fig. 2.2-13 Phase Comparison Ranging

An advanced version of the PRS can also be used for orbit update and as a navigation aid for landing. For orbit update, position and velocity can be obtained through the use of one interrogator on the vehicle and one transponder on the ground at a precisely known location. Direction to the ground transponder can be obtained by using the same three antennas used for rendezvous. The ambiguity in orientation about the direction vector to the ground transponder can be resolved through a single star sighting or by using the horizon sensor to control to the local vertical. For landing, range and range rate can be obtained through the use of one or more transponders at the landing site and altitude and vertical velocity can be obtained through triangulation with three or more transponders on the ground near the landing strip. These same landing site transponders can be used for the orbital update if they fall within range of the interrogator on the orbiter during orbit passes.

Cubic has in production for the Air Force the Model CR 100-1 (CIRIS) unit, which has a limited range of 200 miles. This model could be used unchanged for rendezvous and landing purposes but would not be adequate for orbital update. Since the primary use of the PRS is for rendezvous, with it being used as backup to conventional ILS (horizontal test flights) or the microwave scanning beam (vertical flights) for landing, it is proposed that the existing limited range model CR 100-1 be used on the early Mark I vehicles and the longer range CR 100-4 model be incorporated when that system becomes available. Table 2.2-7 lists the expected errors when PRS is used for approach and landing navigation.

e. Attitude Control Propulsion System (ACPS) and Thrust Vector Control (TVC) Electronics. The ACPS and TVC electronics used on the orbiter will be similar to those used with great success on more recent Agena and other Lockheed space programs. The same depth of technology developed in the electronic and packaging design, parts application, and materials and processes selection areas for these programs will be applied to the orbiter electronics.

Table 2.2-7

MODEL CR 100-4 RANGE AND RANGE RATE ERROR BUDGET
(APPROACH AND LANDING MODE)

Error Source	1σ Magnitude
Range	
Random Error	
Ranging Error Due to Finite Signal-to-Noise Ratio and Equipment Added Noise	0.3 ft
Phase Shift Over Dynamic Range of Ranging Operations	0.5 ft
Phase Shift of Interrogator Due to Vibration, Shock and g-Loading	Negligible
System Error Due to Craft Dynamics (2000 ft/sec) and 1000 ft/sec^{-2}	0.2 ft
Multipath Error in Ground-to-Air Range Links	3 cos ϕ ft
Digitization Error	0.3 ft
RSS Total	$(3 \cos \phi)^2 + (0.7)^{2-1/2}$
where ϕ = elevation angle	
Bias Error	
Calibration (Equipment)	1.0 ft
Phase Shift With Temperature	0.5 ft
Scale Factor	
Stability of Crystal Oscillators	0.1 ppm
Uncertainty in Velocity of Light	0.5 ppm
Propagation	
N Approximation	50.0 ppm
End Point Correction	10.0 ppm
Range Rate	
Velocity Independent	
Rate Error Due to Finite Signal-to-Noise Ratio and Equipment Added Noise	0.01 ft/sec
System Error Due to Craft Dynamics, a = 1000 ft/sec^2	0.001 ft/sec
Digitization Error	0.014 ft/sec
Multipath	0.01 ft/sec
RSS Total	0.02 ft/sec
Velocity Dependent	
Stability of Crystal Oscillator	1 ppm
Uncertainty in Velocity of Light	0.5 ppm
Propagation	
N Approximation	50.0 ppm
End Point Correction	10.0 ppm

The ACPS electronics for the orbiter performs much the same functions as the electronics used on the Air Force P467 program to drive the monopropellant (hydrazine) thrusters on this large space vehicle. The main differences between the two applications are that the P467 electronics drive 8 thrusters and the orbiter electronics must drive a total of 32 thrusters; and that the logic involved in thruster selection for the shuttle vehicle will be significantly more complex because of the number of thrusters and the interaction between translation and rotation created by the lack of pure couples. Figure 2.2-14 shows the combinations of thrusters required for rotational and translational control. Figure 2.2-15 shows a typical ACPS electronics channel.

The TVC electronics for the orbiter is similar to that used for the P110 Air Force Agena Program and will derive most of its circuit and packaging techniques from that unit. The same basic circuits were used previously on the Agena target vehicle for the Gemini program. Figure 2.2-16 shows the signal phasing associated with gimballing the four engines and Fig. 2.2-17 is a block diagram of the Orbiter TVC electronics.

THRUSTER GROUP	ENGINE QUANTITY	TRANS			ROTATION		
		X	Y	Z	θ	ψ	ϕ
1	2			+	+		+
2	3	+				-	
3	2			-	-		-
4	3	-				+	
5	3		-			+	
6	2			+	+		-
7	3	+				+	
8	2			-	-		+
9	3	-				-	
10	3		+			-	
11	3	$-\Delta$		$+\Delta$	+		
12	3	$+\Delta$		$-\Delta$	-		
	32 TOTAL						

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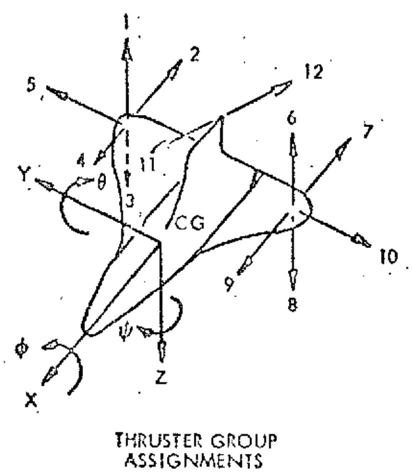

Fig. 2.2-14 ACPS Thruster Matrix

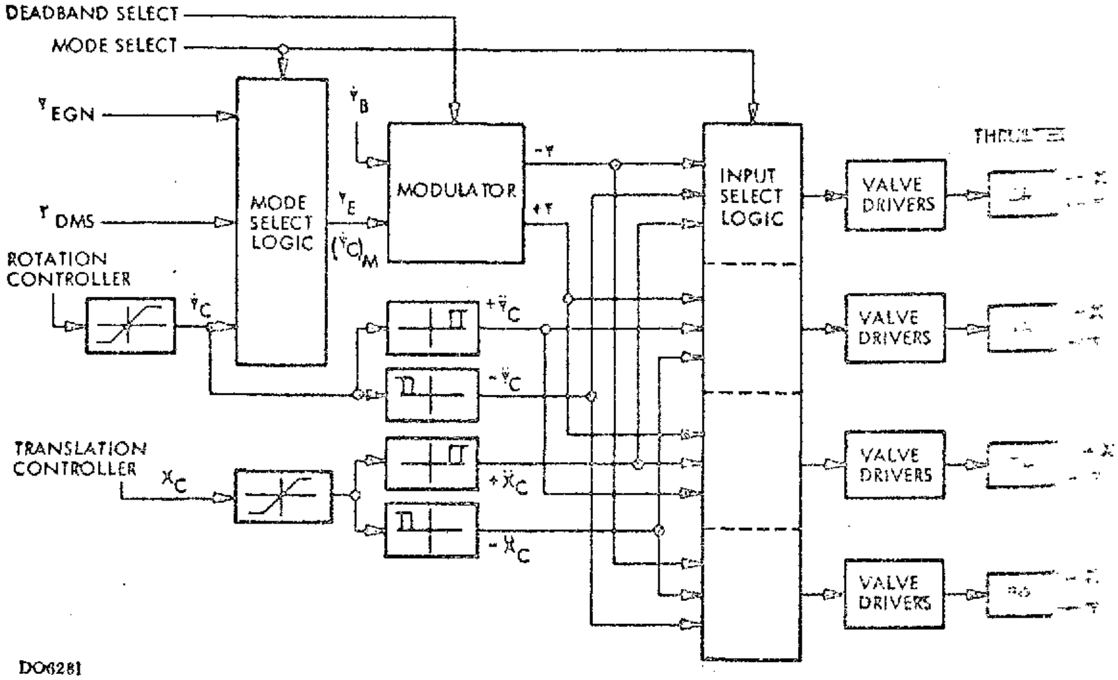

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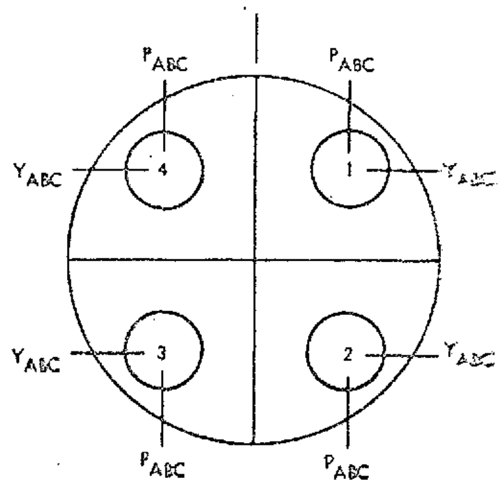
Fig. 2.2-15 Typical ACPS Drive Electronics (X Translation and Yaw Rotation)

EXTEND ACTUATOR	ATTITUDE PHASING		
	θ	ψ	φ
1P	-		-
1Y		-	+
2P	+		+
2Y		-	-
3P	+		-
3Y		+	+
4P	-		+
4Y		+	-



DO6243

Fig. 2.2-16 Orbiter TVC Phasing

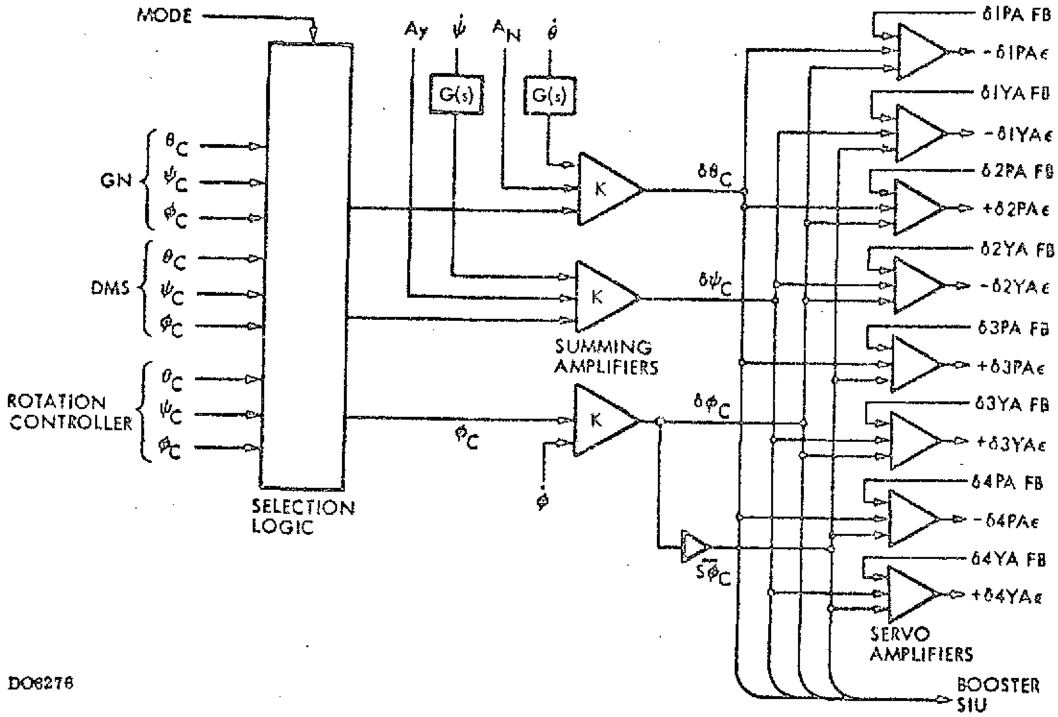


Fig. 2.2-17 Orbiter TVC Electronics

f. Equipment Summary. A list of the orbiter spacecraft GN&C system equipment is shown in Table 2.2-8, together with the part number, weights, present or planned usage and the quantity required by program phase.

Table 2.2-8

ORBITER SPACECRAFT GN&C EQUIPMENT

EQUIPMENT	PART NO.	WEIGHT PER EACH	PRESENT OR PLANNED USAGE	QUANTITY/EFFECTIVITY				
				MARK I				MARK II
				FHF	FVFM	FVFUN	OPER	
INTERIAL REFERENCE UNIT	CAROUSEL IV CAROUSEL VB OR HONEYWELL 448	53 LB 58 LB 38 LB	747 THIC AGENA		2	2	3	3
DIGITAL COMPUTER	UNIVAC 1832	126 LB	S-3A		1	1	1	1
STAR TRACKER	BENDIX ATM	77 LB	SKYLAB				2	2
HORIZON SENSOR	BARNES 13-166 QUANTIC MOD IV OR LMSC LAHS	25 LB 18 LB 10 LB	AGENA — —				2	2
ORBIT ALTIMETER	GE 7631111G1 DERIVATIVE OF ABOVE UNIT	45 LB 40 LB	SKYLAB GEOS		2	2	3	3
RENDEZ. AND LANDING DME (PRS)	CUBIC CR 100-1 OR CR 100-4	25 LB	—				2	2
MAIN ENGINE GIMBAL SERVO (TVC) ACTUATOR PKG	MOOG	50 LB	SIVB		8	8	8	8
TVC ELECTRONICS	SIMILAR TO AGENA P110 TVC ELECTRONICS	30 LB	AGENA		3 8 CHANNELS EACH	3	3	3
ACPS ELECTRONICS	SIMILAR TO A/F P467 ACPS ELECTRONICS	50 LB	P467 ACPS		3 EACH HAS DRIVERS FOR 12 ENGINES	3	3	3
SUBSYSTEM INTERFACE UNITS	—	55 LB	NEW		3	3	3	3
TOTAL	—				22	22	30	30

2.2.1.2 Communications, Tracking, and Navigational Aids. The communications subsystem for the Mark I orbiter (Fig. 2.2-18) comprises the equipment necessary for data and voice up-link and down-link, crew intercom system, and recovery beacon for all phases of shuttle operations.

For atmospheric flight operations (horizontal flight test, ferry, and approach and landing following reentry), standard aircraft UHF voice communications is provided through the UHF transceivers, antenna switch, and combination UHF/S-band scimitar-notch (SCIN) antennas. Voice utilizes the S-3A audio panel and is switched through the panel controls and the S-3A communications SIU to the UHF transceivers. Use of the VHF beacon applies primarily to atmospheric flight test vehicle recovery operations.

Spacecraft Communications. For exoatmospheric voice communications, the Apollo S-band equipment (premodulation processor, unifield S-band equipment, S-band power amplifier, and antenna switch, and the combination antennas) are utilized. Voice again utilizes the audio panel and communication SIU for proper switching.

Aircraft/Spacecraft Communications. Telemetry (PCM and FM) utilizes the S-band equipment for both aircraft and spacecraft applications to transmit developmental and operational instrumentation data, as well as coded data from the data management system. Crew intercommunication equipment is contained in the audio panel for both applications.

Equipment Selection. All equipment is selected based on existing spacecraft and aircraft applications when high reliability has been proven for voice data and command communication. The S-band equipment is Apollo-proven and the UHF audio and interface equipment is from the S-3A aircraft. A complete list of equipment is given in Appendix D.

The Apollo unified S-band equipment combines voice and data communication, tracking, ranging, and telemetry on a single (rest) frequency of 2272.5 ±.455 MHz.

The VHF recovery beacon transmitter frequency is 243 MHz with 30 percent modulation with a 1000 Hz square wave, which is keyed on two out of five seconds on a cyclic basis.

The UHF equipment is tuned through the communication SIU to the selected channel. The antennas are placed on the upper and lower surfaces of the orbiter to give full 360-deg coverage. Each is a notched scimitar type to give excellent efficiency on both S-band and UHF frequencies. The antenna placement is shown in Fig. 2.2-19. The Apollo equipment is internally redundant on a modular basis and requires no external duplication of equipment. The UHF equipment is not internally redundant but is highly reliable; therefore, dual redundance is considered adequate.

<u>Tactical Air Communication and Navigation (TACAN) System.</u> TACAN systems provide the pilot and co-pilot with bearing and distance indications to selected ground TACAN stations. Audio tone identification (Morse code) of the selected ground stations is provided to the pilot and co-pilot headphones through the orbiter interphone system. In addition, the pilot and co-pilot may select and fly courses about the selected ground stations by following deviation indications on their HSIs. The pilot HSI will indicate how far the orbiter is away from a preselected course as well as indicate the position of the ground station from the aircraft. Bearing indications up to 360 degrees is indicated on HSIs with distance up to 300 nautical miles being indicated on the same indicators. The orbiter is equipped with two complete and independent TACAN systems whose components are located as follows:*

- Two RT units, located in avionics equipment bays
- Two antennas, located on the bottom fuselage
- Two antennas, located on the upper fuselage
- Two control panels, located on the center console.

*Each system contains three circuit breakers (C/Bs); a 28-Vdc, a 115-Vac and a 26-Vac, all of which are located on the C/B panel.

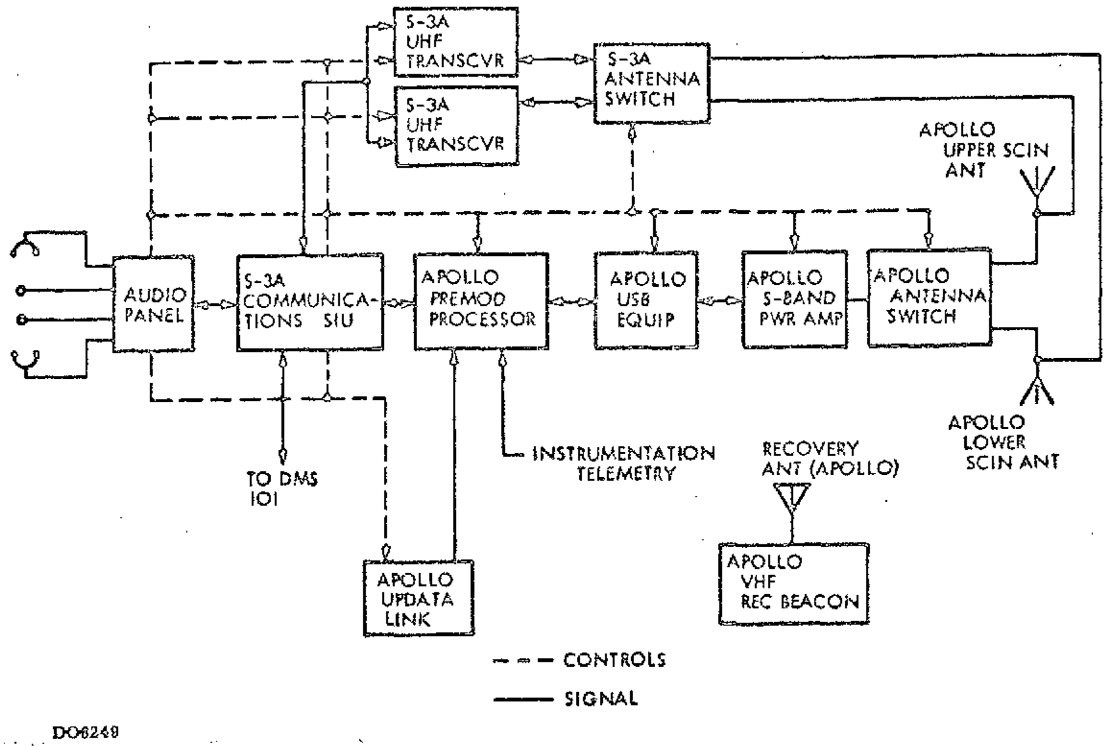


Fig. 2.2-18 Communications Subsystem

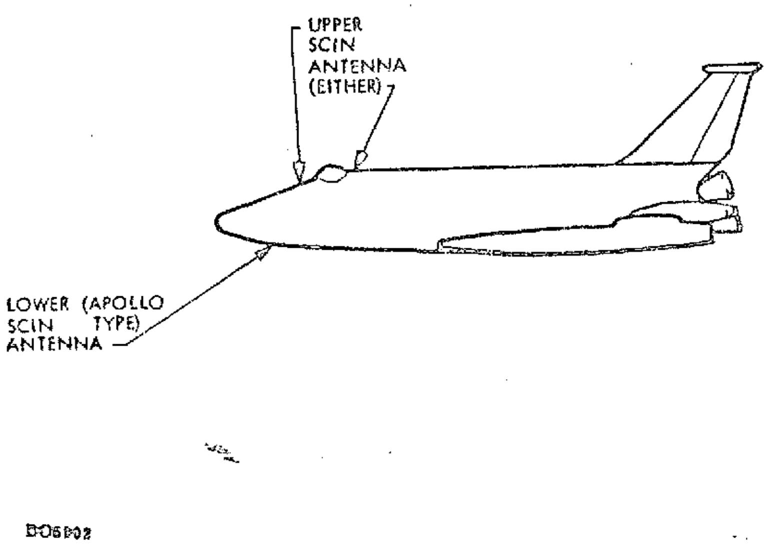


Fig. 2.2-19 Antenna Placement

The control panel contains controls for pilot operation of the system. Control functions are as follows:

- The knob marked VOL controls the audio volume level of the identification tone.
- The MODE switch allows selection of 126 channels in "X" mode and an additional 126 channels in "Y" mode when the surface beacons gain this capability.
- The TENS knob controls the first two digits and the units knob controls the last digit in the channel indication window.
- The indicator displays channels 1 through 126.
- TOP or BOTTOM antenna can be selected manually or automatically by the ANT switch.
- The center control switch controls off-on power and reciever-transmitter functions. Transmitter ranges of 70-200 and 300 nautical miles can be selected.
- The TEST switch energizes BIT circuits in the RT unit that automically checks the status of the system.
- GO/NO-GO lights indicate system status.
- Reference to the fault indicators or the RT unit locates the fault to a particular unit, i.e., RT unit, control panel.

Range, bearing, course deviation, and validity, are displayed on the pilot's and co-pilot's HSI when selected on the integrated navigation display selector. A desired course, referenced to magnetic heading, is set-in by the COURSE SET knob. This course is displayed in the COURSE window and by the course arrow. The deviation bar indicates deviation of the orbiter left or right of this desired course. Deviation signals are switched through the flight director computer (FDC).

The bearing pointer, outside the compass card, indicates bearing to the selected TACAN station. Distance to the station is displayed in the MILES window; the window is also masked when range information is invalid. The TO-FROM indicator points "to" the head of the course arrow when the selected course is within 90 degrees of the bearing to the station as indicated by the bearing pointer. The TO-FROM indicator points "from" the head of the course arrow when the selected course is more than 90 degrees from the bearing to the station, as indicated by the bearing pointer.

The receiver-transmitter performs the major functions of transmitting and receiving coded RF pulse pairs and single RF pulses, decoding the received pulses to recover bearing and range information, and performing bearing and range computations. Transmission and reception of signals is accomplished over a common antenna system (not part of the navigational set). The antenna system consists of two antennas that are switched at a 0.1-cps rate when the antenna selector switch is set for automatic operation and the system is not locked in range or bearing. The signal for switching antennas is developed by the receiver-transmitter.

TACAN units were selected as navigation aids for their built-in compatibility with the data management computer through the navigation data repeater and converter. This interface conveys system status and display information. The former is processed in the GPDC and the latter in the navigation data repeater and converter (NDRC) for selection by the pilot/co-pilot on the navigation display selectors.

Instrument Landing System (ILS). Provisions were made in the orbiter avionics for conventional instrument landing system (ILS) receivers as a navigation aid for horizontal ferry flights and for use throughout the initial phases of horizontal flight test. Several alternatives to the conventional ILS were investigated including:

- Autonomous Radar Approach Systems
- Precision Ranging Systems
- Microwave Landing Guidance Systems

The conventional ILS approach was adopted initially for the following reasons:

- The autonomous radar approach requires a radome of significant size and associated equipment of sufficient cost and complexity to rule them out on the basis of the cost and weight of the equipment, the software additions, the redundancy required, projected against reliabilities, and system development costs. The C-5A is the only aircraft planned to implement this automatic mode. There are operational requirements on the C-5A requiring it to land without ground aids. For this concept to hold true the onboard radar system must have exceptional resolution to "see" the end of the runway. The C-5A system now uses corner reflectors on the runway end for accurate radar approach calculations. For the shuttle application this approach was considered outside the realm of off-the-shelf philosophy.

- The precision ranging system approach requires the terminal facility to have at least three geometrically positioned transponder stations. These are at present located at no facilities to our knowledge. The cubic 100-4 is not presently in production. The positions of the airports across the CONUS, which can expect a shuttle visit are unknown at this time, so pricing the ground station costs would be indeterminate. At present the PRS is projected for spacecraft-type operation, requiring an antenna on top behind the cockpit area. Its use as a landing aid would require an additional antenna mounted on the bottom of the orbiter. This requirement, in addition to the added costs for redundancy and transponder stations, makes it an unattractive alternate. In addition, there has been no previous development on such a landing aid and it would therefore require extensive development (which violates the off-the-shelf principle).
- The microwave landing guidance system (MLGS) is the most attractive alternative, but has the fundamental drawback of disuse throughout the National Landing System. To be of use, numbers of ground stations would be necessary throughout the CONUS, at all selected sites for shuttle recovery. There is also no consensus on the FAA's part for a preferred MLGS. Studies are under way now on this and it is presently estimated that the MLGS will be operational within the next 10 years in the United States.

A survey of existing and planned landing guidance systems was completed at GELAC, in conjunction with RTCA SC-117 committee activities, and it points out that several development models for a microwave landing guidance system (MLGS) are available for use. These include the Honeywell STATE, LFE PAALS, Bell SPN-42, AIL AILS, AIL C-SCAN, and the AIL SHORSCAN systems. Any one of these systems is capable of multipath guidance; however, none have been throughly tested against the requirement for the microwave landing guidance system (MLGS) preferred by the RTCA.

In the absence of a readily available alternative, ILS receivers meeting the requirements of ARINC Characteristic 578 were chosen for the initial horizontal flight tests. To prove the high glideslope power-off autoland prior to FVF, inclusion of a development model MLGS at a selected ground facility, e.g., Edwards AFB would be recommended, based on trade study results evaluating several candidate systems against SC-117 committee requirements. As soon as they become available, the production MLGS should be incorporated in the shuttle and at all prime and alternate ground facilities.

2.2.1.3 Electrical Power. The orbiter electrical power subsystem (EPS) consists of two elements: (1) power generation and (2) power distribution, including conditioning and control. Power is generated to provide vehicle average loads, peak loads, and the total mission energy requirement. EPS design requirements and the approach to the phased development of the Mark I orbiter are described below. The groundrules for the EPS design are listed in Table 2.2-9.

2.2.1.3.1 Power Requirements. Initial Mark I shuttle test flights will consist of horizontal atmospheric flights and subsequently lead to vertical flights into orbit. The horizontal flight avionics requirements are less than for orbital flight; hence only aircraft subsystems are planned for use in order to reduce annual costs. The aircraft EPS configuration selection depends on the spacecraft orbital requirements and the orbital EPS configuration.

User power requirements are summarized in Fig. 2.2-20 for a 7-day orbital mission and include a payload power requirement provided by NASA for the LMSC Alternate Avionics System Study. While payload power requirements have not been well established, a design that anticipates the payload power requirement is of interest. This requirement is defined as 3 kW average, 6 kW peak for periods of orbiter low power requirements and 500 W average, 800 W peak for periods of orbiter high power requirements. The spikes in Fig. 2.2-20 indicate worst-on-worst additions of peak power requirements.

Table 2.2-10 shows average power requirements of the major users for horizontal flight and orbital missions. Aircraft subsystems and airbreathing engine fuel booster pumps are major power users during atmospheric flight. Spacecraft subsystems and pumps for circulating main rocket engine cryogenic propellants are major power users during launch and ascent phases of shuttle operation. On orbit, spacecraft subsystems and the payload are the major power users. The lower portion of Table 2.2-10 shows total electrical energy required for three different missions. The 12-hr and 7-day missions include payload power requirements described above.

Table 2.2-9

ELECTRICAL POWER SUBSYSTEM DESIGN GROUNDRULES

- Emphasize low-cost minimum technology design
- First horizontal flight may omit spacecraft equipment required only for orbital flight
- FO/FS design not required
- Make maximum use of existing spacecraft and aircraft equipment
- Control environment of aircraft equipment that is used in the orbiter
- Minimize extent of required equipment design modification or development
- Select Mark I design for ease of growth to Mark II design
- Mark II development not to be over 50 percent more in cost than Mark I

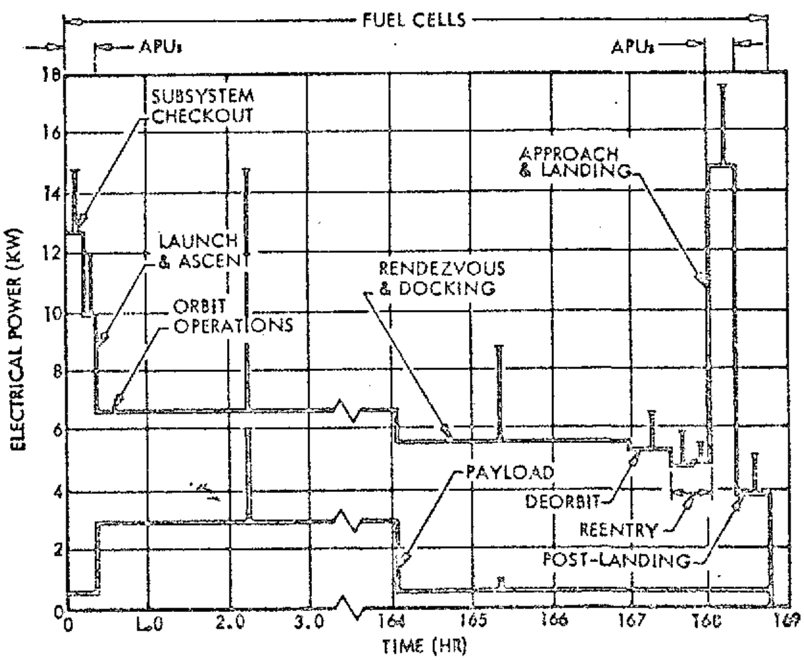


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Fig. 2.2-20 User Electrical Power Requirement Summary – 040A Orbiter

Table 2.2-10

ELECTRICAL POWER SUBSYSTEM REQUIREMENTS

POWER USERS	HORIZONTAL FLIGHT	ORBITAL FLIGHT		
		LAUNCH AND ASCENT	ORBIT OPERATIONS	APPROACH AND LANDING
AIRCRAFT SUBSYSTEMS	5 KW			5 KW
ABES FUEL BOOSTER PUMPS	8 KW			8 KW
ROCKET ENGINE PROPELLANT CIRCULATION FOLLOWING LOADING		4.5 KW		
SPACECRAFT SUBSYSTEMS		4 KW	4 KW	
PAYLOAD			3 KW	

MISSION		ENERGY	
3 HR	HORIZONTAL FLIGHT	42	KW-HR
12 HR	ORBITAL PAYLOAD LAUNCH	106	KW-HR
7 DAY	LOGISTICS	1196	KW-HR

2.2.1.3.2 Aircraft EPS. Figure 2.2-21 shows the aircraft EPS configuration for the first Mark I Orbiter horizontal flight. The power generation subsystem is composed of three 200/115 Vac spray oil-cooled generators, driven directly by three auxiliary power units (APUs) which also drive hydraulic pumps for aero-control surface movements. Generator speed is maintained within ±5 percent of the nominal 12,000 rpm, and they are not operated in parallel. One generator will support the total load requirement, and switching to replace failed units is employed. Two 10 amp-hour NiCd batteries are provided for emergency EPS control power when generators are not running.

The power distribution, conditioning, and control subsystem includes generator control units that provide generator protection, control, and voltage regulation. Three-phase, four-wire ac power is supplied to the ac generator distribution unit, which distributes power to the airbreathing engine fuel booster pumps and three transformer-rectifier (TR) units rated at 200 amps each. The TR units supply 28 Vdc power to the main dc distribution unit which, in turn, distributes power to the power distribution units throughout the vehicle. One TR unit can support the vehicle load. Four static three-phase

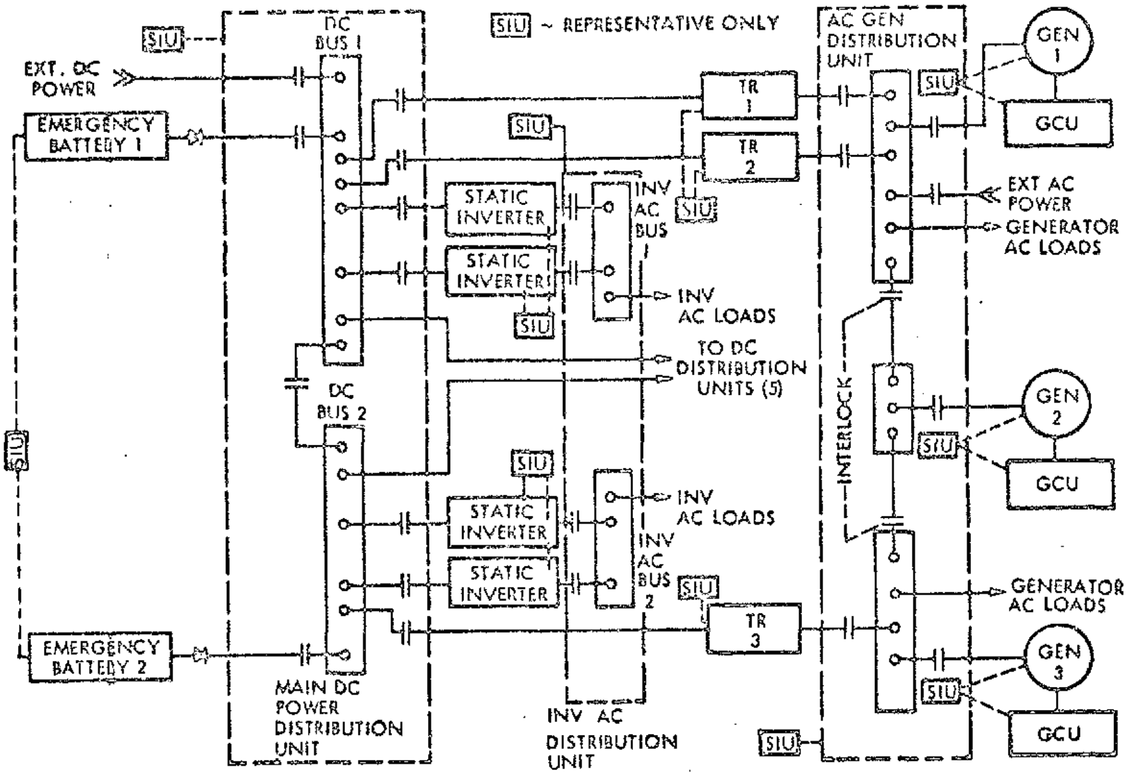

Fig. 2.2-21 Electrical Power Subsystem Schematic – Aircraft

115 Vac, inverters supply centralized conditioned ac power to the inverter ac distribution unit for avionics users. The units do not operate in parallel, and one unit in each set of two is required to support the load.

Distribution units for dc, inverter ac, and generator ac power contain contactors and circuit breakers that are controlled by hardwires and used to connect power sources and conditioning equipment to buses and to protect buses from distribution system and user faults. Reverse current relays are contained in these units to protect fuel cells and transformer rectifiers. Sensing devices are also located in the distribution units to automatically switch defective equipment off buses and place backup equipment on buses.

Two redundant buses each are used for dc distribution, inverter ac distribution, and generator ac distribution systems. Subsystem Interface Unit (SIU) interfaces indicated in Fig. 2.2-21 will be located in one SIU forward and one SIU aft to handle combined power generation and power distribution, conditioning, and control subsystems. Since the EPS functions are safety critical, most of the controls and instrumentation will be hardwired to the crew stations for manual override of automatic controls. The data management subsystem will provide selected EPS sequencing and configuration changing functions, using EPS software inputs and the SIU interface with the EPS control components. Operational instrumentation will be used by the data management subsystem to provide redundancy management for selected EPS components.

2.2.1.3.3 Spacecraft EPS. The spacecraft EPS is designed to meet increased requirements of orbital flight by the addition of equipment to the horizontal flight configuration. As shown in Fig. 2.2-22 , three H_2-O_2 fuel cells with reactant tankage are added to the horizontal flight EPS configuration to provide a lightweight primary power source for the shuttle orbital mission. Three fuel cells provide 28 Vdc power to the orbiter from prior to liftoff to the completion of the landing phase. Each fuel cell is rated at 8 kW continuous power. A fuel cell can provide up to 100-percent overload power for short periods and, thus, fuel cell fail operational/fail safe capability is provided. The cryogenic tankage provides supercritical storage of the H_2 and O_2 reactants in a dual set of tanks that will provide fail safe capability for the 7-day shuttle mission.

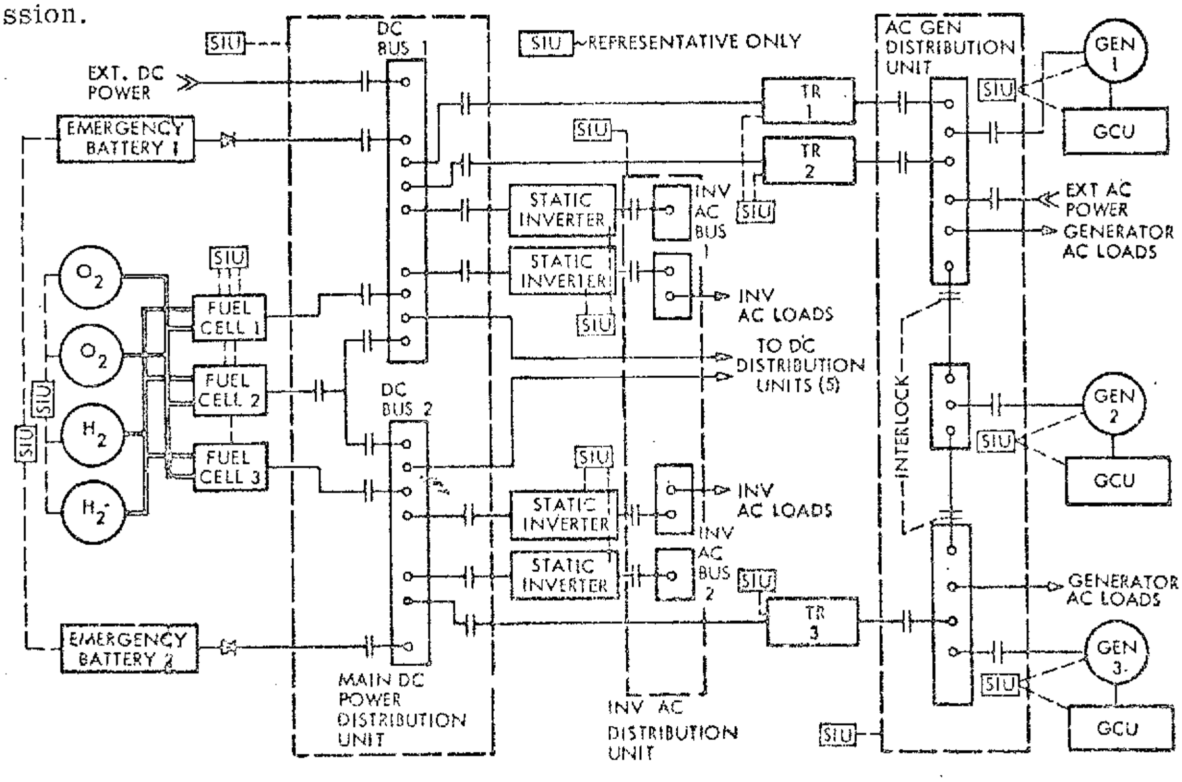

Fig. 2.2-22 First Vertical Flight EPS – Mark I Orbiter

The high average power requirements shown in Fig. 2.2-20 at the beginning and end of the orbital mission were previously identified with: (1) main rocket engine propellant circulation (four J-2S H_2-O_2 engines) from propellant loading to orbiter ignition, and (2) airbreathing engine fuel booster pump operation during orbiter landing operation. Rather than sizing the fuel cells for these high average requirements, the ac generators on the Auxiliary Power Units (APUs) are used to supply these loads. The APUs are operated during the short booster phase to assure readiness for orbiter rocket engine gimballing. The APU operation during approach and landing is similar to that for the horizontal flight test program.

Transformer-Rectifier (TR) units used in the aircraft EPS are not used in the orbital missions, but remain in the spacecraft EPS for ferry missions. The orbiter ferry mission is accomplished by using the main dc distribution system to provide power to much of the same equipment used in entry and landing phases of the orbital mission. To conserve the available operating life of the fuel cells, the ac generators on the APUs are used to supply energy to the TRs which convert ac to dc and supply the main dc power distribution system. The APUs are running during the ferry mission to provide hydraulic power and to drive the ac generator for jet engine fuel booster pump operation.

Use of inverters, power distribution units, and buses is the same as in the aircraft EPS. The interfaces with hardwired displays and controls and with the data management subsystem are increased to include the fuel cells and cryogenic storage system.

2.2.1.3.4 EPS Equipment Selections. Alternate approaches and rationale for selection of major EPS equipment are summarized in Table 2.2-11.

Fuel Cell. A new design fuel cell was selected over the startup of the 1.4 kW Apollo fuel cell module program. At similar cost between the two candidates, a new low cost-minimum technology 2000-hr-life, 8-kW fuel cell module can be developed. The fuel cell characteristics are shown in Table 2.2-12.

Table 2.2-11

EPS EQUIPMENT ALTERNATE APPROACHES AND SELECTION

CANDIDATES	RATIONALE
APOLLO FUEL CELLS $\boxed{\text{NEW DESIGN FUEL CELLS}}$	COST OF PROGRAM TO START UP APOLLO SIMILAR TO DEVELOPMENT OF LARGER MODULE DESIGN
APOLLO REACTANT TANKS $\boxed{\text{AAP REACTANT TANKS}}$	COST OF INDIVIDUAL TANKS SIMILAR BUT FEWER OF THE LARGER AAP DESIGN ARE REQUIRED
400 Hz STATIC INVERTERS, 115 VAC $\boxed{\text{3}\phi\text{ APOLLO - 1250 VA}}$ 3ϕ PERSHING - 500 VA 1ϕ L-1011 - 750 VA	3ϕ POWER DESIRED FOR LIGHTER MOTORS AND EXISTING CRYOGEN STORAGE SYSTEMS. LARGE VA MODULE DESIRED, PARTICULARLY WITH NO PARALLEL OPERATION CAPABILITY; SPACE-QUALIFIED FOR APOLLO
TRANSFORMER-RECTIFIERS (28V) LARGE NO. OF AIRCRAFT AIR-COOLED DESIGN $\boxed{\text{200-AMP - P3C DESIGN}}$	LOW WEIGHT (18 LB) AND BUILT TO MILITARY SPECIFICATIONS
OIL-COOLED BRUSHLESS AC GENERATORS $\boxed{\text{MODIFIED S-3A - 40 KVA}}$ 4QN 60F - 75 KVA (S-3A) 28B-282-2 - 60 KVA (F-14)	20 KVA MODULE SIZE DESIRED

Table 2.2-12

FUEL CELL SYSTEM CHARACTERISTICS

Description	Data
Type	Capillary matrix, modified Bacon cell
Design Life (hr)	2000
Continuous Power Rating (kW)	8
Voltage at 8 kW (Vdc)	30
Reactants	Oxygen and hydrogen
Number of Cell Sections	32
Cooling Mode	Fuel cell liquid loop and HX in vehicle liquid loop
Operating Pressure (psia)	60
Operating Temperature ($^{\circ}$F)	160 to 200
Dimensions (in.)	15 x 15 x 36 (4.7 ft^3)
Voltage Regulation (2 to 14 kW)	± 12 percent
Module Weight (lb)	320

Cryogenic Reactant Storage. Apollo Applications Program (AAP) reactant tanks were selected over the Apollo tanks for their larger capacity (55 lb H_2 vs 28 lb and 456 lb O_2 vs 320 lb) and the fewer required tanks to provide desired capacity. The status of the discontinued AAP tank program and costs to provide a minimum technology design are still being reviewed with the vendor to support this selection or indicate another choice. The reactant storage system is described in Table 2.2-13.

Static Inverter. The Apollo three-phase inverter was selected for conditioned 400 Hz, 115 Vac power because of its large module power, 1250 Vac. The three-phase power is desired to be compatible with light three-phase motors for the cryogen storage and the fuel module systems. Separate single-phase and three-phase inverter equipment is not desirable from a shuttle simplicity standpoint. Three-phase inverter operation in parallel is not possible with existing equipment, so that a given inverter ac bus can only be powered by one inverter at a time; this results in a large module size requirement with the present using equipment selections. The Apollo inverter has cold plate cooling and has been qualified for space. The inverter ac power requirement to be provided is shown in Fig. 2.2-23.

Table 2.2-13

FUEL CELL REACTANT STORAGE DATA – TWO TANK SETS

Specific Reactant Consumption (SRC) (lb/kW-hr)	0.85
O_2 Required for ECLS (lb)	64
Fuel Cell Energy Required (7-Day Mission With Payload) (kW-hr)	1191
H_2 Required for Power (lb)	112.0
O_2 Required for Power + ECLS (64 lb) (lb)	959.6
Additional H_2 Required for Fail Safe* (lb)	7.0
Additional O_2 Required for Fail Safe (lb)	59.4
Available H_2 Required in One Tank (lb)	59.5
Available O_2 Required in One Tank (lb)	509.5
H_2 Available in 39 in. O.D. Tank (lb)	53.5
O_2 Available in 33 in. O.D. Tank (lb)	468
Allowed Payload Operation Time at 3 kW Average Power With Selected Tanks (hr)	120
Fuel Cell Energy Provided (kW-hr)	1062
Weight of H_2 Loaded, Total (lb)	111
Weight of O_2 Loaded, Total (lb)	936
Weight of H_2 Tanks (Empty) (2 each)	296
Weight of O_2 Tanks (Empty) (2 each)	406

*12 hr in orbit plus deorbit and landing

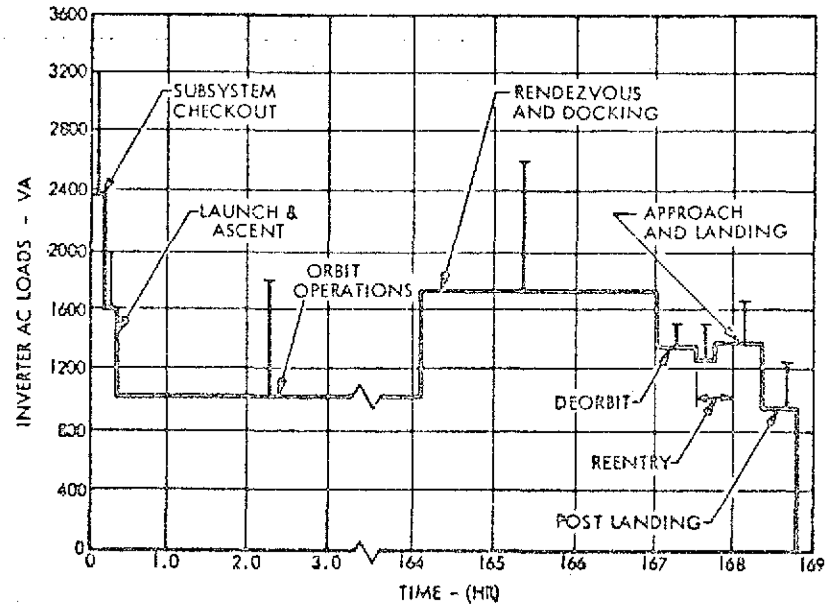

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Fig. 2.2-23 User Inverter AC Requirements – 040A Orbiter

Transformer-Rectifiers. A large number of TR unit designs exist for air-cooled, aircraft operation. The P3C 200-amp unit was selected because it has low weight (18 lb), has its own fan, and is built to military specifications. The TR units can be operated in parallel. These units require installation in an earth atmosphere environment in the orbiter. The characteristics of this unit are shown in Table 2.2-14.

AC Generators. A 40 kva spray oil-cooled brushless generator was selected to provide environment free operation at as close as possible to the desired 20 kva module size. This unit has not yet been flown, but six units have been built and tested by the vendor. The design is based on the 60/75 kva S-3A unit. The generator requires the APU gear box to provide the drive-end bearing and cooling oil.

Generator Control Unit GCU. Several aircraft generator control units are available for use with the orbiter. The design used in the S-3A was selected for compatibility with the selected ac generator. It is powered by the dc bus. One GCU is used with each generator. The GCUs require installation in an earth atmosphere environment in the orbiter.

Table 2.2-14

TRANSFORMER-RECTIFIER CHARACTERISTICS

Manufacturer	Wagner Electric Co.
Model No.	28VS200Y
Input – ac	
Voltage	195 to 210 Vac
Current – Full Load	17 amp
Frequency	380 to 420 Hz
Phase	3-phase, 4-wire
Output – dc	
Voltage – Nominal	28 Volts
Current	200 amp
Ripple	1.5 percent RMS
Overload	250 percent for 1 minute
Efficiency	85 percent minimum
Power Factor	.95 percent
Cooling	Integral fan
Size	11 in. x 6 in. x 7 in.
Weight	18 lb
Transformer, Primary	Wye
Transformer, Secondary	Delta-wye
Rectifiers	Silicon diode

Power Cabling. Aluminum cabling will be used between the fuel cells and the main dc power distribution unit and between the dc ground power receptacle and the main dc power distribution unit. Round wire copper cabling with Kapton insulation will be used elsewhere for power distribution.

Power Switching. Circuit breakers and switches will be located in the cabin atmosphere to control most of the power to individual users. Hermetically sealed contactors will be used to switch major EPS components on and off line. Remote controlled circuit breakers are proposed for dc and generator ac bus protection in the aft portion of the orbiter. Environmental design problems remain to be solved for conventional circuit-breakers; solid state devices will be used where possible.

2.2.1.3.5 Electrical Power System Redundancy. The level of redundancy proposed for the Mark I orbiter is shown in Table 2.2-15. The fuel cells, ac generators, generator control units, and transformer-rectifiers provide fail operational/fail safe capability. The cryogenic tanks, static inverters, and distribution buses provide fail safe capability only.

Table 2.2-15

ELECTRICAL POWER SUBSYSTEM REDUNDANCY

ITEM	NO. REQ'D FAIL SAFE	NO. SELECTED	RATIONALE
FUEL CELLS (8 KW)	2	3	LOWEST RELIABILITY UNIT IN EPS AND HIGHEST RISK OF CURTAILED MISSION – LIMITED DATA
O_2 REACTANT TANK	2	2	HIGH RELIABILITY – EACH TANK HAS FAIL-SAFE RESERVE LOADED – TANKS ARE EXPENSIVE
H_2 REACTANT TANK	2	2	
3ϕ STATIC INVERTERS (CENTRAL) – 1250 VA	4	4	FAIL-SAFE – ACCEPTABLE WITH LIMITED AVAILABLE INVERTER MODULE SIZES
AC GENERATORS AND GCUs – 40 KVA	2	3	3 APUs AVAILABLE – COST FOR FO/FS NOT SIGNIFICANT
TRANSFORMER RECTIFIERS (200 AMPS) (HORIZONTAL TEST AND FERRY FLIGHT ONLY)	2	3	FO/FS SOURCE OF DC POWER POWER DESIRABLE
AC AND DC BUSES	2	2	TWO BUSES PROVIDE HIGH RELIABILITY

D05710

Cryogenic Tanks. Sufficient reserve reactants are loaded in each O_2 and H_2 tank to provide sufficient energy for crew safety, should one tank fail. The high reliability of the tank systems plus the high cost of these systems justifies the fail safe design rather than FO/FS.

Static Inverters. Four static three-phase inverters are used because available module sizes are smaller than desired for the inverter ac power requirements. Since available three-phase inverters cannot be operated in parallel, the load is divided between two sets of buses. Each set is supplied by one of two inverters. If one inverter in a set fails the other inverter is placed in operation to provide fail safe capability. This mode of operation was selected over increased weight and cost of six units for FO/FS design.

Power Distribution Buses. Two redundant buses each are provided for dc, inverter ac, and generator ac distribution for fail safe operation, due to the high reliability of the buses and the high weight of bus systems.

The location and number of the major EPS components in the Mark I orbiter are shown in Fig. 2.2-24. The ac generators are located on the APUs. The APUs are located aft in the orbiter to be near the hydraulic power users. The aircraft type generator control units (GCU) are located in the forward controlled environment area as are the aircraft type transformer-rectifiers. The GCUs are low-power solid-state devices and require little cooling. The transformer-rectifier units will require significant air cooling in horizontal flight. The five dc distribution units are located to serve centers of significant power use. The physically separated dc buses are routed to serve each dc distribution center.

The electrical power system weights are summarized in Table 2.2-16.

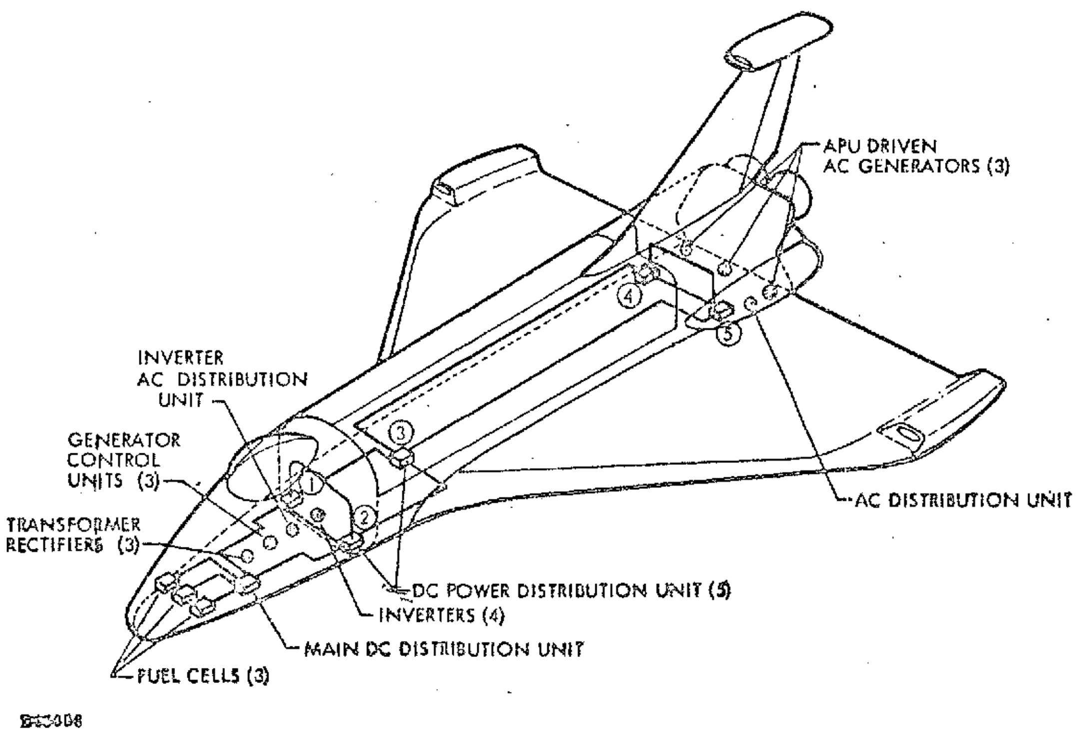

Fig. 2.2-24 Electric Power Generation and Distribution Configuration – O4OA Orbiter

Table 2.2-16

MARK I ORBITER AVIONICS WEIGHTS SUMMARY

Subsystem	FHF Wt. (lb)	FVF Man Wt (lb)	FVF Unman Wt (lb)	Operational Wt (lb)
Guidance, Navigation and Controls	1119	2018	2018	2275
Communication and Navigation Aids	316	436	436	531
Electrical Power Generation, Control and Distribution	2332	3696	3696	3696
Displays and Controls	751	1049	1049	717
Data Management	555	665	665	665
Instrumentation	493	664	693	376
Sub-Total	5566	8528	8528	8260
Installation (10%)	556	853	853	826
TOTAL	6122	9381	9381	9086

2.2.1.4 Controls and Displays. The objective of the control and display (C&D) system is to significantly reduce pilot workloads, as shown by Figs. 2.2-25 and 2.2-26. Three alternate concepts were developed (Table 2.2-17); the baseline concept (Alternate C) is described in this paragraph. This selected baseline system offers a Space Shuttle configuration that is flyable by a two-man crew in the operational phase and one-man emergency modes. Also, the configuration permits measurable reduction of the total instrument panel area and eliminates the requirement for a third crewman.

The other two alternatives (A and B) are discussed briefly in following paragraphs and are presented in more detail in Part D, Appendixes 2 and 3. Each alternative was evaluated against the basic data management system, flight dock volume, crew size and complement, degree of onboard autonomy, inflight checkout, redundancy, power, ground support requirements, operability, and developmental requirements.

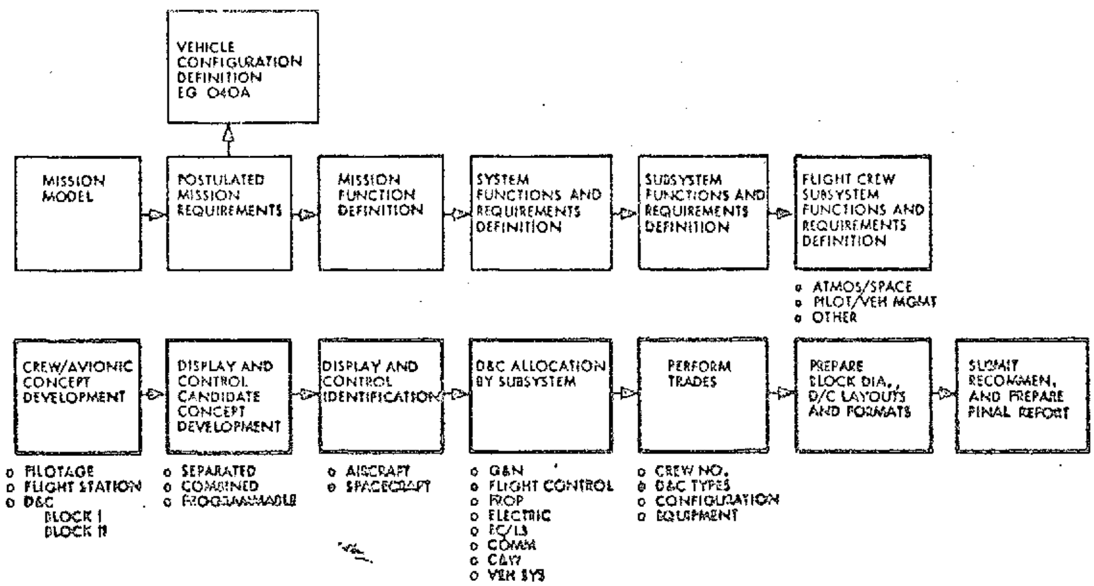

Fig. 2.2-25 Control and Display Functional Approach

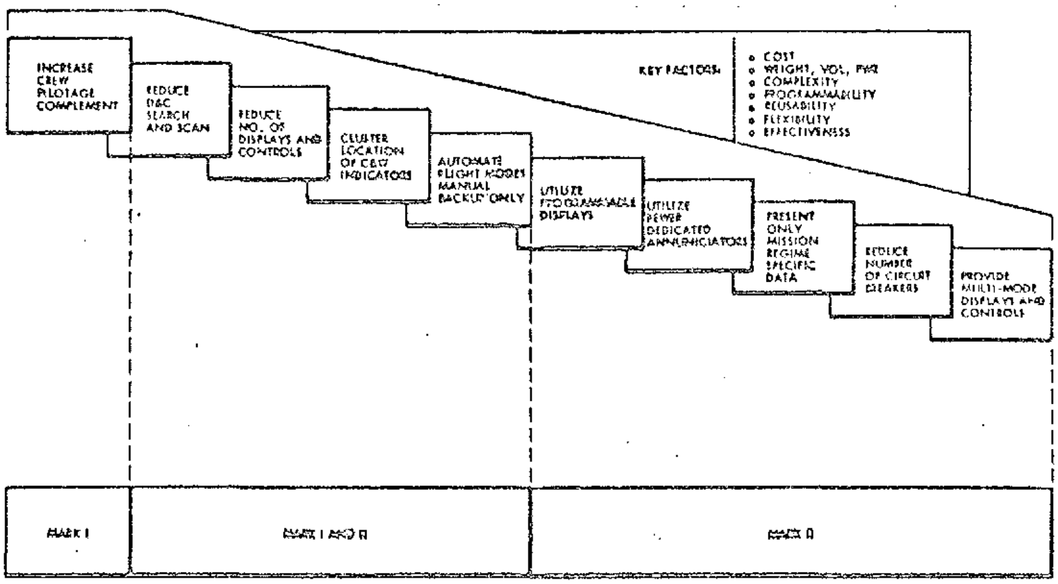

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Fig. 2.2-26 Considerations for Pilot Workload Reduction

Table 2.2-17

CANDIDATE CONTROL AND DISPLAY AVIONIC CONCEPTS

ALTERNATE C	BASELINE – PROGRAMMABLE CONTROL AND DISPLAY STATIONS o AIRCRAFT/SPACECRAFT OFF-SHELF INSTRUMENTS o ADVANCED AIRCRAFT AND SPACECRAFT TYPE INSTRUMENTS o INCREASED DATA BUS AND COMPUTER CAPABILITY o CATHODE RAY TUBE RANDOM ACCESS DATA o AMALGAMATED CONTROL AND DISPLAY CONFIGURATION o DEDICATED CONTROL AND DISPLAY MANUAL MODE BACKUP
ALTERNATE A	SEPARATE AIRCRAFT AND SPACECRAFT CONTROL AND DISPLAY STATIONS o SEPARATE STATIONS o SAME STATION – SEPARATE PANELS o SAME STATION – SEPARATE SUBPANELS o AIRCRAFT/SPACECRAFT OFF-SHELF INSTRUMENTS o NO DATA BUS, MINIMUM COMPUTER CAPABILITY o DEDICATED CONTROLS AND DISPLAYS
ALTERNATE B	COMBINED AIRCRAFT AND SPACECRAFT CONTROL AND DISPLAY STATIONS o MIX OF AIRCRAFT AND SPACECRAFT CONTROLS AND DISPLAYS IN STATION o AIRCRAFT/SPACECRAFT OFF-SHELF INSTRUMENTS o SOME DATA BUS AND COMPUTER CAPABILITY o DEDICATED CONTROL/DISPLAY MANUAL MODE BACKUP o SOME DUAL MODE INSTRUMENT USAGE/CAPABILITY

D02483

2.2.1.4.1 Rejected Alternate Concepts. The Alternate A concept for separate aircraft and spacecraft C&D stations employs (1) off-the-shelf large aircraft and (2) Apollo LM/CM C&D instruments (Fig. 2.2-27) in a dedicated system approach without programmable display capabilities. This requires increased C&D instruments and associated greater panel area and results in the highest amount of pilot workload of the three alternates considered.

The combined aircraft/spacecraft C&D station approach (Alternate B) is similar to the Alternate A concept but includes a malfunction, detection, analysis, and recording (MADAR) onboard checkout capability, as on the C-5A aircraft, with dedicated controls and displays (Fig. 2.2-28). All critical components are duplicated and hardwired; a mixture of off-the-shelf aircraft and spacecraft instruments are used for the C&D panel. This approach significantly reduces (over Alternate A) the quantity of dedicated caution and warning indicators needed for pilotage, panel area requirements, and the pilot workload level, but is still less desirable than the Alternate C approach selected as baseline.

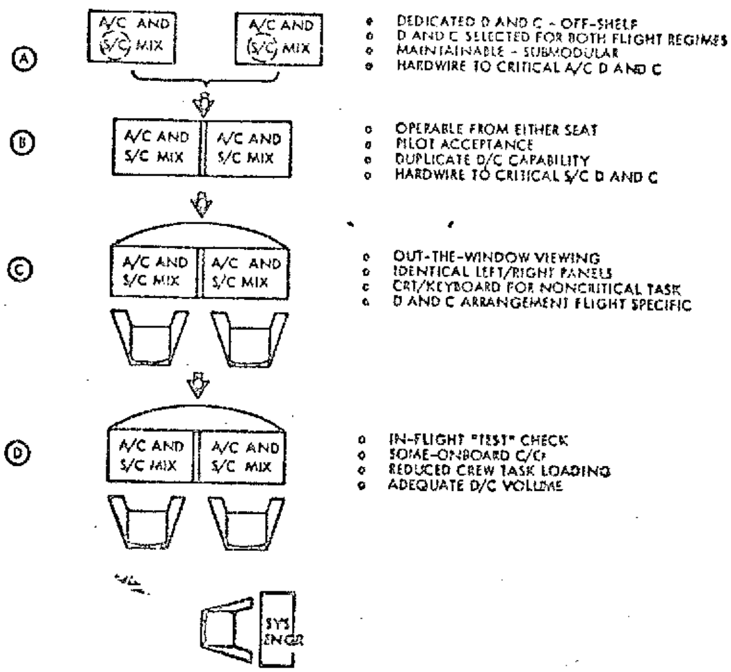

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Fig. 2.2-27 Alternate A – Separated Control and Display Approach

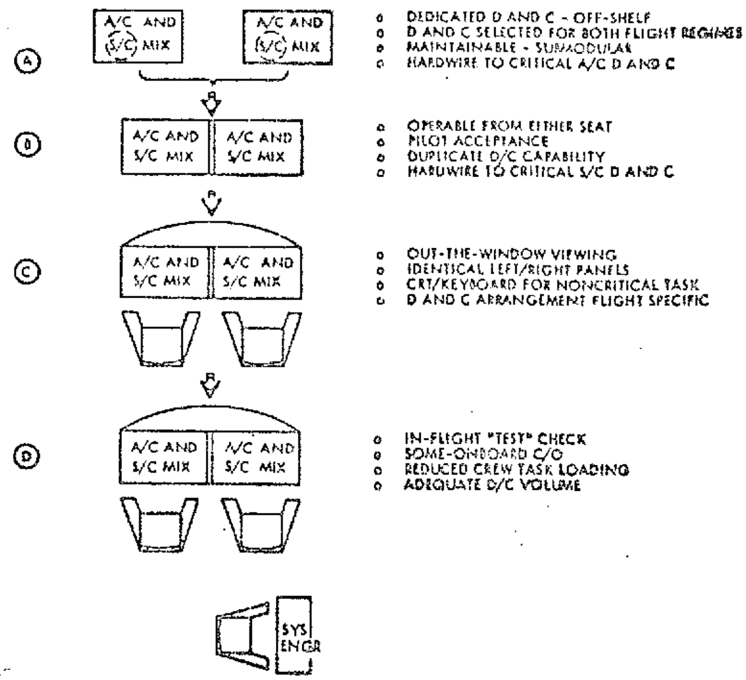

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Fig. 2.2-28 Alternate B – Combined Control and Display Approach

2.2.1.4.2 <u>Baseline (Alternate C) Approach.</u> The selected baseline concept provides programmable controls and displays (Fig. 2.2-29) and the increased flexibility for instrument use. Panel area is considerably less than for the other two concepts, and the third crew member (System Flight Test Engineer) is not required for flight operations. Also, the panel area and crew-size reduction reduces the total dedicated volume and number of panels within the flight-deck area. The baseline configuration features, resulting in selection over the other two candidates, include the following:

- Flown by a two-man crew in the operations phase; one-man in emergency modes
- Measurably reduced total instrument panel area and requirement for a third crewman
- Decreased C&D instrument weight
- Permitted onboard checkout, fault isolation, and redundancy management (COFIRM)

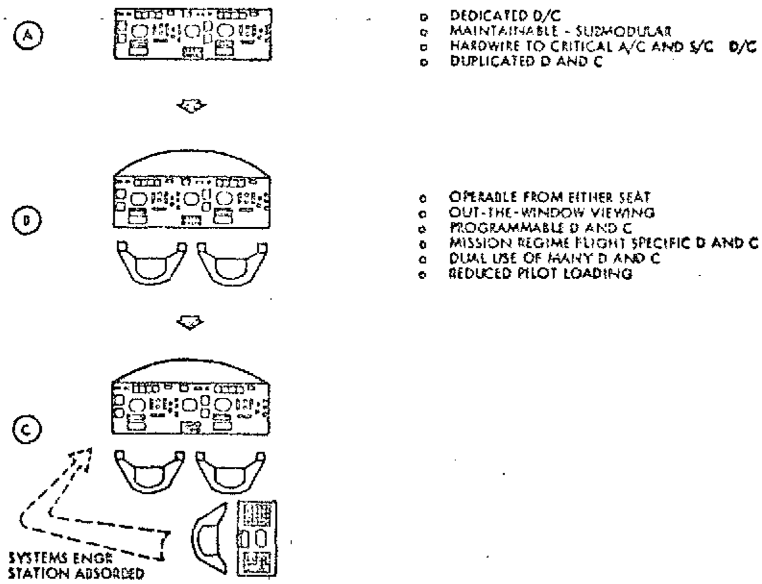

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Fig. 2.2-29 Programmable Control and Display Approach

- Simplified integration with the basic S-3A data management system
- Achieved an FO-FS redundancy for crew safety
- Permitted excellent utilization of computer capability for C&D programmable functions
- Utilized nearly all instruments as totally off-shelf items
- Provided extensive inflight and growth flexibility for "softly" defined missions
- Had compatibility with data obtained in NASA-sponsored Space Shuttle simulation programs at LMSC

The baseline concept, on the basis of using off-the-shelf S-3A aircraft data management system hardware, is considered to have minimal risk; particularly since all but two major C&D instruments are flight-proven. Spacecraft instruments from the

Apollo LM/CM can be employed and also are flight-proven. Aircraft instruments are readily available from the L-1011, S-3A, C-5A, YF-12, and other existing military and commercial systems.

Cathode ray tubes (CRTs), which are fully operational on the S-3A, employ alpha-numeric, pictorial, diagram, chart, map, procedural, real-world image, etc. to communicate data to the pilot. The flight management CRT data are used for the first time as a real inflight information tool. These data will be available four years ahead of the first orbiter horizontal flight test date. Data entry keyboards (e.g., Apollo, C-5A, and S-3A) provide a common inflight device with extensive software programs and routines. Thus, with ground test, inflight test, and operation flight experience gained in previous developments, the level of confidence and availability of these instruments can be assured and C&D subsystem equipment requirements can be met totally within the state-of-the-art.

2.2.1.4.3 Requirements for the Control and Display Subsystem - Mark I. The C&D development approach necessitates generation of basic subsystem drivers and requirements. Table 2.2-18 presents the initial Mark I Orbiter C&D subsystem drivers; of these, only flight-crew ejection for the test/verification flights remain "soft" because of the lack of specifications in the test program. (Note - if intact abort is imposed on the test/operations program, crew ejection will not be required.) Additional Mark I requirements are listed in Table 2.2-19.

The basic Mark I C&D subsystem with subsequent growth to Mark II will be pilotable by a two-man flight crew during all flight regimes and flyable by one crewman under emergency conditions. For this subsystem, the basic underlying requirement is the development of an approach and ultimately the hardware to permit sufficient capability and redundancy for providing safe crew return and general mission success. Additional requirements for Mark II are presented in subsequent paragraphs.

Table 2.2-18

MARK I AVIONIC CONTROL AND DISPLAY SUBSYSTEM DEVELOPMENT DRIVERS

	FLT TEST	OPS
1. FLYABLE FROM EITHER SEAT	X	X
2. MISSION CRITICAL C&D HARDWIRED	X	X
3. NO PILOT/COPILOT TRANSLATION IN:	X	X
ATMOSPHERE: o TAKE-OFF; o FINAL APPROACH; o LANDING; o ROLL-OUT		
EXO-ATMOSPHERE: o FINAL COUNT-DOWN; o LAUNCH AND ORBIT INSERTION; o TERMINAL RENDEZVOUS AND DOCKING; o RE-ENTRY		
4. DUPLICATE MISSION CRITICAL C&D INSTRUMENTS	X	X
5. CONFIGURATION GROWTH POTENTIAL TO BLOCK II AVIONICS		X
6. MODULAR DESIGN	X	X
7. INITIAL REQT - ATMOS C&D: PROVISION FOR SPACE C&D	X	
8. OUT-THE-WINDOW VISION REQUIRED (ATMOS); DESIRED (EXO-ATMOS)	X	X
9. FLIGHT CREW EJECTION (HORZ & VERTICAL TEST)	X	
10. USE OF OFF-SHELF C&D INSTRUMENTS	X	X
11. CREWMAN STRAPPED IN SEAT; HIGH G LOADS PROHIBIT EXTENSIVE ACCURATE ARM/HAND MOVEMENT	X	X

D05587

Table 2.2-19

MARK I CONTROL AND DISPLAY REQUIREMENTS

1. Flight Control

- o Manual override of all flight critical functions (crew safety) and modes
- o Automated functions to the degree required to meet safety, performance accuracies, energy management economy, reaction time, etc.
- o Selected automatic or manual control by flight crew for nominal or non-nominal flight modes or contingencies
- o Ability of shifting flight control authority from flight deck to other stations for on-orbit operations (e.g., docking or payload handling)
- o Selected manual or automatic functioning of MAINS, OMS, RCS, APU, AND ABES by crew
- o Selected manual or automatic vehicle configuration control by crew
- o Selected autopilot/land flight mode and regain of manual control at any time by crew

2. Flight Data Presentation

- o Flight management information automatically programmed by mission regime
- o Anomalies automatically called to the crew's attention on C&W indicators and, then, more definitive data presented selectively elsewhere
- o Data not mission-regime specific not presented but capable of being requested by the crew
- o Routine and recurring data not mission-critical/specific located out of the critical visual cone or provided in form for callup
- o All major subsystems (10 to 12) with dedicated annunciators for C&W
- o Mission-critical display data presentable in summary form on other than dedicated displays
- o Critical flight information always within the easy scan and forward-viewing visual envelope of the pilot/copilot

Table 2.2-19 (Cont'd)

3. Operations Control

- Vehicle configuration control manual backup provided the crew
- Manual control to all crew and flight safety functions provided
- Noncritical repetitive tasks, resulting in crew functional constraints, automated with consideration for keyboard override to degree required
- Any vehicle function to the subsystem level enabled or inhibited by flight crew
- Normal subsystem functions need not be continuously controlled by the crew
- Flight control during on-orbit operations, relative to rendezvous/docking and payload functions, may be remoted depending on vehicle configuration/layout
- Critical function safety interlock provided to prevent accidental operation

4. Crew Caution and Warning

- Malfunctions, deviations, or out-of-tolerance conditions automatically displayed to the crew, thus eliminating crew dedicated/repetitious checkout tasks
- Major subsystem status available to the crew for both inflight and ground operations
- Limited but critical booster status (and vice-versa) presented to the orbiter crew from crew boarding through launch and staging
- All major subsystem malfunctions, deviations, or out-of-tolerance conditions presented to crew (those influencing crew safety or mission success)
- Supplementary diagnostic data and control capability to isolate the problem, ascertain its nature and impact, and effect appropriate action provided by the crew
- Data provided permitting the crew to compare current parameters with discrete range limits for purposes of go versus no-go decision making
- Initial presentation of major subsystem malfunction(s) or out-of-tolerance condition(s) provided for in the direct forward instrument view of the crew
- Major system status provided through dedicated C&W annunciators, located just off-center of the main center-line of the pilot's head axis

2.2.1.4.4 Control and Display Concept Development – Mark I.

Definition of Pilot Functions and Identification of Controls and Displays (C&D). Concurrent with development of basic approaches (Concepts A, B, and C), significant effort was allocated to delineate gross pilot functions and relevant controls and displays. Table 2.2-20 is an example of data developed to specify gross functions for GN&C for space and atmospheric flight regimes. Also, C&D applicable from the Apollo CM/LM are identified. Where instruments are unavailable from either spacecraft/aircraft, an indication "new" is made to identify requirements from other inventories. Table 2.2-21 presents an example of gross spacecraft functions and the identification of related controls and displays from Gemini and Apollo CM/LM. This effort provides the "shopping list" for allocation of C&D made in the development of the data sheets represented in Table 2.2-19.

The three alternate approaches are compared against off-shelf control and display instruments and candidate C&D mode identification and selection performed. Figures 2.2-30 and 2.2-31 present both C&D modes and the candidate instruments considered for each. All items indicated are either off-shelf or currently under development for viable NASA or military contracts. These lists provide the aggregate from which the LMSC control and display instrument baseline was selected. Additionally, the operational L-1011 autopilot/land C&D instrument panels are incorporated, thereby, maintaining the basic off-shelf approach. The only instruments that can be positively identified as needing fairly major modification are the following:

- Attitude director indicator – modify for atmosphere and space use
- Translation controller – modify to operate alternate engines
- Attitude controller – modify by adding disable yaw switch
- Aero-surface indicator – modify for elevon and rudder surfaces only

Minor control and display modifications required are: (1) labels/legends, (2) scales, (3) lamp colors, and (4) mounting mechanisms.

From an examination of postulated mission requirements and by identification of gross crew functions, candidate controls and displays, and mix of instruments, the evolution

Table 2.2-20

SUBSYSTEM FUNCTIONS VS OFF-SHELF S/C & A/C INSTRUMENTS

OCCUPIED SYSTEM	FUNCTION CONTROLLED OF PARAMETER/CONDITION DISPLAYED	SPACE FLIGHT REGIME	LM	CM	ATMOSPHERIC FLIGHT REGIME
GUIDANCE & NAVIGATION	DIGITAL READOUT OF ELAPSED TIME HRS/MINS/SEC	MISSION TIMER DISPLAY	X		IDENTICAL REQUIREMENTS
	TIMER CONTROL FUNCTIONS START - STOP - RESET	TIMER CONTROL SWITCH	X		
	HOURS SLEW CONTROL FUNCTIONS TENS-DISABLE-UNITS	SLEW CONTROL (HOURS) SWITCH	X		
	MINUTES SLEW CONTROL FUNCTIONS TENS-DISABLE-UNITS	SLEW CONTROL (MINUTES) SWITCH	X		
	SECONDS SLEW CONTROL FUNCTIONS TENS-DISABLE-UNITS	SLEW CONTROL (SECONDS) SWITCH	X		
	DIGITAL READOUT OF EVENT ELAPSED TIME-MIN/SEC	EVENT TIMER DISPLAY	X		
	TIMER CONTROL FUNCTIONS START-STOP	TIMER CONTROL SWITCH	X		
	TIMER COUNT-RESET CONTROL FUNCTIONS RESET-UP-DOWN	TIMER RESET COUNT SWITCH	X		
	MINUTES SLEW CONTROL FUNCTIONS TENS-DISABLED-UNITS	SLEW CONTROL (MINUTES) SWITCH	X		
	SECONDS SLEW CONTROL FUNCTIONS TENS-DISABLED-UNITS	SLEW CONTROL (SECONDS) SWITCH	X		
	INERTIALLY DERIVED ALTITUDE & RATE OF CHANGE IN ALTITUDE	ALTIMETER/ALTITUDE RATE DISPLAY	X		BACKUP FOR AIRCRAFT C/O BELOW
	RADAR ALTITUDE & RATE OF CHANGE IN ALTITUDE				
	MODE SELECTION FUNCTIONS IMU-RADAR	MODE SELECTOR SWITCH			
	SERVO CONTROLLED BAROMETRIC ALTITUDE				BAROMETRIC ALTIMETER W/INTEGRAL SETTING CONTROL
	MANUAL ALTITUDE SETTING CONTROL				
	RATE OF CLIMB DESCENT				VERTICAL SPEED INDICATOR
	ALTITUDE ABOVE TERRAIN 2500 TO TOUCHDOWN				RADIO ALTIMETER W/DECISION HEIGHT DISPLAY
	DIGITAL AND POINTER FOR DECISION HEIGHT 500 TO 0				DECISION HEIGHT INPUT CONTROL
	MANUAL DECISION HEIGHT SETTING CONTROL				AURAL DISPLAY OF DH + 50 TO DH
	AURAL TONE 50 FT ABOVE DECISION HEIGHT				
	TOTAL VEHICLE ATTITUDE ROLL/PITCH/YAW	FLIGHT DIRECTOR ATTITUDE INDICATOR	X	X	
	ATTITUDE RATES ROLL/PITCH/YAW				
	ATTITUDE ERRORS ROLL, PITCH, YAW				
	RENDEZVOUS RADAR SHAFT & TRUNNION ANGLES (PITCH & YAW ERROR DISPLAYS)				
	CAGE ATTITUDE BALL	CAGE ATTITUDE BALL SWITCH	X	X	
	INTERNAL ILLUMINATION OF FDAI-OFF-INCREASE BRIGHTNESS	BRIGHTNESS CONTROL	X	X	
	TEST RATE INDICATORS & SELF-TEST-OFF	SELF-TEST SWITCH	X	X	
	ALIGN FDAI WITH ATTITUDE GYRO COUPLING UNIT (AGCU)	ALIGN SWITCH	X	X	
	SET ATTITUDE & COMPARE WITH AGCU TO DETERMINE ERRORS-ON-OFF	ATTITUDE SET SWITCH	X	X	CURRENT AIRCRAFT ADI SUITABLE FOR SPACE FLIGHT REGIME WITH MODS
	INSERT ROLL COMMAND	VERNIER ROLL CONTROL		X	
	DISPLAY OF ROLL VERNIER SETTING	ROLL INDICATOR		X	
	INSERT PITCH COMMAND	VERNIER PITCH CONTROL		X	
	DISPLAY OF PITCH VERNIER SETTING	PITCH INDICATOR		X	
	INSERT YAW COMMAND	VERNIER YAW CONTROL		X	
	DISPLAY OF YAW VERNIER SETTING	YAW INDICATOR		X	
	SELECT RATE SCALE FOR FDAI 5°/SEC - 25°/SEC	RATE SCALE SWITCH	X		
	RATE/ERROR MONITORING SELECTION	RATE/ERROR MONITORING SWITCH	X		
	SHAFT/TRUNNION ANGLE RANGE SCALE SELECTION ±5° - ±50° (RENDZ RADAR)	SHAFT/TRUNNION SWITCH	X		
	TURN RATE DURING ATMOSPHERIC FLIGHT				
	SIDE SLIP OR LATERAL ACCELERATION				(B) - POINTER ORG. MUST PROVIDE ENCODED DATA

D05814

EXAMPLE

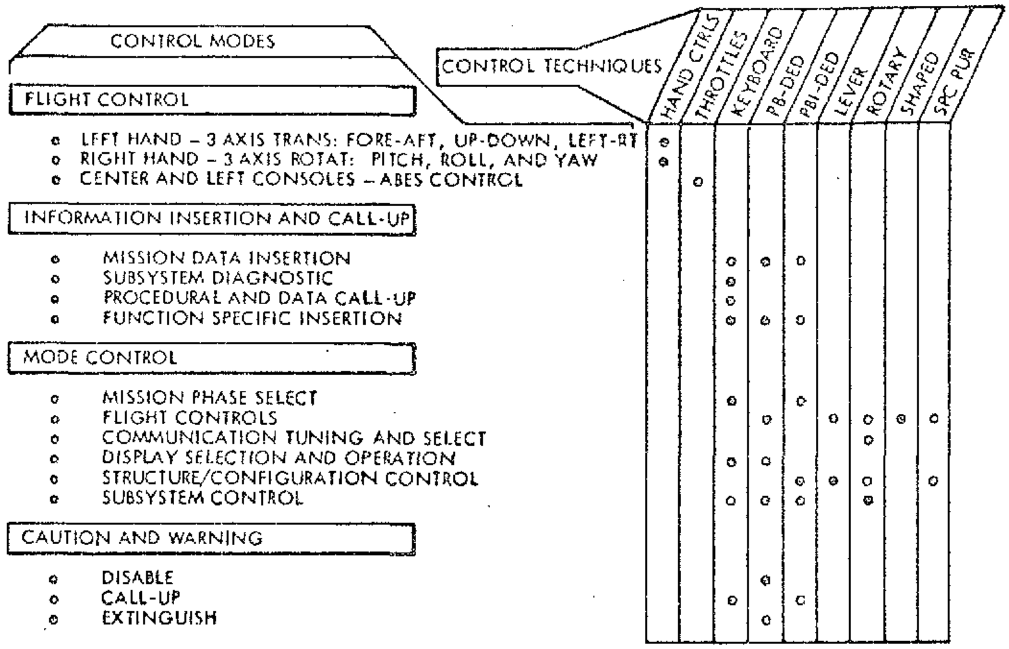
Table 2.2-21

ANALYSIS OF SPACECRAFT FUNCTIONS AND RELATED CONTROLS/DISPLAYS

CONTROL & DISPLAY FUNCTIONS		SPACECRAFT CONTROLS & DISPLAYS		
NO.	FUNCTION & DESCRIPTION	GEMINI	APOLLO CM	APOLLO LM
		FLIGHT DIRECTOR ATTITUDE INDICATOR (2 EA.)	FLIGHT DIRECTOR ATTITUDE INDICATOR (MDC-4)	FLIGHT DIRECTOR ATTITUDE INDICATOR (1 & 8)
	COMMANDED VEHICLE ATTITUDE ROLL/PITCH/YAW	X	X	X
	ATTITUDE RATES ROLL/PITCH/YAW	X	X	X
	ATTITUDE ERRORS ROLL/PITCH/YAW	X	X	X
	RENDEZVOUS RADAR SHAFT & TRUNNION ANGLES		X	X
	CAGE ATTITUDE BALL	X	X	X
	REFERENCE AXIS	X	X	X
	PITCH/YAW ANGLE COMMAND		X	X
	POWER SWITCH		X	X
	MODE SWITCH (COMPUTER COMMAND)	X	X	X
		EVENT TIMER	EVENT TIMER (MDC-2)	EVENT TIMER
	DIGITAL TIME READOUT	X	X	X (1)
	TIMER START/STOP/RESET SWITCHES	X	X	X (2)
		ALTIMETER	ALTIMETER (MDC-?)	(SEE P.3 BELOW)
	REFERENCE BARO ALTITUDE TO 80,000 FT	X	X	
			FLIGHT COMBUSTION STABILITY MONITOR (MDC-8)	
	SPS ENGINE THRUST TERMINATION (APOLLO CM PECULIAR)	ACCELEROMETER DISPLAY	ACCELEROMETER (MDC-8)	ACCELEROMETER (1)
	NORMAL & MAXIMUM + G IN X AXIS	X	X	X
		MASTER WARNING ANNUNCIATORS	MASTER WARNING ANNUNCIATORS (MDC-2)	CAUTION/WARNING LIGHT PANEL GROUPS (1 & 2)
	MASTER ALARM SWITCH LIGHTS & AUDIBLE ALARMS	X	X	X
		ABORT COMMAND	ABORT COMMAND (MDC-2)	ABORT COMMAND (1)
	ABORT REQUESTED BY RANGE SAFETY OR MISSION CONTROL	X (2 EA.)	X (2 EA.)	X
			LAUNCH VEHICLE RATES LIGHT (MDC-2)	
	ANGULAR RATES IN ROLL/PITCH/YAW EXCEEDED		X	
			LAUNCH VEHICLE GUID. FAILURE LIGHT (MDC-2)	
	GUIDANCE SYSTEM PLATFORM FAILURE/LOSS OF ATTITUDE CONTROL		X	
			LAUNCH VEHICLE ENGINE THRUST LIGHT (MDC-2)	
	ENGINE THRUST BELOW REQUIRED VALUE		X	
			ATTITUDE VERNIER CONTROL/DISPLAYS (MDC-?)	
	VERNIER ATTITUDE THUMBWHEELS, ROLL/PITCH/YAW		X	
	DIGITAL READOUT OF VERNIER ATTITUDE SETTINGS		X	
			ENGINE GIMBAL CONTROLS/DISPLAYS (MDC-?)	
	ENGINE THRUST VECTOR ALIGNMENT, PITCH/YAW		X	
	THRUST CHAMBER POSITION INDICATORS, PITCH/YAW			

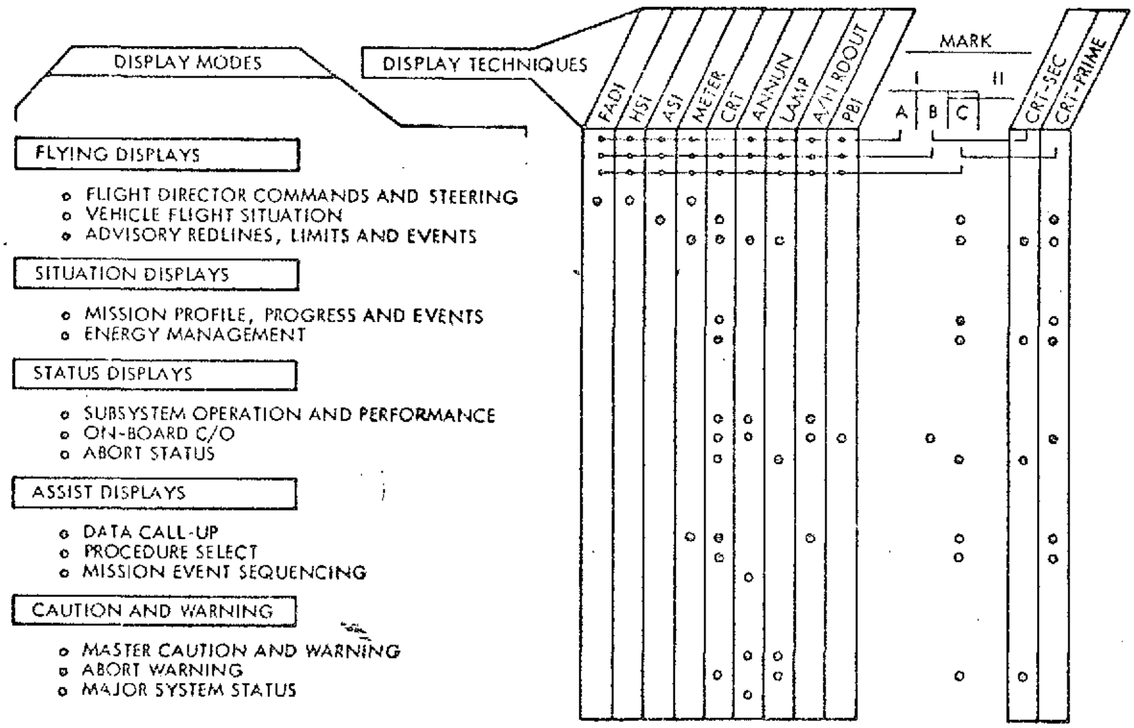
EXAMPLE

D05931



D05645

Fig. 2.2-30 Control Mode Identification and Control Selection



D05644

Fig. 2.2-31 Display Mode Identification and Display Selection

Evolutionary Approach to Programmable Controls and Displays. Recognition of the need for reduced pilot workload, lesser dedicated C&D panel area, and reduced flight-deck volume indicate the necessity to consider programmable controls and displays. Therefore, the approach conceived is for a C&D concept, which logically and systematically evolves from the purely dedicated (Alternate A) concept to the highly programmable baseline (Alternate C) concept. Figure 2.2-32 illustrates the evolutionary process. Appropriate to this process is the question of "why the transition from hardwired (dedicated) to programmable C&D" - which question is addressed in Table 2.2-22. Consideration is given then to programmable C&D and to development of a logic decision thread for their application in the basic subsystem – considering such elements as (1) crew safety and mission success, (2) critical phases and events, (3) frequency and concurrency of use, (4) mission regime assignment, (5) decision rules, and (6) fail conditions and redundancy. Figure 2.2-33 illustrates this decision thread for the selected baseline in Mark I.

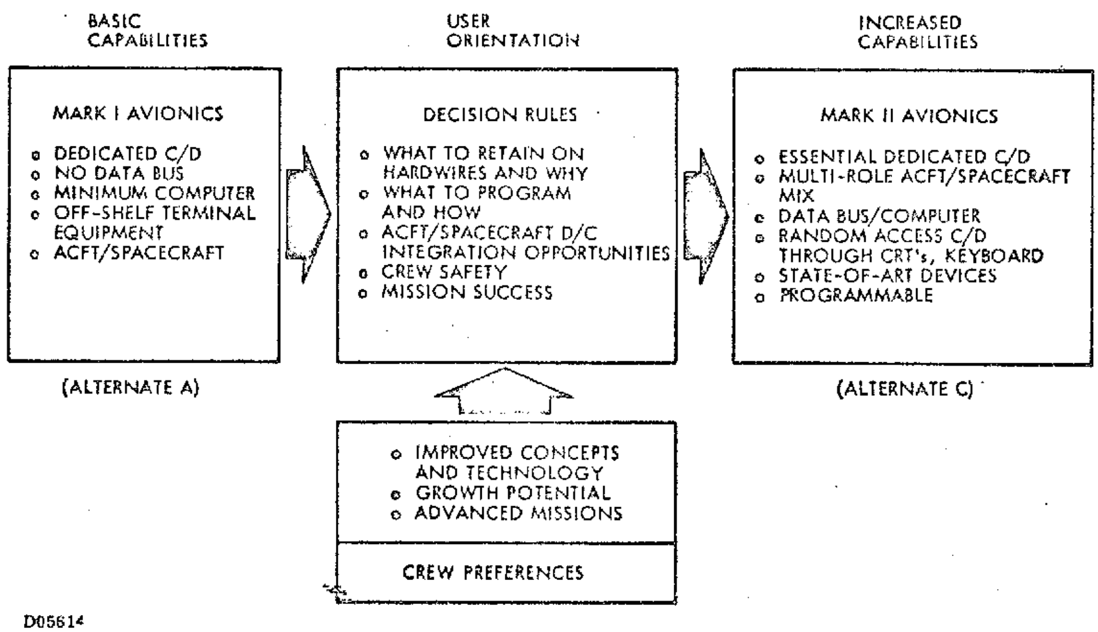


Fig. 2.2-32 Evolution of Programmable Control/Display Concept

Table 2.2-22

DEDICATED VS PROGRAMMABLE CONTROLS AND DISPLAYS

Q WHY TRANSITION FROM HARDWIRED TO PROGRAMMABLE?

DEDICATED

A
1. HIGH WEIGHT/POWER/PANEL SPACE REQUIREMENTS
2. INFLEXIBLE/NOT ADAPTABLE TO MISSION PHASE
3. DATA NOT ESSENTIAL TO MISSION PHASE-OVERLOADS CREW
4. C/D EXTEND BEYOND PRIMARY VISUAL/MANUAL ENVELOPE
5. 3RD CREWMAN TO MANAGE SUBSYSTEMS
6. MAINTENANCE/REPAIR/LOGISTICS MORE COMPLEX

PROGRAMMABLE

A
1. FLEXIBLE/ESSENTIAL DISPLAYS ALWAYS IN PRIME SPACE
2. LESS PANEL SPACE/LOWER POWER AND WEIGHT
3. BETTER ALLOCATION OF WORKLOAD/REDUCTION IN CREW SIZE
4. GREATER GROWTH POTENTIAL

D066AA

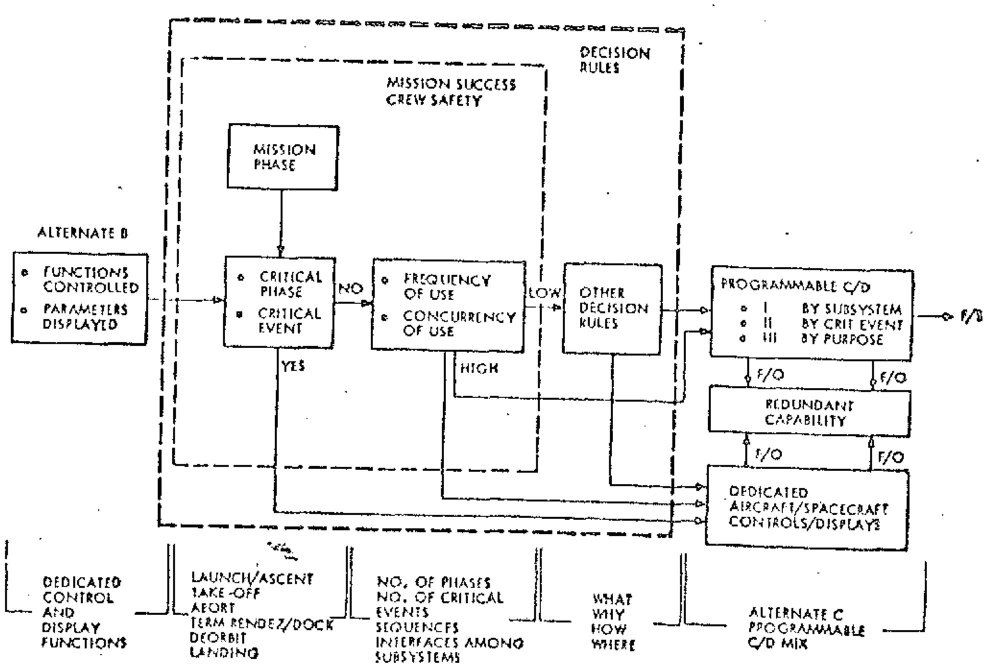


Fig. 2.2-33 Dedicated Vs Programmable Control/Display Decision Thread

and decision logic thread are traced through to a programmable concept, with sufficient redundancy to provide the level identified for crew safety. This logic thread is illustrated in Fig. 2.2-34. From this effort, dedicated versus programmable C&D decision rules are generated as basic to the development of the control and display subsystem (Table 2.2-23).

Control and Display Baseline Description. A pictorial diagram of the baseline (Alternate C) is presented in Fig. 2.2-35. Obviously, not all C&D are discretely identified, although major categories are included. The key to the C&D subsystem is the utilization of the "off-shelf" S-3A basic data management system and its inherently flexible potential (para. 2.1.5). Shown in Fig. 2.2-35 is the hardwire and programmable interface for C&D. All flight-critical (crew safety and mission success) controls and displays are hardwired and, as shown in Fig. 2.2-36 (main instrument panel), certain of these are duplicated (one set at the pilot's station and one set at the copilot's station) for redundancy. Figure 2.2-37 illustrates the eyebrow/overhead panel located in the overhead between the crew.

- Basic Flight Control and Displays – Crew Safety and Mission Success. As the C&D subsystem definition developed, it was necessary to determine those instruments critical to crew safety and mission success and which need to be hardwired and duplicated. (See Table 2.2-24.)
- Multifunction Display Units (MDU). Key to the programmable information presentation approach is the incorporation of three multifunction display units (MDUs incorporating CRTs: one each for the pilot and copilot and one located between for common sharing. The pilot and copilot MDUs are mission-regime sequenced so that only pertinent data for a particular mission phase are presented unless overriden by the crew.

Representative data presented on the three MDUs is indicated in Table 2.2-25. Information can be switched to any unit. Typical types of data presented are: (1) pictorial or real image, (2) alphanumeric, (3) graphic, (4) tabular, (5) map, (6) procedural, (7) caution and warning, (8) schematic, and (9) symbology. Three modes for grouping data on the MDUs are selected as illustrated in Table 2.2-26. Mission-phase and single-purpose display format examples are presented in Figs. 2.2-38 and 2.2-39a and b.

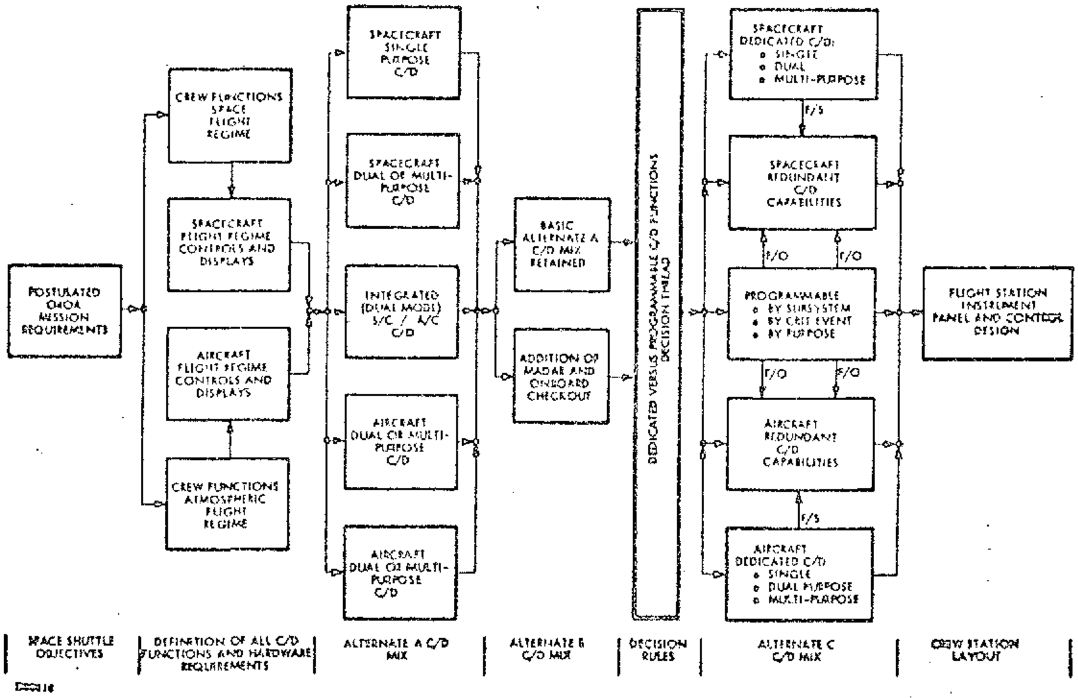


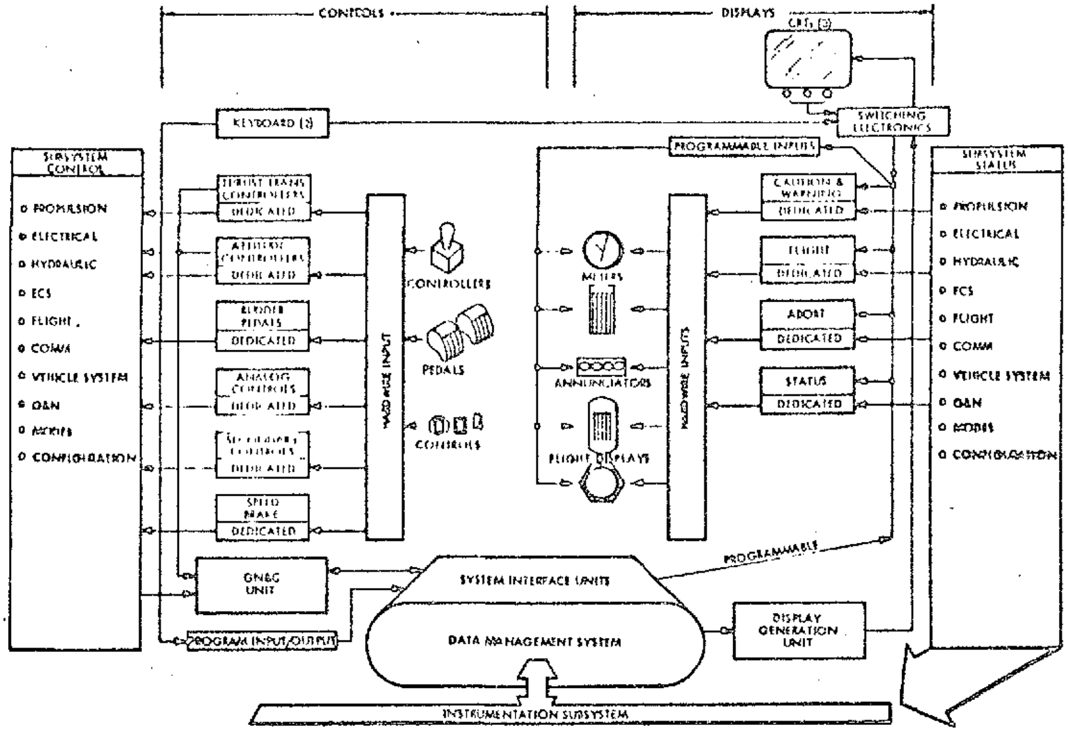
Fig. 2.2-34 Evolution of Programmable Controls and Displays

Table 2.2-23

DEDICATED VS PROGRAMMABLE CONTROL/DISPLAY DECISION RULES

PRIORITY	DECISION RULE (C/D EXAMPLES IN PARENTHESES BELOW)	DECISION DEDICATED	DECISION PROGRAMMED
I	1. ULTIMATE BACKUP C/D HARDWARE (LAST REDUNDANCY) FOR ESSENTIAL FUNCTIONS OF FLIGHT CONTROL, G & N, PROPULSION, EPS, COMM. (FDAI, ALT/VERT SPEED, AIRSPEED/MACH/α, HSI)	X	
	2. ULTIMATE BACKUP C/D HARDWARE FOR MISSION COMPLETION (RANGE/LOS/RANGE RATE, RADAR ALTIMETER, EMER O_2 FLOW, ETC.)	X	
	3. CATASTROPHIC FAILURE, CRITICAL CONDITION, EMERGENCY DISPLAYS (ABORT, MASTER WARNING, CAUTION AND WARNING ANNUNICATORS)	X	
II	4. ANALOG/CONTINUOUSLY VARIABLE CONTROLS (ATTITUDE CONTROLLER, THROTTLE CONTROLS, RUDDER/BRAKE PEDALS, POTENTIOMETERS, ETC.)	X	
	5. MANUAL CONTROL FUNCTIONS REQUIRING IMMEDIATE RESPONSE TO CORRECT CONDITION OR AVERT HARDWARE DAMAGE/CREW INJURY (ABES AND APU SHUTDOWN, RCS QUAD DISABLE, EJECTION SEQ INITIATION)	X	
	6. FUNCTIONAL (OR INTEGRAL) CONTROLS FOR DEDICATED DISPLAYS (TEST, MODE SELECT, SCALE, PARAMETER ADJUST, COMMAND SETTING, ETC.)	X	
	7. VEHICLE CONFIGURATION C/D (POSITION OF AERODYNAMIC CONTROL SURFACE TRIM, SPEED BRAKES, DOORS OPEN, GEAR AND ABE DEPLOYED, ETC.)	X	
	8. C/O PANELS FOR PACKAGED STATE-OF-THE-ART SUBSYSTEMS (MADAR, TACAN, ATC TRANSPONDER, UHF COMMUNICATIONS, ETC.)	X	
	9. SINGLE MISSION PHASE OR CRITICAL EVENT MONITORING (BOOSTER ATTITUDE RATE AND PROPULSION ABNORMALTIES RENDEZVOUS/DOCKING)		X
	10. ROUTINE MONITORING & RECONFIGURATION OF SUBSYSTEM (ECS, ELECT PWR DIST, PROPELLANT/PRESS PARAMETERS, CROSSFEED)		X
	11. SIMULTANEOUS DISPLAY OF INFORM. REL. TO MISSION PHASE		X
	12. MALFUNCTION ISOLATION IN PARALLEL WITH ON-BOARD C/O		X

D05563



D06251

Fig. 2.2-35 Control and Display Diagram

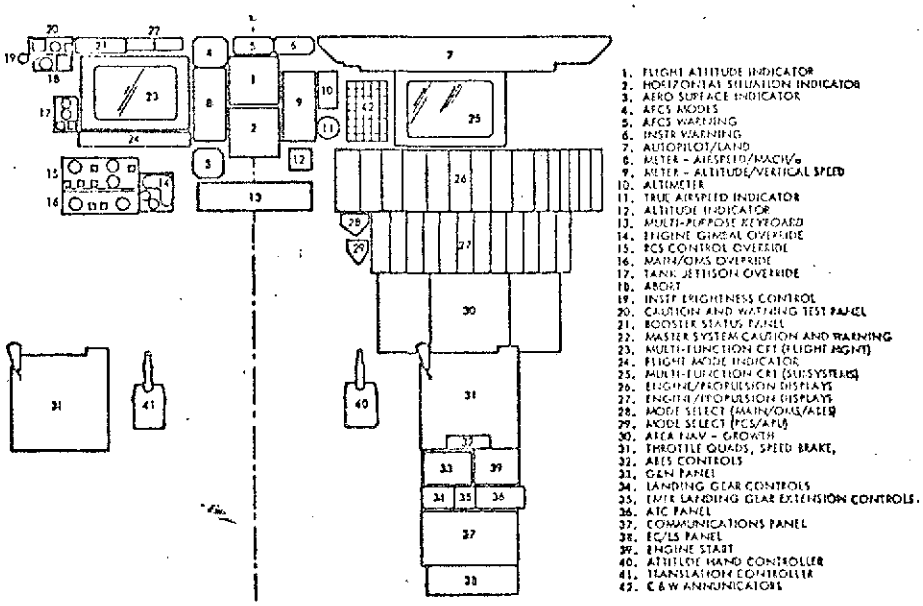
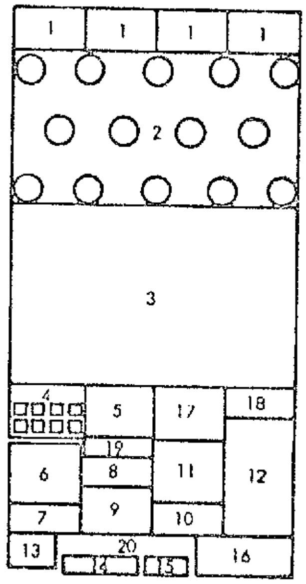


Fig. 2.2-36 Main Instrument Panel



1. ENGINE FIRE CONTROL PANELS
2. EC/LS GAS SUPPLY OVERRIDE VALVES
3. ELECT. PWR GENERATION AND DIST
4. ELEVON DISABLE
5. RUDDER DISABLE
6. SAS, PITCH, ATS, AND TRIM EMER. CONTROLS
7. ANTISKID CONTROLS
8. SENSOR HEAT CONTROLS
9. PFCS MON., RUDDER AND ELEVON EMER. CONTROLS
10. RUDDER LIMITER
11. ENGINE START
12. APU ENGINE CONTROLS
13. CABIN LIGHTS
14. MISSION TIMER
15. EVENT TIMER
16. EXTERIOR LIGHTS
17. GROWTH
18. GROWTH
19. GROWTH
20. GROWTH

D05907

Fig. 2.2-37 Eyebrow/Overhead Panel

Table 2.2-24

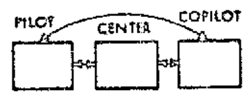
BASIC DEDICATED FLIGHT CONTROLS AND DISPLAYS

- FLIGHT ATTITUDE DIRECTOR INDICATOR
- HORIZONTAL SITUATION INDICATOR
- AIRSPEED (MACH) AND METER
- ALTITUDE, VERTICAL SPEED METER
- PROPULSION/ENGINES METERS
- THROTTLES - QUADS
- ATTITUDE AND TRANSLATION CONTROLLERS
- RUDDER PEDALS
- COMMUNICATION PANEL
- INSTRUMENT LAMPS
- EC/LS CONTROLS
- ELECTRICAL POWER CONTROLS
- ANTISKID CONTROLS
- VEHICLE CONFIGURATION CONTROLS
- FIRE CONTROL
- ABORT CONTROL/LAMP
- MAJOR SYSTEM ANNUNCIATORS
- BOOSTER SYSTEM STATUS ANNUNCIATORS
- AEROSURFACE POSITION INDICATOR
- RUDDER LIMITER
- SPEED BRAKE
- ELEVON CONTROLS
- ABES CONTROLS
- LANDING/EMERGENCY CONTROLS
- ENGINE START - ABES AND APU
- SAS, PITCH, AND TRIM EMERGENCY CONTROLS
- MISSION AND EVENT TIMERS/CONTROLS

D08428

Table 2.2-25

TYPICAL ALLOCATION OF PROGRAM DATA TO PILOTAGE CRT DISPLAYS

PILOT CRT	CENTER CRT	COPILOT CRT
1 MISSION PHASE DATA o PREDICTED PERFORMANCE ENVELOPES o FLIGHT REGIMES o COMPOSITE G&N SITUATION DISPLAY o COMPOSITE CONFIGURATION DISPLAY 2 CRITICAL EVENTS o HORIZONTAL FLIGHT ✓ TAKE-OFF (SPEEDS, CONFIG, PERFORM) ✓ LANDING (SPEEDS, ALT, CONFIG, RATES) o VERTICAL FLIGHT ✓ LAUNCH (IGN, LIFTOFF, 1st STAGE, BURN, BOOST) ✓ ASCENT (STAGE, IGN, PERF, TRAJ, SEP) ✓ INSERT (TRAJ, PERF, SYS TRANSITION ✓ RENZ (TRAJ, PERF, SENSORS, MODES) ✓ TERM R&D (ACQ, SENSORS, TRAJ, MODES) ✓ RETRO/DEORB (G&N, CONFIG, THERMAL) o ABORT ✓ MISSION PHASE ✓ SPECIFIC FAILURE ✓ SEQUENCED ✓ GROUND/RANGE SAFETY	1 SUBSYSTEM STATUS o KEY SUBSYSTEM PARAMETERS BY SUBSYSTEM 2 SUBSYSTEM CONFIGURATION o WHICH HARDWARE ON LINE o WHICH OPERATING MODES IN MULTI-MODE SUBSYSTEMS o REAL-TIME RECONFIGURATION TALK-BACK 3 PREFLIGHT CHECKOUT PROCEDURES 4 MALFUNCTION ISOLATION IN RESPONSE TO C&W INDICATION 5 POSTFLIGHT CHECKOUT PROCEDURES  CRT DATA SWITCHING POTENTIAL	1 MISSION PHASE DATA 2 CRITICAL EVENTS 3 MISSION PLANNING DATA 4 MISSION PHASE CHECKLISTS

D06244

Table 2.2-26

CRT DISPLAY PROGRAMMING APPROACH

THREE MODES OF GROUPING DATA FOR CRT DISPLAYS:

1. BY SUBSYSTEM
   - o GROUP AND DISPLAY S/S CONFIGURATION AND OPERATING PARAMETERS
   - o "WALK THROUGH" THE S/S
   - o FOR S/S "MANAGEMENT," HANDLING CONTINGENCIES, MALFUNCTION ISOLATION AND TROUBLE SHOOTING
2. BY MISSION PHASE AND CRITICAL EVENT
   - o DISPLAY "WHAT NEEDS TO BE USED" – SEVERAL DISPLAYS
   - o INTER-RELATED DATA FROM VARIOUS SUBSYSTEMS SIMULTANEOUSLY
   - o "ALWAYS USED" DISPLAYS/CONTROLS ALWAYS AVAILABLE = HARDWIRED
3. SINGLE-PURPOSE
   - o ONE TYPE OF DISPLAY OR INSTRUMENT AT A TIME
   - o FLY-TO ENVELOPE OF COMMAND EXECUTION
   - o CHECK LIST   o BACKUP PROCEDURES

D03563

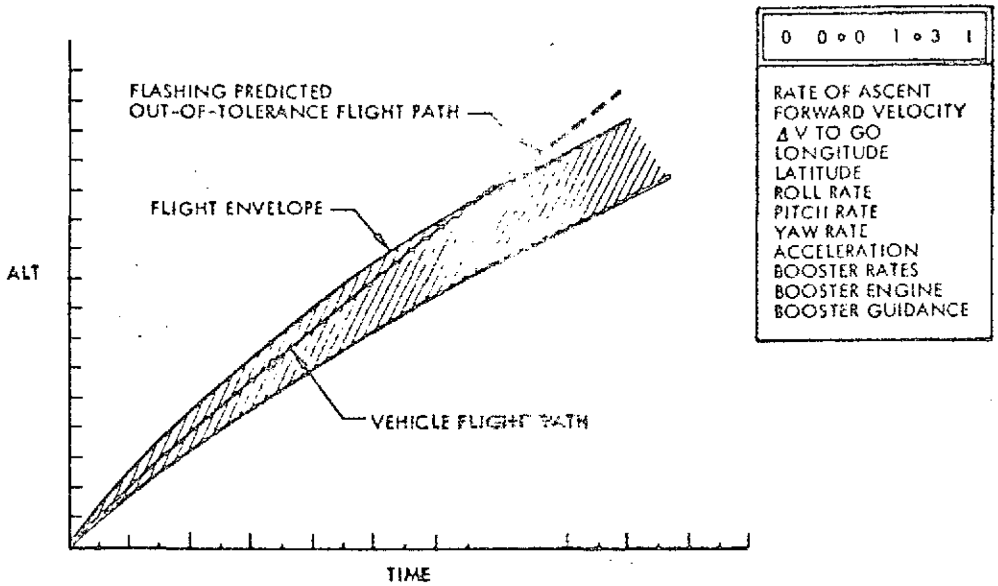
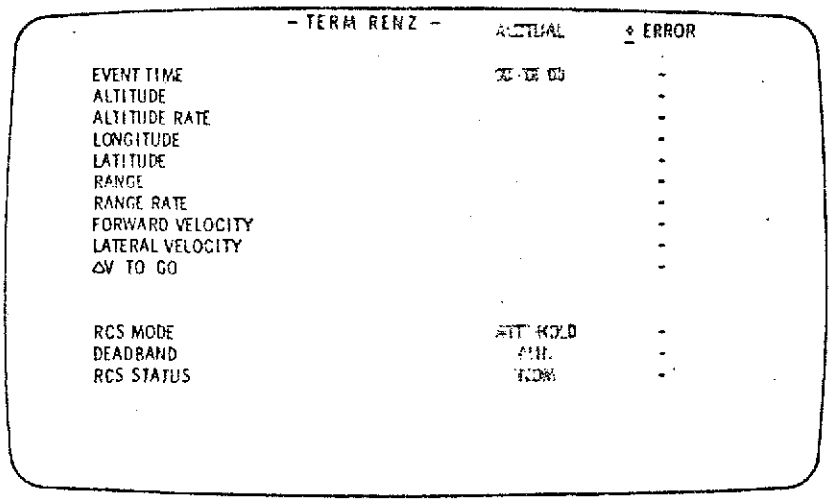
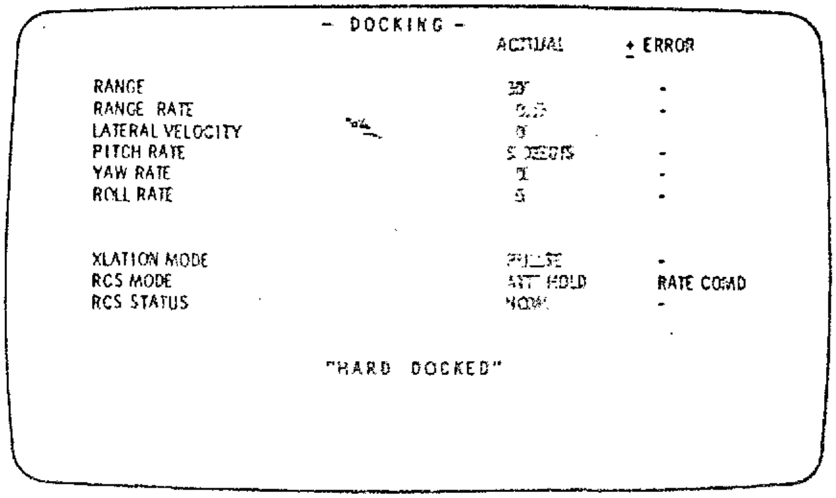


Fig. 2.2-38 CRT Programming By Mission Phase: Launch



D05132

a. Terminal Rendezvous



b. Docking

Fig. 2.2-39 CRT Programming By Mission Phase: Terminal Rendezvous and Docking

Multifunction display unit information is presented in consistent formats for either crewman and always in subsystem groups for clarity and ease of information scan and interpretation. Major system, subsystem, and "unique" data are available for call-up through the pilot or copilot keyboards. Data switching among the three DMUs must be accomplished by manual override to eliminate inadvertant data loss. Information is called-up in transcending levels and, in general, the lower levels provide greater detail.

Malfunction, out-of-tolerance, or anomalies can be presented on the MDUs and crew attention effected through flashing and increased brightness techniques. To aid the crewman, nominal values are presented in the ranges of acceptance to permit the crew to determine the extent of the problem. The center MDU is keyed to the caution and warning system for automatically presenting information relative to the malfunction being indicated on the caution and warning annunciators. In instances where more than one malfunction occurs, MDU area sharing is accomplished through the data management system (DMS).

The off-shelf MDUs recommended for the baseline are S-3A units with the nomenclature of "Indicator, Tactical IP-1053/ASA-82". Specific performance data characteristics for the baseline MDU are presented in Table 2.2-27. Usable display area is 6 by 9 inches. The three MDUs are equipped with contrast and brightness controls and are designed to operate well within the expected Space Shuttle flight deck ambient environment (approximately 4000 to 5000 ft candles).

- <u>Manual Controls.</u> Several types of manual controls were selected (see Fig. 2.2-30). Standard throttle quads are recommended and allocated to the pilot and copilot. For both pilot and copilot use, attitude controllers with yaw disable are planned for the righthand positions and translation controllers are planned for lefthand positions. Figure 2.2-40 presents thrust/translation and attitude control for space and atmospheric flight regimes; the spacecraft controllers illustrated are typical of previous devices employed to perform these functions. Rudder pedals and attitude controllers appear applicable from existin programs. However, the translation controller for main and OMS engine control may require development, or at best, major modification from the current Apoll

Table 2.2-27

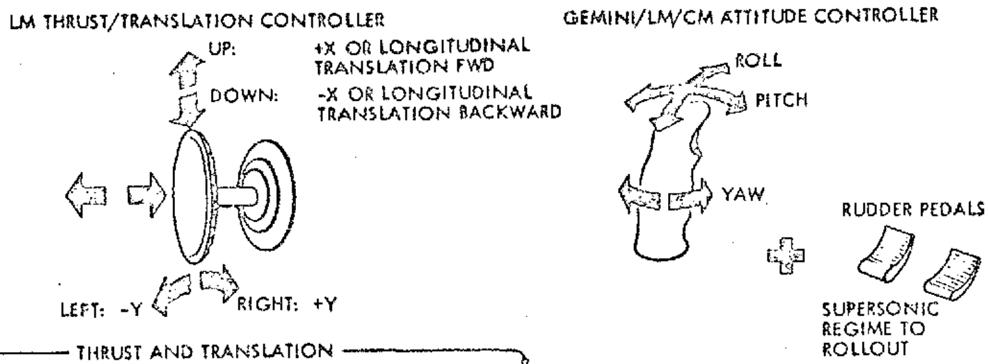
MULTIFUNCTION DISPLAY UNIT (MDU) PERFORMANCE CHARACTERISTICS

Parameter	MDU	Parameter	MDU
Cathode Ray Tube:*		Worst Case:**	
Shape	Rectangular	Settle (within grid) 1 percent, μsec	6
Usable Rectangle, in.	6.45 +0.00 -0.125 X	0.1 percent (one-grid position), μsec	8
	9.3 +0.00 -0.125	Linearity (geometry), %	1.0
Phosphor	P-31	Character Size (nom), in.	1/8 high
Spot Size, mil	9	Aspect Ratio (nom)	3/4
Brightness, ft-L (Exclusive of filter)	100	Character Generation Time (max), in./sec	3.2
		Writing Speed (max), in./sec	565,000
Ambient, ft-candles	5500	Video Bandwidth (TDS system), MHz	DC to 37
Gray Tones	8		
Deflection Random Positioning, bits	10 (digital)		

*The CRT faceplate radius of curvature shall be greater than 24 in.
**Deflections less than 0.5 in. shall take less than 3.5 μsec, and deflections greater than 0.5 in. shall take a time proportional to distance (not to exceed 8 μsec).

CM translation controller. Among the difficulties associated with the translation controller are problems affecting (1) control of two dissimilar thrusting modes, (2) mode selection, (3) mixer boxes, and (4) consideration of wire backup-linkage coupling.

- Multifunction Keyboard. The basic DMS interface is through the multifunction keyboard (MFK). Two such keyboards are provided; one for the pilot and the other for the copilot. These MFKs are located directly in front of the crewmen to permit ease of operation and for clustering of important functions within the main instrument panel area. The keyboards are utilized for the following typical tasks:



THRUST AND TRANSLATION

AXIS	SPACE FLIGHT REGIME			ATMOSPHERIC FLIGHT REGIME			SPACE	AXIS
	MAIN	OMS	RCS	ABES		AEROCONTROL SURFACES	RCS	
+X	✓	✓	✓	✓		ELEVATORS BODY FLAP (PITCH TRIM)	✓	PITCH UP
-X			✓	HYPERSONIC	RCS	HIGH LIFT DEVICES	✓	PITCH DN
+Y			✓		RCS	ELEVATORS OR ELEVONS	✓	ROLL L
-Y			✓		RCS		✓	ROLL R
+Z			✓		RCS	RUDDER	✓	YAW L
-Z			✓		RCS		✓	YAW R

ATTITUDE CONTROL

D05612

Fig. 2.2-40 Control Function Interaction By Mission Phase

- Flight command entry
- Guidance and navigation computation and data entry
- Discrete event initiation
- Initiate preflight and postflight vehicle checkout functions – automated
- System and subsystem interrogation
- Callup and interrogate the COFIRM program
- Vehicle configuration control
- Callup subsystem information
- Callup nominal and emergency procedures
- Callup predictor display
- Subsystem configuration control and sequencing
- Consumables management control and interrogation
- Abort mission planning

Several types of keyboards have been examined and tradeoffs conducted. Table 2.2-28 presents the basic requirements considered essential for the keyboard.

Table 2.2-28

KEYBOARD REQUIREMENTS

OPERATIONAL CONSIDERATIONS	DATA MANAGEMENT CONSIDERATIONS
OPERABLE UNDER g LOADS	COMPUTER COMMUNICATION PERMITTED
MINIMIZED ATTENTION FOR ACTUATION	REPROGRAMMABLE ONBOARD
VERY FINE AND DISCRETE MOVEMENTS REDUCED	AVAILABLE SOFTWARE PROGRAMS
RAPID DATA ENTRY	ADAPTABLE TO VARIOUS FLIGHT REGIMES/MISSIONS
MINIMUM ADDITIONAL SUPPORT EQUIPMENT	**UTILIZATION CONSIDERATIONS**
MINIMUM COMPLEXITY	VARIETY OF USE FLEXIBILITY
MINIMUM OPERATOR ERROR ENTRY POTENTIAL	MINIMUM VOLUME
PILOT ACCEPTANCE	ALPHA/NUMERIC CAPABILITY
INSTRUCTIONS MINIMIZED	EASE OF USE IN PLANNED LOCATION

D06429

Six basic types of keyboards were evaluated against the requirements noted in Table 2.2-28. Both the key-select matrix and page-overlay keyboards met the requirements. Major consideration is being given to the use of the S-3A "INCOS TACCO TRAY" keyboard, since it is functionally integrated with the S-3A DMS, and therefore, available software is avionic applicable to the Space Shuttle. However, vehicle functions not specific to avionics also must be considered. Although out of the scope of this study, these nonavionics functions will have direct bearing on the keyboard functional and software requirements. Thus, in addition to the S-3A keyboard, a page-overlay keyboard is being considered as an alternate for the Mark I Orbiter baseline.

Other manual controls employed are toggles, rotary switch controls, pushbuttons, and special-purpose and shaped controls (speedbrakes, trim controls, fire pull handles, valves, etc.); these are all off-shelf items with no major modifications anticipated. Generally, these types of controls are provided to both pilot/copilot in instances where sharing is not practical, when rapid and error-free actuation is necessary, or when "feedback feel" is required.

- General Displays. Several types of displays, in addition to the MDUs, are incorporated. Caution and warning (C&W) annunciators are provided to alert the crew to all emergencies, malfunctions, and out-of-tolerance conditions that affect crew safety or mission success. The prime C&W annunciator is located directly above the basic instrument group (Fig. 2.2-36, Item 22) within the prime cone of vision envelope. This annunciator displays via message format the particular system that has malfunctioned, gone out-of-limits, etc. A basic matrix of annunciators broken down to subsystem level (Fig. 2.2-36, Item 42) illuminates further pinpointing of the problem. These annunciators are hard-wired for crew safety purposes. In response to the annunciator signal, the center MDU (CRT) presents further status information regarding the problem. At this point, the copilot will, in general, continue to interrogate the subsystem in question using the MFK (keyboard) and MDU to isolate the problem and to determine the most effective resolution (assuming a nonautomated changeover to a redundant subsystem or equipment item).

  Other displays – such as digital readouts for time, annunciators of status, light-emitting diodes (LEDs) for quantitative readout, mechanical readouts (communication panel), lamps (abort), and mechanical flags (Apollo-type) are required. Additionally, standard instruments -- such as indicated in Table 2.2-24 and illustrated in the flight instrument cluster shown in Fig. 2.2-36 – complete the display complement.

- Control and Display Test and Checkout. Utilization of the basic S-3A data management system and inherent BITE equipment and interface units permits continual test of the C&D instrumentation subsystem to the LRU level. In essence, the test capability is built into each major equipment item, and the DMS is constantly interrogating the equipment to determine health status. The status is not called to the attention of the crew unless a malfunction, anomaly, or out-of-tolerance condition is sensed. As discussed previously, C&W annunciators, subsystem status annunciators, and the MDU (CRT) are employed to facilitiate checkout for automatic or manual interrogation. This system is basic to the S-3A and has proven to be a major enhancement to both flight- and ground-crew personnel. Furthermore, the technique lends itself specifically to the VAST type of ground checkout which has the potential for a significant reduction in GSE checkout equipment.

- <u>Control and Display Subsystem Maintainability.</u> The basic flight-deck control and display subsystem has been designed from the outset (1) to facilitate maintenance, (2) permit instrument or component replacement, and (3) to ensure that subsystem integrity has not been compromised. Maintenance considerations include:

  (1) Access

  - Front access for all control and display instruments
  - Instruments all removable from the front
  - Track and rail installation for removal/replacement of "heavy" instruments
  - Service loops to facilitate front removal/replacement
  - Lamps replaceable from the front

  (2) Design

  - Electrical disconnects at the panel
  - Panel-structure captive mounted instruments
  - Module, control, or display device removal/replacement
  - Service loops in reel-roll configuration
  - Groundline to assure crew safety
  - EMI protection
  - Modules are LRU configuration

All instrument panels are designed with respect to consoles, seats, and associated equipment to permit adequate installation and removal/replacement volume. Instruments can be panel demated and removed without decoupling the connectors to facilitate on-site servicing, inspection, and checkout. Also, instruments are provided with a second connector interface (capped) for initial flight testing and subsequent ground checkout; this precludes the requirement to pull the unit simply to run service, continuity, or basic checkout while on the ground.

2.2.1.4.5 Baseline Rational – Controls and Displays – Mark I. Rationale for selection of the baseline C&D included several major considerations. These are delineated below:

- Incorporation of the basic S-3A data management system and system interface units
- Inclusion of S-3A and compatible L-1011 and C-5A control/display instruments
- Incorporation of the S-3A "BITE" to reduce inflight or ground checkout man-hours and complexity
- Inclusion of the L-1011 autopilot/land instruments and subsystems
- Utilization of the off-shelf-developed basic flight displays for both atmospheric and exoatmospheric flight modes
- Use of the basic flight displays for on-line operational and backup (dedicated)
- Incorporation of multifunction display units (MDUs) with CRTs for programmable data and for presenting only mission-regime specific information
- Reduction of displays through use of the multifunction display units
- Elimination of the Flight Systems Engineer through:
  a. Reducing checkout and test instrumentation and displays
  b. Presenting only mission-regime specific data to the pilot/copilot
  c. Reducing displays and, hence, scanning which reduces workload
  d. Introducing the autopilot/land subsystem which frees considerably the workload from the pilot/copilot, thus permitting monitoring of other subsystems with greater attention
  e. Incorporating the third MFD and allocating it primarily for subsystem status, sequencing display, and COFIRM functions
  f. Locating flight management information on the MFD in front of the pilot/copilot
  g. Dual use of several instruments, thus reducing panel area and providing additional space for displays/controls at the pilot/copilot station panels
  h. Providing for computer-aided vehicle configuration, consumable conservation, abort-mission planning, and vehicle sequencing

- Duplication of basic flight display and control instruments at both the pilot/copilot panels to facilitate ease of takeover or flight control from one position in emergencies

The basic S-3A DMS and associated SIUs currently incorporate programmable and displays controls including the use of three CRTs. Thus, the LMSC baseline takes advantage of existing and proven capability, thereby, nearly eliminating any risk factors associated with a programmable C&D approach. Furthermore, considerable software applicable to flight displays currently exists and can be utilized.

2.2.1.4.6 Panel Area, Crew Complement, and Control and Display Weight Factors.

Panel Area Layout. Development of the basic main instrument panel layout is predicated on standard crew-function allocation factors and anthropometric considerations. Primary flight C&D are provided to both the pilot and copilot (1) to facilitate flying the vehicle from either seat position and (2) to provide redundancy of C&D considered critical to crew safety and mission success. The selected C&D layout grouping groundrules are pictorially presented in Fig. 2.2-41. This arrangement is traditional and presents no violation of accepted practices. Control and display layout within this grouping is achieved as per Fig. 2.2-36.

Crew Complement and Location. Pilot and copilot locations are shown in Fig. 2.2-42. Also included are system/payload and tele-operator work stations. During the initial horizontal and vertical test flights, it is strongly recommended that a third crewman and work station be provided; this position will greatly off-load typical test and checkout functions from the pilot and copilot, thus measurably reducing workload. Upon completion of the test flights, this "Flight Systems Test Engineer" station is reconfigured into the payload monitoring and checkout work station. Thus, no scar penalty is recognized in growth from the test vehicle to the operational vehicle.

The tele-operator work station is provided for payload manipulation and satellite deployment or capture; remote manipulator C&D is required at this station to facilitate these functions. Additionally, it is strongly recommended that attitude and translation controls as well as range, range rate, etc., displays be incorporated at this station to facilitate vehicle maneuvers associated with payload activities. Inclusion of these

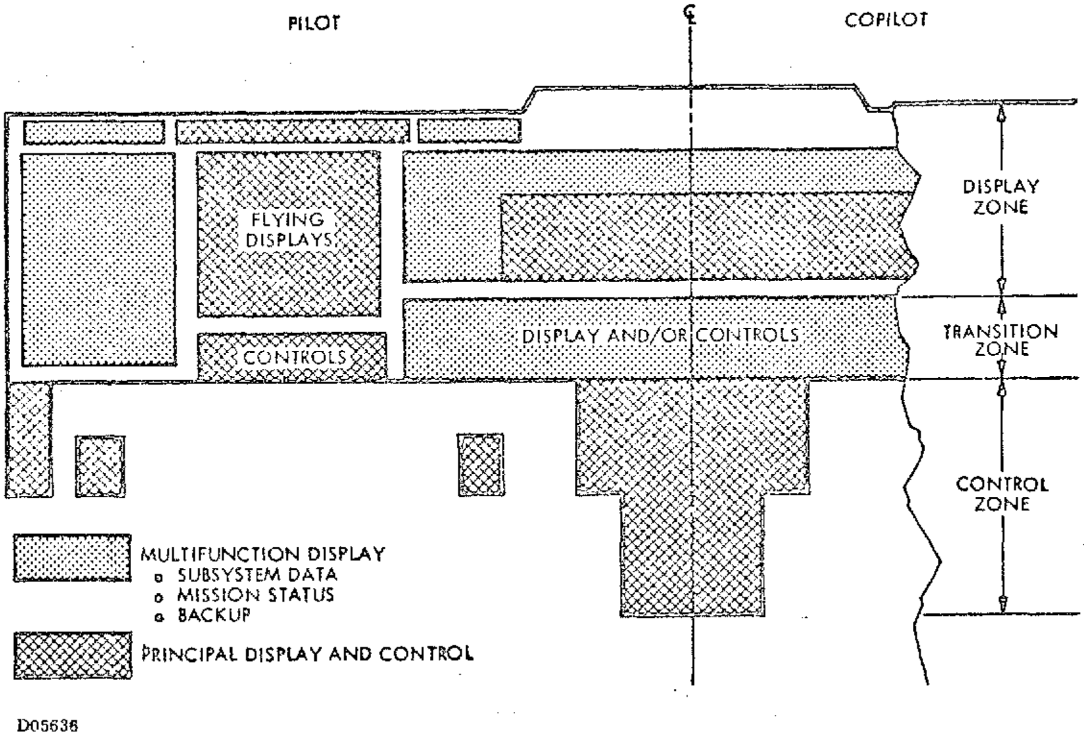


Fig. 2.2-41 Control and Display Layout Grouping Groundrules

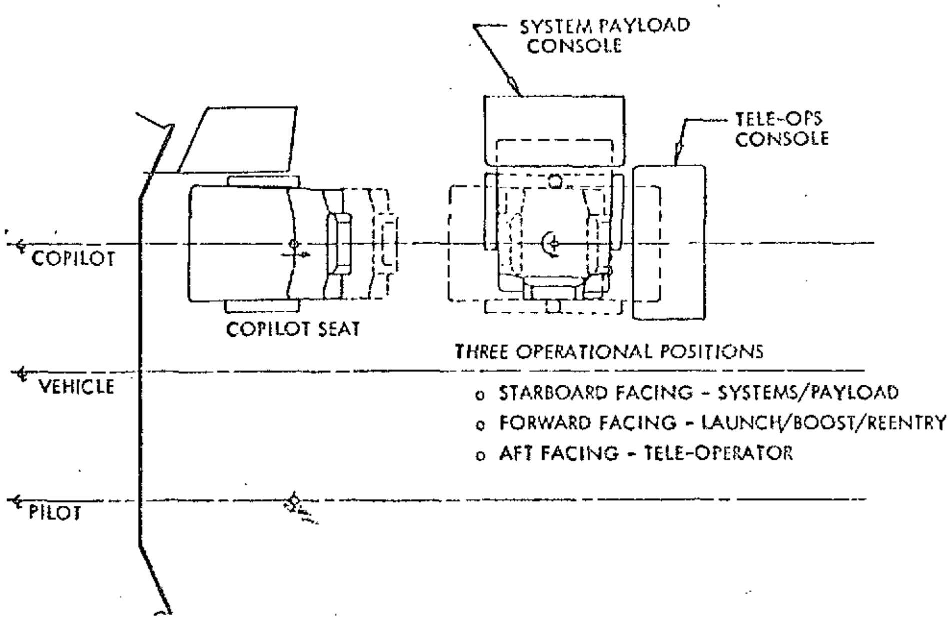


Fig. 2.2-42 Systems/Payload Monitoring and Tele-operator Console/Seat Positions

additional flight C&D necessitates positive techniques for transfer of these functions from the pilot/copilot station aft to this work station (if located on the flight deck). Figure 2.2-43 presents a simplified concept of this arrangement. Transfer of flight control and visibility are two potential areas for major investigation and analysis.

Control and Display Weights and Panel Areas. A comparison analysis was made of aircraft and spacecraft panel areas and control/display weights. Figure 2.2-44 presents these data in summary form. The third crew station is recommended for the test flights only. Thus, C&D weight for the Mark I operational vehicle is determined as 717 lb. It is important to note that 189 lb of the total weight is allocated to the CRTs. Display generation and computers associated with the C&D subsystem are charged to the data management subsystem.

Panel area for the operational Mark I vehicle is 13.5 sq ft, realizing that the systems engineering station (8.3 sq ft) requirement drops out after the test program. Other aircraft and spacecraft panel areas are included for comparative purposes. Of interest is the display and control panel area versus weight-density plot presented in Fig. 2.2-45. It shows that for the baseline Mark I control and display subsystem, maximum control and display panel density (53 lb per sq ft) is realized, thus, tending to indicate reasonably efficient use of panel area.

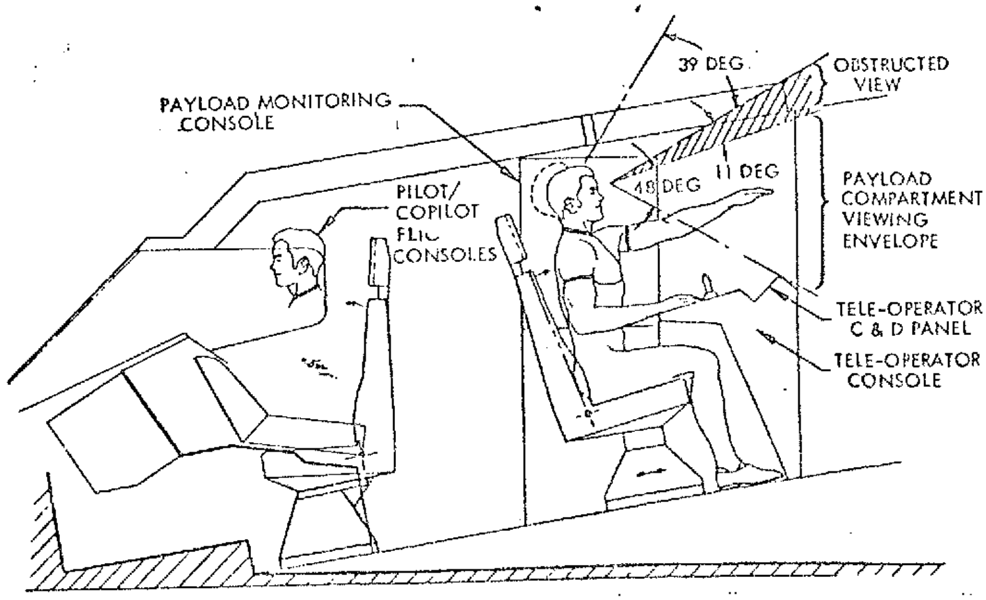


Fig. 2.2-43 Systems/Payload Monitoring and Tele-Operator Consoles

	AIRCRAFT			SPACECRAFT			SPACE SHUTTLE			
CREW SIZE	4 MEN	3 MEN	2 MEN	2 MEN	2 MEN	3 MEN	3 MEN	3 MEN	3 MEN	2 MEN
VEHICLE	C-5A	1011	S-3A	GEMINI	LM	CM	ALT A	ALT B	ALT C (TEST)	ALT C (OPS)
TOTAL PANEL AREA* – SQ FT	57	26.75	9.3	19.4	16	27.1	28.1	23.9	21.8	13.5
TOTAL C AND D WEIGHT – LBS	790	313	400	260	290	344	[illegible]	810	705/988	717

	PILOT/COPILOT STA		SYS ENGR STA		CRT'S		SUMMARY	
	WEIGHT (LB)	AREA (SQ FT)	WEIGHT (LB)	AREA (SQ FT)	NO	WEIGHT (LB)	WEIGHT (LB)	AREA (SQ FT)
ALTERNATE A	602	13.5	276	14.6	0	0	878	28.1
ALTERNATE B	602	13.5	208	10.4	1	63	810	23.9
ALTERNATE C_T (HORZ/VERT TEST)	514/797	13.5	189	8.3	1	63	705/988	21.8
ALTERNATE C_O (OPERATIONS)	528	13.5	0	0	3	189	717	13.5

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Fig. 2.2-44 Control and Display Panel Area and Weight Comparison

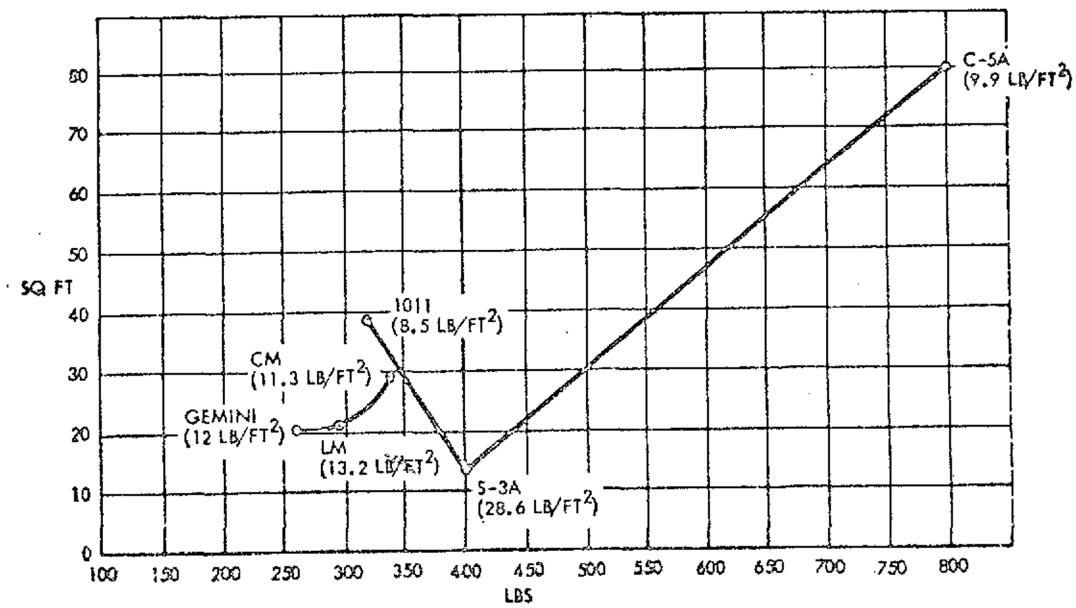

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Fig. 2.2-45 Control and Display Panel Area Vs Weight Density (Nonnormalized)

2.2.1.4.7 Control and Display Tradeoff Analyses. Several tradeoff studies were conducted to support the development of the control and display configuration, definition, and layout process. Additional supplementary analyses also are included to present the range of C&D data and impact factors considered. Both trades and analyses are briefly discussed in the following paragraphs.

Circuit Breaker Analysis. Circuit breaker estimates (approximately 500 or more) are made for the vehicle. From the newly developed concept of "remote circuit breakers," the estimated panel area is placed at approximately 16 sq ft. Table 2.2-29 presents a circuit breaker area-panel comparison between representative spacecraft and aircraft, thus further substantiating the dedicated panel area relegated to these necessary C&D items.

Display Device Tradeoffs. A review of potentially applicable displays was made to determine status and availability; Table 2.2-30 presents an encapsulated summary of this review. Holograms, lasers, and grid-type displays are all in the

Table 2.2-29

CIRCUIT BREAKER PANEL AREA COMPARISON

AREA / VEHICLE		PILOT/COPILOT OVERHEAD	PILOT/COPILOT/FLT ENGR OVERHEAD-AFT	FLT ENGR RIGHT-REAR	ASTRONAUT LEFT-SIDE	ASTRONAUT RIGHT-SIDE	NAVIGATOR FRONT RIGHT	NAVIGATOR RIGHT SIDE	AVIONIC COMPARTMENT						TOTAL SQ IN	TOTAL SQ FT
C-5A							761	2560							3321	23
1011		198	165	1144											1507	10.25
S-3A									677						677	4.7
GEMINI		192			72	72									336	2.3
LM					425	425									850	6
CM					120	240									360	2.5

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Table 2.2-30

DISPLAY TRADEOFFS

TYPE	FEATURES	S/S USE	STATUS
DISCRETE			
o INCANDESCENT	HIGH BRIGHTNESS/SIMPLE	COUNTERS/ BARGRAPHS	COMMERCIAL AND MILITARY JETS
o GAS DISCHARGE	MEDIUM BRIGHTNESS/LIFE		AVAILABLE COMMERCIAL INDICATORS
o LIGHT-EMITTING DIODES	HIGH BRIGHTNESS/LIFE		PROTOTYPE
o ELECTROLUMINESCENT	SLOW FAILURE/MULTI-COLOR		APOLLO
o FLUIDIC	HIGH AMBIENT USE/ REFLECTIVE		EXPERIMENTAL
o LIQUID CRYSTAL	FIXED MESSAGE/ REFLECTIVE		EXPERIMENTAL
FIXED STORAGE			
o FILM	LARGE INFORMATION CONTENT	MAPS AND INSTRUCTIONS	AIRCRAFT MAP DISPLAYS
o HOLOGRAM	LARGE INFORMATION CONTENT		EXPERIMENTAL
SERVOS	UNAMBIGUOUS DISPLAY	INSTRUMENTS ADI, HSI, ENG	APOLLO, COMMERCIAL AND MILITARY JETS

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TYPE	FEATURES	S/S USE	STATUS
CRT			
o 1-GUN B&W	HIGH BRIGHTNESS/RESOLUTION	MULTI-PURPOSE DISPLAYS	1011, SST, F14
o 3-GUN COLOR	LOW BRIGHTNESS/COLOR CONTRAST		COMMERCIAL TV
o 1-GUN COLOR	MEDIUM BRIGHTNESS/COLOR CONTRAST		PROTOTYPE
o 7-GUN	WRITING SPEED		PROTOTYPE
o STORAGE TUBE	VARIABLE PERSISTENCE		COMMERCIAL AND MILITARY JETS
o 4-GUN STORAGE TUBE	SIMULATED PERSISTENCE/MODE		PROTOTYPE
o LASER	NO VACUUM/MULTI-COLOR		EXPERIMENTAL
GRID			
o PLASMA	MEMORY/HIGH BRIGHTNESS	MULTI-PURPOSE DISPLAYS	EXPERIMENTAL
o THERMO-CHROMIC	HIGH AMBIENT/CONTRAST/ REFLECTIVE		EXPERIMENTAL
o MAGNETO OPTIC	HIGH AMBIENT/CONTRAST/ REFLECTIVE		EXPERIMENTAL

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experimental stage and are not considered sufficiently state-of-the-art devices to consider their inclusion as candidate displays.

Keyboard Tradeoffs. Six basic keyboards were examined (Table 2.2-31) relative to three specific criteria areas: (1) data management, (2) utilization, and (3) operational. Analysis results indicate that, for the requirements and constraints considered, only two keyboards generally meet the criteria. The key-select matrix keyboard incorporates fixed function keys and four-position variable indicators, thus providing considerable flexibility and functional capability. One POK folio can provide between 300 and 400 discrete functions. This technique appears promising and is a strong candidate for inclusion in the C&D subsystem.

Hand Controllers. Eight types of hand controllers were examined for their applicability to flight control (see Fig. 2.2-46). Analysis of criteria concerning these candidates results in the development of the data produced in Fig. 2.2-47. The two controller candidates, which generally meet the criteria acceptance envelope, are the dual- and single-side arm controllers (starred).

Propulsion Displays. The propulsion-engine system is composed of five separate subsystems: (1) main engines, (2) orbital maneuvering system (OMS), (3) reaction control system (RCS), (4) airbreathing engines (ABES), and (5) auxiliary power units (APU). The display of information on a dedicated hardwire basis to the crew promotes several problems in terms of panel area consumption. Analysis of these potential problems results in the development of a discrete series of steps and resultant information. Table 2.2-32 presents propulsion parameters by the five subsystems with the total parameters indicated for each. The minimum number of displays required by each subsystem (which totals 80 for the sum of the five subsystems) is indicated in Table 2.2-33 To determine the need by mission regime for each of the subsystems, requirements by mission phase are established on a time-line basis and are illustrated in Fig. 2.2-48. Next, the types and numbers of displays required, as indicated by the time-line, are developed for the five subsystems (see Fig. 2.2-49). Thus, two rows of meters, as seen in Fig. 2.2-50, are required; by sharing (hardwire) displays, engine status can be displayed based on the time-line for all parameters

Table 2.2-31

KEYBOARD TRADEOFF SUMMARY

KEYBOARD TYPES		DATA MGMNT CONSIDERATIONS				UTILIZATION CONSIDERATIONS				
		PERMITS COMPUTER COMMUNICATION	REPROGRAMMABLE ON BOARD	AVAILABLE SOFTWARE PROGRAMS	ADAPTABLE TO VARIOUS FLIGHT REGIMES AND MISSIONS	HIGH REL ATTAINABLE	VARIETY OF USE FLEXIBILITY	MINIMUM VOLUME	ALPHA/NUMERIC CAPABILITY	EASE OF USE IN PLANNED LOCATION
STANDARD NUMERIC	DEDICATED	EXTREMELY LIMITED	NO	NO	NO	YES	EXTREMELY LIMITED	YES	NO	YES
NOUN/VERB	DEDICATED	YES	NO	NO	LIMITED	YES	YES	YES	NO	YES
FIXED NOMENCLATURE	DEDICATED	VERY LIMITED	NO	NO	VERY LIMITED	YES	VERY LIMITED	LIMITED	YES	YES
KEY-CURSOR	NONDEDICATED	YES	YES	LIMITED	YES	SLIGHTLY LIMITED	SLIGHTLY LIMITED	YES	YES	LIMITED
KEY-SELECT MATRIX	NONDEDICATED	YES	YES	NO	YES	YES	SLIGHTLY LIMITED	SLIGHTLY LIMITED	YES	YES
PAGE OVERLAY	NONDEDICATED	YES	YES	NO	YES	YES	YES	YES	YES	YES

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KEYBOARD TYPES	OPERATIONAL CONSIDERATIONS								
	OPERABLE UNDER HIGH-G LOADS	MINIMIZED ATTENTION FOR ACTUATION	VERY FINE & DISCRETE MOVEMENTS REDUCED	RAPID DATA ENTRY	MINIMUM ADDITIONAL SUPPORT EQUIPMENT	MINIMUM COMPLEXITY	MINIMUM OPERATOR ERROR ENTRY POTENTIAL	PILOT ACCEPTANCE	INSTRUCTIONS MINIMIZED
STANDARD NUMERIC	YES	YES	YES	YES	YES	YES	YES	YES	NO
NOUN/VERB	YES	SLIGHTLY LIMITED	YES	YES	YES	LIMITED	SLIGHTLY LIMITED	YES	NO
FIXED NOMENCLATURE	YES	YES	YES	YES	YES	YES	YES	YES	VERY LIMITED
KEY-CURSOR	EXTREMELY LIMITED	VERY LIMITED	VERY LIMITED	LIMITED	LIMITED	LIMITED	LIMITED	YES	LIMITED
KEY-SELECT MATRIX	YES	YES	YES	YES	YES	YES	YES	YES	LIMITED
PAGE OVERLAY	YES	YES	SLIGHTLY LIMITED	SLIGHTLY LIMITED	YES	YES	YES	YES	YES

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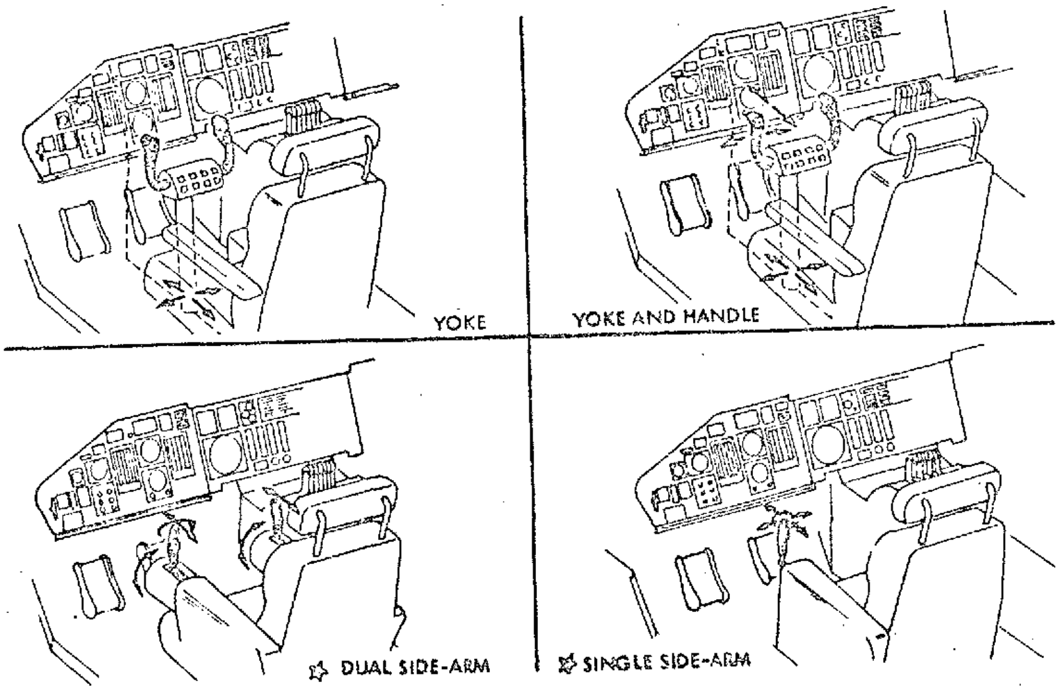

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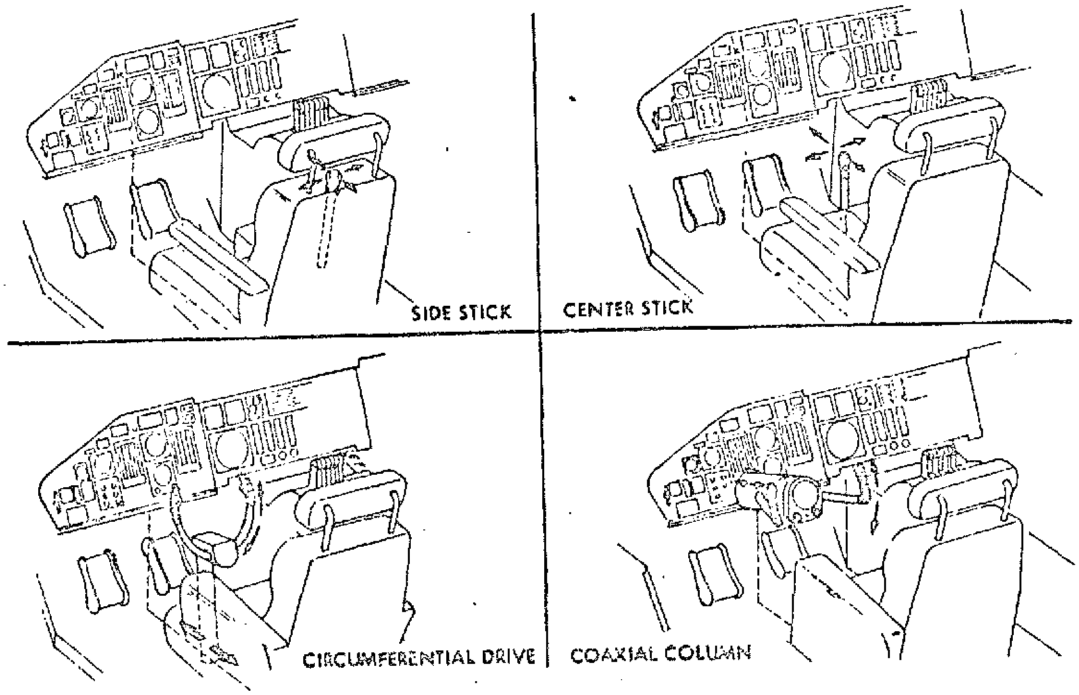

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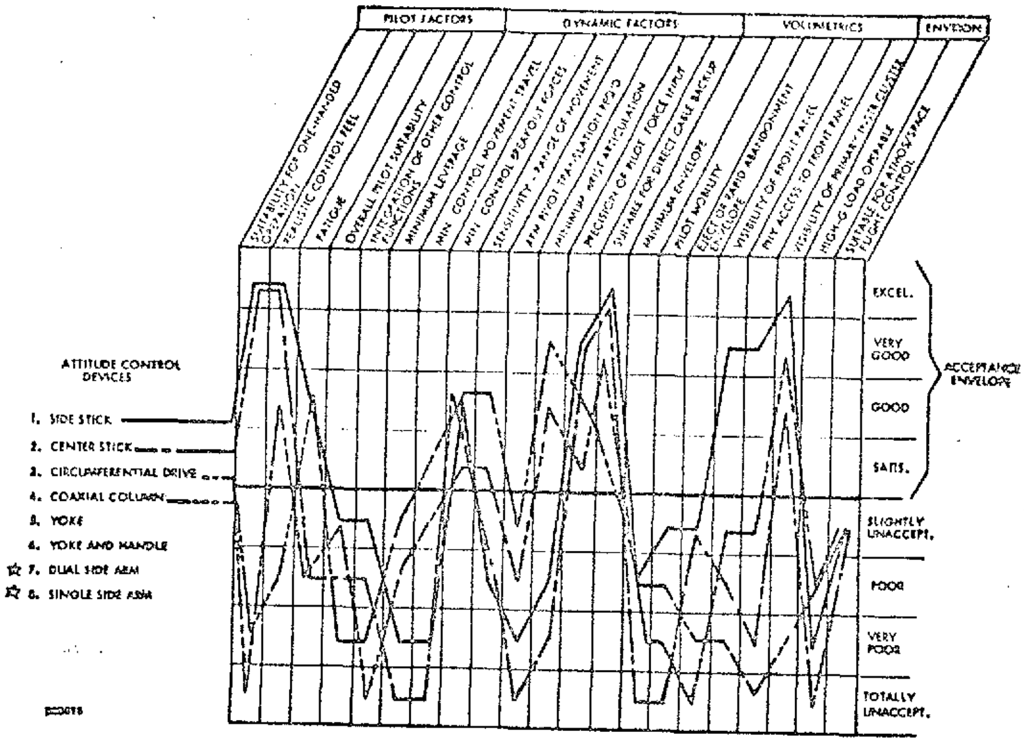
Fig. 2.2-46 Candidate Translation/Attitude Controllers



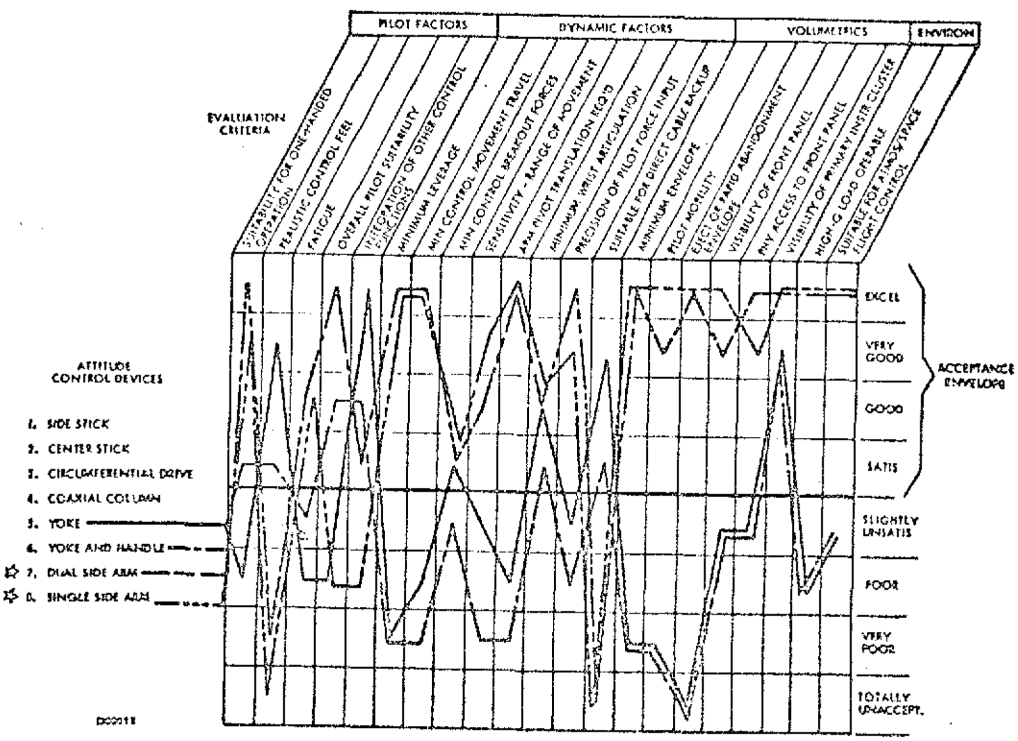


Fig. 2.2-47 Attitude Control Device Tradeoffs

Table 2.2-32

PARAMETERS BY SUBSYSTEM - PROPULSION EXAMPLE

ITEM	PARAMETER/CONDITION DISPLAY REQUIREMENT	UNITS TO BE MONITOR'D	PROPULSION SUBSYSTEM MAIN	APU	OMS	RCS	ABES
1	SUBSYSTEM IDENTITY	—	✓	✓	✓	✓	✓
2	PERCENT THRUST COMMANDED	ENG.	4	—	2	—	—
3	PERCENT THRUST BEING DELIVERED BY ENGINES	ENG.	4	—	2	—	—
4	ΔV REMAINING	TOTAL	✓	—	✓	—	—
5	ENGINE GIMBAL POSITION IN PITCH	ENG.	4	—	2	—	—
6	ENGINE GIMBAL POSITION IN YAW	ENG.	4	—	2	—	—
7	THRUST VECTOR ALIGNMENT ERROR IN PITCH	ENG.	4	—	2	—	—
8	THRUST VECTOR ALIGNMENT ERROR IN YAW	ENG.	4	—	2	—	—
9	FUEL/OXIDIZER MIXTURE RATIO (% FUEL)	ENG.	1	—	—	—	—
10	FUEL QUANTITY REMAINING OR % FUEL REMAIN.	TANKS	1	6	2	3	2
11	OXIDIZER QUANTITY REMAINING OF % OX REMAIN.	TANKS	1	(MONO. PROP)	2	3	(MONO-PROP)
12	FUEL TEMPERATURE	TANKS	1	6	2	3	2
13	OXIDIZER TEMPERATURE	TANKS	1	—	2	3	—
14	FUEL PRESSURE	TANKS	1	6	2	3	2
15	OXIDIZER PRESSURE	TANKS	1	—	2	3	—
16	HELIUM PRESSURE	TANKS	—	6	2	6	—
17	HELIUM QUANTITY	TANKS	—	6	2	6	—
18	ENGINE RPM	ENG.	—	3	—	—	—
19	COMBUSTOR OUTLET TEMPERATURE	ENG.	—	3	—	—	—
20	PERCENT RPM N_1	ENG.	—	—	—	—	4
21	PERCENT RPM N_2	ENG.	—	—	—	—	4
22	TURBINE BLADE TEMPERATURE	ENG.	—	—	—	—	4
23	FUEL FLOW	ENG.	—	—	—	—	4
24	ENGINE OIL TEMPERATURE	ENG.	—	—	—	—	4
25	ENGINE OIL PRESSURE	ENG.	—	—	—	—	4
							4
26	ENGINE OIL QUANTITY	ENG.	—	—	—	—	4
27	ENGINE VIBRATION - FAN STAGE	ENG.	—	—	—	—	4
28	ENGINE VIBRATION - TURBINE STAGE	ENG.	—	—	—	—	4
		TOTAL PARAMETERS	32	36	29	30	42

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Table 2.2-33

MINIMUM DISPLAYS REQUIRED BY PROPULSION SUBSYSTEM

INFORMATION TYPE	MIN. NO. OF DISPLAYS REQUIRED	PROPULSION SUBSYSTEM PARAMETER: MAIN	MAIN SCALES REQ'D 4	3	2	1	APU	APU SCALES REQ'D 4	3	2	1	OMS	OMS SCALES REQ'D 4	3	2	1	RCS	RCS SCALES REQ'D 4	3	2	1	ABES	ABES SCALES REQ'D 4	3	2	1
QUANTITY (PROPELLANTS, HELIUM, OIL)	12	2			✓		12	✓				6	✓		✓		12		✓			6	✓		✓	
TEMPERATURE (PROPELLANTS, HELIUM, OIL, OTHER)	10	2			✓		9	✓				4	✓				6		✓			10	✓		✓	
PRESSURE (PROPELLANTS, HELIUM, OIL, OTHER)	12	2			✓		12	✓				6	✓		✓		12		✓			6	✓		✓	
RPM	8	—					3	✓				—					—					8	✓			
FLOW	4	—					—					—					—					4	✓			
POSITION OF ENGINES & ERROR	16	16	✓				—					8			✓		—					—				
THRUST (COMMAND & ACTUAL)	8	8	✓				—					4			✓		—					—				
ΔV	1	1				✓	—					1				✓	—					—				
MIXTURE RATIO	1	1				✓	—					—			✓		—					—				
VIBRATION	8	—					—					—					—					8				
TOTAL INDIVIDUAL INDICATORS	80	32					36					29					30					42				

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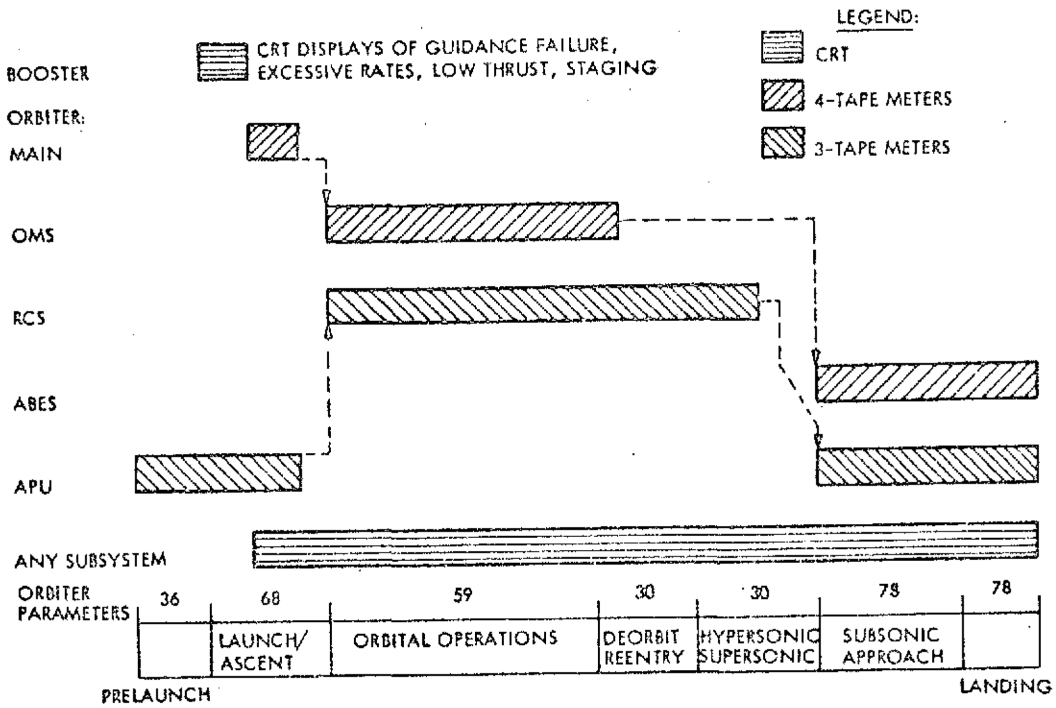

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Fig. 2.2-48 Propulsion Display Requirements by Mission Phase

TYPES OF DEDICATED INDICATORS	PROPULSION SUBSYSTEM					MINIMUM NUMBER REQUIRED
	MAIN	APU	OMS	RCS	ABES	
4-TAPE VERTICAL METER	6		4		9	9
3-TAPE VERTICAL METER		12		10		12
2-TAPE VERTICAL METER	3		6		3	6
SINGLE-FUNCTION METER	2		1			2
TOTAL	11	12	11	10	12	29

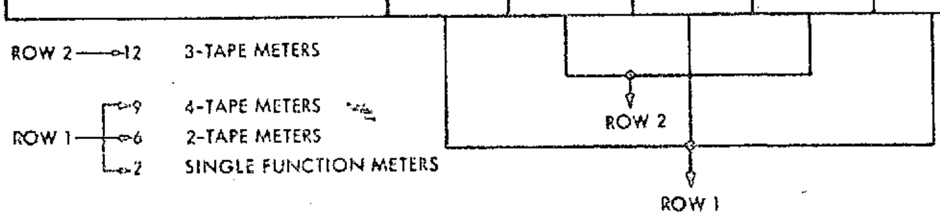

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Fig. 2.2-49 Types and Numbers of Propulsion Displays Required

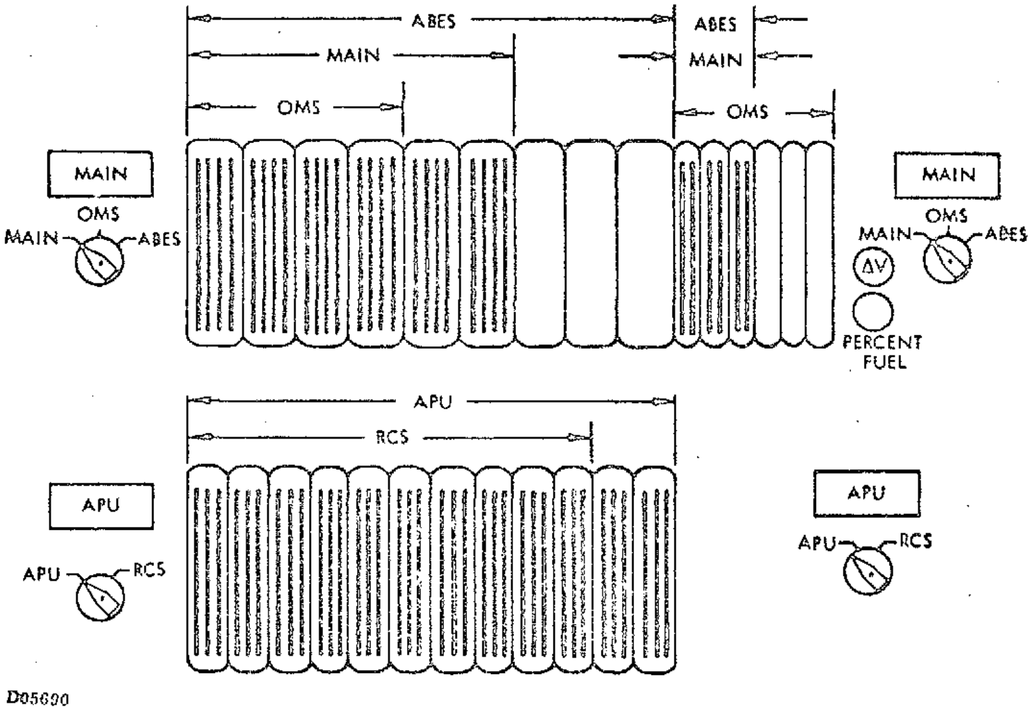


Fig. 2.2-50 Propulsion Subsystem Dedicated Displays

indicated in Table 2.2-31. Accordingly, only 29 displays of the original 80 need be provided, based on this common share principle, while still maintaining a dedicated hard-wire philosophy. Furthermore, the panel reduction can be considered at least 50 percent of the original requirement.

2.2.1.5 Data Management and Onboard Checkout. The baseline Data Management Subsystem (DMS) is fundamental to the accomplishment of the goals of performing pilotage tasks with a two-man crew, meeting reduced turnaround times of 30 days for Mark I and 2 weeks for Mark II, and achieving sufficient autonomy for an airplane type of operation, i.e., be independent of an earth based Mission Control Center (MCC). The baseline DMS is incorporated into the orbiter to assist both the space and ground support crews in decision making in all aspects of the mission, and to perform routine tasks that are attendant to the required mission. A single-thread nonredundant concept identical to the S3A DMS, complete with a hard-wired backup capability for all safety of flight (SOF) functions, was selected as being compatible with program needs while minimizing development and attendant programmatic costs and schedule risks.

The degree of DMS participation in performance of tasks is planned to progress from a low level during early horizontal flight test to nearly complete automatic control with crew override option as a growth capability in Mark II.

Phased acquisition of the DMS is accomplished as shown in Table 2.2-34. This approach eliminates the parallel development of a development flight test instrumentation (DFI) subsystem while simultaneously gaining demonstrated confidence in the operational DMS hardware in its ultimate system environment. Operational functions (including effectivity) accomplished within the DMS are listed in Table 2.2-35.

2.2.1.5.1 Baseline Approach. The baseline DMS is an integrated adaptation of existing, developed subsystems and supporting software. Development is limited to hardware and software modifications at the module level and integration of the subsystems into a versatile system through hardwired interface units.

The developed subsystems integrated into the 040A baseline DMS are: the S3A data management subsystem consisting of a UNIVAC 1832 computer, 6MHz dedicated data transmission lines, and standardized system interface units (SIU); an analog multiplex data gathering subsystem from the LMSC SESP program; and elements of the Apollo Block II communications and data subsystem. The SESP multiplex subsystem and the

Table 2.2-34

DATA MANAGEMENT SUBSYSTEM PHASED ACQUISITION

Phase	DMS Status
Development prior to FHF	Bench integration of S3A computer subsystem, P-50 instrumentation multiplex subsystem and Apollo telemetry subsystems.
FHF	S3A computer subsystem, P-50 instrumentation subsystem, and Apollo TLM subsystem employed as an integrated DFI.
FVF	S3A computer subsystem and P-50 instrumentation employed as operational crew support, flight test engineer support, and integrated DFI. Added P-50 instrumentation multiplexing to support DFI acquisition of data for test phase only. Apollo TLM subsystem for DFI downlink and proof of operational capability.
Mark I	S3A computer subsystem, P-50 multiplexing subsystem (operational data only), and Apollo TLM subsystem as an earth cooperative support subsystem. Automatic control limited to the instrumentation and electrical power subsystem in flight, available for all avionics S/S control between flight tests.
Mark II	S3A computer subsystem and P-50 multiplexing subsystem providing crew support necessary for autonomous operation. Apollo TLM retained as an emergency support option. Automatic subsystem control capability limited by funding only.

Table 2.2-35

DATA MANAGEMENT FUNCTIONS EFFECTIVITY

FUNCTION	HFT	MARK I VFT	OP'L	MARK II
ONBOARD CO/FI AND DATA EXTRACTION	0	0	0	0
INSTRUMENTATION AND ELECTRICAL POWER CONTROL	0	0	0	0
ABORT AIDS	0	0	0	0
GN&C COMPUTATIONS		0	0	0
ONBOARD CO/FI/RM		0	0	0
SYSTEM MANAGEMENT AIDS		0	0	0
AVIONICS CONFIGURATION CONTROL			0	0
CONSUMABLES MANAGEMENT			0	0
RENDEZVOUS COMPUTATION			0	0
PAYLOAD MANAGEMENT				0
A/C AND S/C FLIGHT CONTROL				0
NONAVIONICS CONFIGURATION CONTROL				0
MISSION PLANNING				0

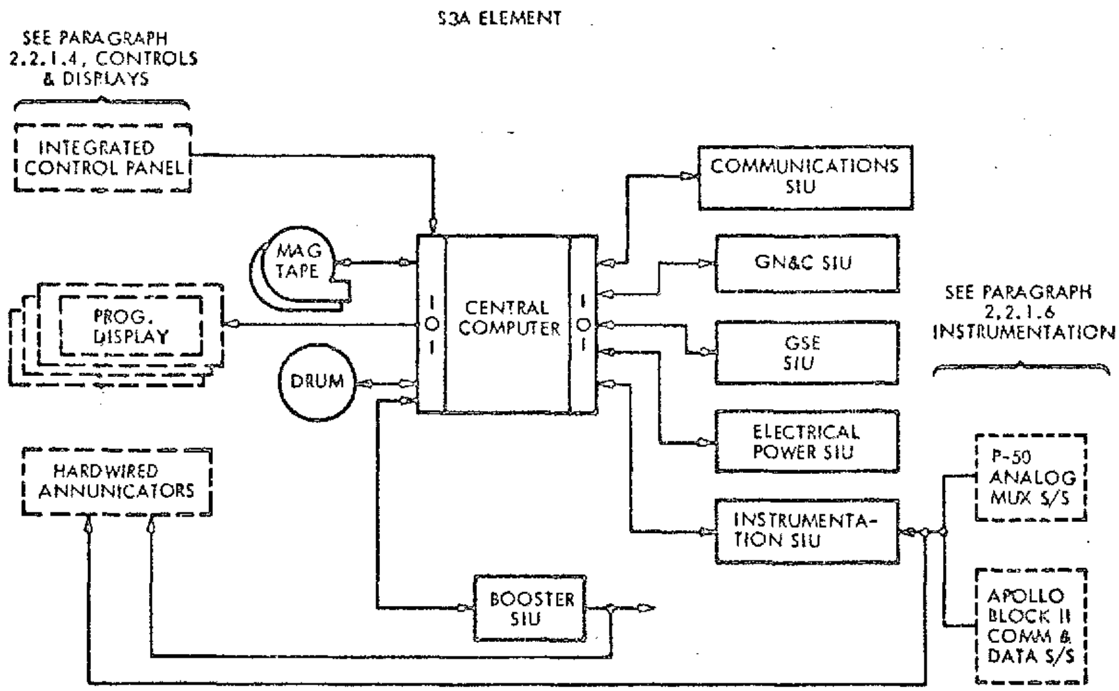
Apollo communications and data subsystem elements are discussed in par. 2.2.16, Instrumentation. The S3A adaptation which forms the heart of the on-board DMS is detailed in the following paragraphs. An overview block diagram of the S3A DMS baseline is shown in Fig. 2.2-51.

The DMS is integrated around a general purpose digital computer system (UNIVAC 1832) with dual central processors used in the S3A avionics system. Each central processor has access to two banks of deposited film memory, each containing 32,768 32-bit words, with effective cycle times of 750 μsec. If operands and instructions are in different banks, instruction overlap is possible, yielding add times (including access) of 1.3 μsec and multiply/divide times of 12.1 μsec.

Input/Output (I/O) is provided through the Input-Output Controller (IOC) and Input Output Interface (IOI) units (Fig. 2.2-52). The IOC units provide the control logic which interprets and carries out I/O operations. A self-contained command repertoire allows direct access to and from core memory, without interruption of central processing. Each of the IOC units operates independently, and may gain access to either memory bank. Each IOC may control up to eight bi-directional serial channels and one parallel channel, yielding a capability (with both IOCs active) of data transfer rates of 1.3 million words per sec. The IOCs accept and generate various types of interrupts.

The IOI units (Figs. 2.2-52 and 2.2-53) provide signal level matching, serial/parallel conversion, line receivers and transmitters, and storage necessary for buffered communication with external avionics equipment through SIUs. Each IOI operates independently and may operate with either IOC. IOI design is modular by channel; no single failure will disable more than one channel.

The IOI units communicate with the other subsystems through SIUs, which terminate the 6 MHz serial shielded twisted-pair lines that are the standard communication means of the system. Each SIU contains a receiver/transmitter pair, response control logic, and storage for the standard 36-bit communications word (Figs. 2.2-54 and 2.2-55). The remaining portion of the SIU is tailored to the particular subsystem interface required.



DO6411

Fig. 2.2-51 S3A Elements of Data Management – Mark I Orbiter

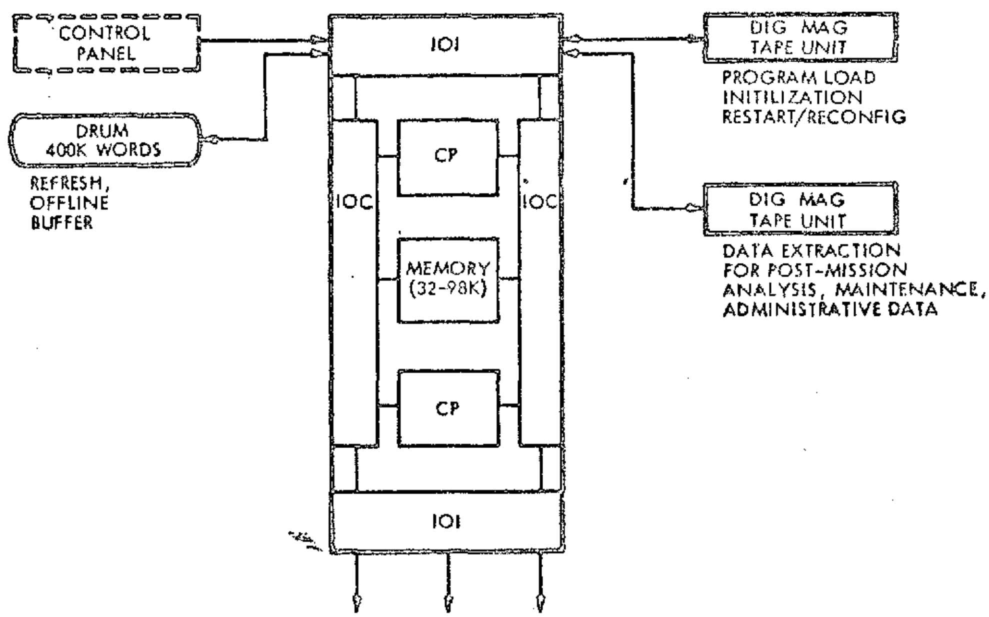


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Fig. 2.2-52 DMS Central Computer Subsystem

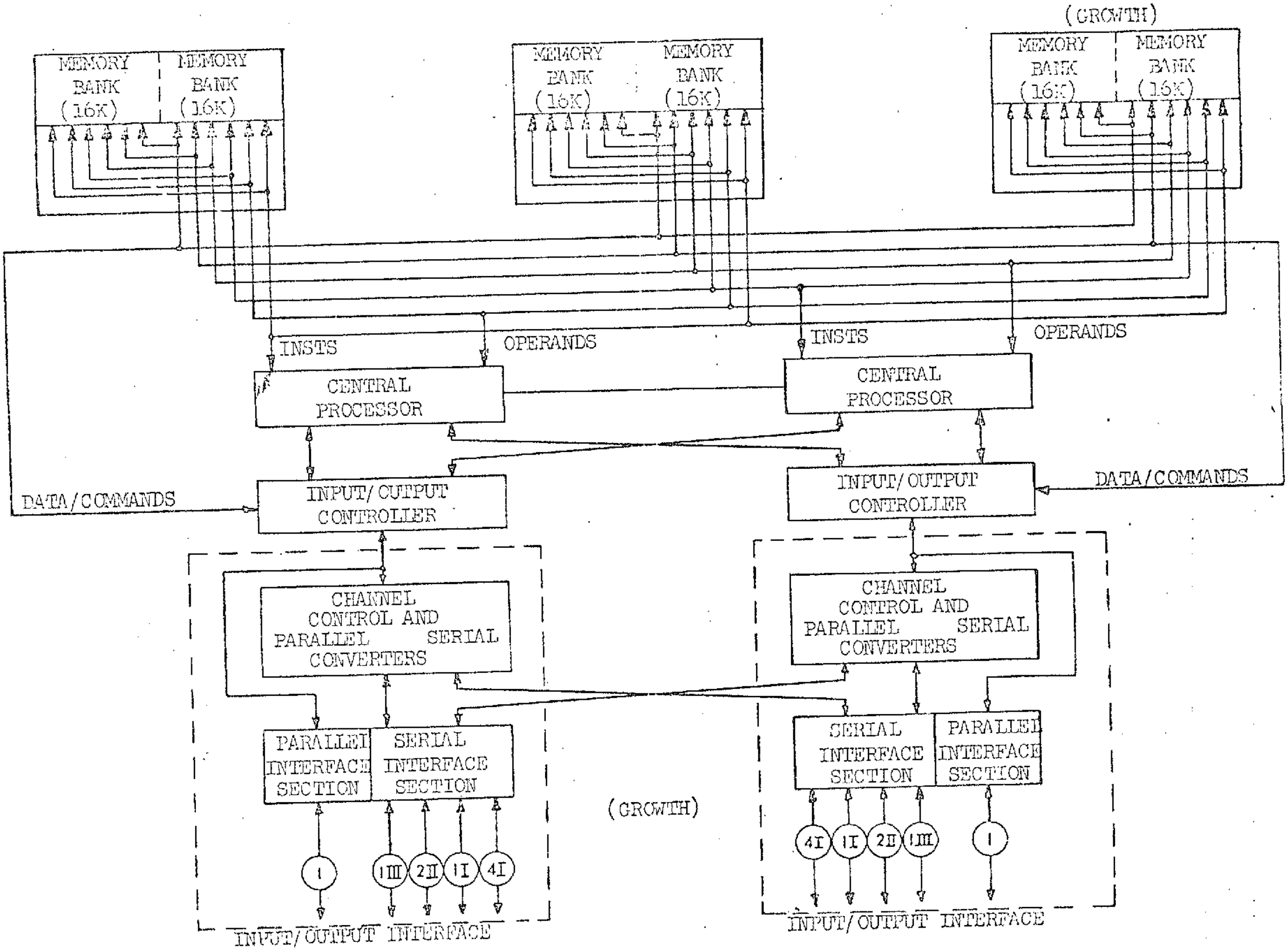

Fig. 2.2-53 GPDC Block Diagram

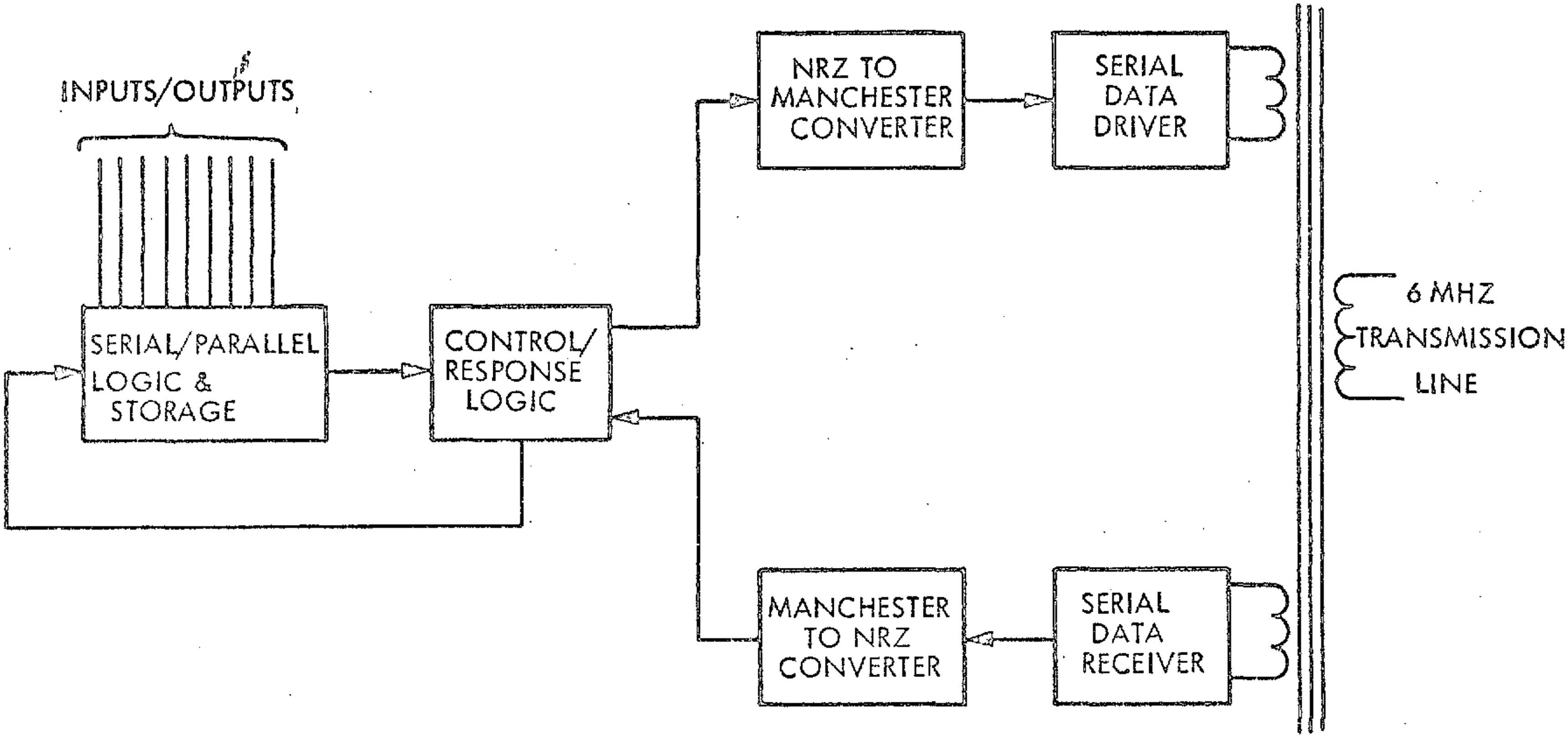

Fig. 2.2-54 SIU/Transmission Line Interface

The primary Mark I SIUs are adapted from the S3A program to interface with the avionics subsystems, i.e. communications and tracking, electrical power, guidance/navigation and control, control/display, and instrumentation. The hardware capability for monitoring and control exists within this basic Mark I configuration; however, only the electrical power subsystem will be operationally commanded in Mark I.

The existing S3A capability for automatic checkout of flight readiness is employed for the avionics subsystems through their SIU interfaces with the DMS. Those signals required to activate internal BITE and to initiate subsystem action attendant to checkout for flight readiness are either originated in the SIU through computer command or implemented by personnel at the flight station through operational controls on cue from the DMS via programmable display.

Passive monitoring for analysis of nonavionic subsystems is provided through the instrumentation interface into the DMS central computer, control being supplied by crew action if necessary for checkout and fault isolation.

An interface is provided for tying the GSE automatic checkout facility and the on-board DMS into a single unified checkout and fault isolation system for maintenance and launch operations.

The IOIs also communicate more directly with the magnetic drum and digital magnetic tape mass storage devices. The drum has capacity for 400,000 32-bit words with average access time of 12.5 msec. Each tape has capacity for 3.5 million 32-bit words, giving total system mass storage of 7,400,000 32-bit words.

An integrated control panel is also serviced through the IOI, allowing direct manual man-machine communication. The operation of the keyboard is software-interpreted, allowing key function to change from mission mode to mode. The programmed displays are versatile cathode ray tube devices that allow the presentation of both graphic and alphanumeric information to the crew. Both operations peculiar and supplemental trouble-shooting data are accessed as required.

The IOI communicates with booster subsystems through the booster SIU, which is contained in the orbiter.

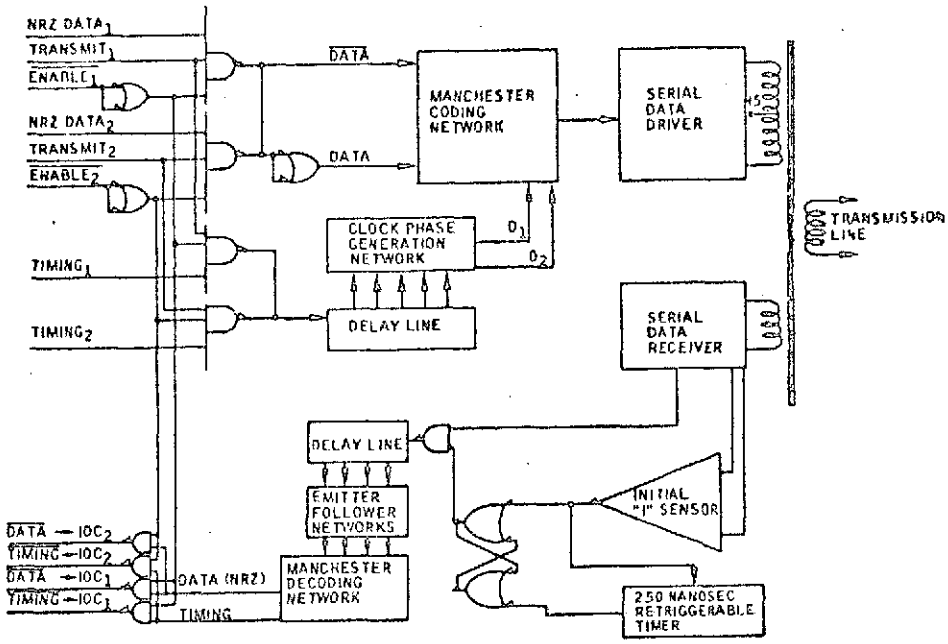

Fig. 2.2-55 S3A Serial Driver/Receiver – Coder/Decoder Block Diagram

Hardwired flight-critical annunciators bypass the DMS by being wired in parallel from the input to instrumentation and booster SIUs for Safety-of-flight on-board checkout and abort warning functions.

2.2.1.5.2 Composite Aircraft/Spacecraft Avionics. The merging of aircraft and spacecraft functions in the DMS occurs primarily in the areas of instrumentation and in maintenance and mission logging on the digital magnetic tape. The integrated control panel and programmatic displays perform functions in either areas, as the software directs.

The S3A program has developed the techniques and hardware necessary to successfully integrate avionics and electrical/electronic units into a centralized DMS through the SIU, using a combination of standard and specialized interface circuits. The baseline DMS employs identical methods of consolidating available BITE outputs from the S3A, L-1011, Apollo, and other manned program hardware selected for the avionics system. For the limited units which do not include BITE sufficient for preflight readiness validation, the signal/logic validation techniques developed on the C5A MADAR are implemented to achieve a self-contained checkout and fault isolation capability within the orbiter for avionics subsystems in the Mark I phase.

An interface between the DMS computer and the GSE automatic checkout complex permits ground crew access to all data within the DMS computer and provides a capability for failure analysis by the flight crew during the mission.

A conceptual overview of the DMS in the operational environment is displayed in Fig. 2.2-56, 2.2-57, and 2.2-58.

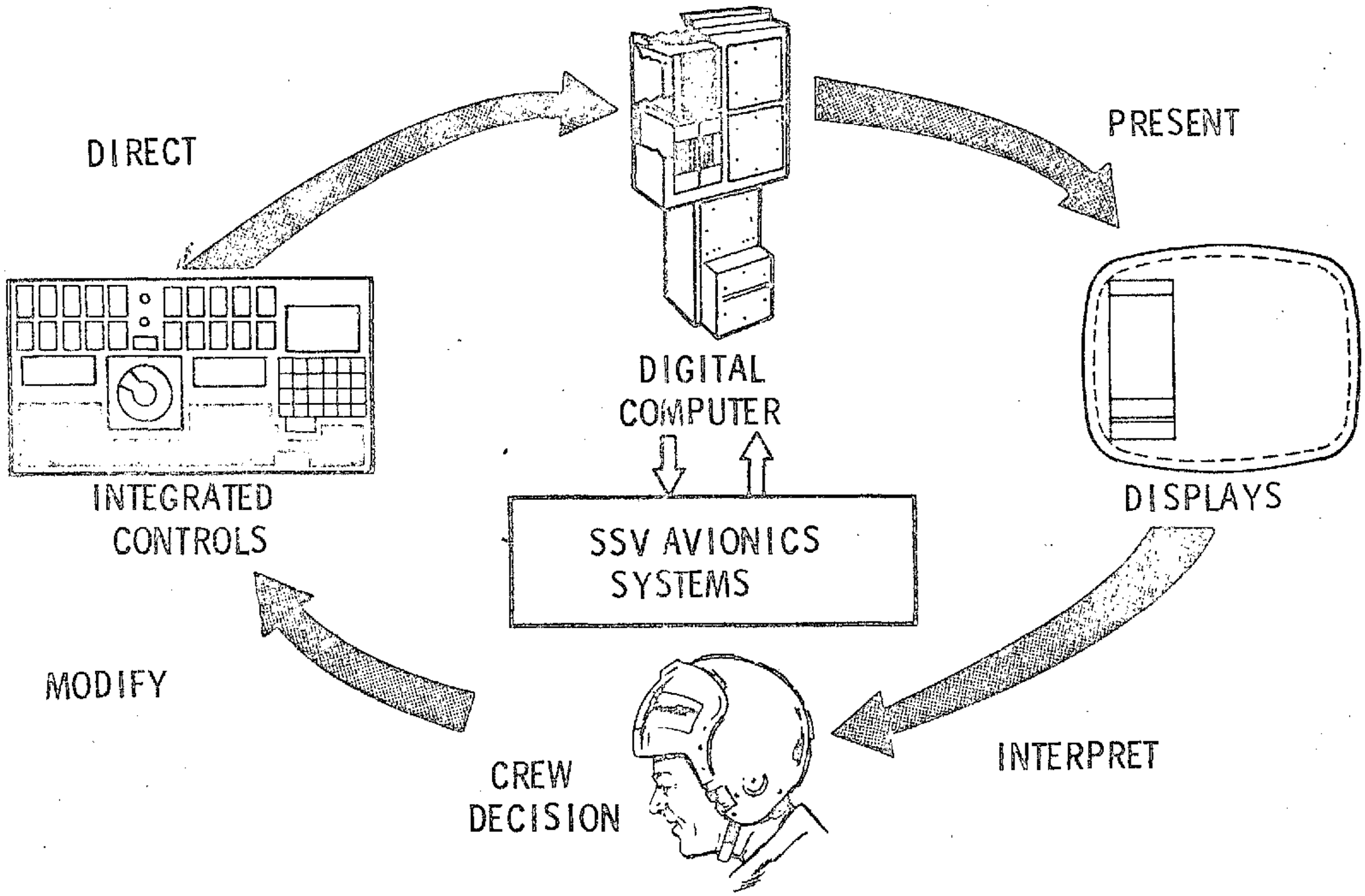

Fig. 2.2-56 DMS Operational Concept

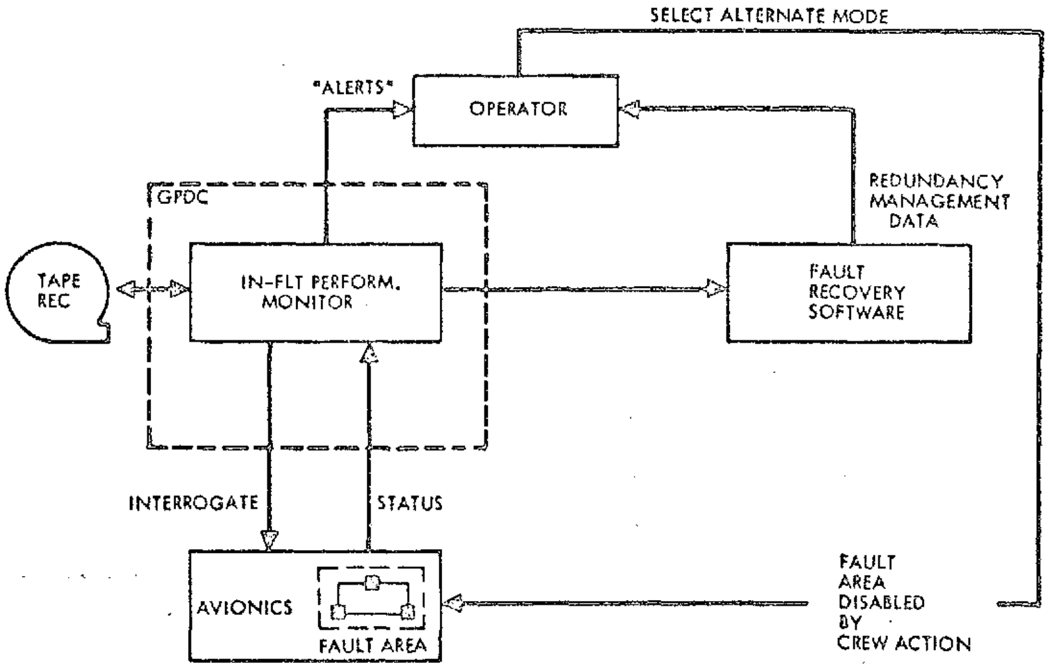

Fig. 2.2-57 DMS Inflight Test Operations

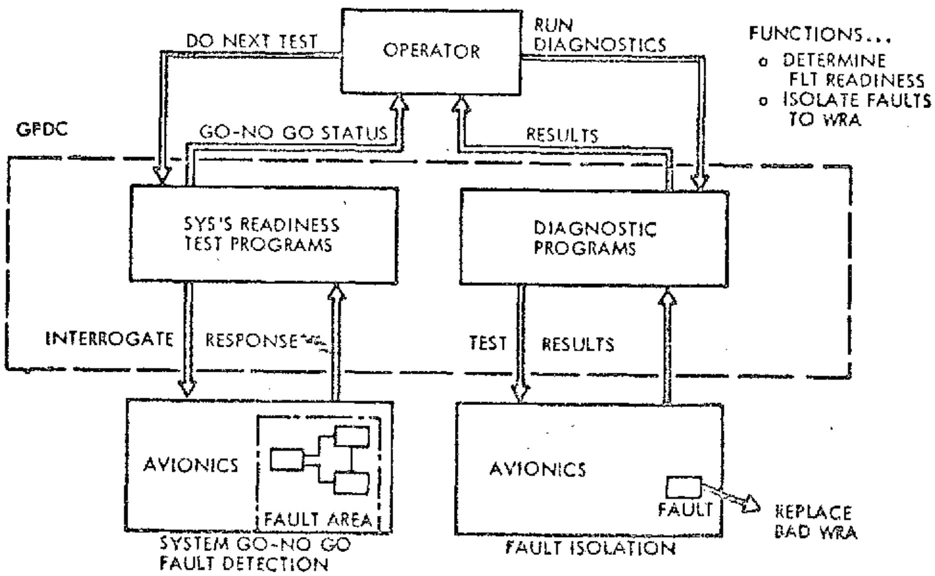

Fig. 2.2-58 DMS Preflight Test Operations

2.2.1.6 Instrumentation. The baseline instrumentation subsystem for the Mark I Orbiter comprises transducing, signal conditioning, and multiplexing equipment; telemetry formatting controls; FM recording equipment; a flight data recorder; and a time code generator. The instrumentation subsystem includes both operational and developmental flight instrumentation (DFI) in its formatting and recording capabilities to meet the data requirements summarized in Table 2.2-36. DFI multiplexing and cabling is separated from the operational so that the DFI is easily removable with minimum labor and scar effect.

2.2.1.6.1 Multiplexing and Telemetry Formatting. The multiplexing and formatting of instrumentation data is under control of the data management subsystem through the instrumentation SIU. Communication between the DMS computer and the SIU is by 36-bit words as in the standard S-3A 6 MHz-line interface. SESP type multiplexer cards are used for the first level of multiplex; each card can switch one of 32 lines to a single line under control of a 5-bit address. The address is furnished to the specific card from the DMS computer via the instrumentation SIU. The selected Apollo Block II PCM telemetry equipment has multiplexing capability for 365 analog high level points, 264 discrete inputs, and one 40-bit serial word. The handling of the various classes of instrumentation data is discussed in the following paragraphs and diagrammed in Fig. 2.2-59.

Analog Data Multiplexing. Approximately 200 analog points are considered to be safety-of-flight items. These are fed directly into 200 points of the Apollo analog multiplexer. For the noncritical points both analog and discrete, 2560 can be multiplexed using SESP 32-point multiplexers on 80 points of the Apollo PCM multiplexer and 80 SESP multiplexer cards. Eighty-five channels are available for growth and/or faster sample rates.

Discrete Data Multiplexing. There are 264 discrete data lines into the digital multiplexer of the Apollo Block II PCM telemeter. Assuming 100 of the discrete signals are flight critical, these are tied directly to the discrete input line. Another 164 fast-sample rate high priority discrete points can be tied to the direct discrete inputs, as well as being digitized and transmitted through the analog channel for redundancy.

Table 2.2-36

O4OA ORBITER INSTRUMENTATION SUMMARY

Subsystem Equipment	FHF Oper	FHF Dev	FVF Man Oper	FVF Man Dev	FVF Unman Oper	FVF Unman Dev	Operational Mark I
Communications	9	1	10	-	10	-	10
Instrumentation	2	7	2	7	2	7	2
(S/C) GN&C Attitude Control			410	64	410	64	552
(A/C) Navigation	54	34	-	-	-	-	-
(A/C) Flight Control	254	192	254	42	280	44	254
Display and Controls	20	20	20	-	20	-	20
Nav Aids (S/C and A/C)	32	26	70	6	70	6	78
Data Management	90	46	90	-	90	-	90
ECLSS	72	-	72	-	48	-	106
Propulsion	32	102	308	466	308	466	308
Struct/Mechanical	86	538	143	301	143	301	97
Hydraulic Power	60	89	60	6	60	6	60
Elect. Pwr Distribution	130	66	130	60	130	60	130
Elect. Pwr Generation	18	20	104	71	104	71	104
Total	859	1141	1673	1023	1675	1025	1811
	2000		2696		2700		1811

Note: Oper = Instrumentation points required for basic operation
Dev = Instrumentation points required for development flight test (DFI)

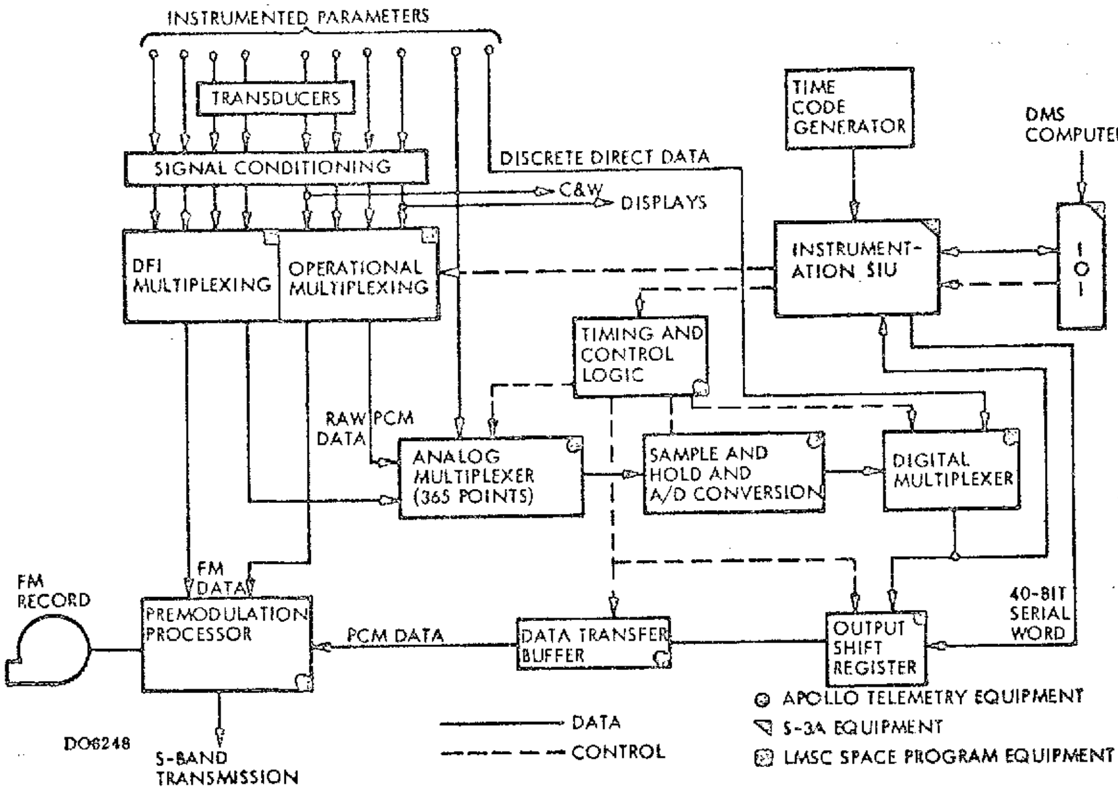

Fig. 2.2-59 Baseline Instrumentation Subsystem

Serial Data Transmission. The 40-bit serial data word input is used for direct digital transmission on the PCM link between the DMS computer and receiving station (e.g., for time codes and crew entered data).

FM Data Transmission. Data not suitable for digital transmission are transmitted via the FM data input to the Apollo Block II premodulation processor of the system. Formatting is accomplished again by the instrumentation multiplexing. Biomedical data may be transmitted on available bandwidth voice transmission channel, under control of the communications SIU.

2.2.1.6.2 Integrated Vs Overlay DFI. The multiplexing hardware used for instrumentation consists of 32-point SESP multiplexers and Apollo Block II equipment. Because of the inherent capability of the system, it was decided to integrate the DMS bit stream and the SESP instrumentation system at the input to the Apollo equipment, rather than duplicate the computer control and telemetry system. The DFI first level multiplexing, wiring, and signal conditioning would be installed as an overlay for minimum scar effect and minimum cost of removal.

Changes in software are limited to changing only the instrumentation tables and the telemetry format.

The DFI system, then, is neither fully integrated nor fully overlayed, but a combination that fits within the instrumentation/DMS capability at minimum cost.

2.2.1.7 Baseline Orbiter Avionics Weight Summary. Available equipment has been identified for mechanizing each subsystem. The subsystem weights are summarized in Table 2.2-37 for each major development phase. The final operational avionics weight is 9086 pounds, including a 10-percent assessment for equipment installation. The detailed listings of equipment, unit weights, and prior program application are given in Tables 2.2-38 A through F. These lists identify the required sequence for adding equipment. It should be noted that the major change is from first horizontal flight test to the vertical flight program. However, the use of FTV-1 as a passenger carrier to accumulate operating experience on FTV-2 equipment should result in a gradual buildup of capability; i.e., the indicated step function will not occur.

Table 2.2-37

MARK I ORBITER AVIONICS WEIGHTS SUMMARY

Subsystem	FHF Wt. (lbs)	FVF Man Wt (lbs)	FVF Unman Wt (lbs)	Operational Wt (lbs)
Guidance, Navigation and Controls	1119	2018	2018	2275
Communication and Navigation Aids	316	436	436	531
Electrical Power Generation, Control and Distribution	2332	3696	3696	3696
Displays and Controls	751	1049	1049	717
Data Management	555	665	665	665
Instrumentation	493	664	693	376
Sub-Total	5566	8528	8528	8260
Installation (10%)	556	853	853	826
TOTAL	6122	9381	9381	9086

Table 2.2-38A MARK I ORBITER AVIONICS BASELINE EQUIPMENT LIST

Guidance, Navigation and Control Equipment	Unit Wt. (lbs)	FHF Quan.	FHF Total Wt.	FVF Man Quan.	FVF Man Total Wt.	FVF Unman Quan.	FVF Unman Total Wt.	Oper. Quan.	Oper. Total Wt.	Program Source
2.1.1										
IMU	53			2	106	2	106	3	159	747
Digital Comp. (1832)	126			1	126	1	126	1	126	S-3A
Star Tracker	77							2	154	Bendix ATM
Horiz. Sensor	25							2	50	Agena
Navig. Data Repeater Converter	55			2	110	2	110	2	110	S-3A
TVC Electronics	30			3	90	3	90	3	90	Agena
ACPS Electronics	50			3	150	3	150	3	150	Agena
M. E. Gimbal Actuator	50			8	400	8	400	8	400	S IV B
Subtotal					982		982		1239	
2.1.1.2										
Direct Gyro	15	2	29							C-5A
Compass Coupler	9	2	17							C-5A
Compass Controller	1	2	2							C-5A
Flux Valve	1	2	4							C-5A
Vertical Gyro	15	2	29							C-5A
Mag Compensator	1	2	2							C-5A
Subtotal			83							
2.1.1.3										
Rudder Servos	48	2	95	2	95	2	95	2	95	C-5A
Elevon Servos	48	8	384	8	384	8	384	8	384	C-5A
PFCS Servos	5	10	50	10	50	10	50	10	50	Stol
Trim/Backup Servos	1	10	10	10	10	10	10	10	10	C-141
Speed Brake Servo	10	4	40	4	40	4	40	4	40	New
Elevon Computer	23	4	92	4	92	4	92	4	92	L-1011
Rudder Computer	23	2	46	2	46	2	46	2	46	L-1011
Air Data Sensor Assy / Central Air Data Comp.	29	2	59	2	59	2	59	2	59	YF-12 / S-3A

Table 2.2-38A MARK I ORBITER AVIONICS BASELINE EQUIPMENT LIST

Guidance, Navigation and Control Equipment	Unit Wt. (lbs)	FHF		FVF Man		FVF Unman		Oper		Program Source
		Quan.	Total Wt.	Quan.	Total Wt.	Quan.	Total Wt.	Quan.	Total Wt.	
2.1.1.3 Continued										
Pitch Rate Gyros	1.8	4	7	4	7	4	7	4	7	L-1011
Roll Rate Gryos	2	4	7	4	7	4	7	4	7	L-1011
Yaw Rate Gryos	2	4	7	4	7	4	7	4	7	L-1011
Normal Accel.	1.1	4	4	4	4	4	4	4	4	L-1011
Lateral Accel	1.1	4	4	4	4	4	4	4	4	L-1011
Longitudinal Accelerometer	1.1	2	2	2	2	2	2	2	2	L-1011
Pitch APFDS Computer	28	2	56	2	56	2	56	2	56	L-1011
Roll APFDS Computer	28	2	56	2	56	2	56	2	56	L-1011
Eng. Spd. Contr. Sys.	15	4	60	4	60	4	60	4	60	AH 56
Speed Computer	27	1	27	1	27	1	27	1	27	L-1011
Anti-skid Controller, Touch Down SW., Wheel Spd. Sensor	30	1	30	1	30	1	30	1	30	New
Subtotal			1,036		1,036		1,036		1,036	
TOTAL			1,119		2,018		2,018		2,275	

Table 2.2-38B MARK I ORBITER AVIONICS BASELINE EQUIPMENT LIST

Communications and Tracking Equipment	Unit Wt. (lbs)	FHF Quan.	FHF Total Wt.	FVF Man Quan.	FVF Man Total Wt.	FVF Unman Quan.	FVF Unman Total Wt.	Oper Quan.	Oper Total Wt.	Program Source
2.1.2										
Pre- mod Proc.	11	1	11	1	11	1	11	1	11	Apollo
Unif. S-Band Eq.	38	1	38	1	38	1	38	1	38	Apollo
S-Band P.A.	32	1	32	1	32	1	32	1	32	Apollo
UHF SCVR	33	2	66	2	66	2	66	2	66	S-3A
Up Data Link	22	1	22	1	22	1	22	1	22	Apollo
S-Band Ant. Switch	3	1	3	1	3	1	3	1	3	Apollo
UHF Ant. Switch	2	1	2	1	2	1	2	1	2	S-3A
Antenna	2	2	4	2	4	2	4	2	4	Apollo
ATC Transponder	12	2	24	2	24	2	24	2	24	C-5A
ATC Antenna	3	2	6	2	6	2	6	2	6	C-5A
VHF Rec. Beacon	13	1	13	1	13	1	13	1	13	Apollo
VHF Rec. Beacon Ant.	1	1	1	1	1	1	1	1	1	Apollo
Tacan	37	2	74	2	74	2	74	2	74	S-3A
ILS RCVR	10	2	20							C-5A
AILS RCVR	15			2	30	2	30	2	30	C-Scan
Radar Altimeter	10			2	20	2	20	2	20	S-3A
Orbit Altimeter	45			2	90	2	90	3	135	Sky Lab
Prec. Ranging System	25							2	50	Ciris
TOTAL			316		436		436		531	

Table 2.2-38C MARK I ORBITER AVIONICS BASELINE EQUIPMENT LIST

Electrical Power Equipment	Unit Wt. (lbs)	FHF		FVF Man		FVF Unman		Oper		Program Source
		Quan.	Total Wt.	Quan.	Total Wt	Quan.	Total Wt.	Quan.	Total Wt.	
<u>2.1.3</u>										
Electric Power Distribution										
Gen. Ctrl. Unit	8	3	24	3	24	3	24	3	24	S-3A
SFMR/Rectifier	18	3	54	3	54	3	54	3	54	P3V
DC Bus	595	2	1,190	2	1,190	2	1,190	2	1,190	New
AC Bus	290	2	580	2	580	2	580	2	580	New
Static Inverter	40	4	160	4	160	4	160	4	160	Apollo
DC Distribution Units	10	6	60	6	60	6	60	6	60	New
AC Distribution Units	10	2	20	2	20	2	20	2	20	New
Subtotal			2,088		2,088		2,088		2,088	
Electric Power Generation										
Fuel Cell (2K Hr)	298			3	894	3	894	3	894	P&W Dev.
Cryo H_2 Supply (Tank only)	97			2	194	2	194	2	194	AAP
Cryo O_2 Supply (Tank only)	138			2	276	2	276	2	276	AAP
Battery (Emerg.)	62	2	124	2	124	2	124	2	124	Agena
A.C. Generator	40	3	120	3	120	3	120	3	120	Mod. S-3A
Subtotal			244		1,608		1,608		1,608	
TOTAL			2,332		3,696		3,696		3,696	

Table 2.2-38D MARK I ORBITER AVIONICS BASELINE EQUIPMENT LIST

Displays and Controls Equipment	Unit Wt. (lbs)	FHF Quan.	FHF Total Wt.	FVF Man Quan.	FVF Man Total Wt.	FVF Unman Quan.	FVF Unman Total Wt.	Oper Quan.	Oper Total Wt.	Program Source
Eyebrow/Overhead Panel:										
Engine Fire Control Panels	8.0	1	8.0	1	8.0	1	8.0	1	8.0	L-1011
EC/LS Gas Supply Override Valves	22.3	-	-	1	22.3	1	22.3	1	22.3	-
Elect Pwr Generation and Dist	6.9	1	6.9	1	6.9	1	6.9	1	6.9	-
Elevon Disable	2.5	1	2.5	1	2.5	1	2.5	1	2.5	-
Rudder Disable	2.8	1	2.8	1	2.8	1	2.8	1	2.8	-
SAS, Pitch, ATS, and Trim Emer Controls	3.9	1	3.9	1	3.9	1	3.9	1	3.9	L-1011
Antiskid Controls	2.3	1	2.3	1	2.3	1	2.3	1	2.3	L-1011
Sensor Heat Controls	1.9	1	1.9	1	1.9	1	1.9	1	1.9	L-1011
PFCS Mon, Rudder and Elevon Emer Controls	3.2	1	3.2	1	3.2	1	3.2	1	3.2	L-1011
Engine Start	2.4	1	2.4	1	2.4	1	2.4	1	2.4	L-1011
APU Engine Controls	4.7	1	4.7	1	4.7	1	4.7	1	4.7	-
Cabin Lights	2.0	1	2.0	1	2.0	1	2.0	1	2.0	L-1011
Mission Timer	2.5	1	2.5	1	2.5	1	2.5	1	2.5	CM or LM
Event Timer	1.9	1	1.9	1	1.9	1	1.9	1	1.9	CM or LM
Exterior Lights	2.8	1	2.8	1	2.8	1	2.8	1	2.8	L-1011
Rudder Limiter	2.1	1	2.1	1	2.1	1	2.1	1	2.1	L-1011
Sub-Total			40.90		72.20		72.20		72.20	

Table 2.2-38D MARK I ORBITER AVIONICS BASELINE EQUIPMENT LIST

Displays and Controls Equipment (cont'd)	Unit Wt. (lbs)	FHF Quan.	FHF Total Wt.	FVF Man Quan.	FVF Man Total Wt.	FVF Unman Quan.	FVF Unman Total Wt.	Oper Quan.	Oper Total Wt.	Program Source
Main Instrument Panel:										
Flight Attitude Indicator	6.94	2	13.88	2	13.88	2	13.88	2	13.88	S-3A
Horizontal Situation Indicator	8.0	2	16.00	2	16.00	2	16.00	2	16.00	S-3A
Aero Surface Indicator	2.25	2	4.50	2	4.50	2	4.50	2	4.50	L-1011
AFCS Modes	2.75	2	3.50	2	3.50	2	3.50	2	3.50	L-1011
AFCS Warning	1.88	2	3.76	2	3.76	2	3.76	2	3.76	L-1011
Instr Warning	1.88	2	3.76	2	3.76	2	3.76	2	3.76	L-1011
Autopilot/Land	20.2	1	20.2	1	20.2	1	20.2	1	20.2	L-1011
Meter - Airspeed/Mach/α	8.2	2	16.4	2	16.4	2	16.4	2	16.4	C-5A
Meter - Altitude/Vertical Speed	8.2	2	16.4	2	16.4	2	16.4	2	16.4	C-5A
Altimeter	3.0	2	6.0	2	6.0	2	6.0	2	6.0	S-3A
True Airspeed Indicator	1.5	2	3.0	2	3.0	2	3.0	2	3.0	S-3A
Altitude Indicator	1.5	2	3.0	2	3.0	2	3.0	2	3.0	S-3A
Multi-Purpose Keyboard	22.0	-	-	2	44.0	2	44.0	2	44.0	S-3A
Engine Gimbal Override	1.95	-	-	2	3.90	2	3.90	2	3.90	CM
RCS Control Override	2.75	-	-	2	5.50	2	5.50	2	5.50	-
Main/OMS Override	1.20	-	-	2	2.40	2	2.40	2	2.40	-
Tank Jettison Override	1.90	-	-	2	3.80	2	3.80	2	3.80	-
Abort	1.10	2	2.20	2	2.20	2	2.20	2	2.20	-
Instr Brightness Control	0.75	2	1.50	2	1.50	2	1.50	2	1.50	-
Caution and Warning Test Panel	1.10	2	2.20	2	2.20	2	2.20	2	2.20	-
Booster Status Panel	2.90	-	-	2	5.80	2	5.80	2	5.80	-
Master System Caution and Warning	2.97	2	5.94	2	5.94	2	5.94	2	5.94	-
Multi-Function Crt (Flight Management)	63.0	-	-	2	126.0	2	126.0	2	126.0	S-3A
Flight Mode Indicator	2.80	2	5.60	2	5.60	2	5.60	2	5.60	-
Multi-Function Crt (Subsystems)	63.0	-	-	1	63.0	1	63.0	1	63.0	S-3A

Displays and Controls Equipment (cont'd)	Unit Wt. (lbs)	FMF Quan.	FMF Total Wt.	FVF Man Quan.	FVF Man Total Wt.	FVF Unman Quan.	FVF Unman Total Wt.	[illegible] Quan.	[illegible] Total Wt.	Program Source
Main Instrument Panel (cont'd):										
Engine/Propulsion Displays	8.5,6.2, 1.2	9,6,2	104.1	9,6,2	104.1	9,6,2	104.1	16	99.2	-
Engine/Propulsion Displays	6.2	12	74.4	12	74.4	12	74.4	-	-	-
Mode Select (Main/OMS/ABES)	1.1	2	2.2	2	2.2	2	2.2	2	2.2	-
Mode Select (RCS/APU)	1.1	2	2.2	2	2.2	2	2.2	2	2.2	-
Area Nav - Growth	(63)*	-	-	-	-	-	-	-	-	L-1011
Throttle Quads, Speed Brake & Elevons Control	38.0	2	76.0	2	76.0	2	76.0	2	76.0	C-5A
ABES Controls	1.1	1	1.1	1	1.1	1	1.1	1	1.1	C-5A
G&N Panel	2.6	1	2.6	1	2.6	2	2.6	1	2.6	-
Landing Gear Controls	4.2	1	4.2	1	4.2	1	4.2	1	4.2	C-5A
Emer Landing Gear Extension Controls	1.5	1	1.5	1	1.5	1	1.5	1	1.5	C-5A
ATC Panel	2.0	1	2.0	1	2.0	1	2.0	1	2.0	C-5A
Communications Panel	14.0	1	14.0	1	14.0	1	14.0	1	14.0	S-3A
EC/LS Panel	3.9	1	3.9	1	3.9	1	3.9	1	3.9	-
Engine Start	2.9	1	2.9	1	2.9	1	2.9	1	2.9	C-5A
Attitude Hand Controller	8.0	2	16.0	2	16.0	2	16.0	2	16.0	CM
Translation Controller	8.2	2	16.4	2	16.4	2	16.4	2	16.4	-
C&W Annunicators	5.6	2	11.2	2	11.2	2	11.2	2	11.2	-
Sub-Total			458.54		724.54		724.54		645.24	

Table 2.2-38D

MARK I ORBITER AVIONICS BASELINE EQUIPMENT LIST

Displays and Controls Equipment	Unit Wt. (lbs)	FHF		FVF Man		FVF Unman		Oper		Program Source
		Quan.	Total Wt.	Quan.	Total Wt.	Quan.	Total Wt.	Quan.	Total Wt.	
Sys Engr Panel	-	8.3 ft^2	189	8.3 ft^2	189	8.3 ft^2	189	0	0	
CRT	63	1	63	1	63	1	63	0	0	
			252		252		252		0	
TOTAL			751.44		1048.74		1048.44		717.44	

Table 2.2-38E MARK I ORBITER AVIONICS BASELINE EQUIPMENT LIST

Data Management [illegible]	Unit Wt. (lbs)	FHF Quan.	FHF Total Wt.	FVF Man Quan.	FVF Man Total Wt.	FVF Unman Quan.	FVF Unman Total Wt.	Oper Quan.	Oper Total Wt.	Program Source
[illegible]										
Digital Computer	255	1	255	1	255	1	255	1	255	S-3A
Drum Storage	70	1	70	1	70	1	70	1	70	S-3A
Dig. Mag. Tape	20	2	40	2	40	2	40	2	40	S-3A
Display Gen. Unit	80	1	80	1	80	1	80	1	80	S-3A
Comm. SIU	40	1	40	1	40	1	40	1	40	S-3A
Instr. SIU	10	1	10	1	10	1	10	1	10	S-3A
GN&C SIU	80			1	80	1	80	1	80	S-3A
ECLS SIU	15	1	15	1	15	1	15	1	15	S-3A
El. Power SIU	15	1	15	1	15	1	15	1	15	S-3A
Booster SIU	30			1	30	1	30	1	30	S-3A
CSE/LCC SIU	30	1	30	1	30	1	30	1	30	
TOTAL			555		665		665		665	

Table 2.2-38F MARK I ORBITER AVIONICS BASELINE EQUIPMENT LIST

Instrumentation Equipment	Unit Wt. (lbs)	FHF Quan.	FHF Total Wt.	FVF Man Quan.	FVF Man Total Wt.	FVF Unman Quan.	FVF Unman Total Wt.	Oper Quan.	Oper Total Wt.	Program Source
2.1.6										
Transducers	0.5	290	145	460	230	490	245	280	140	
Sig. Cond. & Wire	0.2	730	146	1160	232	1230	246	700	140	
Flight Recorder	48	2	96	2	96	2	96	2	96	C-5A
Time Code Gen.	15	1	15	1	15	1	15			C-5A
Film Camera	7	1	7	1	7	1	7			A/C
PCM TLM Equipment	44	1	44	1	44	1	44			Apollo
Fm Wide Band	40	1	40	1	40	1	40			L1011
TOTAL			493		664		693		376	

2.2.2 Electrical Ground Support Equipment

The electrical ground support equipment for the baseline avionics system is required to support a shuttle system that is composed of an interim, unmanned booster in conjunction with a 040A Mark I Orbiter that has on-board data management capability. Avionics on-board checkout, fault isolation, and all abort indications are controlled and monitored from the oribter. This provides a near-autonomous shuttle operation during prelaunch checkout and launch activities for the avionics system.

To support the baseline avionics system, it is proposed to modify and utilize existing facilities and electrical ground support equipment (EGSE) located at Kennedy Space Center. These facilities and equipment together will provide the necessary means to refurbish, maintain, checkout and launch the shuttle avionics system.

2.2.2.1 Checkout and Maintenance. Complete avionics subsystem checkout and fault isolation are performed with the on-board checkout system; the basic philosophy of maintenance is "on demand", i.e., replacing units only when malfunctioning. This concept is viable because of real time checkout on-board during all mission phases and allows the line replaceable units (LRU) to be immediately dispatched to their respective subsystem laboratories after landing, where detailed and positive fault isolation takes place on automatic test equipment. Those problems external to line replaceable units require additional on-board trouble-shooting conducted by utilizing carry-on equipment such as VTVMs, oscilloscopes, etc. presently in inventory at KSC.

Electrical ground support equipment, to satisfy the peculiar bench checkout requirements of each subsystem and LRV, will be maintained in dedicated laboratories in the MSOB. Each dedicated subsystem laboratory is in turn connected to a common computer controlled test equipment complex by way of a data switching unit, all within the MSOB. Each subsystem laboratory after establishing the necessary soft-ware test routines, can conduct checkout and fault isolation to the replaceable module on any of the laboratory line replaceable units.

The existing GE computer complex consists of a general purpose computer, memory unit, input/output console, mag-tape transports, teletypewriter, tape reader, and tape punch to which will be added a data switching unit interconnected to stimulus and measurement equipment sections (created from existing test hardware). The resultant test tool is identical in function to the versatile avionics system test concept used for shipboard support of the S-3A avionics.

The data switching unit is the communications element in the total test interface. It enables the automatic test equipment operator to exercise supervisory or direct control of the test program, monitor subsystem laboratory requests for support in real time (shared), and select the modes of operation. The computer, acting through the data switching unit monitors, controls, and synchronizes all data and instruction transfers between common test equipment and the user subsystem laboratories.

To accommodate the on-board system certification of newly installed components, the Launch Control Center is connected into a data link terminal that is located in each orbiter and booster cell in the Vertical Assembly Building. This allows final system certification of each vehicle prior to vertical assembly on the mobile launcher.

2.2.2.2 System Checkout and Launch. The total shuttle system, composed of the booster, tank, and orbiter, is assembled in the Vertical Assembly Building on the platform of the mobile launcher. The orbiter DMS system is then connected to the Launch Control Center (through the GSE SIU) by way of a high speed data link transmission line; voice communications and external power are through other electrical connectors. This puts the LCC computer and the ML computer in contact with the on-board data management system and prepares the vehicle and facility for integrated systems checkout and simulated flight tests. Checkout is initiated and monitored from the orbiter cockpit with the on-board data management system calling up the necessary test routines and monitoring the test and data responses under GSE cueing. System test is controlled from the ground test center in the LCC.

When integrated checkout and simulated flight tests are complete, the mobile launcher and shuttle are moved to the launch site and installed on the launch pad. Identical electrical and data link connections are made and the same computer-controlled automatic checkout of the shuttle avionics is performed.

The computer complex that supports this phase of the operation is composed of two general purpose computers and their peripheral equipment. One computer is located in the Launch Control Complex while the other is stationed in the base of the mobile launcher. Both computers are RCA-110As and are connected in tandem to perform their system support functions. Fueling is controlled independently of the avionics support and is not considered in this study. Existing peripheral equipment includes a line printer, card reader, card punch, paper tape reader, tape punch, mag tape transports, and digital display equipment.

2.2.2.3 Integrated Test Facility (ITF). Components of the shuttle avionics system that are not repairable at the launch operations facility are sent to an integrated test facility for depot-level maintenance. This facility (Fig. 2.2-60) is the existing S-3A ITF modified for shuttle compatibility and has the capability of complete component repair and also total system checkout with respect to any individual component. The facility consists of three supporting elements.

- Integration Bench Setup (IBS)
- Integration Test Facility (ITF)
- Laboratory Computer Facility (LCF)

The IBS contains the necessary equipment to repair and checkout the individual avionics components. This includes the work benches, hand tools, simulators, and mechanical and electrical lab equipment associated with shuttle avionics.

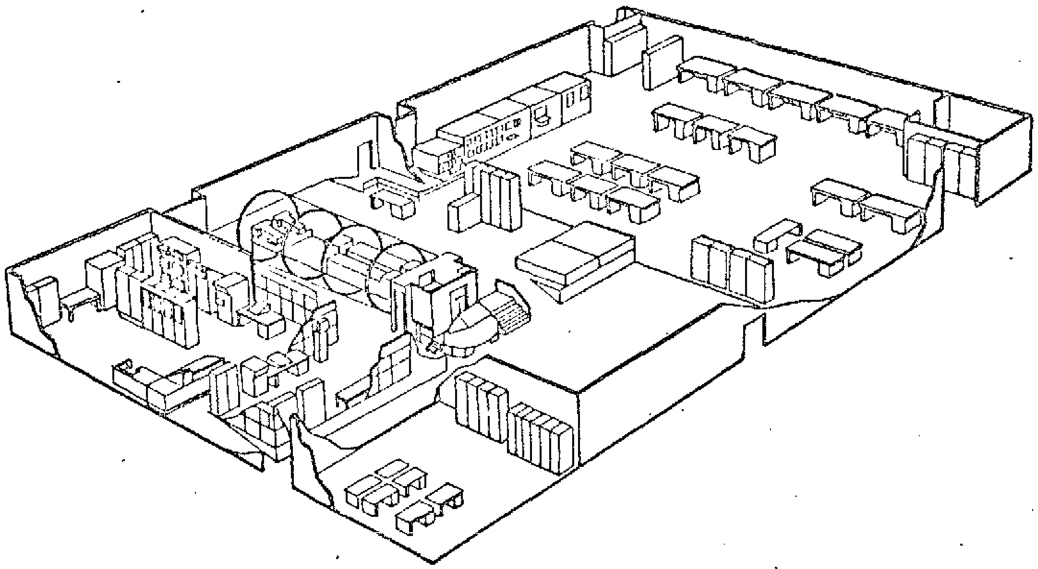

Fig. 2.2-60 Integrated Test Facility/Integrated Test System
Rye Canyon Bldg. 229

The ITF houses the physical mockup of the avionics system. Each repaired component is installed in the mockup and subjected to an integrated systems test.

The LCF contains the computer system and all of its peripheral equipment to support the integrated system run that is conducted in the adjacent ITF.

2.2.2.4 Mission Support. During the development phase of the program, the use of the existing Mission Control Center (MCC) modified for SSV support and an earth based tracking network will be required for shuttle operation. It can be seen (Fig. 2.2-61) that as the program progresses from the development phase through the early operational phase to the more mature operational phase, fewer active stations will be required to support the orbiter and its operations (reference EM L4-02-05-04-M1-1 SGLS Coverage by Ground Station - 100 nm Polar Orbit).

Each station in the network is assumed to be selected specifically to support the shuttle operation. Emergency voice contact to MCC is assumed to be available at any time through the Air Force tracking network.

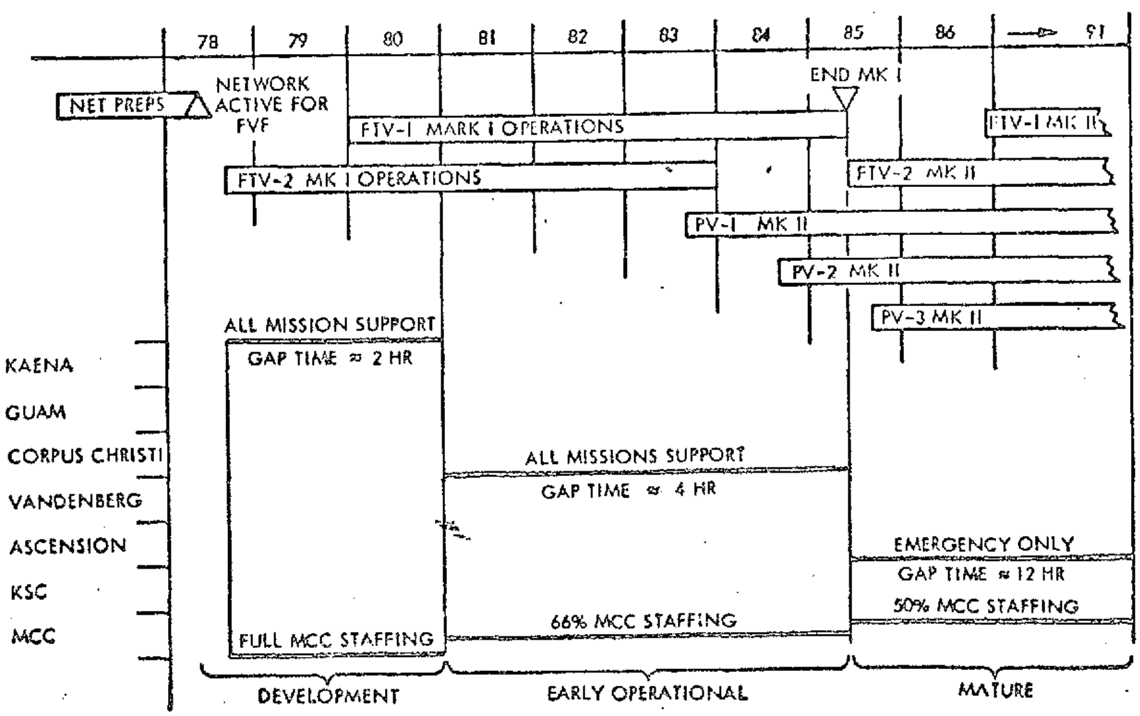

Fig. 2.2-61 040A Avionics Baseline - Orbital Support

2.2.3 Software

The principal implications of the selected avionics hardware baseline on shuttle software definition and development may be summarized as follows:

- The extensive adoption of S-3A data management subsystem features offers a high degree of commonality between shuttle software needs and those currently existing or under prior development within the Federal Government inventory.
- Flexible (CRT/KEYSET) man interface and DMS highspeed, digital access to all LRUs without resort to a complex data bus and offers straightforward economical methods of addressing shuttle-unique, combined aircraft-spacecraft instrumentation/checkout/control problems.
- Compatible ground software development facilities necessary for support of early contractor simulation, development, and integration of orbiter mission software already exist so that short-term expenditures for new development are minimal.

In effect, the selected avionics baseline has been drawn from a maximum S-3A application alternative and, with respect to shuttle software in particular, the original alternate objective remains intact.

To the extent that the above implications are realizable, the possibility exists for the shuttle software program to gain the most capable, proven system contemporarily available at a cost competitive with the least-capable system tailored to a minimum, shuttle-peculiar requirement. This possibility is the principal motivation behind the functions, structure, and segmentation of the Mark I shuttle software baseline as described below.

2.2.3.1 Functional vs Chronological Analysis Summary. Assuming an application of S-3A software concepts to at least an aircraft-dedicated configuration for early horizontal flight, it is of interest to show:

a. That all functional requirements are met

b. By what sensible manner the configuration might be progressively updated to achieve a satisfactory Mark I/Mark II orbital capability.

c. How this approach compares with other possible approaches.

To determine a maximum application, it is, of course, necessary to relate shuttle functional requirements to specific capabilities of the existing S-3A avionics system. Further, it is necessary that functions (or capabilities) be prioritized in a manner that facilitiates proper differentiation with respect to relative importance and logical order of development.

2.2.3.1.1 Functional Analysis Approach. As a part of the overall alternate avionics concept study, a detailed functional analysis resulted in the consideration of three possible variations of functional capability for Mark I orbiter avionics based on the following categorization:

a. SAFE. This variation is a "barebones" approach which yields a safe, acceptable Mark I configuration at absolute minimum cost, but with some sacrifice in mission success probability. All functions and features assigned a "safety of flight" (SOF) criticality (Rank 1) are included. This variation does not support all functions and features assigned a "mission success" (MS) criticality (Rank 2).

b. Mission Success. All functions and features assigned either an SOF or MS level of criticality (Rank 1 or Rank 2) are included. This variation is predicated on achieving a high probability of Mark I mission success, but does not include functions and features dedicated to growth and improved adapability for early Mark II development testing and evaluation.

c. Improved Capability. All functions and features considered technically justifiable for a full-capability Mark I avionics system are included.

Software-oriented, major functional capabilities categorized in this manner by avionics subsystem are given in Table 2.2-39. Capabilities are cumulative in going from one variation to the next such that the "improved capability" variation includes all functions in the table.

Table 2.2-39

MARK I FUNCTIONAL ANALYSIS SUMMARY

	CONTROL AND DISPLAYS	GN&C	COMMAND TRACKING	ELECTRICAL POWER	INSTRUMENTATION	DATA MGT AND SOFTWARE
SAFE	CONFIGURATION CHECKLIST HARDWIRED ANNUNICIATORS HARDWIRED SWITCH CONTROLS SENSOR-DRIVEN FLIGHT INSTRUMENTS	AFT/HDG REF RADAR ALT CADS TACAN SAS INS GUID COMPUTER	VOICE RADAR AND INTERCOM TM DOWNLINK L-BAND XPDR COMMAND UPLINK	DC SOURCE AC SOURCE DISTR AND CONTROL	DFI (HFT) SIG COND AND TM INTERFACE FLIGHT REC (CRASH) AUTO-ABORT	HARDWIRED MASTER SEQ. CONTROL LOGIC UNIT
MISSION SUCCESS	SIU-DRIVEN ANNUNCIATORS SIU-PROCESSED SWITCH CONTROLS AUTO PILOT-DRIVEN FLIGHT INSTRUMENTS HARDWIRE GN&C CRT DISPLAY	A/C AND S/C AUTOPILOTS ONBOARD UPDATE ORB ALTIMETER DMS/SIU C/O, MON MODE CONTROL ILS	DMS/SIU C/O, MON., MODE CONTROL SEARCH RADAR EVA CREW TO VEHICLE TO GRD VOICE/DATA	DMS/SIU C/O, MON., MODE CONTROL	TIMING SUBS DMS/SIU C/O, MON., MODE CONTROL ANALOG TAPE REC.	GEN. PURPOSE COMPUTER ONBOARD C/O SYS MGT ANTENNA/OPTICS POSITIONING
IMPROVED CAPABILITY	PROGRAMMABLE CRT DISPLAY AND CONTROL	DMS/SIU GUID (INCLUDING RENDEZVOUS AND DOCK) DOCKING SENSOR	VEH/DETACH MODULE VOICE AND DATA S-BAND MSFN VOICE/DATA	DMS/SIU POWER MANAGEMENT	———	GN&C COMPUTATION AUTO MISSION CONTROL CALL-UP CATALOG

103506 (1) 1 = EHFT 2 = EVFT 3 = LVFT 4 = MK II

The "safe" variation, while perhaps presenting the least cost and complexity for orbiter avionics does not present an acceptable level of mission success probability and is greatly dependent upon extensive onboard hard instrumentation and ground systems tie-in for both checkout and operations. The "mission success" level of capability greatly reduces data point instrumentation and ground dependency, but is relatively limited in terms of suitability for extended missions and expanded payloads capabilities. The "improved capability," while technically justifiable in the long run, may present too great a cost and complexity.

Comparing current S-3A software-supported avionics capabilities with those indicated in Table 2.2-39 leads to the conclusion that a maximum application would generally result in an "improved capability" level in all subsystem areas, especially so in controls and displays and data management. If such an application proves as cost-effective or more cost-effective than other possible approaches offering less capability, then the major question would be whether or not the relative complexity of the approach increases or lowers program risks. This question leads to a look at the chronological aspects of shuttle development in the light of related previous experience.

2.2.3.1.2 Effects of Progressive Development. From the viewpoint of minimizing program risks, a progressive approach to shuttle development might well initially target on a set of paths (as depicted in Table 2.2-39) for a four-step, incremental development. There are, of course, problems to be reckoned with in such an event, namely, how to avoid: (1) costly onboard (early) instrumentation and ground systems development that might ultimately prove wasteful or too expensive to maintain, and (2) crippling constraints on upgrading avionics designs due to unforeseen costly retrofits. Without proper end-point planning, lower short-term risks gained by keeping the configuration simple for earlier development phases can result in much greater risks later in the program.

From the viewpoint of initial hardware/software design for an early horizontal flight test (FTV-1) configuration, requirements are necessarily influenced by ultimate program objectives with respect to: (1) systems development test and integration, (2) operational concepts of onboard checkout/monitoring and ground maintenance, and (3) operational reliability and redundancy.

The selected Mark I avionics baseline viewed in the light of these considerations suggests a reasonable set of end-point constraints for bounding early equipment and software specifications, namely:

- The ultimate shuttle orbiter avionics will be a moderately integrated system.
- All subsystems will, to the individual LRU level, provide:
  1. A BITE-based capability for onboard DMS checkout and in-flight performance monitoring
  2. A built-in test capability consistent with an automatic test equipment ground maintenance philosophy.
- All subsystems will provide a standard interface tie-in with the DMS for purposes of BITE logic access and for in-flight monitoring of equipment mode/configuration status. This standard interface will conform to the S-3A DMS input/output communications scheme, which employs a serial, 6 MHz, biphase-Manchester-coded, duplex interface.

- Where complexity of subsystem redundancy and functions warrant, the DMS subsystem interface unit (SIU) will provide the capability for remote, integrated control (via DMS programmable keyset) of subsystem configuration and modes so as to provide:

    1. Minimum essential flexibilities for integration/verification/maintenance testing without requiring extensive use of special test equipment.
    2. Automatic sequencing for DMS directed go-no/go checkout prior to flight and during extended stationkeeping operations on orbit.

It is important that this latter capability of DMS (or automatic test equipment) programmable control of subsystem configurations and modes is well justified for test purposes alone and does not necessarily imply crew-dependence upon this method of subsystem control during flight operations. Past experience in this regard fully supports this position in that: (1) the amount of special test equipment (and vendor involvement) required during integration testing can be held to cost-effective levels, and (2) the extensive variations in software/hardware configurations and modes necessary for in-depth verification testing are readily programmed and implemented without resort to lengthy and tedious procedures or the involvement of large numbers of support personnel and special test equipment.

The ultimate method of assuring an adequate level of operational reliability must also influence early hardware/software design. If the ability for flexible, software (keyset-DMS) control of subsystem equipment operations is already justified for test purposes, then this available feature of the system becomes an important consideration in providing essential redundancy and sharing of resourses. In effect, the integrated controls, displays, DMS computer, and SIU tie-in with the LRUs constitute a software path for the crew to manage and control the system. Subsystem-dedicated redundancy above the LRU level, then, need not be duplicated to provide fail-safe operations or triplicated to provide fail-operational. In the case of mission critical functions of GN&C (see Fig. 2.2-62), the dedicated processor provides a crew-control path that is completely paralleled by DMS and manual modes and need not be duplicated to provide fail-operational capability. The same applies to other subsystems as well, such that

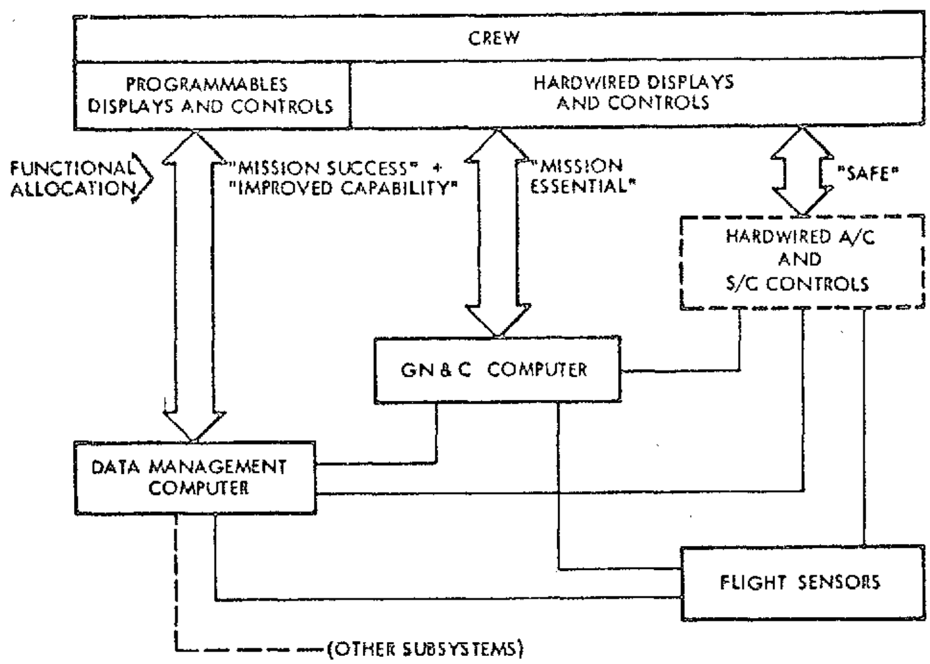

Fig. 2.2-62 Computer System Multipath Control Concept for Mark I Baseline

considerable quantities of equipment, which might otherwise require parallel redundancy, may be safely left out of the system by providing the crew with the minimum software and keyset controls to accomplish the same functions via the DMS. The diagram of Fig. 2.2-62 can be drawn for each of the major subsystems to reflect this type of multipath redundancy, as appropriate.

In conjunction with early horizontal and vertical flight testing, substantial development instrumentation that will not carry over into operational use will be required. As a combined aircraft/spacecraft, even the operational instrumentation problem will be much more severe than in previous systems. The number and variety of transducers, cables, signal conditioners, indicators, and control switches is linely to be prohibitive to the point where conventional, parallel hardwire approaches to data/control point instrumentation cannot be tolerated. Again, the BITE/MODE monitoring and controls available via the DMS/SIU/LRU route offer advantages. Temporary overlay, development flight instrumentation can be held to a minimum and the amount of parallel, hardwired indicator and control panels can be kept at tolerable levels.

The programmable displays and controls can be effectively utilized for selected monitoring of both graphic and discrete parameters, depending on current test flight emphasis and suspect areas. Limit checking and formatting for down-link PCM telemetry is then accommodated by the DMS software. Again, the potential perturbating effects on component design due to early versus end-point configuration requirements are significantly alleviated.

These and other considerations affecting the degree of ground equipment/support dependency, flexibilities for test, redundancy, growth and interface control, and overall program development/operational costs greatly influence the choice of interim avionics configurations and associated software capabilities leading to an effective Mark I end-point capability.

2.2.3.1.3 Functional Baseline and Effectivity. The selected functional baseline for Mark I data management software is depicted in Fig. 2.2-63. Function effectivity versus interim Mark I configurations is shown in Table 2.2-40 suggested by Fig. 2.2-64, primary-mode mission critical GN&C computations are provided for by software residing in the subsystem-dedicated computer.

Data management software effectivity for HFT does not include functions peculiar to orbital operations per se; however, programmable displays and controls and on-board software capabilities are in keeping with the preceding functional analysis and the principal objectives listed below.

- Provide for flexible and extensive test and checkout capability for each subsystem to the individual LRU level
- Provide early hardwire-independent crew monitoring and control of subsystem operations
- Permit alleviation of early, parallel instrumentation/control overlay problem

Adequate time spans for implementation of these objectives are readily accommodated with low risk to the program, since most of the avionic equipment required for HFT is compatible with existing software providing these same functions for S-3A.

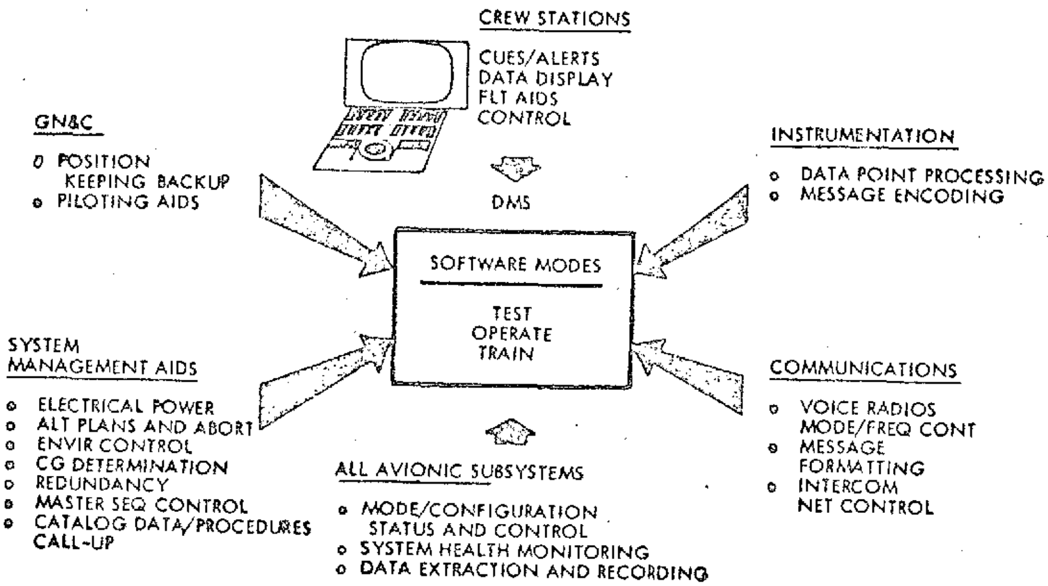

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Fig. 2.2-63 Shuttle DMS Software Functions – Mark I Baseline

Table 2.2-40

MARK I BASELINE SOFTWARE FUNCTIONS EFFECTIVITY

FUNCTIONS	MARK I		
	HFT	VFT	OP'L
ONBOARD CO/FI AND DATA EXTRACTION	O	O	O
AIRCRAFT INSTRUMENTATION AND COMMUNICATIONS PROCESSING AND CONTROL	O	O	O
AVIONICS SUBSYSTEMS PERFORMANCE AND MONITORING AND SELECTIVE CONTROL	O	O	O
GN&C COMPUTATIONS		O	O
SPACECRAFT INSTRUMENTATION AND COMMUNICATIONS PROCESSING AND CONTROL		O	O
SYSTEM MANAGEMENT AIDS		O	O
AVIONICS CONFIGURATION CONTROL			O
CONSUMABLES MANAGEMENT			O
RENDEZVOUS COMPUTATION			O

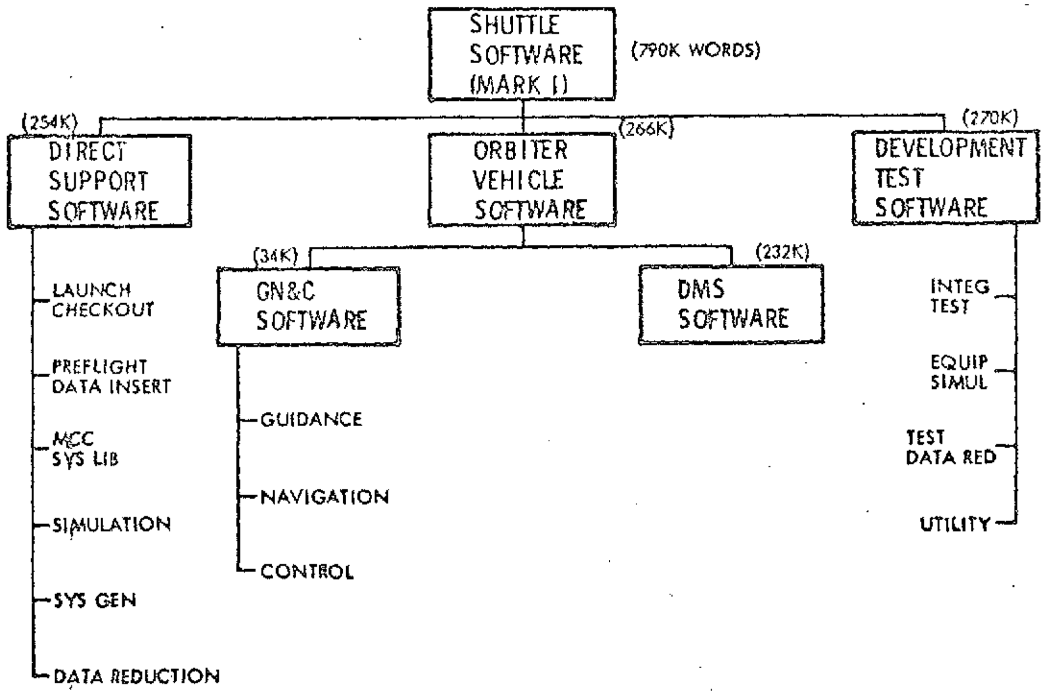

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Fig. 2.2-64 Mark I Baseline Software Configuration

For VFT, GN&C software is added for mission critical spacecraft control functions. Programs residing in this dedicated, single-CPU version of the 1832 computer provide a DMS-independent capability for all computations and command generation necessary for both on-board autonomy and command uplink control of flight path under crew supervision. In addition, DMS software functions are expanded to include the following:

- GN&C backup
- Spacecraft communications processing and control
- System management aids
    1. Electrical power control
    2. Environmental control
    3. Alternate plans
    4. Degraded-mode recovery

During late HFT and prior to Mark I operational missions, the DMS software is further updated to include all functions called for in Fig. 2.2-64. This end-point Mark I software configuration has been sized in accordance with the following objectives:

- Full utilization of the moderately sized DMS for assisting the crew in managing the non-flight-critical functions of the combined system, including a high degree of on-board autonomy for OBCOFIRM. Crew-elective utilization of the available DMS OSCOFIRM access to subsystem path for monitoring and control of critical subsystem functions.
- Restriction to proven needs and a manageable degree of sophistication versus time.

2.2.3.1.4 Orbiter Baseline Impact on Ground Support Software. The extensive on-board autonomy features of the baseline configuration considerably lessens the degree of early sophistication and criticality of direct support software needed on the ground. This is true regardless of whether early vertical flight tests are manned or unmanned, although the latter alternative would significantly increase the degree to which software would be pacing overall shuttle system development.

Functional requirements, sizing, and costing of ground support software are based on the selection of S-3A, off-the-shelf components, both in the DMS/GN&C areas and in the subsystem interfaces with the DMS. For this reason, S-3A concepts are generally adopted for early contractor development, including the automatic test equipment approach to long-term ground maintenance of major avionics components.

Although the baseline configuration postulates both dual and simplex versions of the S-3A 1832 computer, other inboard (CP and IOC) elements for these applications could be readily adapted from comparable, available systems. The S-3A input/output interface and associated communication scheme is basic to the approach and would require modification to I/O designs if other machine elements are ultimately selected. In this connection, it is important to note that use of a higher-order language such as XCMS-2 is essential where such flexibility is required. Although other higher-order languages (such as SPL and HAL) may be as well or better suited, the baseline choice of XCMS-2 is predicated on: (1) the recent, extensive improvements and large-scale application to the S-3A and other (AEGIS/RCA and LHA/Litton) Navy operational command and control systems; and (2) the availability of substantial, readily-applicable ground support and on-board programs currently supporting the S-3A program.

The consideration of other candidate computer systems will imply the development of a Code Generator to the UYK-7-hosted, CMS-2 compiling system that will accommodate object code generation for the selected machine. The creation and insertion of a new "target machine", object code generator does not represent an uncharted path, since the CMS-2 system has been constantly expanding in this regard for more than a decade. Should the final computer selection also require software development in a ground computational facility other than the one now dedicated to S-3A, modifications would also be required to the Object Loader Program to ensure compatibility with CMS-2 object output format and conventions. Although much can be written on this subject based on current S-3A experience and approaches, the nature and potential magnitude of the software problem for the shuttle warrants further study based on a finite set of: (1) possible trades with respect to the degree of centralized versus decentralized processing and control of subsystem functions; and (2) candidate computer systems and programming languages, both ground and vehicle. Because of the wide range of variables involved, including software support as well as hardware, comparison of candidate computer systems requires a most careful analysis.

For the selected baseline and associated costing of ground support software, utilization of existing S-3A programming languages and compilation facilities have been assumed in order to achieve the lowest cost and risk factors. Alternate costing is also supplied, wherein other existing, programming launguages (such as HAL) are utilized along with ground computational facilities other than those presently used on S-3A.

2.2.3.2 Selected Baseline Structure. The selected baseline structure for total shuttle system software is illustrated in Fig. 2.2-64. Sizing estimates in thousands of 32-bit words are given in the parenthesis appearing with each major area. Components comprising each of these areas are briefly described below.

2.2.3.2.1 Ground Support Programs. These programs are identifiable with the following facilities and basic functions:

- Software Development Facility – initially run by contractor and turned over to NASA for MCC direct support operations prior to FVF. This facility (as SDF and MCCO provides the means of pre-flight and post-flight processing of orbiter vehicle tapes as required to support development (SIL), flight test, and operational activities (on-board DMS and GN&C programs). This software includes: (1) compilers, assemblers, loaders, and necessary SDF operating system utilities; (2) system generation program as required to build orbiter system tapes for various ground and in-flight uses; (3) post-mission data reduction program which interfaces with on-board data extraction and ground recorded telemetry to permit mission reconstruction and analysis prior to subsequent flight.
- System Integration Laboratory – initially run by contractor and turned over to NASA (perhaps as part of MCC) for life-cycle-sustaining development support operations prior to first production vehicle flight. This facility (as contractor-maintained SIL) provides: (1) necessary power, cooling, cabling, special test equipments, and physical mock-ups for component-, subsystem-, and system-level avionics test and integration; (2) development tool, integration test programs for each avionics subsystem as required for initial, laboratory-peculiar bench and integration levels of hardware testing; (3) an S-3A level of on-line, avionics equipment simulation capability; (4) a system test and early flight test data reduction program for use in contractor development-phase evaluations. This facility will have a dedicated laboratory computer facility (LCF) and computer interface unit (CIU) for direct on-line tie-in with on-board computers (DMS and GN&C) for purposes of avionics equipment simulation and stimulation under software and manual (keyboard) control. The necessary operations system and utilities software for the LCF are also included in the total shuttle system software estimates.
- Launch Control Complex – facilities for avionics-associated pre-launch checks and countdown related voice/data interfaces with on-board

instrumentation, communications, GN&C, and DMS subsystems. The shuttle-peculiar software for use at this facility is developed by the contractor in cooperation with and at NASA-Kennedy. All necessary development support computer time and operating system software facilities are presumed GFE, including languages, compilers, and utilities.

- Systems Development Simulation Facilities – located at NASA, Houston and devoted to avionics design-phase requirement definition/experimentation. Simulation software is primarily directed to GN&C functions and will complete model shuttle-peculiar flight control dynamics in conjunction with the baseline flight computers and associated man/machine interfaces. For purposes of sizing and costing, all computer systems, associated support software, and operations personnel are assumed to be supplied GFM; however, estimated computer-hour costs are included as required for contractor development of avionics-related simulation software development.
- Avionics Maintenance Test Facilities; presumably to be located at or near launch site, are assumed to employ automatic test equipment of the S-3A VAST type. This integrated maintenance concept for the major avionics LRU's will require the development of extensive test software for isolating faults down to pluggable card level. S-3A is currently developing such software for 65 WRAs (LRU equivalents). For baseline costing, a level of 80 shuttle avionics LRUs are assumed to be covered by this type of integrated ground maintenance. As of this writing, these LRUs are not individually identified and estimates are based on a 1.2 complexity factory with respect to similar S-3A development. Although included for costing purposes, this software is assumed to be provided GFM and is not treated explicitly in the baseline software structure.

Direct Support Software. A more detailed description of those components of direct support software based on existing S-3A capabilities follows. Direct support software includes all preflight and post flight software systems necessary to develop and maintain mission software, generate mission software data bases and process the extracted mission event data.

- Compiler - Monitor System 2 Extended (XCMS-2) – the extended compiler-monitor system 2 provides support software services necessary for initial compilation and maintenance of mission subprograms. The UYK-7 based XCMS-2 system includes the MS-2 monitor, the CMS-2 compiler, a librarian, UYK-7 loaders, tape utility routines and a flow charter. The XCMS-2 compiler is a three phased language processor that analyzes a users program and generates absolute or relocatable reentrant object code for the 1832. The UYK-7 and 1832 computers are instruction-repertoire compatible.

- ULTRA/32 Macro Assembly System – the ULTRA/32 Macro assembly system for the AN/UYK-7 computer is an independent, self-supporting software system consisting of an assembler, loader, librarian and utilities component which operates within the AN/UYK-7 executive environment. Each component represents an individual product which is interfaced with a system control program and a centralized input/output program. The assembler system products operates on the AN/UYK-7 computer and generates object code for the AN/UYK-7 which is compatible with the 1832 computer.

  The Macro assembly system provides support software services necessary for initial generation and maintenance of operational control subprograms (executive, initialization and recovery), input/output controller chains, in-flight performance monitoring subprograms (IFPM and data extraction) and system common routines.

- Systems Generation (SYS GEN) – system generation is the process that enables the user to generate an 1832 operational software system tailored to the specific requirements of an installation by processing a master system library tape and a set of installation control cards through the systems generation program. The systems generation program design accommodates a number of independent systems, i.e., operational program, system readiness test program, in-flight training program. The end product is a digital magnetic tape unit (DMTU) cartridge with the tailored software recorded on tracks 3 and 5.

C3

Each systems tape (DMTU cartridge) contains one or more independent program systems. A selection loader program provides a method for initiating the loading of any particular system into the 1832 main memory. The selection loader and each particular program system loader references tables built by SYS GEN for program system location in 1832 main memory and system segments contained on the systems tape.

Each program system is comprised of one or more segments which contain instructions and/or data. Each of these segments is "built" using output from one or more ULTRA/32 assemblies and/or XCMS-2 compilations. The contents of these segments are determined by the specifications input via the control card deck.

- Preflight Data Insertion – the preflight data insertion program operates at both the MCC and launch facility. This program generates, formats and records the mission-peculiar information and specific operational parameters on DMTU track 7. Preflight data is comprised of two major categories:

  1. Historical Data
  2. Modifiable Data

Historical data comprises that data recorded for subsequent use by the post-flight data reduction program. Hence, historical data items are not loaded by the operational program loader. Historical data includes the following:

1. Exercise Name
2. Exercise Phase
3. Operational Commander
4. Base Designation
5. Month, Day, Year
6. Scheduled Time of Lift-off or Take-off (ZULU)

Modifiable data includes configuration parameter data which may vary from mission to mission. The following categories of modifiable data are identified:

1. GN&C Preflight Parameters
2. Communications Link MSFN Participating Unit List Parameters
3. Communication Configuration Parameters
4. Ultra High Frequency CH Frequency Assign Parameters
5. Abort Parameters

- Post-Mission Data Reduction – the post flight data reduction program operates in the carrier based computer center and reduces, for mission analysis and reconstruction, the mission event data that are extracted on DMTU tracks 2, 4 and 6 during a mission by the data extraction subprogram.

Development Test Software. Development test software is associated with contractor SIL needs during initial avionics integration. Descriptions of the S-3A based programs included in the baseline configuration are given below.

- Integration Test Programs; before installation of the avionics system in the orbiter vehicle, each subsystem will be functionally checked as a single unit, operating in a totally integrated, laboratory environment. To provide this capability for controlled testing at a subsystem level, a set of integration test programs are developed to enable the following functional capabilities.

  1. To verify the communication of each subsystem with the DMS 1832 computer
  2. To test the functional capabilities of each subsystem by transmitting commands, by checking I/O, by requesting status words, and by checking faults which will be reported to the DMS as interrupts
  3. To provide sufficient information on-line (display or printout) to enable the test engineers to analyze hardware and/or software faults

- Avionic Equipment Simulation System (AESS) – an on-line avionics equipment simulation capability, employing the LCF 1230 computer system, will include dynamics model and device simulation modules for each major subsystem interfacing with the on-board 1832 computers. These capabilities are provided to permit: (1) early debugging of software in the 1832 with highly controlled but simulated avionics hardware interface dynamics (including fault insertion), (2) continuation of meaningful software integration in the absence of or malfunction of peripheral avionic equipments, (3) increased interface dynamics using canned data generation, and (4) increased fault isolation capability through on-the-spot software simulated test experiments.

An early AESS capability at the contractor SIL is considered essential to economical verification testing and quality control. Also, this software system will form the basis for parallel development and support of the NASA Systems Development Simulation Facilities at Houston.

- Development Test Data Reduction and Analysis Program – the test data reduction and analysis program (DRAP) will consist of subsystem analysis programs, written as the need arises. At minimum the basic DRAP will consist of a chronological print of events, navigational data plots, and data which is to be stored in a data bank for historical purposes.
- Utility Programs – a minimum SIL-located-1832 operating system utility package which is peculiar to Shuttle avionics test and integration needs will consist of the following:

    1. Loader
    2. Corrector
    3. Recorder
    4. Debugging aids
    5. I/O routines

2.2.3.3 Orbiter Vehicle Software. As shown in Fig. 2.2-64, Orbiter Vehicle Software consists of those programs which operate in the two 1832 computers included in the Mark I Avionics Baseline. Since GN&C dedicated functions are also included as "GN&C backup" under the DMS software hierarchy of Fig. 2.2-65, separate discussion is not necessary.

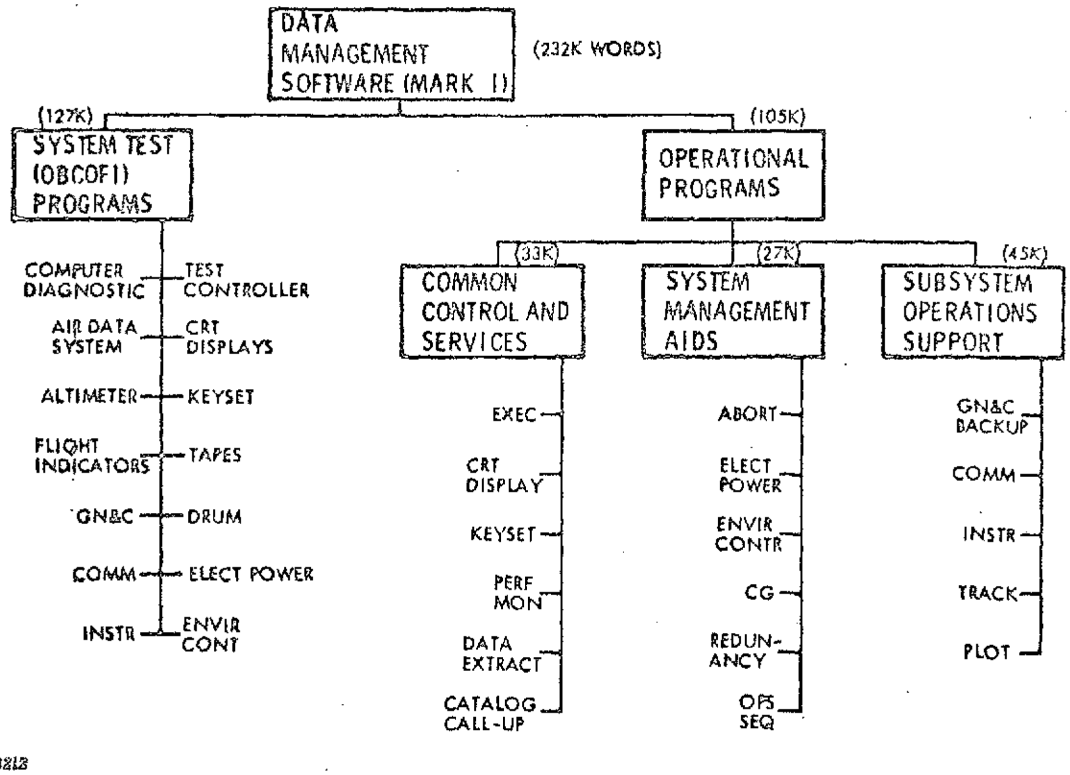

Fig. 2.2-65 Mark I Baseline DMS Software Configuration

Data management subsystem software is divided into System Test (OBCOFIRM) and Operational Programs.

2.2.3.3.1 System Test Programs. As part of the on-board software, System Test Programs are required to establish the operational readiness of the Orbiter avionics system during preflight and to provide a capability to isolate system failures to a LRU level. To meet these requirements, a set of System Readiness Test (SRT) subprograms and a set of Diagnostic Test (DT) subprograms are provided. These programs operate under a test controller program that is independent of the normal operational system executive. A brief description of the functional requirements for the SRT and subprograms is given below.

- System Readiness Test Subprograms – The SRT subprograms are designed for operation during prelaunch and will consist of tests that enable the go-no/go status determination of each DMS interfacing subsystem. Upon detection of a no/go status each SRT shall exit to the System Test Controller to provide an operator option for the selection of appropriate dianostics to enable isolation of an occurring fault to a LRU level.

- Diagnostic Test Subprograms – The DT subprograms are designed to detect and isolate system malfunctions to the non-ambiguous LRU within each avionics subsystem that interfaces with the DMS. In some subsystems, isolation to the quick replaceable assembly (QRA) is provided. DT subprograms will consist of several test routines and, where applicable, arranged in a unique order or sequence. The DT subprogram will isolate the malfunctions within each avionics subsystem by having the DMS test the operational functions, initiate active BITE circuitry, or cue the operator via the MPD to initiate manual BITE circuitry. The DT subprograms are capable of providing amplified, plain language information to the crew regarding functional capabilities remaining and may be selectively utilized during extended orbit operations to determine and implement appropriate degraded submodes available within operational software and equipment provisions.

2.2.3.3.2 Operational Programs. Orbiter Operational Programs include all instructions and data executable on or used by the DMS 1832 in performing the Shuttle mission. This software initially resides on the Digital Magnetic Tape Units (DMTU) and is loaded into main memory and onto magnetic drum storage (MDS) for execution during various phases of the mission. The three general types of subprograms comprising the Operation Programs are: Common Control and Services, Subsystems Operations Support, and System Management Aids. Brief descriptions of these programs are given below.

Common Control and Services. The programs, as listed in Fig. 2.2-65, consist of the following:

- EXEC. The Executive, Initialization and Recovery Subprograms consist of those routines which direct and coordinate the operation of DMS task-state programs, initialize the operating system (hardware and software) upon start-up, and provide for the reinitialization of the operating system during the ASW mission operation. Major executive components include the following:

1. Scheduling Component - Schedules tasks (task mission subprogram tasks) for subsequent priority ordered execution.

2. Dispatcher Component - Selects the highest priority task awaiting execution, establishes its operating environment, and releases control to the selected task.

3. Retrieval-Allocation Components - Obtains task mission subprograms and data, as required, from the drum and dynamically allocates core memory to the retrieved task.

4. Interrupt Component - Provides default or processing options for various equipment and software conditions. Provides entrance and control for Input/Output operations and Executive Service Requests (ESRs).

5. Executive Service Request Component - Enables subprogram tasks to gain the attention of the executive to provide for communication between tasks, initiation of input/output operation, scheduling of other tasks, and activation of various executive service functions.

6. Centralized Input/Output Component - Provides queuing for device and channel requests, reduction of monitor interrupts, wait/no-wait and scheduling of other task options, and a framework of control for subsystem errors.

7. Initialization and Recovery Subprogram - The initialization component of the Initialization and Recovery Subprogram provides for initialization of both hardware (1832 and all interfacing subsystems) and associated software upon initial system load, operator request, or as a result of system recovery.

- CRT Display. The CRT display program processes data to be presented to the crew in symbolic form. All data presentations are categorized into one of two general types: either data presentation in the form of alpha-numeric or data presentation in the form of diagrams, graphs or pictorial video. These two categories of data are referred to as peripheral data and plot data, respectively. Display control routines are provided for use by other functional tasks as required, and for manual selection by the display operator. Display operator control functions are assigned to programmable switch matrices which are grouped into several categories.

- Keyset Processing and Control (KEYPAC) – The Keyset Processing and Control Subprogram services the Integrated Control Subsystem (INCOS) which provides the crew with the capability of exercising control over the s system by translating operator decisions and actions into a digital form for input to the DMS and action by the Mission Subprograms. The function activation signals are received by the 1832 on an interrupt basis from the INCOS whenever a switch is depressed.

- In-Flight Performance Monitoring (IFPM) – The IFPM program monitors the performance of the avionics subsystems throughout the mission and, upon malfunction detection, identifies the failed subsystem and informs the Executive program of this identity. The IFPM program comprises the following elements:

  1. GPDC Self-Test
  2. BITE Monitoring and Maintenance System Status Tableau
  3. Active BITE Initiation

- Data Extraction (DEP) – The Data Extraction program is responsible for recording, in a predefined format, data pertinent to the Shuttle mission. This information is recorded on the DMTU to relieve manual recording by the crew and to assist in mission reconstruction. Collected mission data is processed by the Post-Flight Data Reduction Subprogram. At minimum, the program initiates data extraction at the following times:

  1. Initial Inventories and Equipment Status (Tableaux)
  2. Periodically to record GN&C flight dynamics parameters
  3. Upon the occurrence of a defined event
  4. Whenever an operator requests extraction of a displayed tableau.

- Catalog Procedures/Data Callup – This program provides for storage and retrieval of text and symbolic data for use by the crew in performing troubleshooting and mission planning functions.

2.2.3.3.3 Subsystem Operations and Support Programs. These programs, as shown in Fig. 2.2-65, consist of the following:

- GN&C Backup – The GN&C Backup program operating with the Inertial Navigation System, the Flight Display System, TACAN, Central Air Data System, Radar Altimeter and other flight sensors to perform the following functions:

  1. Maintains shuttle position in geographic and vehicle reference frames.
  2. Maintains a stabilized GN&C backup display upon request.
  3. Displays tabular GN&C parameters when invoked.
  4. Permits manual entry and modification of GN&C parameters.
  5. Calculates and optionally provides steering commands to the Automatic Flight Control System and Horizontal Situation Indicators. These commands direct the aircraft to selected geographical points of reference. This program is constantly operating in real-time but is invoked only upon failure or degradation of the GN&C computer system.

- Communications Processing and Control (COMPAC) – The Communication Processing and Control program provides for a computer-aided operation of the UHF radios and intercom system. Switching logic for radio sets and intercom is so mechanized that the operator has the option of complete manual control or computer-assisted control of voice communications. The Communication Managgement functions provide the operator with assistance in selecting the correct radio frequencies for various flight situations in accordance with effective communication plans entered into the program for each mission. Operating in conjunction with the S-Band Voice/data link, the Communication Processing and Control Subprogram performs the decoding and work formatting required to permit the orbiter to communicate via the Link as a picket unit or as a Data Net Control Unit (DNCU).

- Instrumentation Processing and Control (IPAC). This program provides for selected data point, limit check processing, and message encoding for data transmission via the PCM down link.
- Track and Plot. These programs operate independently of the GN&C real-time backup program to provide optional piloting aids and are drawn from the GN&C repertoire of available routines which are non-flight-critical.

2.2.3.3.4 System Management Aids. Baseline DMS software functions in this category consist of the following:

- Abort Caution and Response Cues based on preflight inserted parameters. Provides preplanned mechanisms for software/hardware reconfiguration necessary for proper abort procedures
- Electrical Power Management in accordance with preplanned, algorithmic formulation and sensed power consumption and source status. May be over-ridden by manual intervention at any time. Also presents manual response cues at selected breakpoints for crew verification prior to executing configuration changes.
- Environmental Control System. Cautions and alerts. Provides current inventory data on consumables and preplanned checklist and control procedures.
- CG Determination. Based on sensed expendables data and/or operator input.
- Redundancy Management. Aids in the form of cautions, alerts, and alternate reconfigurations available. These aids are based on periodic examination of the system status table (maintained by IFPM and Executive programs), trend analysis, and stored information on all possible data and control paths.
- Operations Sequencing. Aids are provided for limited automatic sequencing of system operations and reconfiguration. Frequent breakpoints for manual crew verification or override prior to further sequencing.

2.2.3.4 Development and Integration. A systematic and controlled sequence of software development leading to the timely, economical satisfaction of system requirements is an obvious crucial task of the shuttle program. No facet of systems development in recent years has received more attention in hindsight than this most difficult task. For highly integrated systems such as shuttle, software integration becomes nearly synonomous with systems integration and thus necessitates a corresponding emphasis on planning and coordination spanning a broad spectrum of technical detail.

This realization was registered in S-3A facilities planning accomplished during the proposal period and in the preparation for and conduct of the second Software Design Review. Out of these early and extensive efforts to define the necessary facilities and capabilities for software integration testing, a compatible set of detailed objectives, plans and controls were formulated in pursuit of a sure-footed, step-by-step achievement of the program milestones. An examination of the software integration problem afforded by this brief study suggest a similar approach for the shuttle.

For the orbiter avionics system, an extensive parallel flow of design and development effort threatens to produce a critical bottleneck at integration. Because software, unlike hardware, must intimately incorporate a total system design at an early stage and yet be volatile enough at later stages to adjust to discovered component and system level idiosyncrasies, its development requires a special view of the integration process. From the viewpoint of software, integration consists of a series of controlled experiments with which to determine what really produces desired results and what does not. The "debug" process is literally a planned "design tuning" process, hopefully under laboratory-controlled conditions at all times. Properly considered software test requirements therefore have a healthy influence on test design and, for the shuttle, must play the major role in determining the methodology and sequence of testing.

These considerations have been developed in detail by Lockheed Electronics Company personnel working closely with MSC as is reflected in two recent reports: "Space Systems Integration Laboratory Development Study," dated October 1971, and "Space Shuttle Simulation Program Draft Report," dated November 1, 1971.

In keeping with the facilities descriptions and discussions given in preceding paragraphs, the baseline software development and integration concepts drawn from this study may be succinctly described by the diagrams of Figs. 2.2-66 and 2.2-67.

2.2.3.5 Conclusions. As a long-life, reusable, multipurpose, combined aircraft/spacecraft, the shuttle vehicle imposes an unusually high degree of onboard reliability and complexity in the areas of instrumentation and controls essential for thorough preflight checkout and in-flight safety. The number of potentially critical data points and amount of annunciation necessary to cover a reasonable spectrum of "Mission Success" and "Safe" requirements, when combined with the speed with which the flight situation can deteriorate, will certainly exceed the human, manual envelope. Total dependence on hardwired annunciation, manual sequencing and parallel instrumentation for data point collection and conditioning for GSE tie-in or downlink telemetry transmission would result in an unwieldy, costly proliferation of test and status equipments on the one hand, and a massive amount of electrical wiring on the other.

The S-3A concepts and configuration for data management, displays and controls, and BITE oriented onboard checkout and in-flight performance monitoring are especially well suited to the combined aircraft/spacecraft features of the shuttle vehicle.

As for the disadvantages of a "data bus" in terms of the complexity introduced, it is stressed that the S-3A DMS/SIU communications scheme is not a "bus" in that it does not act as a general relay medium for lateral signal communications between many low-level components in the system and does not, therefore, present a supercritical path from the viewpoint of flight safety or mission success reliability.

The approach generally abounds with flexibility, yet offers a number of hard and proven points of departure. To the extent that the DMS and programmable controls/displays features of S-3A are applied, the approach is especially amenable to ground simulation. During the approximate period of two years allowable prior to commitment to detailed designs for S/C avionics, much could be learned from such simulation in support of trades analysis and software requirements definition/development. The

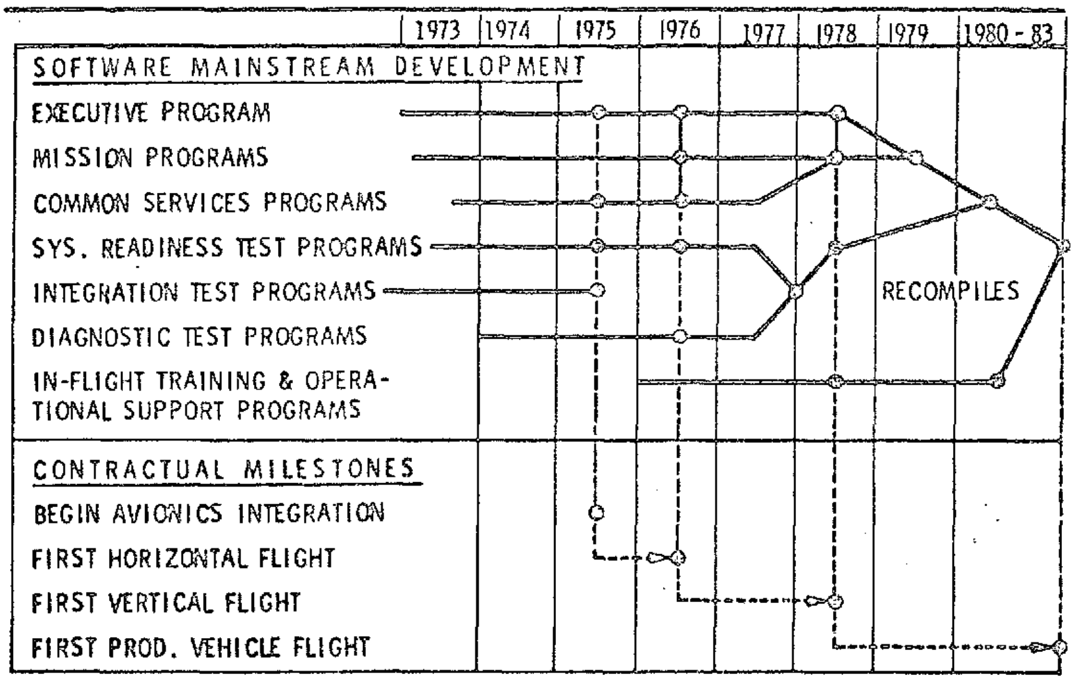

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Fig. 2.2-66 Mark I Baseline Orbiter Software Development

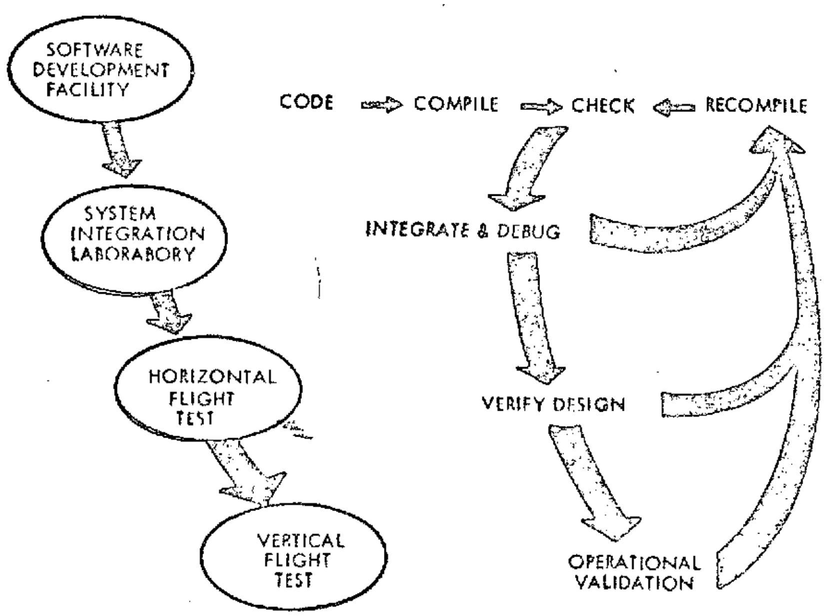

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Fig. 2.2-67 Shuttle Software Integration shortening and removal of overlap of successive development spans avoids difficult to manage and potentially wasteful periods of heavy peak activity and permits more careful analysis, both for initial design choices and for later changes necessary to resolve encountered problems.

Application of the S-3A management experience with large scale, command and control software/hardware development and integration can also be employed to advantage. Successfully applied, prime contractor techniques for parallel integration of this type of multiple vendor/facility activity are worthy of repetition and many of the personnel, plans, facilities and procedures can be applied in a direct and timely manner. The S-3A Software Development Plan, for example, is readily translatable to the shuttle program. It is suggested that such a translation represents one of the logical next steps in the further pursuit of this alternative.

2.3 DELTA IMPACT FOR FIRST VERTICAL FLIGHT – UNMANNED

The delta impact of performing the first vertical flight (unmanned) was determined by identifying those requirements unique to the unmanned vehicle and, then, estimating the additional equipment, software, testing, training, and modifications plus schedule impacts. It was assumed that the unmanned flight would have the same basic complement of Mark I Orbiter avionics onboard as for the manned flight in order to evaluate equipment performance as well as vehicle performance.

The primary change in requirements is to automate all onboard functions or, as a minimum, provide for remote control originating from the ground or from a chase plane. In addition, provisions for monitoring the equipment performance during unmanned flights will be furnished. Range safety requirements impose the need for a command-destruct receiver onboard the vehicle.

Automatic control provisions for nonavionic functions (e.g. ABES deployment/start/propellant control, landing gear deployment, hydraulic power control, time noncritical redundancy management) and for expanded guidance capability to perform aerodynamics flight-path steering were estimated to cost $3 million. In addition, another $3 million cost for TV cameras, video recorders, lighting, and a command-destruct system was estimated.

Additional software development was estimated as follows:

Orbiter	90K	Words for onboard automation
LCC	40K	Words for increased checkout
MCC	160K	Words for increased monitor/control/data handling
Total	290K	Words

A backup "drone control" mode, using ground controllers and a chase plane, will incur additional costs as follows:

Controller training at two stations	$16.0M-yr
Chase plane equipment modification	$0.5M
Chase plane operation	$0.3M

Ten additional two-hour horizontal flights for verification and demonstration of system operation are estimated to cost an additional $1 million.

The delta cost for first vertical flight (unmanned) is then:

Orbiter equipment	$ 6.0M
Software	8.7
Horizontal flight tests	1.0
Ground support/installation	5.9
	$21.6M

No schedule impact is anticipated if the decision to fly unmanned is made at the start of the program. The design and development of automatic onboard systems can be performed within the available time span. Additional software development for automatic capability means an earlier program commitment to what otherwise would be a Mark II capability. Flying unmanned represents a decreased risk to the orbiter crew but an increased risk to the vehicle and program, since no man is onboard to make critical real-time decisions in cases of emergency.

2.4 MARK II AVIONICS SYSTEM

Evolution from the Mark I baseline avionics system to the second generation Mark II capabilities will involve basic improvements to subsystem performance, safety and reliability improvements, and expanded software capabilities. Performance and hardware/software quality improvements applicable to Mark II are currently in progress as advanced technology developments under various NASA, Department of Defense, and industry-wide funded programs. These will provide improvements for the Guidance, Navigation, and Control System (GN&C), Control and Display, Communications and Tracking, Instrumentation, Electrical Power, and Data Management and Checkout, Fault Isolation, and Redundancy Management (COFIRM) systems as extensions to the Mark I baseline. Improvements will reduce the Mark I operating costs sufficiently to more than pay for the estimated nonrecurring costs for Mark II. This is accomplished largely by onboard orbiter navigation improvements and the full onboard autonomy of COFIRM which considerably reduce Mission Control Center and remote ground station operations and associated costs.

2.4.1 Mark II Orbiter Avionics Projection

Projections of the Mark II avionic system changes from the Mark I baseline (Table 2.4-1) have been confined to within a 40 to 50 percent nonrecurring cost allowance over that of Mark I. Desired gross subsystem performance and quality improvements were identified and cost estimates prepared. Similarly, the increased software requirements corresponding primarily to an expanded role for the data management system have been estimated. These investigations show that the desired additions and improvements are well within the allowable cost growth. Projected cost and schedule impact of the Mark II avionics system development are discussed in further detail in the subsequent Section 2.5.

2.4.1.1 Mark II Orbiter Guidance, Navigation, and Control (GN&C). Except for the automatic docking and rendezvous with uncooperative target requirements, the Mark I Guidance, Navigation, and Control subsystem meets all space shuttle avionics requirements (as defined in MSC 04075B, including autonomous navigation and automatic approach and landing capabilities).

Table 2.4-1

MARK II AVIONICS CHANGES

Item	Changes
Subsystems	
Guidance, Navigation, and Control (GN&C)	Improve performance and quality of equipment; accuracy improvements reduce ACPS, ΔV propellant use and reduces reentry dispersions.
Control and Display	Add area navigation/autoland CRT display.
Communications & Tracking	Improve performance and quality of equipment.
Instrumentation	Improve performance and quality of equipment.
Electrical Power	Provide 5000 hour life fuel cell. Improve performance and quality of equipment.
Data Management, Checkout, Fault Isolation & Redundancy Management (COFIRM)	Increase onboard COFIRM for nearly complete autonomy for both avionics and non-avionics.
Software	Performance of functional operations through software instead of hardware could significantly increase mission flexibility and decrease change reaction time. Greater reliance on software in flight controls and COFIRM will require advanced management techniques.
Orbiter	
Mark I Equipment Deletions	Horizon sensor and orbit altimeter.
Tracking Satellite	Used to augment navigation.
On-Board Navigation, Data Management and COFIRM Improvements	Minimize dependence on ground control and remote stations.
Safety/Reliability	Improved quality of equipment will increase probability of mission success and enhance safety. More autonomous fault isolation and redundancy mangement will reduce crew workload and decrease corrective action time.

The Mark II GN&C subsystem as proposed in this study will be changed in the following areas:

- Equipment will be upgraded to meet space environments and to improve performance, accuracy, and quality. Redesigning and requalifying equipment will reduce the avionics equipment imposed requirements for pressurized equipment bay volume and forced air cooling.

  Decreased gyro drifts and accelerometer biases will reduce software and test requirements and could result in a reduction in propellant loading requirements, thus resulting in more accurate guidance and navigational capabilities. Improved equipment quality will improve crew safety aspects and the probability of mission success, and operational costs will be reduced. The ability to reduce failures and increase time between preventative-types of maintenance will reduce the need for replacement of equipment.

  Equipment affected would include the horizon sensor, the inertial measurement unit (IMU), and the digital computer. The new horizon sensor (such as the Quantic Model IV) will have an altitude accuracy capability of at least three times better than the Mark I (Barnes 13-166) unit.

  An IMU candidate is the dodecahedron (six-skewed gyro) unit which theoretically increases the reliability of the IMU system by a factor of ten over a triply redundant three-axes system. The six-pack will also allow voting by axis for failures beyond the first, whereas with the triply redundant IMU voting in any given axis is not possible after the first failure in that axis.

  For Mark II vehicles, a fourth-generation computer (such as the Magic 362, CDC 469 or the GE CP24A) would replace the Univac 1832 machine. (Alternatively, a smaller, lighter Univac machine could be employed.) As shown in tabular form under Section 2, of par. 2.2.1.1.2, significant reduction in weight and power (about a factor of five) can be achieved. Discussions with the suppliers indicate that the unit cost could also be reduced by one-fifth ($100K vs $500K). The cost of developing and validating new software, however, would decrease the price advantage.

- Combining aircraft and spacecraft flight control functions in the digital GN&C computer would eliminate a significant amount of equipment for use in the Mark I GN&C system. The primary flight control and automatic flight control computers (ten in all) could be replaced with simple servo-amplifiers, with control law computations being digitally performed. The three ACPS and three thrust vector control (TVC) electronics packages can be simplified to thruster drivers (Schmitt triggers and power stage) and servo amplifiers, respectively. All control logic and computations (gain scheduling, deadbands, and compensation) can be performed in the digital computer.

  To offset the reduction of the analog dedicated flight control equipment, at least one additional level of computer redundancy should be considered.

- Use of tracking satellites to navigate would reduce sensor requirements. The horizon sensor and orbit altimeter could be eliminated, leaving only the precision ranging system (range, range rate and line of sight) and a star tracker, together with an inertial measurement unit (with appropriate redundancies) as the complete complement of navigational sensors for ascent and orbital use. TACAN used on Mark I for approach navigation and the scanning beam ILS used for landing could also be deleted (or amount of redundancy reduced) through the use of the PRS and ground transponders.

- Provisions for automatic docking and rendezvous capability with uncooperative targets could be added; however, these capabilities were not proposed in the study due to a lack of mission definition. Both capabilities could be added by including radar (either microwave or laser) that exists in various stages of development today. By the time equipment is required for Mark II, it is expected that lower weight, lower power, multipurpose radar will be available for shuttle use.

  In each case (except for the addition of automatic docking and uncooperative target rendezvous) the changes proposed for the Mark II program are of an evolutionary nature, rather than abrupt changes in concept and capabilities.

2.4.1.2 Communications and Tracking. The area of communications system development is particularly in a dynamic stage of growth on an industry-wide basis as a result of cooperative informationally-sponsored satellite and ground communications systems development and bandwidth limitations facing future operations and high interest in laser technology. Baseline Apollo-type system improvements can be anticipated as a result of technology improvements, as well as several "near-at-hand" breakthroughs that will offer opportunities for significant operational technique and hardware performance capability improvements. The major area of anticipated new technology developments affecting Mark II operational flight communications and tracking appears to be in the area of laser concepts and applications, potential laser communications hardware development, and satellite networks planned for communications applications that will become operational in time for Mark II use. Also, operational technique improvements will enable even less dependence on remote ground stations by use of direct, real-time contact with the MCC via communications network relay, e.g., synchronous equatorial satellites in line-of-sight of the space shuttle, each other, and the ground station at all times to eliminate remote station data readout and readin functional requirements.

Except in the broad-base general sense, no attempt has been made to project or extensively evaluate the impact of these major changes to the existing Apollo-type communications and tracking capabilities other than using more conventional technology state-of-the-art advances in the hardware concept as a base of estimation for evolution to the Mark II from Mark I design. Technological advances in this area may provide a tangible basis for considerable simplification, performance improvement, and quality increases, with accompanying cost reductions beyond those presently anticipated. For purposes of this study, a more conservative approach has been taken in projection of the Mark I to Mark II evolution on the basis of present technological experience. Hence, the anticipated level of delta costs for communications system improvement has been statistically calculated at approximately $5 million.

General improvements in current technology, performance, and quality can be more optimistically viewed as achievable by the Mark II flight data as a result of industry-wide advancements -- both under NASA/DoD contracts and as a result of concentrated independent development. For example, considerable work has been performed by LMSC alone over the past two years for demonstration (laboratory) of ultra-wideband optical communications (UWOC) electronics for source modulation and receiver detection and discrimination. High performance was demonstrated in the 1 to 2 GHz bandwidths. Work performed in optical systems for UWOC has included development of widebank modulators, and solid state devices for optical frequency up-conversion.

Practical communications and tracking systems (including navigation and navigation aids) using laser technology, including work performed in conjunction with the Apollo program, may considerably change the Mark II subsystem development approach in this area as these fields reach maturity and are flight-qualified in other applications.

2.4.1.2.1 Mark II Communications. The communications equipment for the Mark II orbiter will remain essentially the same. The S-band equipment will be modified to be compatible with S-3A BITE configurations, and all elements will be updated to incorporate new technology where performance or reliability may be improved.

2.4.1.3 Electrical Power (Mark II). The Mark II orbiter will use the basic Mark I electrical power system functional components and electrical inter-relationship between these components. The location of components and their physical description will change with the following proposed component performance improvements. These are described below.

- Fuel Cell. The fuel cell design life will be extended from 2000 to 5000 hours through a development program that will add to the initial development program. A 2000-hours demonstration program on the Mark II design will limit costs to less than 50 percent of the Mark I design development cost.

- Cryogenic Storage System. Improvements are anticipated in insulation efficiency and ancillary controls, but the Mark I tank selection and development will not be repeated for Mark II.

- Static Inverters. Static AC inverters will be developed to provide the desired module size and reduced weight.

- Transformer-Rectifiers and Generator Control Units. These Mark I aircraft designs will be modified to cold plate cooling and vacuum operation to provide freedom of space allocation in the Mark II orbiter.

- Power Switching. The application of solid-state switch/circuit breaker units able to handle higher currents and provide increased circuit protection capability will be increased.

- EPS/DMS Interface. The control of the power system will be modified to incorporate increased computer control of on-board checkout, of power system operation, sequencing, and both normal and emergency configuration control. EPS equipments will be modified to interface with this increased computer control capability.

2.4.1.4 Mark II Control and Display Configuration. The basic S-3A data management system and interface units are included in the recommended LMSC baseline. This provides an initial Mark I control and display capability which, for the most part, is compatible with and sufficiently flexible for the projected Mark II configuration. Major capability improvement envisioned for Mark II is in the area of software and the extension of certain "automated or programmed" functions, thus further offloading workload from the crew. Representative software capability recommended for growth from Mark I to Mark II include the following:

- Increased software sequencing of control functions
- More automated vehicle configuration
- Increased onboard mission planning through computer operation
- Increased payload operations planning for complex missions
- Greater consumables management computer analysis

It is expected that the control and display devices currently planned for Mark I will be sufficiently flexible to accommodate this increased software capability. The only equipment growth anticipated for Mark II is the addition of area navigation (RNAV) controls and displays. The prime operational applications of area navigation permit reduction of flight distance between two points in a route structure; preorganized arrival and departure flight paths in terminal areas, reducing pilot and controller workload; and permit instrument approaches (within limitations) to airports and runways not equipped with landing aids. Considering the number of flights per year, normal ferry operations between deorbit landing sites and refurbishment centers, and potential abort situations, area navigation is a capability very likely to be incorporated in the Mark I orbiter.

The RNAV CRTs and the existing DGU can also perform ancillary functions such as the following:

- Orbit map displays
- Complex rendezvous and docking maneuver plots
- Backup for the MDU subsystem display (CRT)
- Ancillary source for displaying computer calculations requested by the crewman
- Addition source for procedures display, abort, COFIRM, operations, payload, etc.

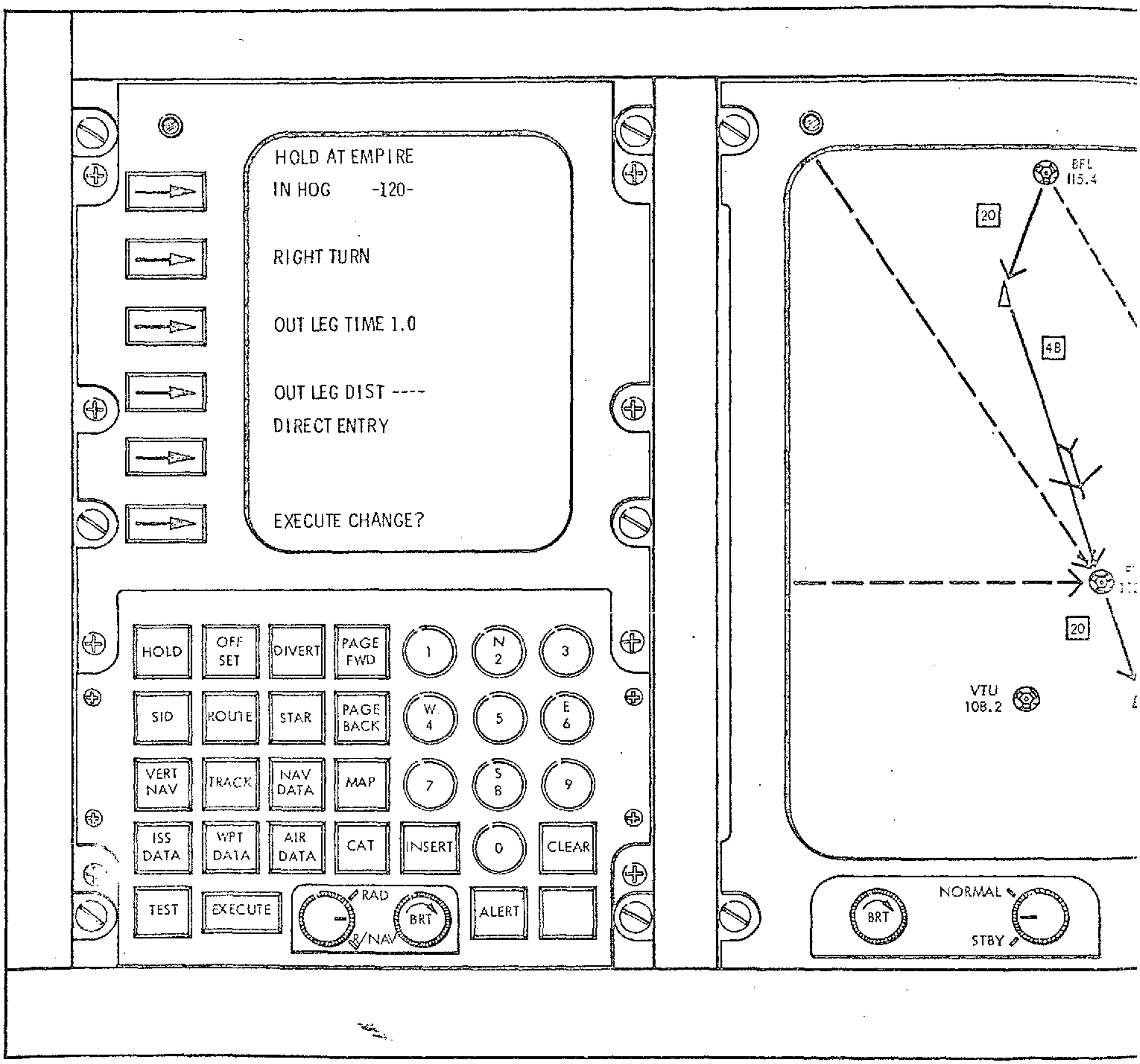
Figure 2.4-1 illustrates the general configuration of the control and display units (2) and the electronic map display unit. Growth volume was provided in the Mark I main instrument panel layout for the possible inclusion of these instruments. When the RNAV is combined with the autopilot/autoland subsystem, considerable capability exists for inclement weather operations for atmospheric flight.

2.4.1.5 Mark II Data Management and Onboard Checkout. The DMS grows in capability through the various phases of Mark I to the operational Mark II, as shown in Table 2.4-2. While the Mark I system is largely passive (except in the area of electrical power control and onboard checkout), the operational Mark II system takes an active part in all phases of effective mission accomplishment. Hardware and software additions and modifications are required for the increased capabilities. Hardware growth requires the necessary modification to SIUs to allow control as well as monitoring of non-avionic subsystems. Manual overrides are retained on flight-critical functions. Addition of control and monitoring capability will extend automatic checkout to the interfaced subsystems with an attendant reduction in ground support. It is anticipated that with the additional onboard capability with the Mark II DMS configuration, ground-based mission support can eventually be reduced to one central station at Cape Kennedy.

Table 2.4-2
DMS FUNCTION PHASING

Phase / Function	Mark I			Mark II
	HFT	VFT	OPNL	
Instrumentation Control	o	o	o	o
Onboard CO/FI	o	o	o	o
Programmed Display Control	o	o	o	o
Maintenance Data Log	o	o	o	o
Electrical Power Control	o	o	o	o
Abort Warning Computation	o	o	o	o
Avionics Sequencing and Configuration	o	o	o	o
Guidance/Nav. Computation		o	o	o
Mission Administration Log		o	o	o
Overall CO/FI/RM		o	o	o
Automatic Configuration Control			o	o
Consumables Management			o	o
Rendezvous Computation			o	o
Payload Management				o
Nonavionics Sequencing and Control				o
A/C and S/C Flight Control				o
Mission Planning				o

o First Usage
o Subsequent Usage


HOLD AT EMPIRE
IN HOG -120-
RIGHT TURN
OUT LEG TIME 1.0
OUT LEG DIST ----
DIRECT ENTRY
EXECUTE CHANGE?
HOLD
OFF SET
DIVERT
PAGE FWD
1
N 2
3
SID
ROUTE
STAR
PAGE BACK
W 4
5
E 6
VERT NAV
TRACK
NAV DATA
MAP
7
S 8
9
ISS DATA
WPT DATA
AIR DATA
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CLEAR
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VTU 108.2
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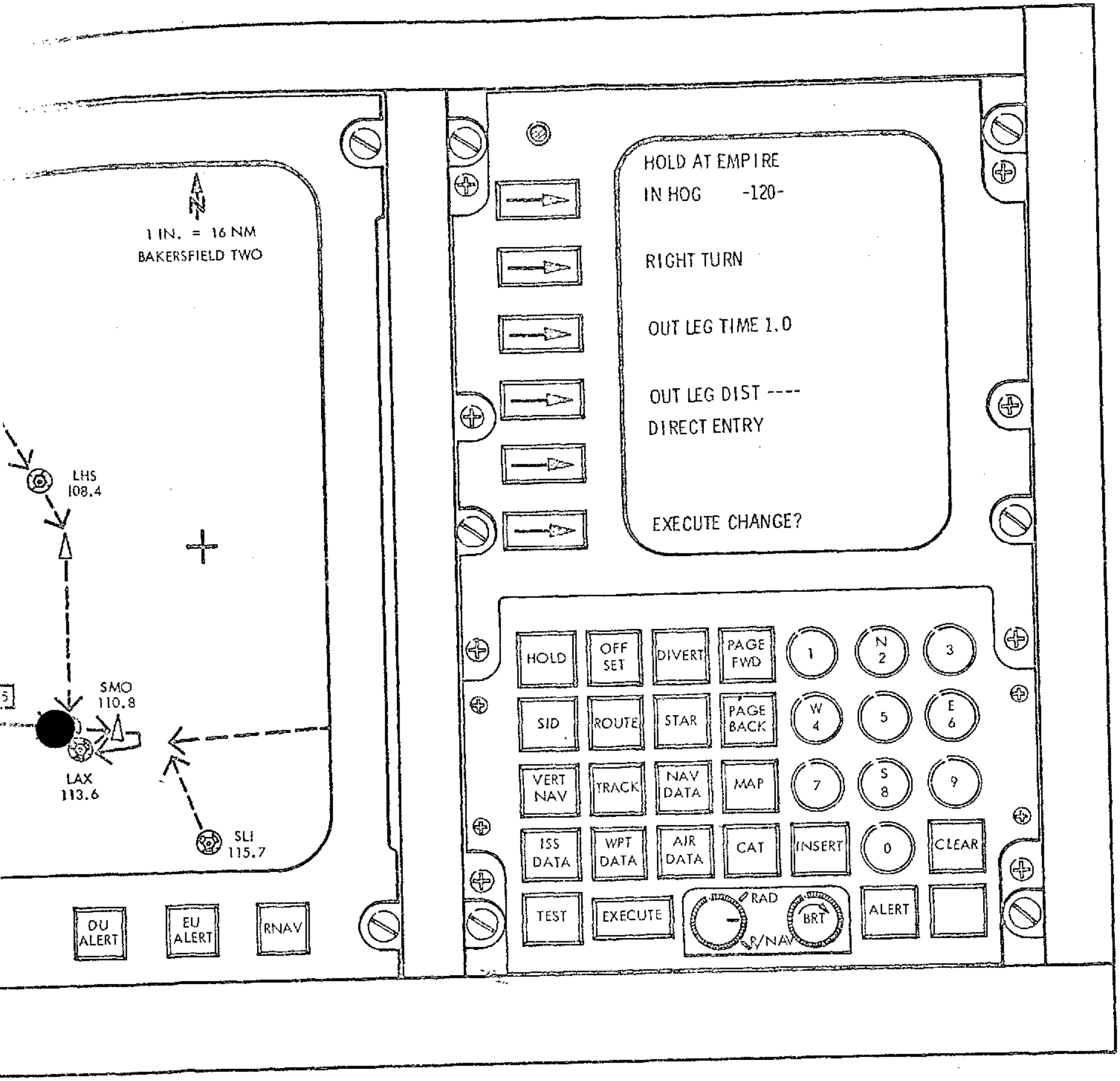


Fig. 2.4-1 Control and Display and Electronic Map Units

2.4.1.6 Mark II Instrumentation. The instrumentation subsystem of the Mark II orbiter will follow the same philosophy as that of Mark I, being controlled by the DMS, with the DFI being an overlay up through the first level of multiplexing. Because of the projected greater integration of control avionics, it is estimated that the operational instrumentation load will not increase.

2.4.2 Ground Support Equipment Mark II

The electrical ground support equipment utilized in the Mark I shuttle avionics system interfaces in Mark II with a shuttle system that has (1) an increase in command capability over each avionics subsystem and (2) a data management system that has extended its control and checkout capabilities to all shuttle systems at each subsystem test to effect the desired rapid two weeks turnaround schedule. Checkout and fault isolation of the line replaceable units for maintenance and checkout will still be accomplished in the avionics subsystem test laboratory with parallel, onboard, circuit troubleshooting as required. The bench test equipment will be similar to that used on the Mark I program with the significant changes centering around the modification/replacement required of the test equipment to match the increased capability of the Mark II shuttle components.

Electrical ground support equipment for integrated systems checkout and launch will undergo a relatively small change. The Launch Control Complex (LCC) equipment will be quite similar to the previous Mark I program, with significant changes being attributed to the software programs needed to accommodate the inclusion of the testing and monitoring of the nonavionic shuttle systems. The major impact is expected to be a reduction in manual control from the cockpit resulting from expanded automatic DMS control capability which can be exercised by the LCC through the orbiter DMS.

2.4.3 Mark II Software

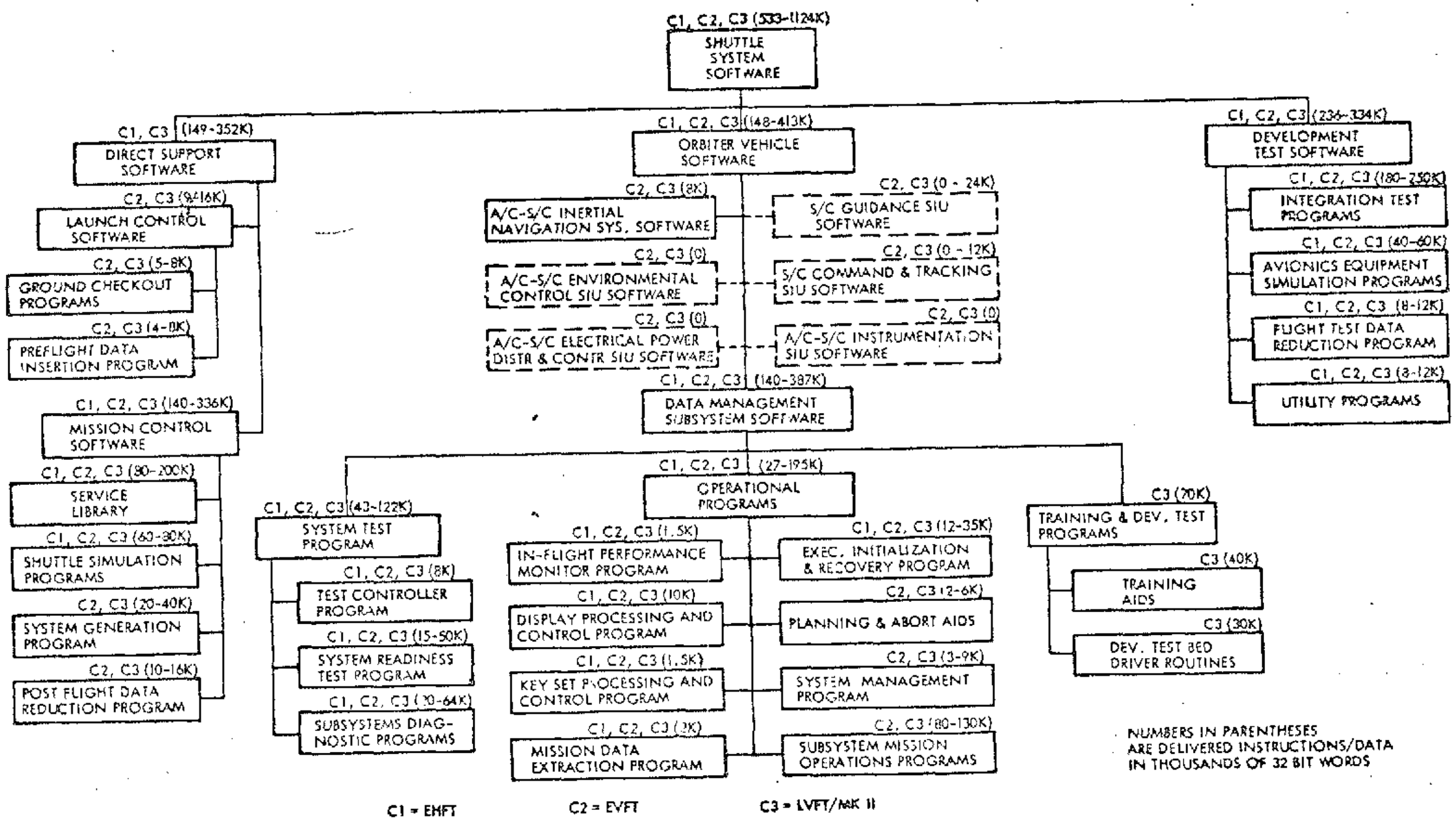
Emphasis is placed on extending DMS and software functions to provide improved capability of a level which may be readily justified as logical and reasonable growth. As operational phases of the shuttle system approach realization, it is assumed that plans will be laid for providing additional assistance to the crew in performing what must eventually become routine operations. The progressive introduction of additional mission capabilities and payloads will inevitably increase crew training and possible on-board mode/procedure proliferation to a point where extensive computer-aided activity is a necessity.

For these reasons, Mark I vehicles should be progressively updated with respect to DMS/software capability at planned intervals of 18 months. This span of time has been found to be an almost forced necessity in operational command and control software applications of any significant scale (NAVY P-3, for example) because of the inevitable refinements that go with extensive test and usage. The capabilities listed in Table 2.4-3 for Mark II represent an initial, planned target for a point in time 36 months after FMOF. The content of the first, 18-month increment directed toward this target is best left to subsequent detailed planning. A representative, on-board software configuration showing major subprogram components is illustrated in Table 2.4-2. Subprograms which have lesser versions available for EHFT (C1) AND EVFT (C2) are so identified. As with progressive hardware development, and perhaps more so with software, it is important at the onset that possible intermediate and ultimate DMS capability objectives be understood. This is essential for proper machine selection and early control program design.

At the stage of the shuttle program being addressed, the need for in-depth training of multiple crews is an item of major concern. The flexibilities of the proposed DMS with regard to simulation offers an especially good approach to this problem. On-board crew training aids can be as nominal or substantial as one wishes to make

Table 2.4-3

SOFTWARE FUNCTIONS EFFECTIVITY



them but, in the instance of a general-purpose, long-life space shuttle, it is assumed that an ultimate objective of substantial on-board training capability is well justified. This assumption is reflected in the software configuration of Table 2.4-2.

An item of on-board software shown in the hierarchy calls for MK II Development Test Bed Driver Routines. Here it is suggested that FTV-2 and, to a lesser extent prior to refurbishment, FTV-1 should be enabled as advanced development test beds for in-flight evaluation/verification of planned additional capabilities, by experimentation, on a basis of noninterference with primary mission objectives, and/or special experimentation flights. In the case of new prototype hardware or software components it may be desirable to test an item under real flight environments without actually coupling into the current operating system. Special driver routines, operating under a background level of the DMS Executive Scheduler, could provide simulated coupling into the operating system and record the results for post-mission analysis. Where man/machine factors are critical, one of the crew stations could be placed off-line in a similar fashion to allow safe trails under actual operating conditions.

The expansion of System Test and Operational programs to include (1) nonavionic subsystems management (including payloads) and mission operations support and (2) automatic A/C and S/C flight path control significantly increases the total sizing of the shuttle software system. The total number of 32-bit words of loadable programs and data range from approximately 500K during early HFT to 790K for LVFT and early Mark II operations, then to approximately 1.1M at a point 36 months after FMOF. Thought the software growth described is represented as first employed on Mark I vehicles, it is directed toward Mark II operational capability, and the increase in development costs associated therewith has been so allocated.

2.5 MARK I/MARK II AVIONICS PHASING

A number of basic avionics changes will be scheduled during the Mark I orbit, and will include the following:

- Addition of spacecraft-type equipment for vertical flight, to augment that aircraft-type equipment initially installed in the test vehicle for horizontal flights
- Addition of mission-critical equipment, such as the GN&C Star Tracker and Horizon Sensors, for transition from test flights to operational flights
- Addition of one or more levels of redundancy
- Refurbishment of the horizontal test vehicle to the operational configuration.

These changes and additions to the avionics system can be incorporated as the need arises. However, the recommended approach is to provide installation well in advance of actual need in order to prove capability and compatibility. Cost penalties to the program should be insignificant, since no additional flights would be necessary; only the timing of equipment installation would be affected. Additional hardware and installation costs should not accrue, i.e., the same hardware would be required for the next phase anyway; only the schedule of installation is changed. Software costs are also not significantly affected by earlier hardware installation; again only timing of software changes is affected. However, the approach would not be capable of completely proving out equipment since actual flight operational environments would not be encountered in the test flight phases. For example, if equipment required for vertical flights are first tested during horizontal flights, the equipment will not be exposed to either the space or high reentry Mach number environments. If determined to be significant, this shortcoming can be minimized by implementation of a phased test program that increases the velocity and altitude in increments until a Mach number of 10 or 12 and an altitude of some 200,000 feet are reached. However, additional test flights would be required. The cost of these additional flights have not been investigated for this study.

As indicated in the par. 2.4, changes proposed in the transition from Mark I to the Mark II avionics configuration are essentially evolutionary in nature. With few exceptions, basic concepts and functions remain unchaged (Primary changes occur in the

PERFORMANCE

- o INCREASED G&N ACCURACIES TO REDUCE ACPS AND V PROPELLANT USE AND REDUCE REENTRY DISPERSIONS

HARDWARE SAFETY AND RELIABILITY

- o IMPROVED QUALITY OF EQUIPMENT TO INCREASE PROBABILITY OF MISSION SUCCESS AND ENHANCE SAFETY
- o MORE AUTONOMOUS G&N AND FAULT ISOLATION AND REDUNDANCY MANAGEMENT TO REDUCE CREW WORKLOAD AND DECREASE CORRECTIVE ACTION TIME

SOFTWARE

- o GREATER RELIANCE ON SOFTWARE IN FLIGHT CONTROLS AND CHECKOUT FAULT ISOLATION AND REDUNDANCY MANAGEMENT (COFIRM) THIS REQUIRES ADVANCED MANAGEMENT TECHNIQUES
- o PERFORMANCE OF FUNCTIONAL OPERATIONS THROUGH SOFTWARE (INSTEAD OF HARDWARE) TO SIGNIFICANTLY INCREASE MISSION FLEXIBILITY AND DECREASE CHANGE REACTION TIME

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Fig. 2.5-1 Projected Mark II Avionics Change Impact

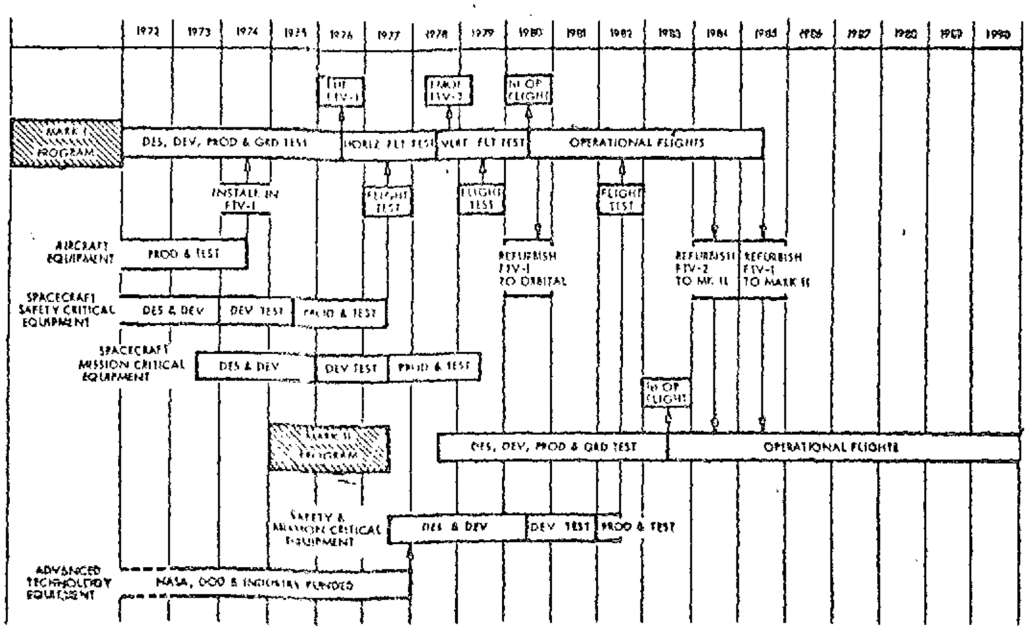

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Fig. 2.5-2 Mark I/Mark II Avionics Phasing method of implementation, performance capability, quality of equipment and scope of coverage of certain capabilities). The autoland/autopilot aircraft functions could be incorporated into the digital GN&C computer. Equipment accuracies could be improved. Other examples of proposed changes include improving the system's capability of withstanding space environments, increasing equipment lifetime, and expanding the capability of checkout, fault isolation, and redundancy management within avionics and other subsystems outside of avionics (e.g., propulsion). The impact of some of these changes is shown in Fig. 2.5-1.

An overview of key Mark I/Mark II milestones, including the change points are shown in Fig. 2.5-2. As in the case of the Mark I program, a flight test using a Mark I vehicle to demonstrate capability and compatibility of Mark II equipment is proposed for accomplishment about one year before the first Mark II flight.

The impact of Mark II avionics changes on cost is shown in Fig. 2.5-3. The accumulated cost for the sum of the Mark I and Mark II programs is $527.1 million. If the Mark II missions are conducted using Mark I avionics unchanged, the total program cost would be $475.6 million.

If the mission ground support effort can be reduced to one center at Cape Kennedy, and equipment is deleted that is no longer required due to use of navigational satellites, avionics costs can be reduced as much as $66.4 million for an actual reduction in cost of some $16 million in going from Mark I to Mark II. A breakdown of the cost increase and decrease is shown in Table 2.5-1. However, to achieve the cost savings it is necessary to increase the on-board checkout, fault isolation, and redundancy management capabilities, i.e., the $58.9 million cost savings is directly tied to the $32 million COFIRM expenditure.

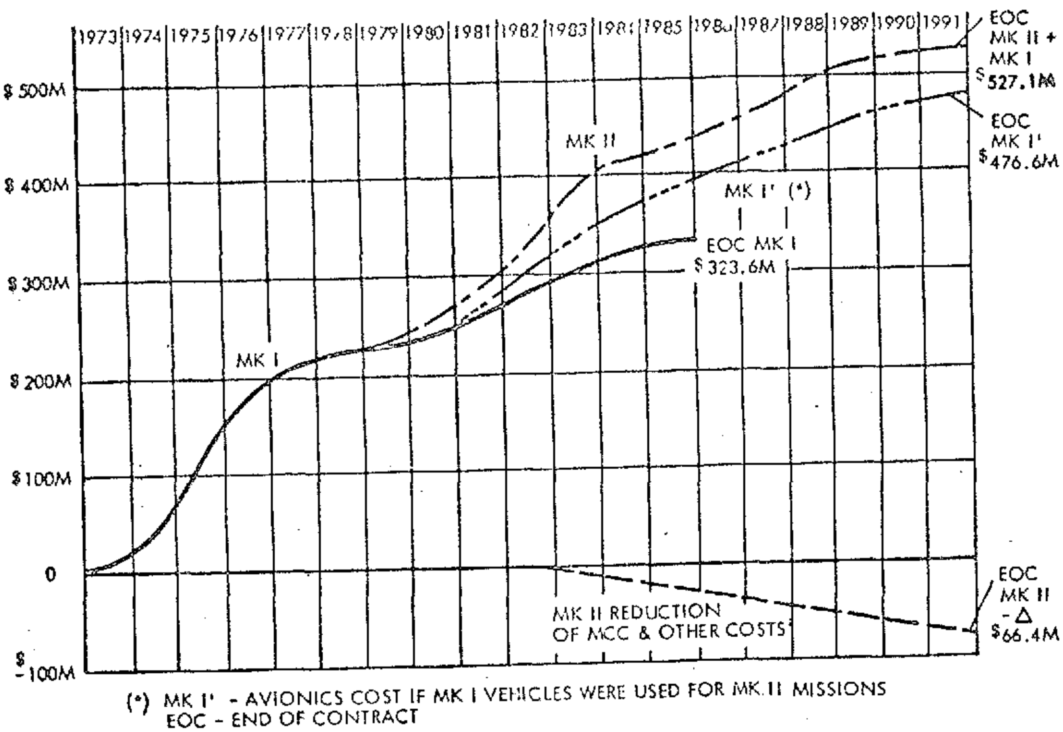

Fig. 2.5-3 Mark II Orbiter Avionics Cost Delta

Table 2.5-1

MARK II AVIONICS COST DELTA BREAKDOWN

Cost Increase		Delta Cost*
GN&C Improve performance and quality of equipment		+ $ 3.5M
Control and Display Add area navigation/autoland CRT display		+ 2.0M
Communications and Tracking Improve performance and quality of equipment		+ 5.0M
Instrumentation Improve performance and quality of equipment		+ 4.0M
Electrical Power 5000-hr life fuel cell. Improve performance and quality of equipment		+ 4.0M
Data Management and Cofirm** Increase on-board support, scope, and coverage of cofirm (both avionics and nonavionics)		+ 32.0M
Total Increase		+ $50.5M
Cost Decrease		
Delete horizon sensor, star tracker and orbit alternate (navigate with tracking satellite)	-$1.5M x 5 =	$-7.5M
Reduce MCC and remote ground station		-58.9M
Total Decrease		-$66.4M

*Difference between using Mark II and using Mark I unchaged
**Cofirm – checkout, fault isolation and redundancy management

2.6 BOOSTER AVIONICS

The booster avionics is defined in Fig. 2.6-1, and in the following text.

2.6.1 Flight Controls (FC)

The FC functions are primarily controlled by the orbiter avionics GN&C subsystem and consist of the following:

- o Interlocks package
- o Barometric altimeter
- o Engine and propellant control unit
- o LITVC drive electronics
- o LITVC valves

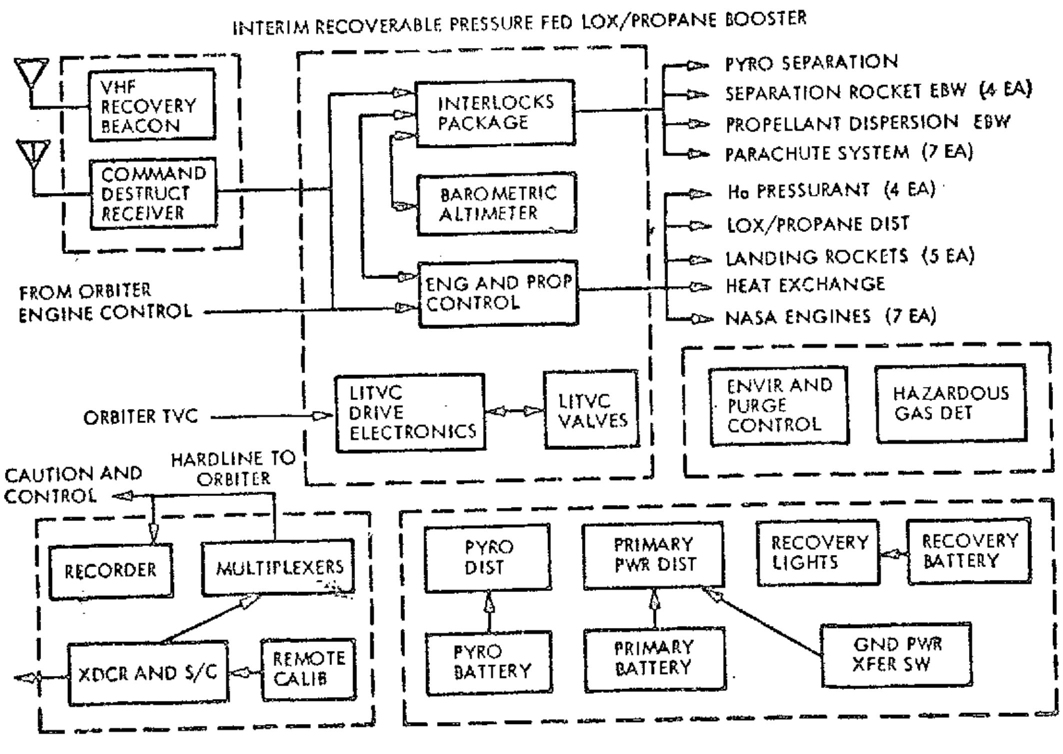

Fig. 2.6-1 Booster Avionics Selected Point Design

The interlocks package receives inputs from the orbiter GN&C subsystem, the command destruct receiver, and the barometric altimeter. It contains an event sequencer which exercises control during and after separation.

The engine and propellant control unit is self-contained, receiving commands from the orbiter GN&C system. Operation is monitored at the GSE and aboard the orbiter through the instrumentation system.

The liquid injection thrust vector control (LITVC) system controls the booster attitude during engine burn. The steering control signals are generated in the orbiter GN&C thrust vector control (TVC) electronics. The orbiter TVC electronics gets its attitude error signals from the orbiter Interial Measurement Unit (IMU) through the GN&C digital computer and the rate and normal and lateral load alleviation acceleration signals from the orbiter rate gyros and accelerometers. These signals are all processed in the orbiter TVC electronics, then sent to the booster electronics as combined attitude error plus rate plus acceleration $(K_1\theta_\epsilon + K_2\dot{\theta} + K_3A_Y + K_4A_N)$ signals to control the booster engine gimballing. Cross-coupling computations for pitch/roll and yaw/roll are performed in the booster TVC electronics, and servo amplifiers drive the liquid injection thrust control valves.

Maximum longitudinal acceleration due to "pogo" effects or due to excessive thrust are controlled through the orbiter GN&C IMU accelerometer and digital computer. Engine "on" and "off" discretes for acceleration control and at achievement of staging velocity are sent from the GN&C computer to the booster engine controller.

Manual overrides of both thrust vector control and engine "off" operations are provided at the orbiter pilot and copilot stations.

The normal inflight cutoff sequence is center engine first, followed by the outboard engines.

In an emergency, the engine will be cut off by any of the following methods: ground support equipment (GSE) command cutoff, prior to umbilical disconnect; range safety command cutoff; "thrust not OK" cutoff; emergency detection system; and outboard cutoff system.

2.6.2 Communications and Tracking

Booster communications consist of a VHF recovery beacon transmitter and a command destruct receiver. The instrumentation telemetry data are transmitted from the orbiter vehicle.

2.6.3 Electrical Power

The booster electrical power subsystem consists of:

- Primary batteries
- Pyro batteries
- Recovery beacon and lights batteries
- Electrical and pyro power distribution
- Recovery lights

Control is exercised by the orbiter before separation and the interlocks package after separation. Control and display functions are performed aboard the orbiter.

2.6.4 Data Management

Booster data management is limited in Mark I to those functions monitored through the booster SIU. Since the booster mission is of short-duration, it is assumed that no DMS system, as such, will be carried onboard the booster itself. In Mark II, growth would again be in the areas of pre-separation propulsion and flight control.

2.6.5 Instrumentation

The instrumentation subsystem monitors functional operations of booster systems and provides signals for vehicle tracking during burn and return. Prior to liftoff, measurements are telemetered by coaxial cable to ground support equipment. During flight, data are transmitted to ground stations over orbiter RF links.

The subsystem consists of:

- o Data acquisition and conditioning units
- o Multiplexer units
- o Remote calibration equipment
- o Flight data recorders

The booster has been assumed to be as complex as the Saturn SIC vehicle, requiring a comparable amount of instrumentation. The test vehicle requires 900 data measurements and the operational vehicle requires 300 data measurements. All of these signals are required for orbiter controlled operation and for telemetry to ground GSE.

2.6.6 Booster/Orbiter Equipment Redundancy and Commonality

The booster equipment is selected from the baseline O4OA orbiter vehicles equipment list as much as possible. However, the limited study conducted in this area shows little commonality between booster and orbiter equipment. Table 2.6-1 lists the equipment and quantities required, weight commonality with orbiter equipment, and prior program application.

Equipment is defined for minimum safe conditions. LITVC injection valves are assumed to be four valves per engine in the configuration shown. In the case of one engine out, the assumption is that the basic NASA design allows for one engine failure with one other shut down for stability control. In the case of one LITVC valve stuck open, the same assumption holds in that another valve could be used to exert a counter force.

Table 2.6-1

BOOSTER AVIONICS WEIGHT SUMMARY

Subsystem Equipment	Quantity Min	Quantity Safe	Total Weight (Safe)	Orbiter Common	Program Application
Communication					
Data Hardline to Orbiter	1	2	10	X	New
VHF Recovery Beacon	1	1	13	X	Apollo
CMD Destruct Receiver	1	2	30	X	Saturn
VHF Recovery Ant	1	1	1	X	Apollo
CMD Destruct Ant	1	1	1	X	Saturn
Subtotal			55		
Flight Controls					
TVC Electronics	7	7	70	X	Agena
Fluid Injection Valves		28	196	No	
Barometric Altimeter	1	3	30	No	
Interlocks Package	1	3	45	No	New
Eng and Prop Control	1	3	45	No	New
Subtotal			386		
Electrical Power					
Primary Batteries	3	5	300	X	Eagle
Recovery Beacon Battery	1	2	20	X	Apollo
Pyro Battery	1	1	15	No	Apollo
Ground Power Transfer Switch	1	1	4	No	New
Ground Power Receptacle	1	1	4	No	Apollo
Main Power Distribution Unit	1	2	20	No	New
Pyro Distribution Unit	1	1	5	No	Apollo
Recovery Lights	1	4	8	No	Apollo
Electrical Harness	1	2	300	No	New
Subtotal			341		
Instrumentation					
Transducers	75	75	38	X	C-5A
Signal Conditioners	200	200	40	X	C-5A
Submultiplexers	5	5	25	X	C-5A
Sequence Control	1	1	37	X	C-5A
Remote Automatic Calibration Unit	5	5	40	No	Apollo
Flight Recorder	1	1	48	X	C-5
Subtotal			308		
AVIONICS TOTAL			1090		

2.7 COST AND RISK FACTORS

The major factors that determine avionics system costs can be associated with total program cost, peak annual cost, or both. The major cost driver for total program cost is the amount of onboard capability provided versus the amount of ground support required to support mission planning, launch control, and mission operations. As the use of ground support is decreased, the cost of supporting facilities and personnel is decreased but the risk to mission success is increased unless equivalent capability is provided onboard the vehicle. For the three system alternates considered in this study a comparison was made of orbiter avionics costs, ground support costs for launch control and mission operations (KSC, MCC, and MSFN), and total costs. Table 2.7-1 summarizes this comparison and shows that the baseline system (Alternate C) provides a net savings in total program cost, although its onboard avionics cost is higher than for Alternates A or B. The risk associated with the baseline system is defined by the longer time intervals between ground station contacts for an orbiting vehicle.

Table 2.7-1

MARK I CONFIGURATION IMPACT ($M)

	Alt A	Alt B	Baseline
Maintenance and LCC	248.6	174.0	149.2
MCC	77.1	77.1	54.0
Remote Sites	5.4	3.5	3.5
Support Cost Totals	331.1	254.6	206.7
Vehicle Program Costs	250.3	289.2	325.2
Impact on Support Costs (Δ From A)	0	-76.5	-124.4
Vehicle Costs (Δ From A)	0	+38.9	+74.9
Net Savings			
B and BL Over A	0	37.6	49.5
BL Over B		0	11.9

Another major cost factor, generally, is the amount of existing/modified equipment used versus the amount of improved state-of-the-art and new development equipment incorporated. Since a common groundrule was applied to all three system alternates (namely, that developed, proven equipment be used), this cost factor reduces to a consideration of what types and what quantities of equipment were used for the three system alternates. Also associated with the quantities is the redundancy level associated with fail-safe or fail-operational/fail-safe. Costs are summarized in Volume III of this document.

The cost of performing the first vertical flight unmanned was determined and may be compared with either a reduced risk to the potential crew or to an increased risk to the vehicle, since no man is onboard to make decisions and to manually override the automatic onboard systems (see Par. 2.3). The cost delta for FVF unmanned was determined but no significant difference in cost was identified among the three system alternates.

The principal factor affecting peak annual funding for a designated set of onboard avionics is the planned phase-in of equipment plus the time spans allotted for DDT&E. Avionics peak annual funding can be reduced by supplying only that equipment needed for each program phase rather than supplying an "all-up" system at the time of first horizontal flight. The latter approach could entail more of a schedule risk than the phased approach.

2.8 CONCLUSIONS

1. A significant reduction in total program cost is achieved through extensive use of developed, available equipment and software, and by use of the designated baseline Mark I Orbiter Avionics System which reduces dependence of the crew and vehicle subsystems on ground support for mission operations and launch. Ground support cost reductions account for major savings to the program.

2. A significant reduction in peak annual funding is achieved by extensive use of developed, available equipment and software and by phasing-in equipment and functional capability as required to support program phases, i.e., horizontal flight test, vertical flight test, and operational phases.

3. Technological risk is reduced by use of existing aircraft equipment and and spacecraft equipment specifically identified in the study. No significant development of equipment is required for the Mark I Avionics System. Environmental protection of most equipment is provided and those equipments exposed to new operating environments will be flight qualified is required. All safety-of-flight functions will be performed using dedicated hardwired equipment configured fail-safe as a minimum.

4. The critical path to first horizontal flight will not be impacted by avionics. Only that equipment required for aircraft flight will be provided initially and such equipment can be available for installation within thirty months after authorization to proceed, allowing another eighteen months until first horizontal flight.

5. Booster/Orbiter avionics commonality is minimal for the case of the interim recoverable ballistic pressure-fed LOX/Propane booster.

6. The selected baseline Mark I Orbiter Avionics System configuration provides the basis for effecting a smooth transition from Mark I to Mark II avionics and vehicle system capability.

7. Selective improvements in performance, equipment quality, and equipment quantity to achieve the Mark II Orbiter Avionics System capability can be made within the study guideline of forty to fifty percent permissible cost growth (nonrecurring) of Mark II over Mark I.

2.9 RECOMMENDATIONS FOR FURTHER STUDY

The completed Alternate Space Shuttle Avionics study has produced a baseline system concept which significantly reduces total program cost and peak annual funding, and reduces technological risk. The principal study objectives have therefore been achieved. However, the short time span allowed for the study precluded detailed study of some aspects of the system.

Recommendations for further study that expand upon the presently defined baseline system are presented below:

- A more detailed study of the application of S-3A avionics system techniques, equipment, and software should be performed. Shuttle avionics equipment interfaces with the data management subsystem, specifically the design of subsystem interface units, should be defined in detail. Redesign and modifications of existing equipment for compatibility with S-3A checkout, fault isolation, and inflight performance monitoring should be analyzed and costed. The extent to which particular S-3A software programs are applicable to the shuttle avionics should be examined; additional software development should be defined and costed in detail. The extent to which existing Lockheed S-3A development facilities (for hardware and software) can be applied to the shuttle avionics development should be determined.
- The fly-by-wire primary flight control system and automatic flight control system (both incorporating stability augmentation) failure modes should be examined in great detail. Mechanizations incorporating separate electrical power sources for each redundant system should be evaluated along with mechanizations employing non-electrical backup mode capability.
- Additional reliability data should be compiled for baseline system equipment, and safety/reliability studies should be performed to evaluate the adequacy of selected equipment in the anticipated operating environment. Computerized optimization studies for each subsystem should be conducted to trade off cost, weight, volume, and component redundancy level according to weighting factors or sensitivity coefficients agreed upon with the NASA.

- All interfaces among avionics subsystems/equipments and between the avionics system and non-avionics subsystems/equipments should be clearly defined. Included are mechanical, electrical, functional, and performance interfaces.
- A thorough packaging and installation study should be completed. Modularity and access for maintenance should be investigated, and impact on the crew station and impact on the environmental control system should be analyzed.
- The baseline avionics system sensitivity to specific performance requirements and to program requirements/constraints should be determined.
- The orbiter/payload interface requirements should be identified for a selected representative group of payloads as agreed upon with the NASA.
- The baseline avionics system definition should be more extensively detailed in all subsystem areas. In addition, redundancy management, automatic configuration control and sequencing, operating modes, and abort with intact vehicle recovery should be examined at the system level.
- The avionics system management plan for design, development, tests, and integration should be prepared. The plan should be keyed to overall program milestones, should identify major avionics tasks, and should state how these tasks will be performed and managed. The experience of Lockheed in aircraft development, spacecraft development, systems management, and avionics integration would be incorporated into the avionics system management plan.

Section 3

COST ANALYSIS SUMMARY

Section 3

COST SUMMARY

Costs were estimated for three Mark I orbiter avionics system configurations:

- Alternate A
- Alternate B
- Alternate C

On the first iteration, all three systems were costed for the Mark I program in terms of orbiter avionics cost and ground-based mission support costs. While Alternate C was found to have the highest on-board avionics cost, it was also found to have the lowest overall cost when the effect of its reduced ground support was taken into account. Alternate C was chosen as the baseline system. Costs for this system were then estimated for Mark II and the initial iteration of this alternate was re-estimated for Mark I. Mark I orbiter avionics costs for all three alternates are summarized in Fig. 3-1.

The final estimates of the baseline system for Mark I are shown in Fig. 3-2, and for Mark II in Fig. 3-3. These are \$323.3 million for Mark I and \$202.6 million for Mark II, for a total program cost of \$525.9 million.

The annual costs for the baseline system are shown in Fig. 3-4.

	BASELINE CONFIGURATION	ALTERNATE A	ALTERNATE B
NONRECURRING	231.2	176.7	210.3
RECURRING	12.8	8.8	8.9
OPERATIONS	81.3	64.8	68.8
TOTALS	325.3	250.3	287.9

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Fig. 3-1 Mark I Orbiter Avionics System Costs Summary

SUBSYSTEM WBS ELEMENT	D&D	UNITS GHT	GRD TEST	FTH (2)	DDT&E	REFURB/ RETROFIT	OPS SPARES
AVIONIC SYS INTEG	2.2	-	-	-	2.2	.7	5.8
GUIDANCE, NAVIGATION & CONTROL	11.8	1.5	4.6	3.8	20.3	2.8	16.1
DATA MANAGEMENT & CO/FI/RM	50.7	1.5	10.0	4.1	65.2	6.4	20.8
DISPLAYS & CONTROLS	13.7	1.5	1.4	1.6	16.8	0.4	5.3
COMMUNICATIONS & NAVAIDS	8.0	1.5	1.8	2.2	12.0	0.4	4.1
INSTRUMENTATION (DEVELOPMENT & OPERATIONAL)	15.4	1.5	2.0	11.9	29.6	2.3	13.1
ELECT POWER DISTRIBUTION & CONTROL	2.9	1.5	0.6	0.5	4.1	-	1.4
SOFTWARE (VEHICLE)	65.8	-	-	-	65.8	-	10.0
ELECTRIC POWER GEN.	8.4	1.5	2.4	2.1	12.9	1.4	3.7
TOTALS					228.9	14.4	80.3

Fig. 3-2 Mark I Orbiter Avionics Costs

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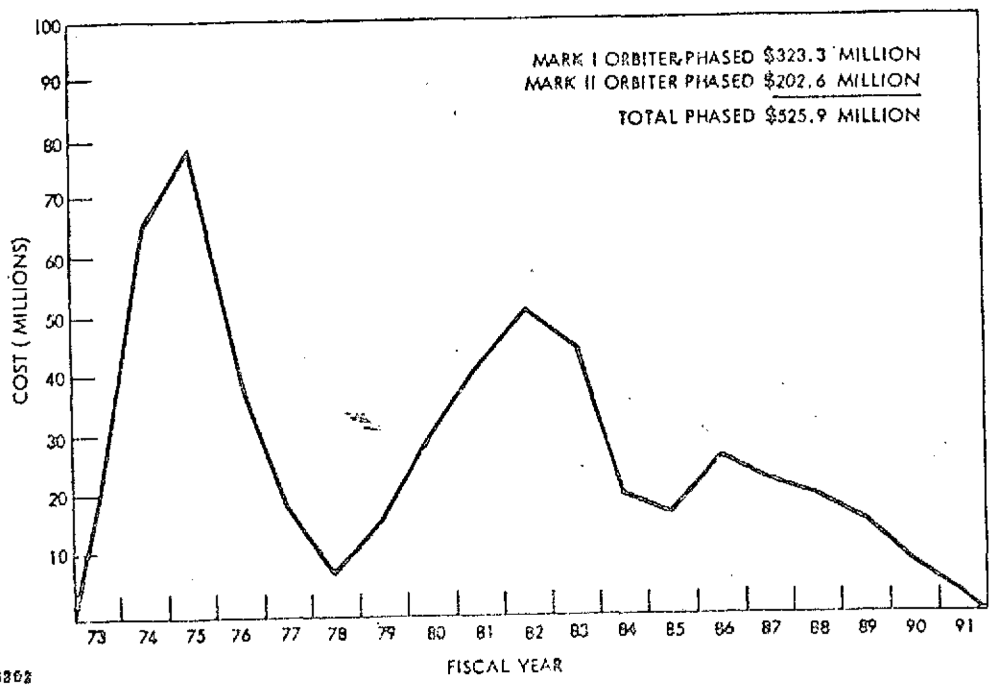
SUBSYSTEM WBS ELEMENT	D&D	UNITS GHT	GRD TEST	FTH (2)	DDT&E	TFU	PROD (3)	REFURB/ RETROFIT (2)	OPS SPARES
AVIONIC SYS INTEG	2.2	-	0	-	2.2	-	-	.4	2.8
GUIDANCE, NAVIGATION AND CONTROL	4.6	1.5	1.3	-	6.0	3.5	10.5	1.0	18.3
DATA MANAGEMENT AND CO/FI/RM	8.2	-	-	-	8.4	3.9	11.7	1.8	30.6
DISPLAYS AND CONTROLS	6.3	1.5	0.7	-	7.0	0.9	2.7	0.4	12.8
COMMUNICATIONS AND NAVAIDS	2.1	1.5	0.1	-	2.2	0.8	2.4	0.1	3.2
INSTRUMENTATION (DEVELOPMENT AND OPERATIONAL)	5.5	1.5	1.4	-	7.0	3.6	10.5	3.7	4.5
ELECT POWER DISTRIBUTION AND CONTROL	1.3	-	-	-	1.3	0.3	0.9	0.1	1.2
SOFTWARE (VEHICLE)	15.0	-	-	-	15.0	-	-	-	8.0
ELECTRIC POWER GEN	4.3	1.5	1.9	-	6.2	2.0	6.0	1.2	3.5
TOTALS					55.3		45.0	17.4	84.9

DO6306

Fig. 3-3 Mark II Orbiter Avionics Costs



DO6302

Fig. 3-4 Mark I and Mark II Avionics Program Annual Costs

Appendix A
REQUIREMENTS ANALYSIS

Attached is the compilation of working materials used for identifying subsystem functional requirements, equipment needs, and minimum redundancy levels. The lists are grouped into five categories:

1. Functions vs Mission Phase for Aircraft
2. Functions vs Mission Phase for Spacecraft
3. Displays and Controls Analysis
4. Aircraft Avionics Equipment Required for Crew Safety vs Mission Phase
5. Spacecraft Avionics Equipment Required for Crew Safety vs Mission Phase

The crew safety sheets were prepared individually for aircraft and spacecraft operational periods to reduce duplication of effort. No sheets were prepared for data management, instrumentation, and displays (programmable), since these subsystems have been established as noncritical to flight safety. The safety-of-flight displays and controls are hard-wired and shared for spacecraft and aircraft functions.

FUNCTIONS VS MISSION PHASE FOR AIRCRAFT

(Pages A-3 through A-7)

AIRCRAFT AVIONICS

VEHICLE ORBITER

SUBSYSTEM PRIMARY FLIGHT CONTROLS

MISSION PHASE

FUNCTIONS	HYPERSONIC FLIGHT	ABES AIR START	DESCENT	APPROACH	LANDING	TAXI	SHUTDOWN AND SAFING	ABES GROUND START	TAKEOFF	CLIMB	HORIZONTAL FLIGHT	REFURBISHMENT	VEHICLE STORAGE
Heading Control	3	1	1	1	1	3			1	1	1		
Attitude Control	3	1	1	1	1				1	1	1		
Yaw Damping	3	1	1	1	1			3	1	1	1		
Pitch Augmentation	3	3	3	3	1			3	1	3	3		
Roll Augmentation	3	1	1	1	1			3	1	1	1		
Speed Stability													

BLANK - NOT REQUIRED

1. CREW/VEHICLE SAFETY
2. MISSION SUCCESS
3. NONCRITICAL

AIRCRAFT AVIONICS

VEHICLE ORBITER

SUBSYSTEM SECONDARY FLIGHT CONTROLS

MISSION PHASE

FUNCTIONS	HYPERSONIC FLIGHT	ABES AIR START	DESCENT	APPROACH	LANDING	TAXI	SHUTDOWN AND SAFING	ABES GROUND START	TAKEOFF	CLIMB	HORIZONTAL FLIGHT	REFURBISHMENT	VEHICLE STORAGE
Pitch Trim	2	2	2	2	3				3	2	2		
Roll Trim	3	3	3	3	3				3	3	3		
Rudder Trim	3	3	3	3	3				3	3	3		
Air Braking			3	3	3								

BLANK - NOT REQUIRED

1. CREW/VEHICLE SAFETY
2. MISSION SUCCESS
3. NONCRITICAL

AIRCRAFT AVIONICS

VEHICLE ORBITER

SUBSYSTEM AUTO PILOT

MISSION PHASE

FUNCTIONS	HYPERSONIC FLIGHT	ABES AIR START	DESCENT	APPROACH	LANDING	TAXI	SHUTDOWN AND SAFING	ABES GROUND START	TAKEOFF	CLIMB	HORIZONTAL FLIGHT	REFURBISHMENT	VEHICLE STORAGE
Automatic Attitude Control		3	3							3	3		
Automatic Navigation		3	3	3						3	3		
Automatic Landing				2	2	3							
Unmanned/Non-Responsive Crew													
Remote Attitude Control	1	1	1	1	1				1	1	1		
Automatic Throttle Control					3				3	3	3		
Unmanned/Non-Responsive Crew													
Automatic Throttle Control			1	1	1	1			1	1	1		

BLANK - NOT REQUIRED
1. CREW/VEHICLE SAFETY
2. MISSION SUCCESS
3. NONCRITICAL

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AIRCRAFT AVIONICS

VEHICLE ORBITER

SUBSYSTEM NAVIGATION

MISSION·PHASE

FUNCTIONS	HYPERSONIC FLIGHT	ABES AIR START	DESCENT	APPROACH	LANDING	TAXI	SHUTDOWN AND SAFING	ABES GROUND START	TAKEOFF	CLIMB	HORIZONTAL FLIGHT	REFURBISHMENT	VEHICLE STORAGE
Compute & Display Course			3	3					3	3	2		
Display Range			3	3					3	3	2		
Display Course Error			3	3					3	3	2		
Display Aircraft Hdg	1	1	1	1	1				1	1	1		
Display ILS Beam Deviation Raw				2	2								
Display Flight Director Cmds			3	3	3				3	3	3		

BLANK - NOT REQUIRED
1. CREW/VEHICLE SAFETY
2. MISSION SUCCESS
3. NONCRITICAL

AIRCRAFT AVIONICS

VEHICLE ORBITER

SUBSYSTEM COMMUNICATIONS

MISSION PHASE

FUNCTIONS	HYPERSONIC FLIGHT	ABES AIR START	DESCENT	APPROACH	LANDING	TAXI	SHUTDOWN AND SAFING	ABES GROUND START	TAKEOFF	CLIMB	HORIZONTAL FLIGHT	REFURBISHMENT	VEHICLE STORAGE
Air Ground Voice	2	2	2	2	2	2	3	3	2	2	2		
Intercommunications	2	1	1	2	2	3	3	2	2	2	2		

BLANK - NOT REQUIRED
1. CREW/VEHICLE SAFETY
2. MISSION SUCCESS
3. NONCRITICAL

FUNCTIONS VS MISSION PHASE FOR SPACECRAFT

(Pages A-9 through A-23)

SPACECRAFT AVIONICS

VEHICLE ORBITER

SUBSYSTEM GUIDANCE, NAVIGATION & CONTROLS

MISSION PHASE

FUNCTION	PRELAUNCH	LAUNCH AND ASCENT	ORBIT INSERTION	RENDEZVOUS	DOCKING/UNDOCKING	ORBIT STAY	DEORBIT	REENTRY	REFURBISHMENT	STORAGE		
Align Platform-Ground	1											
Align Platform-Flight				2	3	2	1					
Local Vertical and Direction												
3-Points-Stars												
3-Points-Earth												
Measure Linear Acceleration		1	1	2	2	2	1	1				
Measure Attitude	1	2	1	2	1	2	1	1				
Measure Angular Body Rates		1	1	2	2	2	1	1				
Integrate Acceleration Information		1	1	2	2	2	1	1				
Integrate Velocity Information		1	1	2	2	2	1	1				
Compute Gravity Effects	1	1	1					1				
Compute Current Inertial Position	1	1	1	2	2	2	1	1				

BLANK - NOT REQUIRED
1. CREW/VEHICLE SAFETY
2. MISSION SUCCESS
3. NONCRITICAL

SPACECRAFT AVIONICS

VEHICLE ORBITER

SUBSYSTEM GUIDANCE, NAVIGATION & CONTROLS

MISSION PHASE

FUNCTION	PRELAUNCH	LAUNCH AND ASCENT	ORBIT INSERTION	RENDEZVOUS	DOCKING/UNDOCKING	ORBIT STAY	DEORBIT	REENTRY	REFURBISHMENT	STORAGE		
Compute Current Inertial Velocity		1	1	2	2	2	1	1				
Update State Vector				2	3	2	1					
Compute Abort Targeting	1	1	1	1	1	1						
Compute Orbit Injection Targeting			1									
Provide Orbiter Engine on Discrete			1									
Provide Orbiter Engine Off Discrete			1									
Provide OMS Engine on Discrete				2	2	2	1					
Provide OMS Engine Off Discrete				1	1	1	1					
Compute Rendezvous Targeting				2		1						
Compute Docking Targeting					2							

BLANK - NOT REQUIRED
1. CREW/VEHICLE SAFETY
2. MISSION SUCCESS
3. NONCRITICAL

SPACECRAFT AVIONICS

VEHICLE ORBITER

SUBSYSTEM GUIDANCE, NAVIGATION & CONTROLS

MISSION PHASE

BLANK - NOT REQUIRED
1. CREW/VEHICLE SAFETY
2. MISSION SUCCESS
3. NONCRITICAL

FUNCTION	PRELAUNCH	LAUNCH AND ASCENT	ORBIT INSERTION	RENDEZVOUS	DOCKING/UNDOCKING	ORBIT STAY	DEORBIT	REENTRY	REFURBISHMENT	STORAGE		
Compute Undocking Targeting					2							
Compute Orbit Ephemeris Change Targeting						2						
Compute Deorbit Targeting							1					
Compute Reentry Targeting								1				
Compute Supersonic Glide Targeting								1				
Provide Inner Loop Compensation		1	1	2	2	2	1	1				
Stabilize Body Bending Modes		1	1					1				
Suppress Slosh Modes		1	1									
Provide Load Alleviation		1						1				
Limit Angular Rates		1	1	2	2	2	2	1				
Limit Axial Acceleration		2	1									
Steer Ascent Trajectory		2	2									

SPACECRAFT AVIONICS

VEHICLE ORBITER

SUBSYSTEM GUIDANCE, NAVIGATION & CONTROLS

Page 4 of 4 Pages

MISSION PHASE

FUNCTION	PRELAUNCH	LAUNCH AND ASCENT	ORBIT INSERTION	RENDEZVOUS	DOCKING/UNDOCKING	ORBIT STAY	DEORBIT	REENTRY	REFURBISHMENT	STORAGE		
Steer Orbital Maneuvers				2			1					
Steer Orbital Attitude					2	2	1					
Steer Docking Maneuvers					2							
Steer Reentry Profile								1				
Provide Vehicle Trim (CG Tracking)		2	2					1				
Provide Mode Selection	2	1	1	2	2	2	1	1				

BLANK - NOT REQUIRED
1. CREW/VEHICLE SAFETY
2. MISSION SUCCESS
3. NONCRITICAL

SPACECRAFT AVIONICS

VEHICLE ORBITER

SUBSYSTEM COMMUNICATIONS

MISSION PHASE

FUNCTION	PRELAUNCH	LAUNCH AND ASCENT	ORBIT INSERTION	RENDEZVOUS	DOCKING/UNDOCKING	ORBIT STAY	DEORBIT	REENTRY	REFURBISHMENT	STORAGE		
Provide Two-way Voice Link (Hard-Line) ORB/GND	2											
Provide Two-way Voice Link (RF) ORB/GND		2	2	2	2	2	2	2				
Provide Two-way Data Link (Hard-Line) ORB/GND	2											
Provide Two-way Data Link (RF) ORB/GND		1	1	2	2	2	1	1				
Provide Two-Way Voice Link (Hard-Line) ORB/STATION					3							
Provide Two-way Voice Link (RF) ORB/STAT.				2	2							
Provide Two-way Data Link (Hard-Line) ORB/STAT.					2							
Provide Two-way Data Link (RF) ORB/STAT.				2	2							
Provide Two-way Data Link (Hard-Line) ORB/Payload	2					2						
Provide Two-way Voice Link (Hard-Line) ORB/Payload	3					3						
Provide Two-way Data Link (Hard-Line) ORB/BSTR	2											
Provide Two-way Voice Link (Hard-Line) ORB Crew	2	2	2	3	3	3	3	2				

BLANK - NOT REQUIRED
1. CREW/VEHICLE SAFETY
2. MISSION SUCCESS
3. NONCRITICAL

SPACECRAFT AVIONICS

VEHICLE ORBITER

SUBSYSTEM ELECTRICAL POWER

MISSION PHASE

FUNCTION	PRELAUNCH	LAUNCH AND ASCENT	ORBIT INSERTION	RENDEZVOUS	DOCKING/UNDOCKING	ORBIT STAY	DEORBIT	REENTRY	REFURBISHMENT	STORAGE		
Fuel Cells Checkout and Start-Up	2					2			2			
Heat Removal	2	1	1	1	1	1	1	1	2			
Water Removal	2	1	1	1	1	1	1	1	2			
Control Operation	2	1	1	1	1	1	1	1	2			
Standby						1			2			
Shutdown						2			2			
Purge	2	1	1	1	1	1	1	1	2			

BLANK - NOT REQUIRED
1. CREW/VEHICLE SAFETY
2. MISSION SUCCESS
3. NONCRITICAL

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VEHICLE ORBITER

SUBSYSTEM ELECTRICAL POWER

MISSION PHASE

FUNCTION	PRELAUNCH	LAUNCH AND ASCENT	ORBIT INSERTION	RENDEZVOUS	DOCKING/UNDOCKING	ORBIT STAY	DEORBIT	REENTRY	REFURBISHMENT	STORAGE		
Fuel Cell Reactant Supply Fill O_2 Tanks (2)	2								2			
Fill H_2 Tanks (2)	2								2			
Expel O_2	2	1	1	1	1	1	1	1	2			
Expel H_2	2	1	1	1	1	1	1	1	2			
Condition O_2	2	1	1	1	1	1	1	1	2			
Condition H_2	2	1	1	1	1	1	1	1	2			
Checkout Storage Sys	2								2			

BLANK - NOT REQUIRED
1. CREW/VEHICLE SAFETY
2. MISSION SUCCESS
3. NONCRITICAL

SPACECRAFT AVIONICS

VEHICLE ORBITER

SUBSYSTEM ELECTRICAL POWER

MISSION PHASE

FUNCTION	PRELAUNCH	LAUNCH AND ASCENT	ORBIT INSERTION	RENDEZVOUS	DOCKING/UNDOCKING	ORBIT STAY	DEORBIT	REENTRY	REFURBISHMENT	STORAGE		
Batteries												
F. C. Emergency Start	2					1			2			
Emergency Power for Avionics												
Static Inverters												
Invert DC Power to 3 Phase AC	2	1	1	1	1	1	1	1				
Generators (AC)												
Provide 3 Phase AC Power	2	1	1					1	2			
Decouple from APU	2	2	2					2	2			
Generator Control Units												
Regulate Voltage	2	1	1					1	2			
Overload Protection	2	1	1					1	2			
Overspped Protection	2	1	1					1	2			
Unbalanced Phase Current Protection	2	1	1					1	2			

BLANK - NOT REQUIRED
1. CREW/VEHICLE SAFETY
2. MISSION SUCCESS
3. NONCRITICAL

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SPACECRAFT AVIONICS

VEHICLE ORBITER

SUBSYSTEM ELECTRICAL POWER

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MISSION PHASE

FUNCTION	PRELAUNCH	LAUNCH AND ASCENT	ORBIT INSERTION	RENDEZVOUS	DOCKING/UNDOCKING	ORBIT STAY	DEORBIT	REENTRY	REFURBISHMENT	STORAGE		
Unbalanced Phase Voltage Protection	2	1	1					1	2			
Bearing Failure Drive Decouple	2	2	2					2	2			
Transformer Rectifiers												
Convert AC Power to DC Power									2			
Electric Power System Control												
Circuit Breaker Control	2	1	1	1	1	1	1	1	2			
Configuration Control	2	1	1	1	1	1	1	1	2			
Component Control	2	1	1	1	1	1	1	1	2			
DC Power and Bus Control												
F. C. Reverse Current Protection	2	1	1	1	1	1	1	1	2			
Fuel Cell Bus Connection	2	1	1	1	1	1	1	1	2			
T-R Unit Bus Connection	2	1	1						2			
Bus Fault Protection	2	1	1	1	1	1	1	1	2			

BLANK - NOT REQUIRED
1. CREW/VEHICLE SAFETY
2. MISSION SUCCESS
3. NONCRITICAL

SPACECRAFT AVIONICS

VEHICLE ORBITER

SUBSYSTEM ELECTRICAL POWER

MISSION PHASE

FUNCTION	PRELAUNCH	LAUNCH AND ASCENT	ORBIT INSERTION	RENDEZVOUS	DOCKING/UNDOCKING	ORBIT STAY	DEORBIT	REENTRY	REFURBISHMENT	STORAGE		
AC Power and Bus Control												
Inverter Bus Connection	2	1	1	1	1	1	1	1	2			
Bus Fault Protection	2	1	1	1	1	1	1	1	2			
Ground Power Interface												
GRD DC Power Connection	2								2			
GRD AC Power Connection	2								2			
AC Malfunction Protection	2								2			
Distribution Units												
House Circuit Breakers												
Provide Conditioning Equipment												
[illegible] Power	2	1	1	1	1	1	1	1	2			
[illegible]												

BLANK - NOT REQUIRED
1. CREW/VEHICLE SAFETY
2. MISSION SUCCESS
3. NONCRITICAL

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SPACECRAFT AVIONICS

VEHICLE ORBITER

SUBSYSTEM DATA MANAGEMENT

MISSION PHASE

FUNCTION	PRELAUNCH	LAUNCH AND ASCENT	ORBIT INSERTION	RENDEZVOUS	DOCKING/UNDOCKING	ORBIT STAY	DEORBIT	REENTRY	REFURBISHMENT	STORAGE		
On-Board Checkout	2	3	3	3	3	3	3	3	2			
Mode Control	3	2	2	2	2	2	2	2	3			
Data Bus Control	2	2	2	2	2	2	2	2	2			
Display Control	2	2	2	2	2	2	2	2	2			
Disposables Management	3	2	2	2	2	2	2	2	3			
Communications Control	3	2	2	2	2	2	2	2	3			
Electrical Power Control	2	2	2	2	2	2	2	2	2			
Propulsion Control	3	2	2	2	2	3	2	2	3			
Abort Warning	2	1	2	2	1	3	1	1	3			
Checkout												
Post Flight Safe												
Stored Data		2	2	2	2	2	2	2	3			
Purge & Cool												

BLANK - NOT REQUIRED
1. CREW/VEHICLE SAFETY
2. MISSION SUCCESS
3. NONCRITICAL

SPACECRAFT AVIONICS

VEHICLE ORBITER

SUBSYSTEM DATA MANAGEMENT

MISSION PHASE

FUNCTION	PRELAUNCH	LAUNCH AND ASCENT	ORBIT INSERTION	RENDEZVOUS	DOCKING/UNDOCKING	ORBIT STAY	DEORBIT	REENTRY	REFURBISHMENT	STORAGE		
Pre-Maintenance									3			
Stored Data Processing									3			
Unscheduled Maintenance Definition									3			
Work Plan Preparation									3			
Repair/Maintenance Verification									3			
Payload/Vehicle Integration	1	1	2	2	2	1			3			
Post Maintenance Verification									3			
Pre-Mate Verification										3		
Orbiter/Tank Verification	1											
Orbiter/LUT Verification	1											
Integrated System Verification	2											
Pre-Tanking Verification	1											
[illegible]	1											

BLANK - NOT REQUIRED
1. CREW/VEHICLE SAFETY
2. MISSION SUCCESS
3. NONCRITICAL

SPACECRAFT AVIONICS

VEHICLE ORBITER

SUBSYSTEM DATA MANAGEMENT

MISSION PHASE

FUNCTION	PRELAUNCH	LAUNCH AND ASCENT	ORBIT INSERTION	RENDEZVOUS	DOCKING/UNDOCKING	ORBIT STAY	DEORBIT	REENTRY	REFURBISHMENT	STORAGE		
Orbital Config. Verification			1									
Mission Peculiar Verification				2	2	2						
Pre-Reentry Verification						1						
ABES Pre-Start Verification												
Atmospheric Flight Verification												
Final Approach Verification												
Fault Isolation												
Unscheduled Maintenance	1								3			
Repair Maintenance	1								3			
Payload Vehicle Integration									3			
Post Maintenance									3			
Pre-Mate										3		
Orbiter/Tank Mate	1											

BLANK - NOT REQUIRED
1. CREW/VEHICLE SAFETY
2. MISSION SUCCESS
3. NONCRITICAL

SPACECRAFT AVIONICS

VEHICLE ORBITER

SUBSYSTEM DATA MANAGEMENT

MISSION PHASE

FUNCTION	PRELAUNCH	LAUNCH AND ASCENT	ORBIT INSERTION	RENDEZVOUS	DOCKING/UNDOCKING	ORBIT STAY	DEORBIT	REENTRY	REFURBISHMENT	STORAGE		
Orbiter/LUT Mate	1											
Integrated System	1											
Pre-Tanking	1											
Pre-Liftoff	1											
S/C Hardware In Flight		1	1	1	1	1	1	1				
A/C Hardware In Flight												
Redundancy Management												
S/C Reconfiguration		1	1	1	1	1	1	1	3			
A/C Reconfiguration								1	3			
S/C Status	1	1	1	1	1	1	1	1	3			
A/C Status	1					1		1	3			

BLANK - NOT REQUIRED
1. CREW/VEHICLE SAFETY
2. MISSION SUCCESS
3. NONCRITICAL

SPACECRAFT AVIONICS

VEHICLE ORBITER

SUBSYSTEM INSTRUMENTATION

MISSION PHASE

FUNCTION	PRELAUNCH	LAUNCH AND ASCENT	ORBIT INSERTION	RENDEZVOUS	DOCKING/UNDOCKING	ORBIT STAY	DEORBIT	REENTRY	REFURBISHMENT	STORAGE		
Data Acquisition	3	3	3	3	3	3	3	3	3			
Signal Conditioning	3	3	3	3	3	3	3	3	3			
Routing	3	3	3	3	3	3	3	3	3			
Formatting	3	3	3	3	3	3	3	3	3			
Analog Multiplexers	3	3	3	3	3	3	3	3	3			
Digital Mux	3	3	3	3	3	3	3	3	3			
Time Code Generation	3	3	3	3	3	3	3	3	3			
Storage	3	3	3	3	3	3	3	3	3			
Flight Recorder	3	3	3	3	3	3	3	3	3			
Data-Digital, Voice, Analog	3	3	3	3	3	3	3	3	3			
Film Cameras	3	3	3	3	3	3	3	3	3			
Redundant Crew Displays	3	3	3	3	3	3	3	3	3			
Calibration	3	3	3	3	3	3	3	3	3			

BLANK - NOT REQUIRED

1. CREW/VEHICLE SAFETY
2. MISSION SUCCESS
3. NONCRITICAL

DISPLAYS AND CONTROLS ANALYSIS

(Pages A-25 through A-35)

ORBITER CONTROLS & DISPLAYS: MARK I AVIONICS PRELIMINARY

25-[illegible]-71
28-10-71
21-[illegible]-71
J.H. DUDDY 12-[illegible]-71

ORBITER SUBSYSTEM	FUNCTION CONTROLLED OR PARAMETER CONDITION DISPLAYED	SPACE FLIGHT REGIME	ATMOSPHERIC FLIGHT REGIME
GUIDANCE & NAVIGATION	DIGITAL READOUT OF ELAPSED TIME: HRS/MINS/SECS TIMER CONTROL FUNCTIONS: START - STOP - RESET HOURS SLEW CONTROL FUNCTIONS: TENS - DISABLE - UNITS MINUTES SLEW CONTROL FUNCTIONS: TENS - DISABLED - UNITS SECONDS SLEW CONTROL FUNCTIONS: TENS - DISABLED - UNITS	MISSION TIMER DISPLAY (LM) TIMER CONTROL SWITCH (LM) SLEW CONTROL (HOURS) SWITCH (LM) SLEW CONTROL (MINUTES) SWITCH (LM) SLEW CONTROL (SECONDS) SWITCH (LM)	IDENTICAL REQUIREMENTS
	DIGITAL READOUT OF EVENT ELAPSED TIME: MINS/SECS TIMER CONTROL FUNCTIONS: START - STOP TIMER COUNT - RESET CONTROL FUNCTIONS: RESET - UP - DOWN MINUTES SLEW CONTROL FUNCTIONS: TENS - DISABLED - UNITS SECONDS SLEW CONTROL FUNCTIONS: TENS - DISABLED - UNITS	EVENT TIMER DISPLAY (LM) TIMER CONTROL SWITCH (LM) TIMER RESET/COUNT SWITCH (LM) SLEW CONTROL (MINUTES) SWITCH (LM) SLEW CONTROL (SECONDS) SWITCH (LM)	
	INERTIALLY-DERIVED ALTITUDE & RATE OF CHANGE IN ALTITUDE RADAR ALTITUDE & RATE OF CHANGE IN ALTITUDE MODE SELECTION FUNCTIONS: IMU - RADAR	ALTIMETER/ALTITUDE RATE DISPLAY (LM) MODE SELECTOR SWITCH	BACKUP FOR AIRCRAFT I/O BELOW
	SERVO-CONTROLLED BAROMETRIC ALTITUDE MANUAL ALTITUDE SETTING CONTROL RATE OF CLIMB, DESCENT		BAROMETRIC ALTIMETER W/ INTEGRAL SETTING CONTROL (L-1011) VERTICAL SPEED INDICATOR DISPLAYS COMBINED IN VERT. SCALE METER
	ALTITUDE ABOVE TERRAIN: 2500 TO TOUCHDOWN DIGITAL AND POINTER FOR DECISION HEIGHT: 500 TO 0 MANUAL DECISION HEIGHT SETTING CONTROL AUDIO TONE 50 FT. ABOVE DECISION HEIGHT RADAR ALTIMETER ANTENNA DEPLOYMENT CONTROL: DEPLOY - [illegible] - RETRACT		RADAR ALTIMETER W/ DECISION HEIGHT DISPLAY (L-1011) DECISION HEIGHT INPUT CONTROL AURAL DISPLAY OF DH + 50 TO DH RADAR ALT DEPLOYMENT CONTROL COMBINED IN ONE METER
	TOTAL VEHICLE ATTITUDE: ROLL/PITCH/YAW ATTITUDE RATES: ROLL/PITCH/YAW ATTITUDE ERRORS: ROLL/PITCH/YAW RENDEZVOUS RADAR SHAFT & TRUNNION ANGLES (PITCH & YAW ERROR DISPLAYS CAGE ATTITUDE BALL INTERNAL ILLUMINATION OF FDAI: OFF - INCREASE BRIGHTNESS TEST RATE INDICATORS: SELF-TEST - OFF ALIGN FDAI WITH ATTITUDE GYRO COUPLING UNIT (AGCU) SET ATTITUDE & COMPARE WITH AGCU TO DETERMINE ERRORS: ON - OFF INSERT ROLL COMMAND DISPLAY OF ROLL VERNIER SETTING INSERT PITCH COMMAND DISPLAY OF PITCH VERNIER SETTING INSERT YAW COMMAND DISPLAY OF YAW VERNIER SETTING SELECT RATE SCALE FOR FDAI: 5°/SEC - 25°/SEC RATE/ERROR MONITORING SELECTION SHAFT/TRUNNION ANGLE RANGE SCALE [illegible] (RNDZ RADAR) TURN RATE DURING ATMOSPHERIC FLIGHT SIDE SLIP OR LATERAL ACCELERATION	FLIGHT DIRECTOR ATTITUDE INDICATOR (LM/CM) CAGE ATTITUDE BALL SWITCH (LM/CM) BRIGHTNESS CONTROL (LM/CM) SELF-TEST SWITCH (LM/CM) ALIGN SWITCH (LM/CM) ATTITUDE SET SWITCH (LM/CM) VERNIER ROLL CONTROL (CM) ROLL INDICATOR (CM) VERNIER PITCH CONTROL (CM) PITCH INDICATOR (CM) VERNIER YAW CONTROL (CM) YAW INDICATOR (CM) RATE SCALE SWITCH (LM) RATE/ERROR MONITORING SWITCH (LM) SHAFT/TRUNNION SWITCH (LM)	FDAI SUITABLE FOR AERODYNAMIC FLIGHT WITH MAJOR MODS. AIRCRAFT ADI SUITABLE FOR SPACE FLIGHT REGIME WITH FEWER MODS. (L-1011) CROSS-POINTER & G METER PROVIDE REQUIRED DATA

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ORBITER CONTROLS & DISPLAYS: MARK I AVIONICS/PRELIMINARY

ORBITER SUBSYSTEM	FUNCTION CONTROLLED OR PARAMETER/CONDITION DISPLAYED	SPACE FLIGHT REGIME	ATMOSPHERIC FLIGHT REGIME
GUIDANCE & NAVIGATION (CONTINUED)	MANUAL CONTROL OF PITCH/ROLL/YAW MANUAL SETTING OF INTERCOM & TELECOMMUNICATIONS (IF INTERCOM USED) DISABLE NORMAL OPERATION OF ACA: ENABLE-DISABLE DISABLE YAW: ENABLE - DISABLE (FOR ATMOSPHERIC FLIGHT)	ATTITUDE CONTROLLER ASSEMBLY (LM) PUSH-TO-TALK SWITCH ON ACA (LM) ACA DISABLE SWITCHES	PITCH AND ROLL CONTROL ONLY IDENTICAL FUNCTION YAW DISABLE SWITCH ENABLE RUDDER PEDALS
	COMPUTER HAS FAILED MALFUNCTION HAS BEEN IDENTIFIED DURING SELF-CHECKING ALLOW OR INHIBIT OF TELEMETERED DATA TO COMPUTER FROM GROUND: ACCEPT - BLOCK ACTIVITY COMPUTER IS ENGAGED IN: UPTEL - COMP DIGITAL DISPLAY OF PROGRAM IN PROGRESS VERB CODE SELECTED DIGITAL DISPLAY DIGITAL DISPLAY OF NOUN CODE SELECTED " " " CONTENTS OF 1, 2 OR 3 REGISTERS/MEM. LOCATIONS BRIGHTNESS OF DIGITAL DISPLAYS DISABLE DSKY OPERATION OF COMPUTER KEYBOARD (DSKY): CLEAR - VERB - NOUN - ENTER - +/- - 0 TO 9	PRIMARY GUIDANCE & NAVIGATION COMPUTER, ... INCLUDING FOLLOWING C/D: COMP FAIL LIGHT CHECK FAIL LIGHT UPTEL SWITCH ACTIVITY LIGHTS: UPTEL & COMP PROGRAM DISPLAY VERB INDICATOR NOUN INDICATOR REGISTER 1, 2 OR 3 DISPLAY(S) BRIGHTNESS CONTROL KEY REL PUSHBUTTON KEYBOARD	
	DISPLAY OF ACCELERATION IN 2 AXES: $\pm G_z$ & $\pm G_x$	G-METER AND/OR THRUST/WEIGHT INDICATOR ...	
	DISTANCE TO TARGET & RATE OF CLOSURE FROM RENDEZVOUS RADAR	RANGE/RANGE RATE DISPLAY (LM)	
	FORWARD AND LATERAL VELOCITIES OR LATERAL ONLY, INERTIAL REF. LINE-OF-SIGHT TO DOCKING TARGET: AZIMUTH - ELEVATION SELECT MODE OF DISPLAY: VELOCITIES - LOS POWER TO X-POINTER FAILED	CROSS POINTER INDICATOR (LM) MODE SEL SWITCH (X-POINTER) (LM) WARNING LIGHT (X-POINTER) (LM)	BACKUP FWD. VELOCITY INDICATION FOR TRUE AIRSPEED DISPLAY/MACH NO.
	ABORT COMMANDED BY RANGE SAFETY/MISSION CONTROL/EMERGENCY DETECTION SYSTEM/GUIDANCE COMPUTER INITIATE ABORT SEQUENCE AS A FUNCTION OF MISSION PHASE	ABORT LIGHT (CM) ABORT/EMER. STAGE PUSHBUTTON (LM)	
	CAUTION/WARNING INDICATIONS TO BE DETERMINED	G & N STATUS LIGHTS (LM/CM)	C&W OVERLAPPING REQUIREMENTS
	HEADING - COMMAND AND ACTUAL DEVIATION COURSE - DIGITAL DISTANCE - DIGITAL FROM DME MALFUNCTION OR OFF INDICIUM		HORIZONTAL SITUATION INDICATOR (HSI) COMPASS WITH ±30 DEG MARKERS DIGITAL DISPLAYS OF COURSE, RANGE
	TACAN CONTROLS & DISPLAYS TACAN AUDIO CONTROLS TACAN ANTENNA DEPLOYMENT CONTROL: DEPLOY - CENTER - RETRACT		TACTICAL AIR NAVIGATION SYSTEM: TACAN CONTROL PANEL TACAN AUDIO SELECT PANEL TACAN ANTENNA DEPLOYMENT CONTROL

ORBITER CONTROLS & DISPLAYS : MARK I AVIONICS / PRELIMINARY

ORBITER SUBSYSTEM	FUNCTION CONTROLLED OR PARAMETER/CONDITION DISPLAYED	SPACE FLIGHT REGIME	ATMOSPHERIC FLIGHT REGIME
FLIGHT CONTROL & STABILIZATION	(G & N FUNCTIONS OVERLAP FC & STAB AND ARE NOT DUPLICATED HERE.)	(G & N C/D ARE ALSO APPLICABLE IN PART TO FC & STAB.)	
	COMMAND VEHICLE TRANSLATIONS VIA THE REACTION CONTROL JETS	THRUST/TRANSLATION CONTROLLER (LM) AND RELATED C/D BELOW:	DISABLED FOR ATMOSPHERIC FLIGHT
	THROTTLE PRIMARY PROPULSION ENGINES (J2/J2S AND LATER THE OMS ENGINES)		
	SELECT X-AXIS/Y-AXIS TRANSLATION	AXIS SELECTOR SWITCH (LM)	
	CONTROL THROTTLE SENSITIVITY	THROTTLE FRICTION CONTROLLER (LM)	
	DISABLE TTCA FOR ATMOSPHERIC FLIGHT REGIME : SPACE – ATMOS	FLIGHT REGIME SEL. SWITCH	
	% THRUST COMMANDED BY GUIDANCE COMPUTER OR MANUALLY BY TTCA : MAIN – OMS	THRUST INDICATOR (LM)	
	% THRUST BEING DELIVERED BY ENGINES : MAIN – OMS		
	SELECT THROTTLE CONTROL MODE : MANUAL – COMPUTER	THROTTLE CONT SELECTOR SW. (LM)	
	INDICATE POWER FAILURE TO DUAL-NEEDLE INDICATOR	WARNING LIGHT (LM)	
	MANUAL INITIATION OF 2 OMS ENGINES (CM ASCENT TYPE)	THRUST 'ON' PUSHBUTTON (CM)	
	MANUAL CAPABILITY TO INITIATE ULLAGE MANEUVER WITH OMS	DIRECT ULLAGE PUSHBUTTON (CM)	
	THRUST CONTROL MODE SELECTION : AUTO – TERMINATE – MANUAL	THRUST MODE SEL. SW. (CM)	
	INDICATE ΔV REMAINING DIGITAL DISPLAY FOR OMS	ΔV REMAINING INDICATOR (CM)	
	MANUAL SETTING UP ΔV INTO DISPLAY : INCREASE – OFF – DECREASE	ΔV SET SELECTOR SW. (CM)	
	" " " MAIN ENGINES GIMBALS ALIGNMENT IN YAW : J2S – OMS	GIMBAL POSITIONS INDICATOR (CM)	
	" " " " " " " " PITCH : "	ERROR INDEXES ON DISPLAYS (CM)	
	INDICATIONS OF GIMBAL POSITIONS IN YAW AND PITCH : "	YAW THUMBWHEEL (CM)	
	ERRORS IN THRUST VECTOR ALIGNMENT IN PITCH AND YAW : "	PITCH THUMBWHEEL (CM)	
	SELECTION OF GIMBAL DRIVE MODE : ENABLE DRIVE – DISABLE DRIVE	ENGINE GIMBAL SWITCH (LM)	
	" " PROPULSION SYSTEM : J2S – OMS	PROP. SYS. SEL. SWITCH	
		STABILIZATION S/S C/D :	
	SELECTION OF RCS CONTROL MODE : MANUAL – COMPUTER	RCS MODE SELECTOR SW. (CM)	
	ENABLE LIMIT CYCLE CIRCUIT TO HOLD ATTITUDE : LIMIT CYCLE – OFF	LIMIT CYCLE SWITCH (CM)	
	SELECTION OF DEADBAND WIDTH IN ATTITUDE HOLD MODE : MIN – MAX	ATT DEADBAND SWITCH (CM)	
	SELECTION OF ATTITUDE IN PITCH AND YAW FOR ENTERING ATMOS : ENTRY – OFF	ENTRY SWITCH (CM)	
	SELECTION OF FLIGHT CONTROL MODE : ATTITUDE – MONITOR – ENTRY (CM) OR OFF – ATTITUDE HOLD – AUTO (TERMS USED IN LM SWITCH)	FLIGHT CONTROL MODE SEL. SW. (CM)	
	DISABLE RCS ENGINE QUADS FOR FREE DRIFT OR MALFUNCTION : ROLL/PITCH, YAW	RCS QUAD DISABLING SW'S. (CM)	RCS UTILIZED TO MAINTAIN VEHICLE ATTITUDE DURING HYPERSONIC FLIGHT REGIME
	SELECT INPUT FOR 3 AXIS ATTITUDE CONTROL : MODE – PULSE – DIRECT	ATTITUDE CONTROL ROLL SW. (LM)	
		ATTITUDE CONTROL YAW SW. (LM)	
		ATTITUDE CONTROL PITCH SW. (LM)	
	CAUTION/WARNING INDICATIONS TO BE DETERMINED	RCS STATUS LIGHTS (LM/CM)	
	" " " " "	OMS STATUS LIGHTS	
	DISPLAY OF COMMANDED ANGLE OF ATTACK	ANGLE OF ATTACK DISPLAY	ANGLE OF ATTACK (BACKUP TO INERTIALLY-DRIVEN DISPLAY)
	INERTIALLY-DETERMINED " " " (Δ = ERROR)		
	MANUAL CONTROL OF FUEL/OXIDIZER MIXTURE RATIO FOR J2/J2S : RICH – CENTER – LEAN	MIXTURE CONTROL	
	DISPLAY OF FUEL/OXIDIZER MIXTURE RATIO : % FUEL	% FUEL INDICATOR	
	INDICATION THAT BOOSTER/LAUNCH VEHICLE GUIDANCE HAS FAILED	L/V GUID FAIL LIGHT (CM)	
	" " " " ATTITUDE RATES ARE BEING EXCEEDED	L/V RATES EXC LIGHT (CM)	
	" " " " ENGINE THRUST IS BELOW REQUIRED VALUE	L/V THRUST LOW LIGHT (CM)	
	" " " " HAS SEPARATED	L/V STAGED LIGHT (CM)	
	" " ORBITER EXTERNAL TANKS HAVE SEPARATED	TANKS STAGED LIGHT	

ORBITER SUBSYSTEM	FUNCTION CONTROLLED OR PARAMETER/CONDITION DISPLAYED	SPACE FLIGHT REGIME	ATMOSPHERIC FLIGHT REGIME
FLIGHT CONTROL AND STABILIZATION	AIRSPEED MACH NUMBER TRUE AIRSPEED ROLL & PITCH MANUAL CONTROL YAW CONTROL DEPLOY HIGH LIFT DEVICES (TO BE DETERMINED) PITCH TRIM " " ROLL " AERODYNAMIC BRAKING VISUAL DISPLAY OF THE POSITIONS OF: RUDDER TRIM FOR YAW ELEVON/BODY FLAP TRIM FOR ROLL & PITCH (±30° & +15/-30°) SPEED BRAKES (0°-50°) (SPOILERS - NEED TO BE DETERMINED) (FLAPS - NEED TO BE DETERMINED) BRAKING VEHICLE ON GROUND DURING TAXI AND LANDING ROLL-OUT PEDAL ADJUSTMENT TO ACCOMMODATE VARYING BODY SIZES ABES THROTTLES - 1 FOR EACH OF 4 AIR BREATHING ENGINES STABILITY AUGMENTATION SYSTEM CONTROLS: ROLL: ENGAGE-OFF PITCH: " " YAW: " " FLIGHT CONTROL MODE SELECT: AUTO - MANUAL (BACKUP) (AUTO = FLY BY WIRE) CONTROL SURFACE HYDRAULIC ACTUATORS: ELEVON: PWR-DISABLE 1L-2L-3L-4L/1R-2R-3R-4R RUDDER: PWR-DISABLE T-B HYDRAULIC SYSTEM PRESSURE: 1-2-3 C & W LIGHTS TO BE DETERMINED INSTANTANEOUS VERTICAL VELOCITY BACKUP FOR PRIMARY ALTITUDE DISPLAY		AIRSPEED/MACH METER (L-1011) DIGITAL TAS DISPLAY (L-1011) 2 AXIS ATTITUDE CONTROLLER - SAME DEVICE FOR SPACE/ATMOS FLIGHT REGIMES RUDDER PEDALS HIGH LIFT DEVICE DEPLOYMENT CONTROLS (TBD) BODY FLAP CONTROL (TBD) ELEVON TRIM CONTROL ROLL " " (ELEVON) SPEED BRAKE CONTROL AERODYNAMIC SURFACE POSITION INDICATOR: RUDDER ELEVON/BODY FLAP SPEED BRAKES (SPOILERS)(TBD) (FLAPS)(TBD) RUDDER PEDALS (WITH BRAKES) PEDAL POSITION ADJUSTMENT CONTROL ABES THROTTLE CONTROLS (4 IN QUADRANT) PRIMARY FLIGHT CONTROL SUBSYSTEM PANEL: SAS ROLL PUSHBUTTON " PITCH " " YAW " FLT CONT MODE SELECT PUSHBUTTON HYDRAULIC SUBSYSTEM CONTROL PANEL FOR ELEVONS: ELEVON ACTUATOR HYD PWR (8) RUDDER " " " (2) HYD PRESS IND (TRIPLE METER) C & W LIGHTS TBD INSTANTANEOUS VERTICAL VELOCITY IND. (L-1011) BACKUP ALTITUDE INDICATOR (L-1011)

ORBITER SUBSYSTEM	FUNCTION CONTROLLED OR PARAMETER/CONDITION DISPLAYED	SPACE FLIGHT REGIME	ATMOSPHERIC FLIGHT REGIME
	(THRUST CONTROL FUNCTIONS INCLUDED UNDER F/C & STAB. S/S)	(THRUST CONTROLS/DISPLAYS INCLUDED UNDER FLIGHT CONTROL & STAB.)	
PROPULSION: ORBITAL MANEUVERING SYSTEM	QUANTITY OF MMH REMAINING IN TANKS: % REMAINING " " NO_4 " " " " " TANK SELECTOR SWITCH: PAIR 1 - PAIR 2 BULK TEMPERATURE OF REMAINING MMH " " " " NO_4 PRESSURE OF REMAINING MMH " " " NO_4 TEMPERATURE/PRESSURE SELECTION FOR DISPLAY SUPERCRITICAL HELIUM PRESSURE DIGITAL DISPLAY SELECT HELIUM CONDITION TO MONITOR ON DISPLAY: PRES TANK 1 — PRES TANK 2 — TEMP	OMS C/D REQUIREMENTS: FUEL QUANTITY INDICATOR (LM) OXIDIZER QUANTITY INDICATOR (LM) TANK SELECTOR SWITCH (LM) FUEL/OXIDIZER TEMPERATURE DISPLAY (LM) FUEL/OXIDIZER PRESSURE DISPLAY (LM) PROPLNT TEMP/PRES MON SWITCH (LM) OMS HELIUM DISPLAY (LM) HELIUM MONITOR SWITCH (LM) DISPLAY POWER FAIL LIGHTS (LM)	
REACTION CONTROL SUBSYSTEM	REMAINING MMH IN RCS TANKS: % REMAINING TANK SELECTOR SWITCH: 1-2-3-4-5-6-TOTAL REMAINING NO_4 IN RCS TANKS: % REMAINING PROPELLANT TEMPERATURE: MMH-NO4 HELIUM PRESSURIZING GAS PRESSURE " QUANTITY RCS CLUSTER DISABLE SWITCHES IN CASE OF MALFUNCTION TALKBACK INDICATING OPERABLE/DISABLED RCS CLUSTERS TO-BE-DETERMINED PANEL PERMITTING OMS FUEL AND OXIDIZER TO BE PUMPED INTO ANY SELECTED RCS CLUSTER/POD OR OREA THAT IS DEPLETED CAUTION/WARNING (IN ADDITION TO DISPLAY POWER FAILURE LIGHTS) TO BE DETERMINED FOR OMS AND RCS OPEN-CLOSE THRUSTER DOORS: NUMBER OF SWITCHES TO BE DETERMINED	REACTION CONTROL C/D REQUIREMENTS DUAL PROPLNT QUAN INDICATOR TANK SELECTOR SWITCH (LM) DUAL PROPLNT TEMP INDICATOR (LM) PRESSURE INDICATOR (LM) HELIUM QUANTITY INDICATOR (LM) RCS CLUSTER DISABLE SWITCHES (LM) DISABLED CLUSTER TALKBACKS (LM) OMS-RCS PROPELLANT CROSSFEED PANEL DISPLAY POWER FAIL LIGHTS (LM) OMS/RCS C&W/STATUS LIGHTS (LM) THRUST DOOR SWITCHES	IDENTICAL FUNCTIONS DURING ENTRY, HYPERSONIC FLIGHT REGIME
AIR BREATHING ENGINE SYSTEM	OPEN-CLOSE ABES ENGINE BAY DOORS TALKBACK INDICATING BOTH DOORS OPEN OR CLOSED DEPLOY - RETRACT AIR BREATHING ENGINES TALKBACK INDICATING ENGINES DEPLOYED OR RETRACTED START ENGINES ON GROUND POWER START ENGINES DURING FLIGHT % RPM FOR SPOOL #1 OF THE F101/F12A3 TURBOFAN ENGINES " " " #2 " " " " " " TURBINE BLADE TEMPERATURE ENGINE OIL TEMPERATURE " " PRESSURE " " QUANTITY ENGINE VIBRATION IN FAN AND TURBINE STAGES ENGINE FIRE AND OVERHEAT WARNING ENGINE FUEL FLOW PRESSURE QUANTITY		ABES C/D REQUIREMENTS: ABES DOORS SWITCH (LM/CM) ABES DOORS TALKBACK (LM/CM) ENGINES DEPLOY SWITCH (LM/CM) ENGINES DEPLOYED TALKBACK (LM/CM) ENGINE START SWITCHES (GROUND) " " " (FLIGHT) % RPM N_1 VERTICAL SCALE INDICATOR (4 TAPE) " " N_2 " " " " TURBINE BLADE TEMP INDICATOR (4 TAPE) ENG OIL TEMP INDICATOR (4 TAPE) " " PRES " " " " QUAN " " LIGHT-EMITTING DIODE DISPLAY MATRIX " " " " " OR C/W IND. FUEL FLOW INDICATOR (4 TAPE) " PRES " " " QUAN " "

ORBITER SUBSYSTEM	FUNCTION CONTROLLED OR PARAMETER/CONDITION DISPLAYED	SPACE FLIGHT REGIME	ATMOSPHERIC FLIGHT REGIME
MAIN ENGINES & PRIMARY SUPPLY	QUANTITY OF LO_2 REMAINING IN TANKS : % REMAINING " " LH_2 " " " " " BULK TEMPERATURE OF REMAINING LO_2 " " " " LH_2 PRESSURE IN LO_2 TANK(S) " " LH_2 " (OTHER REQ'D C/D LISTED UNDER FLIGHT CONTROL & STAB SUBSYSTEM)	DUAL FUEL QUANTITY INDICATOR (LM) FUEL/OXIDIZER TEMPERATURE DISPLAY (LM) FUEL/OXIDIZER PRESSURE DISPLAY (LM)	

ORBITER SUBSYSTEM	FUNCTION CONTROLLED OR PARAMETER/CONDITION DISPLAYED	SPACE FLIGHT REGIME	ATMOSPHERIC FLIGHT REGIME
ELECTRICAL POWER GENERATION AND DISTRIBUTION	FUEL CELL STATUS (3 FCs) SELECTOR SWITCH: 1-2-3 H_2 PUMP STATUS SWITCH AND FEEDBACK: ON-OFF H_2 PURGE VALVE POSITION AND FEEDBACK: OPEN-CLOSE (INITIATE PURGE) O_2 PURGE VALVE POSITION AND FEEDBACK: OPEN-CLOSE H_2 PUMP OUTLET PRESSURE FUEL CELL INTERNAL H_2 PRESSURE " " " STACK TEMPERATURE REACTANT TANK O_2 FILL VALVE POSITION SWITCH AND TALKBACK " " H_2 " " " " " " " " O_2 VENT VALVE POSITION " " " " " H_2 " " " " " " O_2 QUANTITY H_2 " O_2 PRESSURE H_2 " OVERLOAD PROTECTION AND CONTROL REQUIREMENTS TO BE DETERMINED	FUEL CELL FUNCTIONAL C/D GROUP: FC SELECTOR SWITCH (CM) 3 POS H_2 PUMP SWITCHES (CM) 3 EA H_2 PUMP TALKBACK (CM) 3 EA H_2 PURGE VALVE SWITCHES (CM) 3 EA H_2 PURGE VALVE TALKBACK (CM) 3 EA H_2 PUMP OUTLET PRESSURE HIGH TALKBACK (CM) 3 DUAL PRESSURE/TEMPERATURE IND. (CM) O_2 FILL VALVE SWITCH (CM) 3 O_2 " " TALKBACK (CM) 3 H_2 " " SWITCH (CM) 3 H_2 " " TALKBACK (CM) 3 DUAL REACTANT QUANTITY INDICATOR (CM) " " PRESSURE " O_2 VENT VALVE SWITCH AND TALKBACK 3 H_2 " " " " " 3	
	DC SOURCE SELECTOR SWITCH: BATTERIES – RECTIFIER #1 – RECTIFIER #2 – BUS #1 – BUS #2 – DC GROUND POWER DC VOLTS FROM ANY OF ABOVE SOURCES DC AMPS " " " " " TOTAL DC POWER DELIVERED IN WATTS AC SOURCE SELECTOR SWITCH: INVERTERS – APU – AC BUS #1 – AC BUS #2 – GENERATOR – AC GROUND POWER AC VOLTS FROM ANY OF ABOVE SOURCES AC AMPS " " " " " AC WATTS " " " " " OR ALL ON LINE & TOTAL AC FREQUENCY " " " " " BRING BATTERIES ON LINE TO START FUEL CELL(S) IN TURN RECTIFIER #1 & 2 STATUS INDICATIONS: ON-OFF RECTIFIER #1 & 2 TEMPERATURE	BATTERIES/INVERTERS & GENERATORS/RECTIFIERS: DC INDICATOR SOURCE SELECTOR SWITCH (CM) DC VOLTMETER (CM) DC AMMETER (CM) DC WATT METER AC VOLTMETER (CM) AC AMMETER AC WATTMETER AC FREQUENCY METER (CM) AC INDICATOR SOURCE SELECTION SWITCH (CM) BATT ON BUS #1/#2 SWITCH RECT #1 TALKBACK " #2 " DUAL RECT TEMP INDICATOR	
	CONNECT FUEL CELLS TO DC BUSES: BUS #1 – CENTER – BUS #2 " GROUND PWR " " " " " " " BATTERIES " " " " " " " RECTIFIERS " " " " " CONNECT INVERTERS TO AC BUSES: BUS #1 – CENTER – BUS #2 " GENERATORS " " " " " " " GROUND PWR " " " " " "	SWITCH MATRIX: POWER SOURCE TO BUSES: FC 1 TO BUS #1/2 SWITCH (CM) FC 2 TO BUS #1/#2 SWITCH (CM) FC 3 TO BUS #1/#2 SWITCH (CM) GRND DC PWR TO BUS #1/#2 SWITCH (CM) BATT TO BUS #1/BUS #2 SWITCH (CM) RECT 1 " " " " (CM) RECT 2 " " " " (CM) INVERTER 1 & 2 TO AC BUS #1 SWITCH (CM) " 3 & 4 " " " #2 " (CM) GEN 1 " " " " " (CM) " 2 " " " " " (CM) " 3 " " " " " (CM)	

ORBITER SUBSYSTEM	FUNCTION CONTROLLED OR PARAMETER/CONDITION DISPLAYED	SPACE FLIGHT REGIME	ATMOSPHERIC FLIGHT REGIME
ELECTRICAL POWER GENERATION AND DISTRIBUTION	INDICATION OF LOSS OF DC POWER ON BUS #1 " " " " " " " BUS #2 " " " " AC " " BUS #1 " " " " " " " BUS #2 OVERVOLTAGE RESET ON DC BUSES " " " AC " FUEL CELL STATUS FOR #1, #2, #3 EMERGENCY BATTERY STATUS FOR #1, #2 INVERTER STATUS FOR #1, 2, 3, 4 RECTIFIER STATUS FOR #1, #2 GENERATOR STATUS FOR #1, #2, #3 GRND PWR INPUT STATUS – DC, AC	GROUND AC TO AC BUS #1/#2 SWITCH (CM) DC BUS 1 LOSS LIGHT " " 2 " " AC " 1 " " AC " 2 " " DC BUS RESET SWITCH (CM) AC " " " (CM) EPS CAUTION & WARNING INDICATORS: FC 1 C&W LIGHT " 2 " " " 3 " " BATT 1 " " " 2 " " INV 1 " " " 2 " " " 3 " " " 4 " " RECT 1 " " " 2 " " GEN 1 " " " 2 " " " 3 " " GND DC " " " AC " "	
	OPEN N_2H_4 FUEL SUPPLY VALVE: OPEN – CLOSE FUEL VALVE POSITION START UP APU: START – SHUT DOWN TURBINE SPEED: RPM TURBINE OVERSPEED COMBUSTOR OUTLET TEMPERATURE DECOUPLE GENERATOR FROM APU BULK TEMPERATURE OF N_2H_4 HYDRAZINE FUEL FOR APU TANK HEAT: ON – OFF QUANTITY/PRESSURE OF REMAINING FUEL TANK SELECTOR SWITCH: 1 – 2 – 3 – 4 – 5 – 6 HELIUM PRESSURE/TEMP SELECT HELIUM CONDITION TO MONITOR: T1 – P1 – T2 – P2 – T3 – P3 – T4 – P4 – T5 – P5 – T6 – P6 APU LUBE OIL TEMPERATURE		AUXILIARY POWER UNIT/GENERATOR S/S FUEL VALVE POSITION SWITCH (3) " " " TALKBACK (3) APU CONTROL (3) TRIPLE RPM METER (3 TAPES) TURBINE OVERSPEED C&W LIGHT (3) TRIPLE TEMP METER DEC GEN SWITCH (3) FUEL TEMP INDICATOR FUEL TANK HEATER SWITCH TEMP/PRESS GAUGE TANK SELECTOR SWITCH APU HELIUM DISPLAY HELIUM MONITOR SELECTOR SWITCH TRIPLE LUBE OIL TEMP METER (3 TAPES)

ORBITER CONTROLS & DISPLAYS - MARK I AVIONICS/PRELIMINARY

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ORBITER SUBSYSTEM	FUNCTION CONTROLLED OR PARAMETER/CONDITION DISPLAYED	SPACE FLIGHT REGIME	ATMOSPHERIC FLIGHT REGIMES
ENVIRONMENTAL CONTROL AND LIFE SUPPORT	(OXYGEN SUPPLY AND ASSOCIATED C/D ARE LISTED UNDER FUEL CELL C/D IN EPS LIST.) CABIN PRESSURE/TEMPERATURE N_2 PRESSURE/TEMPERATURE CABIN RELATIVE HUMIDITY PARTIAL PRESSURE OF CO_2 IN CABIN ATMOS POTABLE WATER QUANTITY REMAINING ECLS RADIATOR OUTLET TEMPERATURES OF WATER-GLYCOL CABIN TEMPERATURE CONTROL: AUTO-MANUAL " " MANUAL CONTROL: INCREASE AIRLOCK PRESSURE/TEMPERATURE CAUTION & WARNING INDICATORS TO BE DETERMINED PASSENGER COMPARTMENT PRESSURE, TEMPERATURE PAYLOAD BAY/COMPARTMENT PRESSURE, TEMPERATURE AVIONICS BAY TEMPERATURE " " " CONTROL EMERGENCY O_2 VALVE ON FLIGHT DECK	(O_2 C/D LISTED UNDER ELECTRICAL POWER) DUAL PRESS/TEMP INDICATOR (CM) N_2 DUAL " " " (CM) DUAL RH/CO_2 LEVEL INDICATOR (CM) WATER QUANTITY GAUGE (CM) RAD OUTLET TEMP INDICATOR (CM) CABIN TEMP AUTO-MAN SWITCH (CM) " " THUMBWHEEL (CM) DUAL PRESS/TEMP INDICATOR (CM) DUAL PRESS/TEMP INDICATOR (CM) DUAL PRESS/TEMP INDICATOR (CM) C & W ANNUNCIATORS	IDENTICAL REQUIREMENTS AVIONICS BAY TEMP INDICATOR " " " CONTROL EMERGENCY O_2 SWITCH CONTROL

ORBITER SUBSYSTEM	FUNCTION CONTROLLED OR PARAMETER/CONDITION DISPLAYED	SPACE FLIGHT REGIME	ATMOSPHERIC FLIGHT REGIME
TELECOMMUNICATIONS SUBSYSTEM		AUDIO CENTER PANEL (1 FOR EACH CREWMAN)	
	POWER ON TO AUDIO CENTER AND VOICE CONTROL: PUSH TO TALK – OFF – VOX	AUDIO POWER SWITCH (CM)	
	ENABLE HEADSETS TO RECEIVE ONLY OR RECEIVE & TRANSMIT OVER S-BAND EQUIPMENT: T/R – OFF – REC	S-BAND SWITCH (CM)	
	ENABLE HEADSETS TO RECEIVE OR T/R OVER INTERCOM: T/R – OFF – REC*	INTERCOM SWITCH (CM)	
	" " " " " " OVER HARDLINE TO GROUND PRIOR TO LAUNCH: T/R – OFF – REC	GROUND SWITCH	
	ENABLE HEADSETS TO RECEIVE OR T/R OVER HARDLINE TO MANNED BOOSTER DURING PRE-LAUNCH THRU POST-SEPARATION: T/R – OFF – REC	BOOSTER SWITCH	
	SAME FOR SPACE STATION WHILE DOCKED: T/R – OFF – REC	STATION SWITCH	IDENTICAL REQUIREMENTS (AN AIC-13 INTERPHONE SYSTEM USED IN ORBITER)
	ALTER SENSITIVITY OF VOICE-OPERATED RELAY	VOX SENS THUMBWHEEL (CM)	
	BALANCE INTERCOM VIS-A-VIS RF SIGNALS	INTERCOM BALANCE THUMBWHEEL (CM)	
	HEADSET GAIN CONTROL	VOL CONTROL (CM)	
	MAGNITUDE OF RECEIVED S-BAND SIGNAL	S-BAND ANT METER (CM)	
	SELECT S-BAND OPERATING MODES: XPONDER – OFF – XPONDER PWR AMPL	S-BAND SEL SWITCH (CM)	
	" " " POWER: HIGH – LOW	PWR AMPL SEL SWITCH (CM)	
	" " " FREQUENCY: PRIMARY – SECONDARY	OSCILLATOR FREQ SEL SWITCH (CM)	
	" INFORMATION TO BE TRANSMITTED: RANGE PCM/VOICE — RANGE ONLY — TAPE — ANALOG — TV	CONTENT SEL SWITCH (CM)	
	EMERGENCY VOICE COMM MODES: VOICE – OFF – KEY	EMERGENCY SWITCH (CM)	
	ENABLE UP-DATA LINK: OFF – ENABLE	UP DATA SWITCH (CM)	
	S-BAND ANTENNA SELECTION: UPPER – LOWER – AUTOMATIC	S-BAND ANTENNA SEL SWITCH (CM)	
	DATA LINK, ORBITER/GROUND: HARDLINE — RF — OFF	GROUND DATA LINK SWITCH	
	DATA LINK, ORBITER/BOOSTER: HARDLINE – RF – OFF	BOOSTER DATA LINK SWITCH	
	DATA LINK, ORBITER/STATION: HARDLINE – RF – OFF	STATION DATA LINK SWITCH	
	DATA LINK, ORBITER/PAYLOAD: HARDLINE – RF – OFF	PAYLOAD DATA LINK SWITCH	
	VIDEO FROM PAYLOAD TO GROUND: ON-BOARD MONITOR – MON/GROUND – GROUND – OFF	VIDEO LINK SWITCH (CM)	
	SELECT ANTENNA CONTROL MODE: MANUAL – AUTOMATIC	ANT DRIVE SWITCH (CM)	
	SLEW ANTENNA UP – DOWN	SLEW CONTROL SWITCH (CM)	
	" " FORE – AFT	" " (CM)	
	" RATE SELECTOR SWITCH	" RATE " (CM)	
	DEPLOY HIGH-GAIN ANTENNA: OFF – DEPLOY	ANT DEPLOY SWITCH (CM)	
	" " " " TALKBACK	" " TALKBACK (CM)	
	UHF COMMUNICATIONS C/D PANEL TO BE DETERMINED		UHF CONTROL PANEL
	ATC TRANSPONDER CONTROL FUNCTIONS		ATC TRANSPONDER CONTROL PANEL
	UHF ANTENNA SELECTION		UHF ANTENNA SELECTOR SWITCH
	ANTENNA DEPLOYMENT: DEPLOY – CENTER – RETRACT (UHF ANT FOR ATMOSPHERIC FLIGHT)		ANTENNA DEPLOY CONTROL – UHF
	ATC TRANSPONDER ANTENNA SELECTION		ATC TRANSPONDER ANTENNA SEL SWITCH
	" " " DEPLOYMENT: DEPLOY – CENTER – RETRACT		" " " DEPLOY CONTROL

*INCLUDING INTERCOM STATIONS IN DOCKING STATION, AIRLOCK, PASSENGER COMPT., AND PAYLOAD COMPT. AS WELL AS FLIGHT DECK.

ORBITER SUBSYSTEM	FUNCTION CONTROLLED OR PARAMETER/CONDITION DISPLAYED	SPACE FLIGHT REGIME	ATMOSPHERIC FLIGHT REGIME
SEQUENTIAL EVENTS CONTROL	INITIATE/ARM MASTER EVENTS SEQUENCE EVENTS CONTROL LOGIC & COMMAND CIRCUITRY FOR ASCENT, ABORT AND RE-ENTRY FLIGHT PHASES: OFF - ARM (2) ARM CREW ESCAPE SYSTEM SEQUENCE CONTROLLERS: OFF - ARM ARM ORBITER TANK JETTISON SEQUENCE CONTROLLERS: OFF - ARM INITIATE ORBITER TANK JETTISON: MANUAL - AUTO SELECT BOOSTER EMERGENCY DETECTION AND ABORT INITIATION CONTROL MODE: ENABLE L/V EDS - AUTO - MAN - OFF	MASTER EVENTS SEQ CONT SWITCHES (CM) EJECTION SYS SEQ CONT SWITCHES TANK JETT SEQ CONT SWITCHES TANK JETT SWITCH ABORT SYSTEM SWITCH (CM)	→ IDENTICAL REQUIREMENT
CAUTION AND WARNING	RE-SET MASTER ALARM LIGHTS AND AUDIO TONE ALARM C & W LAMP TEST SWITCH C & W FROM ONE DC BUS TO ANOTHER: DC BUS 1 - DC BUS 2	MASTER ALARM ILLUMINATED PUSH-BUTTON (CM) LAMP TEST PUSHBUTTON (CM) C & W PWR SELECTOR SWITCH (CM)	→ IDENTICAL REQUIREMENTS
MISCELLANEOUS SYSTEMS REQUIREMENTS:	INDICATION OF HARD DOCK (ACHIEVED BY CREWMAN PERFORMING DOCKING MANEUVERS FROM CENTER/DOCK STATION IN A CLOCK) INSTRUMENT PANEL LIGHTING: OFF - BRIGHT (ONE FOR EACH CREW STATION) COMPARTMENT FLOOD LIGHTING: OFF - BRIGHT EXTERNAL TRACKING AND DOCKING FLOOD LIGHTING: TRACK - OFF - DOCK LANDING LIGHTS, AERO REGIME: OFF - ON AIRCRAFT-TYPE MARKER/NAVIGATION, CLEARANCE LIGHTS: OFF - ON LANDING GEAR DEPLOYMENT: LOWER - RAISE " " POSITION TALKBACK EMERGENCY LANDING GEAR DEPLOYMENT	HARD DOCKED LIGHT INTERNAL & EXTERNAL LIGHTING: PANEL DIMMER CONTROL (CM) FLOOD DIMMER CONTROL (CM) EXTERNAL LTG SWITCH (LM)	→ IDENTICAL REQUIREMENTS LANDING LIGHTS SWITCH EXTERIOR LIGHTING PANEL LDG GEAR HANDLE " " TALKBACK EMER LDG GEAR HANDLE

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AIRCRAFT AVIONICS EQUIPMENT REQUIRED FOR
CREW SAFETY VS MISSION PHASE

(Pages A-37 through A-43)

AVIONICS EQUIPMENT REQUIRED FOR CREW SAFETY

MISSION PHASE (FHF)

SUBSYSTEM

PRIMARY FLIGHT CONTROLS

Page 1 of 2 Pages

EQUIPMENT	TAKEOFF	CLIMB	HORIZONTAL FLIGHT	DESCENT	APPROACH	LANDING	TAXI	SHUTDOWN AND SAFING			
Rudder Servos	X	X	X	X	X	X	X				
Elevon Servos	X	X	X	X	X	X					
SAS/FBW Servos	X	X	X	X	X	X	X				
Trim/Backup Servos	X	X	X	X	X						
Speed Brake Servos					X	X	X				
PFCS Computers	X	X	X	X	X	X	X				
Roll Rate Gyros	X	X	X	X	X	X					
Yaw Rate Gyros	X	X	X	X	X	X					
Pitch Rate Gyros	X	X	X	X	X	X					
Lateral Accelerometers	X	X	X	X	X	X					
Normal Accelerometers	X	X	X	X	X	X					
Engine Speed Controls	X	X	X	X	X	X					
Pilot Static Probes	X	X	X	X	X	X					

AVIONICS EQUIPMENT REQUIRED FOR CREW SAFETY

MISSION PHASE (FHF)

SUBSYSTEM

PRIMARY FLIGHT CONTROLS

Page 2 of 2 Pages

EQUIPMENT	TAKEOFF	CLIMB	HORIZONTAL FLIGHT	DESCENT	APPROACH	LANDING	TAXI	SHUTDOWN AND SAFING			
Rate of Turn Sensor	X	X	X	X	X	X					
Anti Skid System						X	X				
Touchdown Switch						X					
Wheel Speed Sensor						X	X				
α Sensor				X	X	X					

AVIONICS EQUIPMENT REQUIRED FOR CREW SAFETY

A/C

SUBSYSTEM

NAVIGATION

MISSION PHASE (FHF)

EQUIPMENT	TAKEOFF	CLIMB	HORIZONTAL FLIGHT	DESCENT	APPROACH	LANDING	TAXI	SHUTDOWN AND SAFING			
Tacan Transceiver			X	X	X						
Tacan Control Panel			X	X	X						
Tacan Audio Select Panel			X	X	X						
Tacan Antennae			X	X	X						
Directional Gyro		X	X	X	X						
Compass Controller		X	X	X	X						
Compass Coupler		X	X	X	X						
Flux Valve		X	X	X	X						
Magnetic Compensator		X	X	X	X						
Vertical Gyro	X	X	X	X	X	X					

AVIONICS EQUIPMENT REQUIRED FOR CREW SAFETY

A/C SUBSYSTEM COMMUNICATIONS

MISSION PHASE (FHF)

EQUIPMENT	TAKEOFF	CLIMB	HORIZONTAL FLIGHT	DESCENT	APPROACH	LANDING	TAXI	SHUTDOWN AND SAFING			
UHF Transceiver	X	X	X	X	X	X	X				
UHF Antennae	X	X	X	X	X	X	X				
UHF Antenna Selector	X	X	X	X	X	X	X				
UHF Control Panel	X	X	X	X	X	X	X				
Interphone Stations	X					X					
ATC Transponder	X				X	X					
ATC Transponder Antennae	X				X	X					
ATC Transponder Control Panel	X				X	X					

AVIONICS EQUIPMENT REQUIRED FOR CREW SAFETY

SUBSYSTEM

ELECTRICAL POWER GENERATION AND DISTRIBUTION

MISSION PHASE (FHF)

EQUIPMENT	TAKEOFF	CLIMB	HORIZONTAL FLIGHT	DESCENT	APPROACH	LANDING	TAXI	SHUTDOWN AND SAFING			
AC Generators	X	X	X	X	X	X	X	X			
Generator Control Units	X	X	X	X	X	X	X	X			
Transformer Rectifiers	X	X	X	X	X	X	X	X			
Static Inverters	X	X	X	X	X	X					
Main DC Power Distribution Unit	X	X	X	X	X	X	X	X			
Inverter AC Distribution Unit	X	X	X	X	X	X					
DC Power Distribution Units	X	X	X	X	X	X	X	X			
AC Generator Distribution Unit	X	X	X	X	X	X	X	X			
Inverter AC Bus	X	X	X	X	X	X					
AC Generator Bus	X	X	X	X	X	X	X	X			
DC Bus	X	X	X	X	X	X	X	X			
Panel Circuit Breakers	X	X	X	X	X	X	X	X			
Remote Controlled Circuit Breakers	X	X	X	X	X	X	X	X			

AVIONICS EQUIPMENT REQUIRED FOR CREW SAFETY

A/C

SUBSYSTEM

DISPLAYS AND CONTROLS

MISSION PHASE (FHF)

EQUIPMENT	TAKEOFF	CLIMB	HORIZONTAL FLIGHT	DESCENT	APPROACH	LANDING	TAXI	SHUTDOWN AND SAFING			
ADI	X	X	X	X	X	X					
HSI		X	X	X	X						
Surface Position Indicator	X	X	X	X	X	X	X				
Airspeed Ind.	X	X	X	X	X	X	X				
Altimeter	X	X	X	X	X	X					
Instantaneous Vertical Velocity	X	X	X	X	X	X					
Engine Oil Temperature	X	X	X	X	X	X	X				
Engine Oil Pressure	X	X	X	X	X	X	X				
Engine Fuel Pressure	X	X	X	X	X	X					
Engine N_1 Speed	X	X	X	X	X	X					
Engine N_2 Speed	X	X	X	X	X	X					
[illegible] Fuel Flow	X	X	X	X	X	X					
[illegible]	X	X	X	X	X	X					

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MISSION PHASE (FHF)

SUBSYSTEM

DISPLAYS AND CONTROLS

Page 2 of 2 Pages

EQUIPMENT	TAKEOFF	CLIMB	HORIZONTAL FLIGHT	DESCENT	APPROACH	LANDING	TAXI	SHUTDOWN AND SAFING			
PFCS Control Panel	X	X	X	X	X	X					
PFCS Surface Control Panel								X			
Side Stick Controller	X	X	X	X	X	X					
Rudder Pedals	X	X	X	X	X	X	X				

SPACECRAFT AVIONICS EQUIPMENT REQUIRED FOR CREW SAFETY VS MISSION PHASE

(Pages A-45 through A-48)

AVIONICS EQUIPMENT REQUIRED FOR CREW SAFETY

SUBSYSTEM

SPACECRAFT GUIDANCE,
NAVIGATION AND CONTROL

MISSION PHASE (FVF-M)

EQUIPMENT	LAUNCH & ASCENT	ORBIT INSERTION	ORBIT	DEORBIT	REENTRY	HYPERSONIC FLIGHT	DESCENT	APPROACH	LANDING	TAXI	SHUTDOWN AND SAFING		
GNC Digital Computer	X	X	X	X	X	X							
Inertial Meas. Unit	X	X	X	X	X	X							
Orbit Altimeter				X		X	X						
Subsystem Int. Units	X	X	X	X	X	X	X	X	X	X	X		

AVIONICS EQUIPMENT REQUIRED FOR CREW SAFETY

SUBSYSTEM

SPACECRAFT
GUIDANCE, NAVIGATION & CONTROLS

MISSION PHASE (FVF-M)

EQUIPMENT	LAUNCH & ASCENT	ORBIT INSERTION	ORBIT	DEORBIT	REENTRY	HYPERSONIC FLIGHT	DESCENT	APPROACH	LANDING	TAXI	SHUTDOWN AND SAFING		
Rate Gyros (3)	X	X	X	X	X	X							
Accelerometers - Normal	X	X											
Main Engine Gimbal Actuators	X	X											
TVC Electronics	X	X											
RCS Electronics		X	X	X	X	X							

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S/C AVIONICS EQUIPMENT REQUIRED FOR CREW SAFETY

SUBSYSTEM MISSION PHASE (FVF-M)

COMMUNICATIONS

EQUIPMENT	LAUNCH & ASCENT	ORBIT INSERTION	ORBIT	DEORBIT	REENTRY	HYPERSONIC FLIGHT	DESCENT	APPROACH	LANDING	TAXI	SHUTDOWN AND SAFING		
Antenna	X	X		X	X								
Antenna Switch	X	X		X	X								
S Band Power Amplifier	X	X		X	X								
Unified S-Band Equipment	X	X		X	X								
Pre-Mod Processor	X	X		X	X								
Up-Data Link	X	X		X	X								

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AVIONICS EQUIPMENT REQUIRED FOR CREW SAFETY

SUBSYSTEM

ELECTRICAL POWER GENERATION AND DISTRIBUTION

MISSION PHASE (FVF-M)

EQUIPMENT	LAUNCH & ASCENT	ORBIT INSERTION	ORBIT	DEORBIT	REENTRY	HYPERSONIC FLIGHT	DESCENT	APPROACH	LANDING	TAXI	SHUTDOWN AND SAFING		
Fuel Cells	X	X	X	X	X	X	X	X	X	X	X		
Static Inverters	X	X	X	X	X	X	X	X	X				
Main DC Power Distribution Unit	X	X	X	X	X	X	X	X	X	X	X		
Inverter AC Distribution Units	X	X	X	X	X	X	X	X	X				
DC Power Distribution Units	X	X	X	X	X	X	X	X	X	X	X		
AC Generators	X	X					X	X	X	X	X		
Generator Control Units	X	X					X	X	X	X	X		
AC Generator Distribution Unit	X	X	X	X	X	X	X	X	X	X	X		
Inverter AC Bus	X	X	X	X	X	X	X	X	X				
AC Generator Bus	X	X	X	X	X	X	X	X	X	X	X		
DC Bus	X	X	X	X	X	X	X	X	X	X	X		
Panel Circuit Breakers	X	X	X	X	X	X	X	X	X	X	X		
Remote Control Circuit Breakers	X	X	X	X	X	X	X	X	X	X	X		

Appendix B

ALTERNATE A AVIONICS APPROACH

The Alternate A approach provided for two separate and distinct avonics packages, one for aircraft-type operations and one for spacecraft operations. Figure B-1 shows three groups of avionics for the vehicle: (1) orbiter spacecraft, (2) orbiter aircraft, and (3) the pressure-fed, recoverable booster. The onboard booster avionics are assumed to be minimal and independent of the shuttle configuration; the effect of booster interface variation as compared to the total avionics is negligible.

B.1 EQUIPMENT

The detailed equipment block diagram for the Alternate is given in Fig. B-2. The high level of duplication is particularly evident in the displays and controls and utility

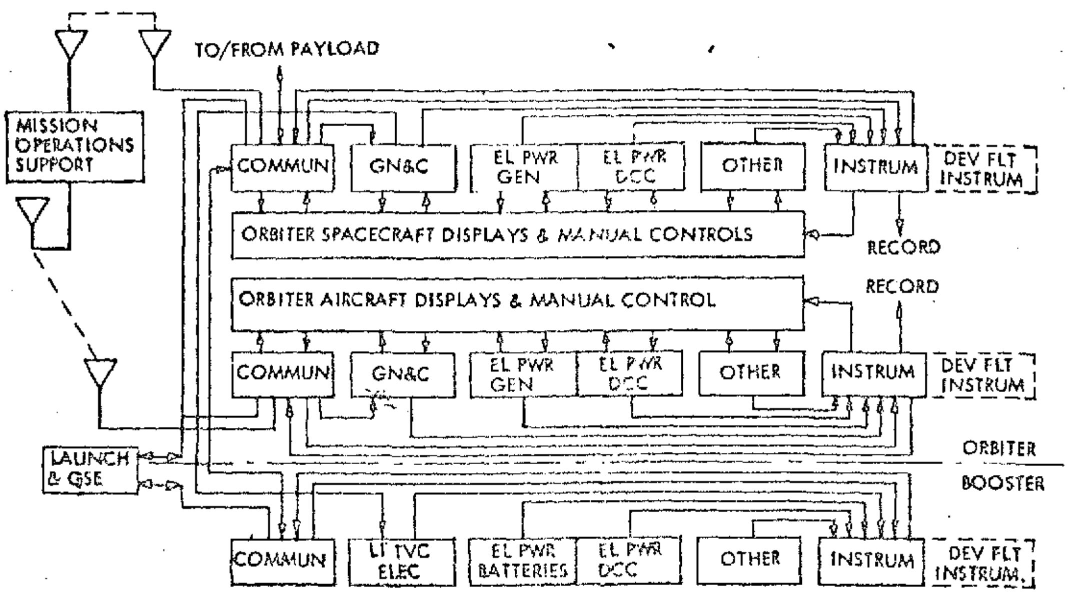


Fig. B-1 Alternate A Avionics Subsystem Diagram functions, e.g., electrical power. (This latter class of subsystems would obviously be combined in a final design; however, they have been kept isolated for the purposes of this study.) The primary feature of this approach is that dedicated equipment is provided for each functional usage. The instrumentation system, operated as a passive monitor, will provide supplementary fault flags to those equipments with built-in test capability. Hence, the crew must perform the redundancy management functions in response to these annunciators. The major consideration for this alternate is that it requires the least development of flight hardware. Conversely, it requires maximum usage of ground support equipment for checkout and maintenance.

Characteristics for Alternate A are summarized as follows:

- Dedicated equipments for each functional usage
- Minimum fault isolation
- Redundancy management performed by crew
- Maximum need for GSE
- Re-design for incorporating improved equipments
- Minimum development of flight hardware

B.2 WEIGHT STATEMENT

The vehicle delta weight statement for this alternate is developed in Table B-1. The equipment identified will replace units with common utilization in the baseline or will increase quantities to provide a similar redundancy level. The total weight increment is 718 lb or an increase of 7.9 percent over the baseline weight. This does not appear to be a significant driver.

Table B-1

ALTERNATIVE A DELTA WEIGHTS

	Baseline Reference						
	Additions			Deletions			
Subsystem	Equip	Quan	Total Wt (lb)	Equip	Quan	Total Wt (lb)	Δ Wt
Guidance, Navigation, and Control	Digital Computer	1	126	Navig Data Repeater Converter	2	110	
	Rate Gyros	12	21				
	Accelerometers	8	8				
	A/C Nav Set	1	83				
	Signal Condr	2	160				
Subtotal			398			110	+288
Communications	Pre-Mod Procr	1	11				
	USBE	1	38				
	S-Band Pwr Amp	1	32				
	Up-Data Link	1	22				
	Audio Cont Panel	2	154				
Subtotal			257				+257
Electrical Power	Emer Battery	2	124				
	AC Gen	3	120				
	Gen Ctrl Unit	3	24				
	Static Inverter	4	160				
	Transformer Rect	3	54				
Subtotal			482				+482
Displays and Controls Subtotal	All		878	All		717	+161
Data Mgmt Subtotal	N/R		--	All		665	-665
Instrumentation	Flight Recorder	2	96				
	Maint Recorder	1	40				
	Time Code Gen	1	15				
	PCM Tlm Equip	1	44				
Subtotal			195				+195
Net Total							+718

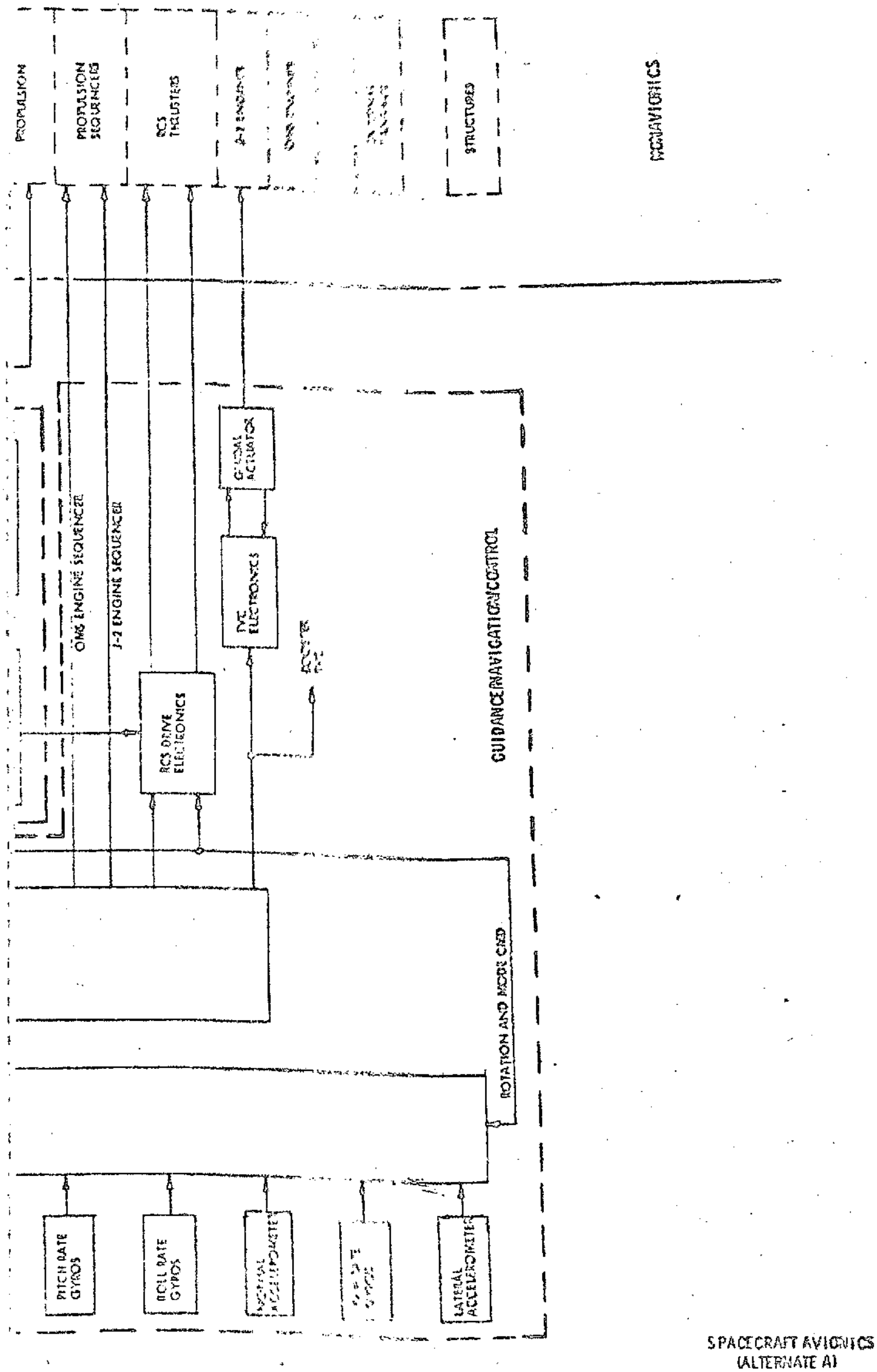
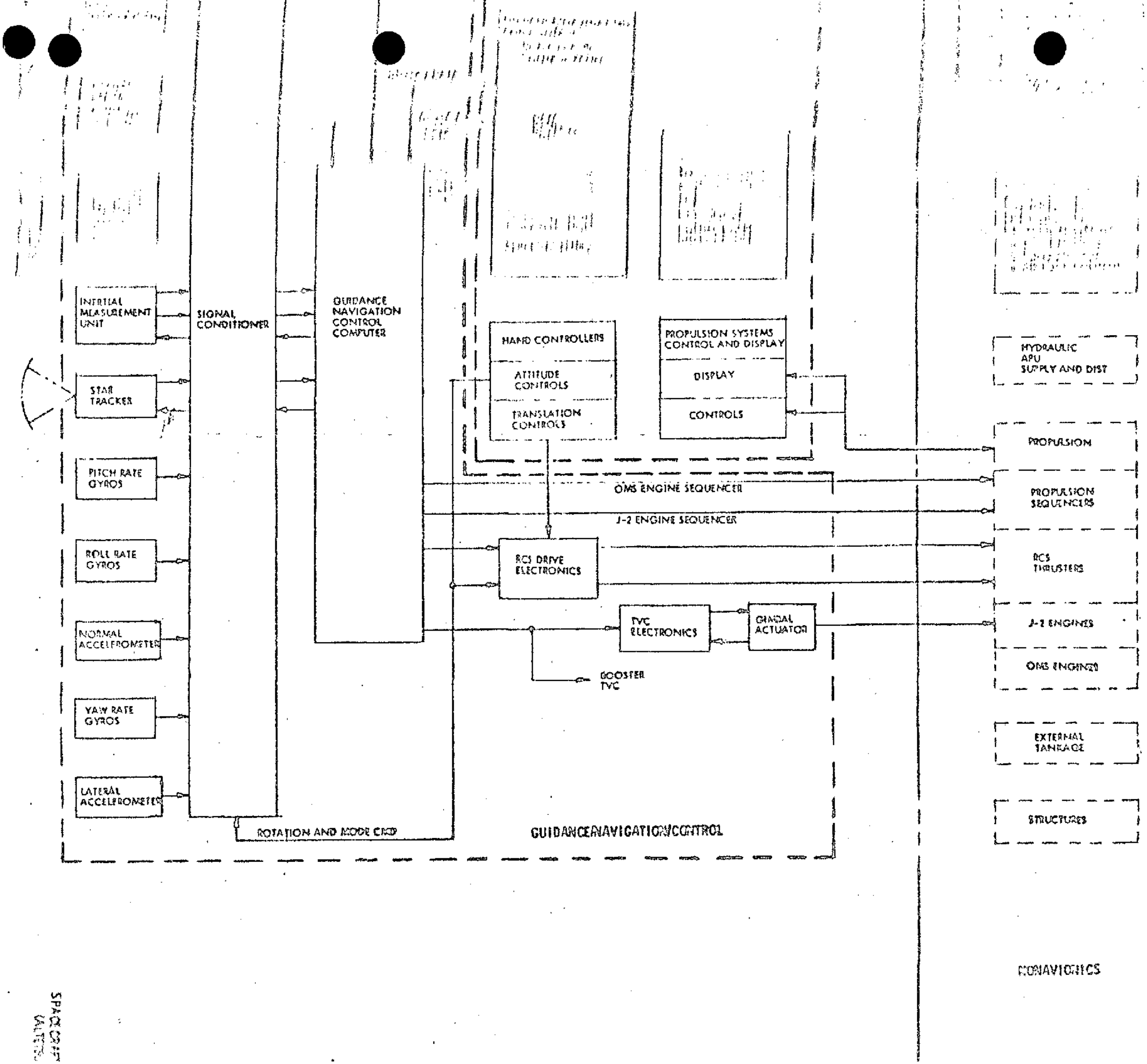
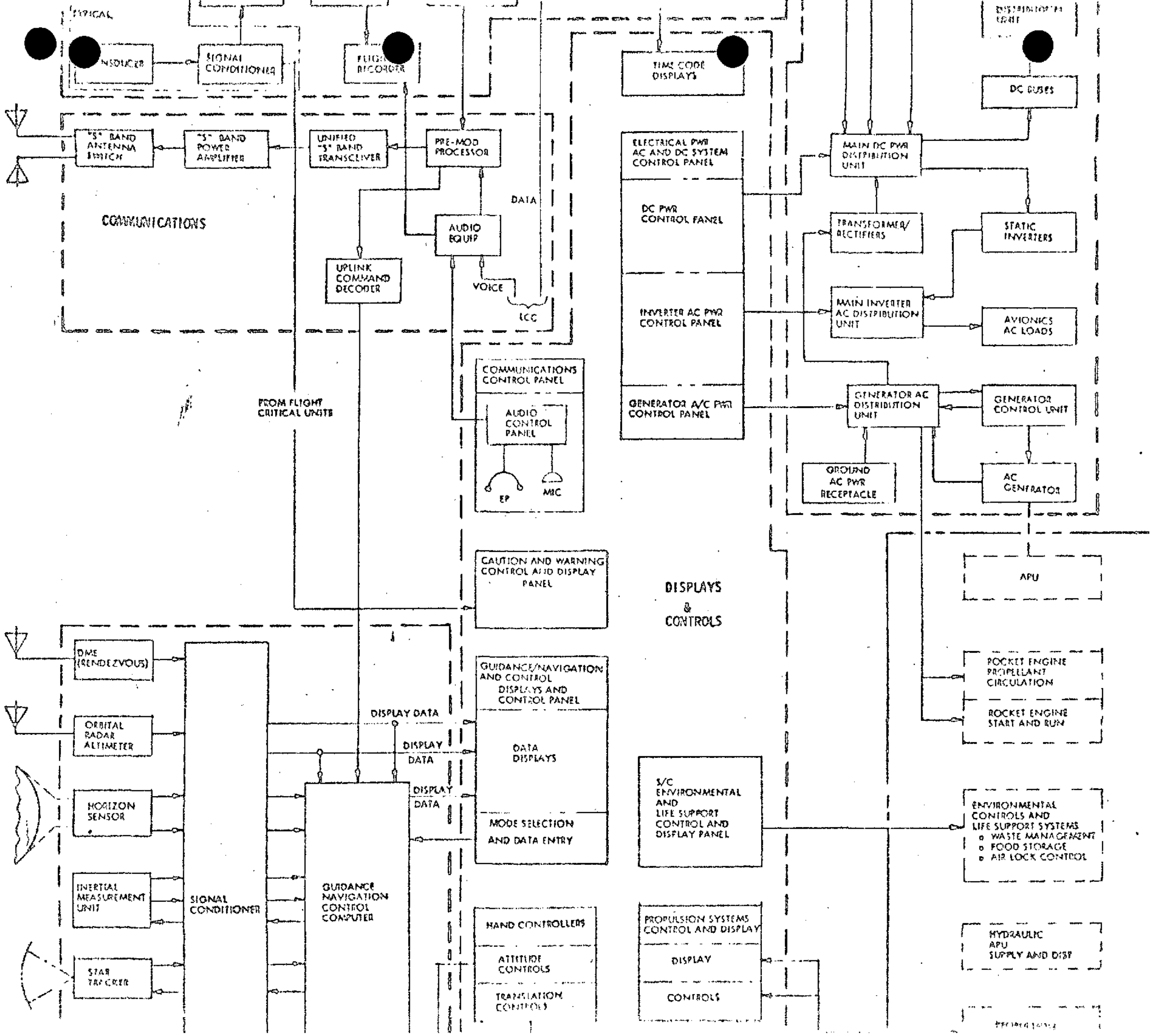


Fig. B-2 Spacecraft Avionics (Alternate A)
Detailed Equipment

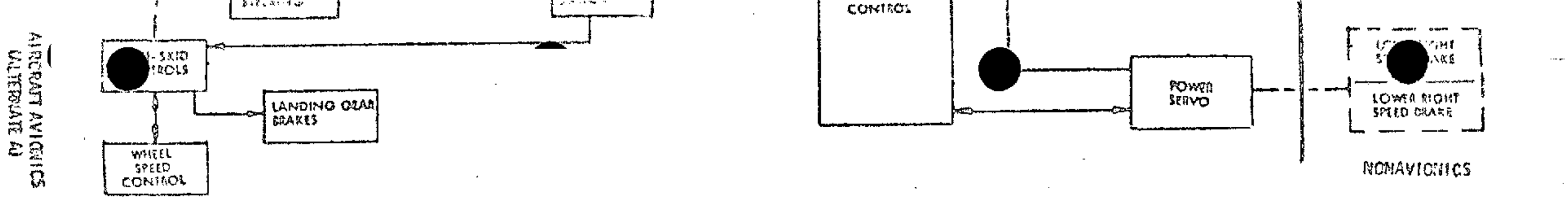


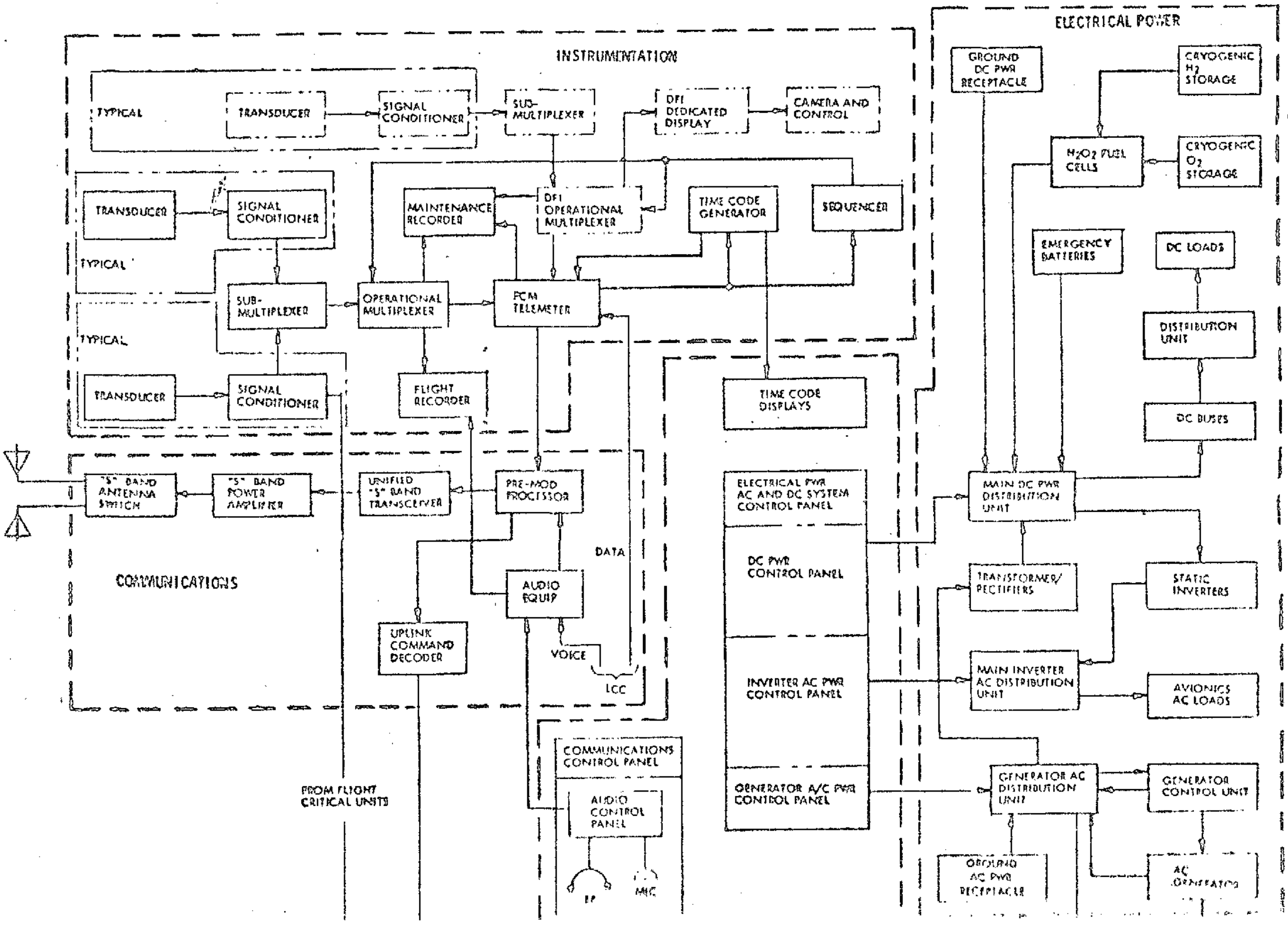
B-6

Fig. B-2 Spacecraft Avionics Detailed Equipment

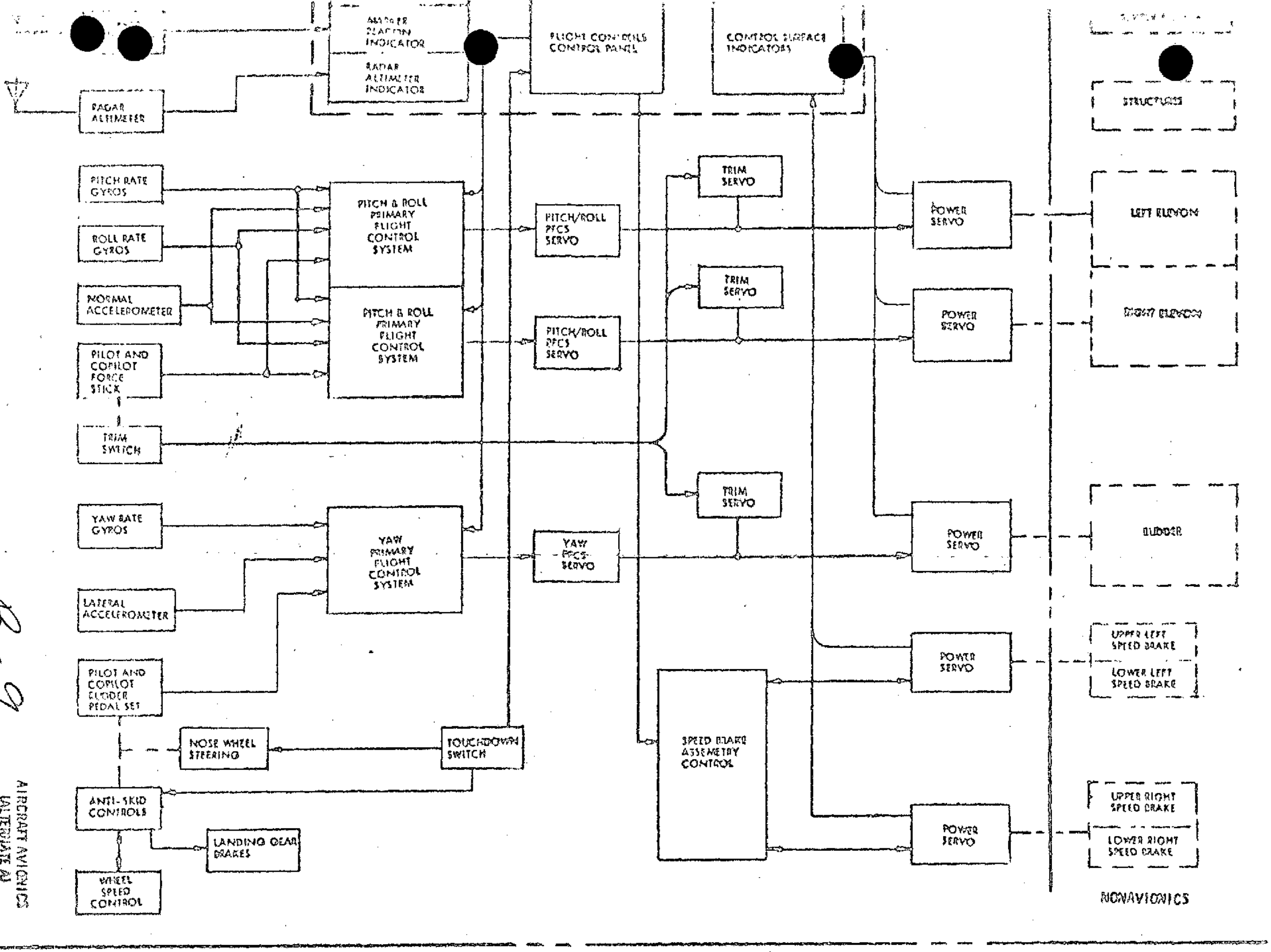

TYPICAL
NSDUCER
SIGNAL CONDITIONER
FLIGHT RECORDER
TIME CODE DISPLAYS
DISTRIBUTION UNIT
DC BUSES
"S" BAND ANTENNA SWITCH
"S" BAND POWER AMPLIFIER
UNIFIED "S" BAND TRANSCEIVER
PRE-MOD PROCESSOR
DATA
AUDIO EQUIP
UPLINK COMMAND DECODER
VOICE
LCC
COMMUNICATIONS
ELECTRICAL PWR AC AND DC SYSTEM CONTROL PANEL
DC PWR CONTROL PANEL
INVERTER AC PWR CONTROL PANEL
GENERATOR A/C PWR CONTROL PANEL
MAIN DC PWR DISTRIBUTION UNIT
TRANSFORMER/ RECTIFIERS
STATIC INVERTERS
MAIN INVERTER AC DISTRIBUTION UNIT
AVIONICS AC LOADS
GENERATOR AC DISTRIBUTION UNIT
GENERATOR CONTROL UNIT
GROUND AC PWR RECEPTACLE
AC GENERATOR
COMMUNICATIONS CONTROL PANEL
AUDIO CONTROL PANEL
EP
MIC
FROM FLIGHT CRITICAL UNITS
CAUTION AND WARNING CONTROL AND DISPLAY PANEL
DISPLAYS & CONTROLS
APU
DME (RENDEZVOUS)
ORBITAL RADAR ALTIMETER
HORIZON SENSOR
INERTIAL MEASUREMENT UNIT
STAR TRACKER
SIGNAL CONDITIONER
DISPLAY DATA
GUIDANCE NAVIGATION CONTROL COMPUTER
GUIDANCE/NAVIGATION AND CONTROL DISPLAYS AND CONTROL PANEL
DATA DISPLAYS
MODE SELECTION AND DATA ENTRY
ROCKET ENGINE PROPELLANT CIRCULATION
ROCKET ENGINE START AND RUN
S/C ENVIRONMENTAL AND LIFE SUPPORT CONTROL AND DISPLAY PANEL
ENVIRONMENTAL CONTROLS AND LIFE SUPPORT SYSTEMS
o WASTE MANAGEMENT
o FOOD STORAGE
o AIR LOCK CONTROL
HAND CONTROLLERS
ATTITUDE CONTROLS
TRANSLATION CONTROLS
PROPULSION SYSTEMS CONTROL AND DISPLAY
DISPLAY
CONTROLS
HYDRAULIC APU SUPPLY AND DISP

7-6


AIRCRAFT AVIONICS
ALTERNATE AJ
LANDING GEAR BRAKES
WHEEL SPEED CONTROL
CONTROL
POWER SERVO
LOWER RIGHT SPEED BRAKE
NONAVIONICS

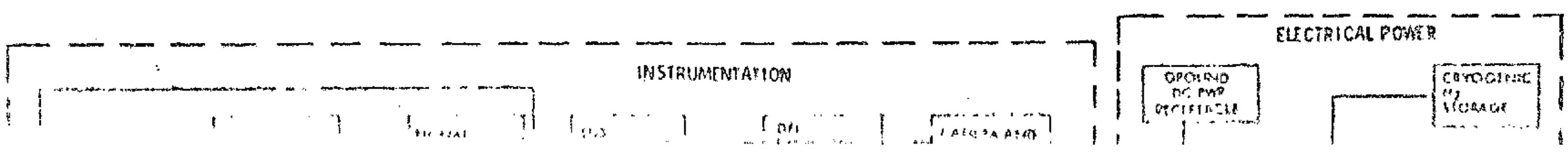

INSTRUMENTATION
TYPICAL
TRANSDUCER
SIGNAL CONDITIONER
SUB-MULTIPLEXER
DFI DEDICATED DISPLAY
CAMERA AND CONTROL
MAINTENANCE RECORDER
DFI OPERATIONAL MULTIPLEXER
TIME CODE GENERATOR
SEQUENCER
OPERATIONAL MULTIPLEXER
PCM TELEMETER
FLIGHT RECORDER
TIME CODE DISPLAYS
"S" BAND ANTENNA SWITCH
"S" BAND POWER AMPLIFIER
UNIFIED "S" BAND TRANSCEIVER
PRE-MOD PROCESSOR
DATA
COMMUNICATIONS
AUDIO EQUIP
UPLINK COMMAND DECODER
VOICE
LCC
COMMUNICATIONS CONTROL PANEL
AUDIO CONTROL PANEL
EP
MIC
FROM FLIGHT CRITICAL UNITS
ELECTRICAL PWR AC AND DC SYSTEM CONTROL PANEL
DC PWR CONTROL PANEL
INVERTER AC PWR CONTROL PANEL
GENERATOR A/C PWR CONTROL PANEL
ELECTRICAL POWER
GROUND DC PWR RECEPTACLE
CRYOGENIC H_2 STORAGE
H_2O_2 FUEL CELLS
CRYOGENIC O_2 STORAGE
EMERGENCY BATTERIES
DC LOADS
DISTRIBUTION UNIT
DC BUSES
MAIN DC PWR DISTRIBUTION UNIT
TRANSFORMER/RECTIFIERS
STATIC INVERTERS
MAIN INVERTER AC DISTRIBUTION UNIT
AVIONICS AC LOADS
GENERATOR AC DISTRIBUTION UNIT
GENERATOR CONTROL UNIT
GROUND AC PWR RECEPTACLE
AC GENERATOR

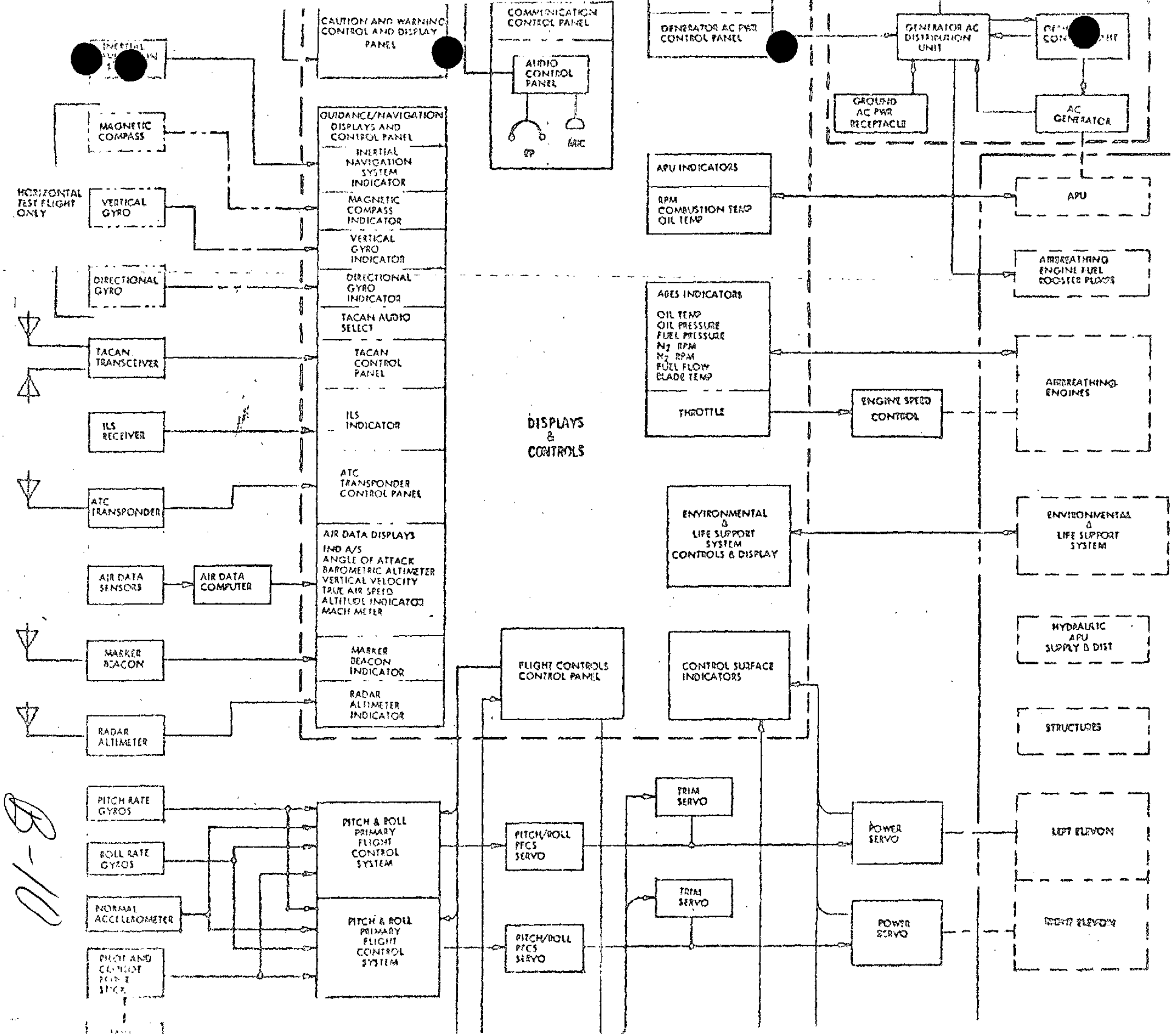
B-7



B-9

AIRCRAFT AVIONICS (ALTERNATE A)

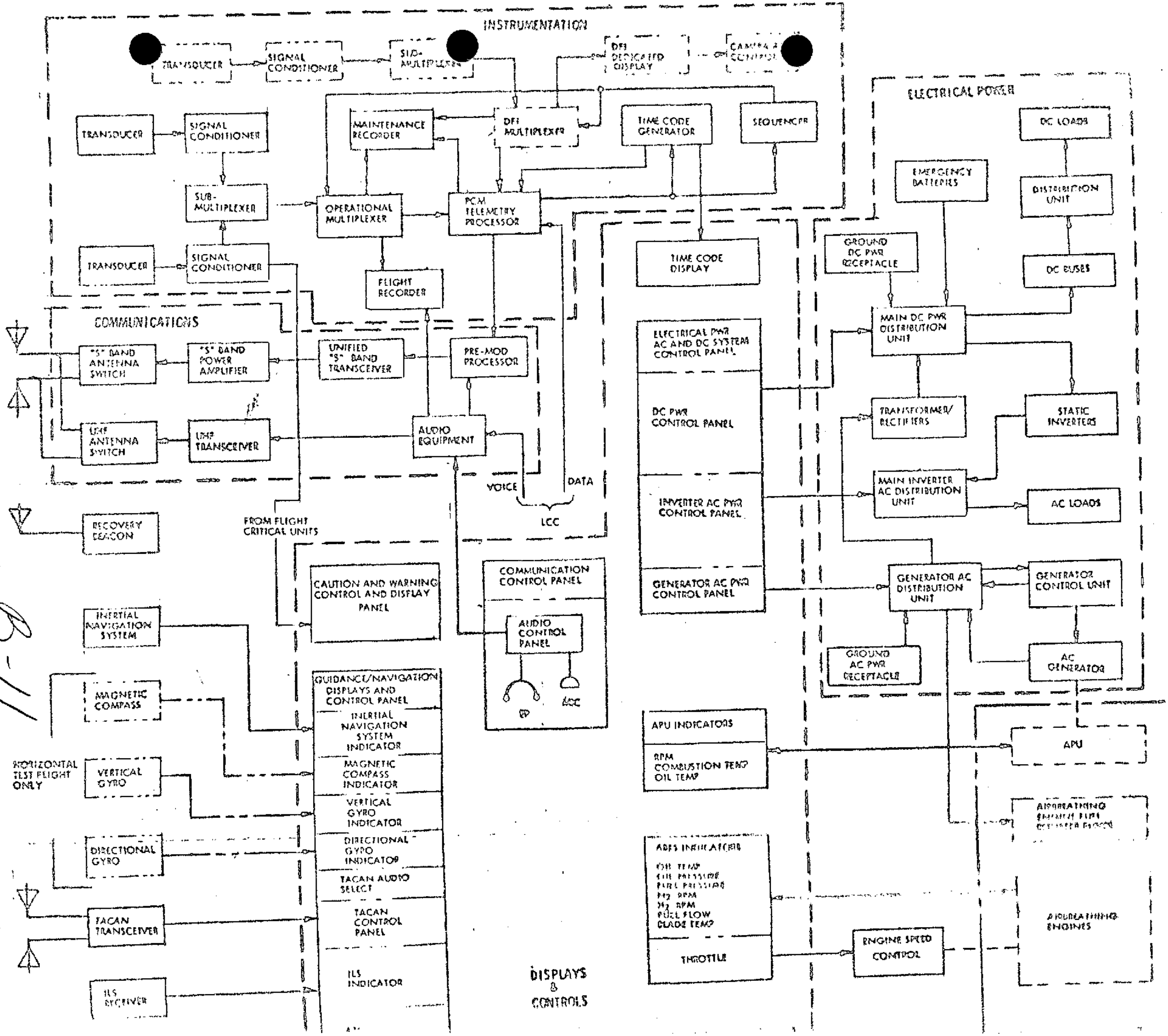




CAUTION AND WARNING CONTROL AND DISPLAY PANEL
COMMUNICATION CONTROL PANEL
AUDIO CONTROL PANEL
MIC
GENERATOR AC PWR CONTROL PANEL
GENERATOR AC DISTRIBUTION UNIT
GROUND AC PWR RECEPTACLE
AC GENERATOR
MAGNETIC COMPASS
GUIDANCE/NAVIGATION DISPLAYS AND CONTROL PANEL
INERTIAL NAVIGATION SYSTEM INDICATOR
HORIZONTAL TEST FLIGHT ONLY
VERTICAL GYRO
MAGNETIC COMPASS INDICATOR
VERTICAL GYRO INDICATOR
DIRECTIONAL GYRO
DIRECTIONAL GYRO INDICATOR
TACAN AUDIO SELECT
TACAN TRANSCEIVER
TACAN CONTROL PANEL
ILS RECEIVER
ILS INDICATOR
ATC TRANSPONDER
ATC TRANSPONDER CONTROL PANEL
AIR DATA SENSORS
AIR DATA COMPUTER
AIR DATA DISPLAYS
IND A/S
ANGLE OF ATTACK
BAROMETRIC ALTIMETER
VERTICAL VELOCITY
TRUE AIR SPEED
ALTITUDE INDICATOR
MACH METER
MARKER BEACON
MARKER BEACON INDICATOR
RADAR ALTIMETER
RADAR ALTIMETER INDICATOR
APU INDICATORS
RPM
COMBUSTION TEMP
OIL TEMP
APU
AIRBREATHING ENGINE FUEL BOOSTER PUMPS
ABES INDICATORS
OIL TEMP
OIL PRESSURE
FUEL PRESSURE
N2 RPM
N2 RPM
FUEL FLOW
BLADE TEMP
THROTTLE
ENGINE SPEED CONTROL
AIRBREATHING ENGINES
DISPLAYS & CONTROLS
ENVIRONMENTAL & LIFE SUPPORT SYSTEM CONTROLS & DISPLAY
ENVIRONMENTAL & LIFE SUPPORT SYSTEM
HYDRAULIC APU SUPPLY & DIST
FLIGHT CONTROLS CONTROL PANEL
CONTROL SURFACE INDICATORS
STRUCTURES
PITCH RATE GYROS
ROLL RATE GYROS
NORMAL ACCELEROMETER
PILOT AND COPILOT FORCE STICK
PITCH & ROLL PRIMARY FLIGHT CONTROL SYSTEM
PITCH & ROLL PRIMARY FLIGHT CONTROL SYSTEM
PITCH/ROLL PFCS SERVO
PITCH/ROLL PFCS SERVO
TRIM SERVO
TRIM SERVO
POWER SERVO
POWER SERVO
LEFT ELEVON
RIGHT ELEVON

11-B

11-8



INSTRUMENTATION
TRANSDUCER
SIGNAL CONDITIONER
SUB-MULTIPLEXER
DFI DEDICATED DISPLAY
CAMERA & CONTROL
TRANSDUCER
SIGNAL CONDITIONER
MAINTENANCE RECORDER
DFI MULTIPLEXER
TIME CODE GENERATOR
SEQUENCER
SUB-MULTIPLEXER
OPERATIONAL MULTIPLEXER
PCM TELEMETRY PROCESSOR
TRANSDUCER
SIGNAL CONDITIONER
FLIGHT RECORDER
TIME CODE DISPLAY
ELECTRICAL POWER
DC LOADS
EMERGENCY BATTERIES
DISTRIBUTION UNIT
GROUND DC PWR RECEPTACLE
DC BUSES
MAIN DC PWR DISTRIBUTION UNIT
COMMUNICATIONS
"S" BAND ANTENNA SWITCH
"S" BAND POWER AMPLIFIER
UNIFIED "S" BAND TRANSCEIVER
PRE-MOD PROCESSOR
UHF ANTENNA SWITCH
UHF TRANSCEIVER
AUDIO EQUIPMENT
VOICE
DATA
LCC
ELECTRICAL PWR AC AND DC SYSTEM CONTROL PANEL
DC PWR CONTROL PANEL
INVERTER AC PWR CONTROL PANEL
GENERATOR AC PWR CONTROL PANEL
TRANSFORMER/RECTIFIERS
STATIC INVERTERS
MAIN INVERTER AC DISTRIBUTION UNIT
AC LOADS
GENERATOR AC DISTRIBUTION UNIT
GENERATOR CONTROL UNIT
GROUND AC PWR RECEPTACLE
AC GENERATOR
RECOVERY BEACON
FROM FLIGHT CRITICAL UNITS
CAUTION AND WARNING CONTROL AND DISPLAY PANEL
COMMUNICATION CONTROL PANEL
AUDIO CONTROL PANEL
INERTIAL NAVIGATION SYSTEM
MAGNETIC COMPASS
GUIDANCE/NAVIGATION DISPLAYS AND CONTROL PANEL
INERTIAL NAVIGATION SYSTEM INDICATOR
MAGNETIC COMPASS INDICATOR
HORIZONTAL TEST FLIGHT ONLY
VERTICAL GYRO
VERTICAL GYRO INDICATOR
DIRECTIONAL GYRO
DIRECTIONAL GYRO INDICATOR
TACAN AUDIO SELECT
TACAN TRANSCEIVER
TACAN CONTROL PANEL
ILS RECEIVER
ILS INDICATOR
APU INDICATORS
RPM
COMBUSTION TEMP
OIL TEMP
APU
ABES INDICATORS
OIL TEMP
OIL PRESSURE
FUEL PRESSURE
FUEL FLOW
BLADE TEMP
THROTTLE
ENGINE SPEED CONTROL
AIRBREATHING ENGINES
DISPLAYS & CONTROLS

Appendix C
ALTERNATE B AVIONICS APPROACH

Alternate B for the space shuttle orbiter avionics provides substantial improvement over the Alternate A concept. This approach permits a single hardware element or subsystem to satisfy the functional requirements in both the spacecraft (exoatmospheric) and aircraft (endoatmospheric) flight regime. The second unique feature of this concept is the inclusion of a passive (monitoring only) data management subsystem to assist the crew in orbital operations and to reduce between-flight turnaround. Figure C-1 is a block diagram of the Alternate B concept.

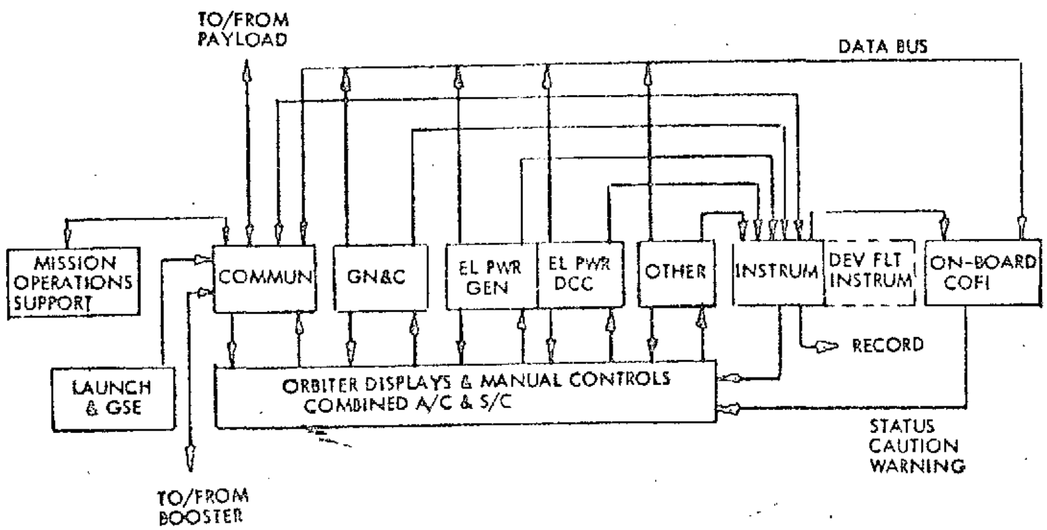


Fig. C-1 Orbiter Avionics Study – Alternate B

C.1 EQUIPMENT

C.1.1 Guidance, Navigation, and Control

The equipment employed in Alternate B is identical to Alternate A (except that the rate gyros and accelerometers for atmospheric flight are deleted). Only the system utilization and interconnections differ. Alternate B takes advantage of the option which permits dual usage (aircraft/spacecraft) of a single hardware element by providing an automatic G&N crew backup mode during atmospheric flight from the Inertial Measurement Unit (IMU) and associated sensors through the autopilot. In Alternate A this interconnect was provided by the crew using display indicators.

C.1.2 Electrical Power Subsystem (EPS)

The EPS is identical in Alternate B, and in the selected baseline (Alternate C), only the interfaces to the data management subsystem change.

C.1.3 Communication Subsystem

The communications hardware is identical in Alternates B and the Baseline; only the interfaces to the data management subsystem change.

C.1.4 Instrumentation Subsystem

The Alternate B instrumentation (Fig. C-2) is essentially identical to Alternate A except for deletion of common airplane equipments. Dedicated hard-wired annunicators and displays are used for all safety of flight information. A multiplex data-gathering subsystem composed of C-5A MADAR analog multiplexers for mission-critical (operational) data, with added units to provide DFI data as an integrated add-on, is used to collect all data except for those non-avionic hardline interfaces required for ground checkout and fault isolation, including fueling operations.

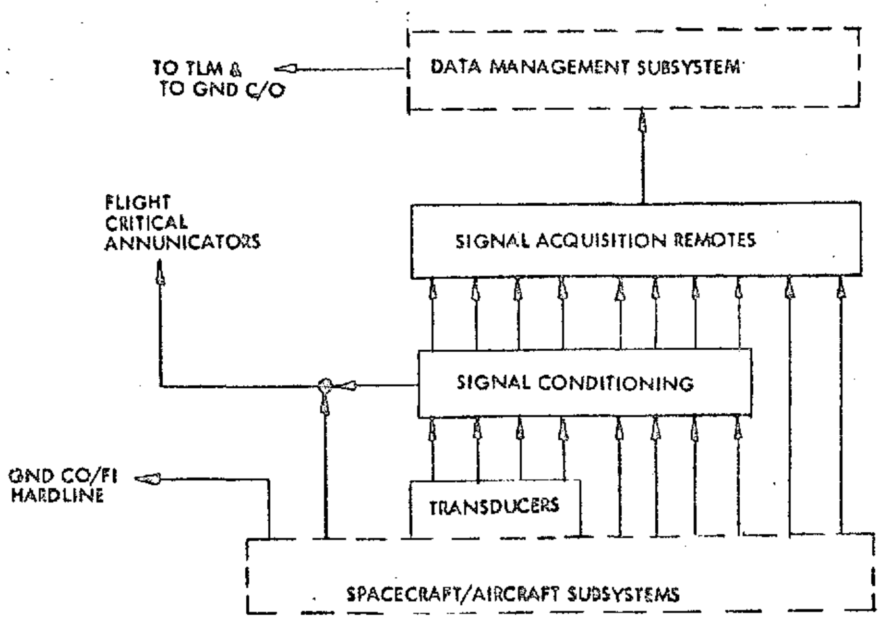


Fig. C-2 Instrumentation — Alternate Orbiter B

A total of 2700 instrumentation points are required for operational and DFI data. Three SAR complexes are used to meet the data acquisition requirements (3840 point capacity) into the DMS. Formatted data from the DMS is input to the communications subsystem for TLM transmission or via hardline interface to the GSE for checkout in all between-flight phases. One-third of the SARs are for DFI and are removed after the development phase is complete, leaving a 1960 point operational instrumentation subsystem, exclusive of SOF and non-avionic hard-wired data points.

C.1.5 Data Management Subsystem

Data management for both airplane and spacecraft subsystems are provided by one system of hardware (Fig. C-3).

Input data to the DMS is from the instrumentation subsystem. The Data Management System (Alternate B) is configured for on-board checkout and is based around the MADAR system as used on the C-5A aircraft. Briefly, the system, under control of

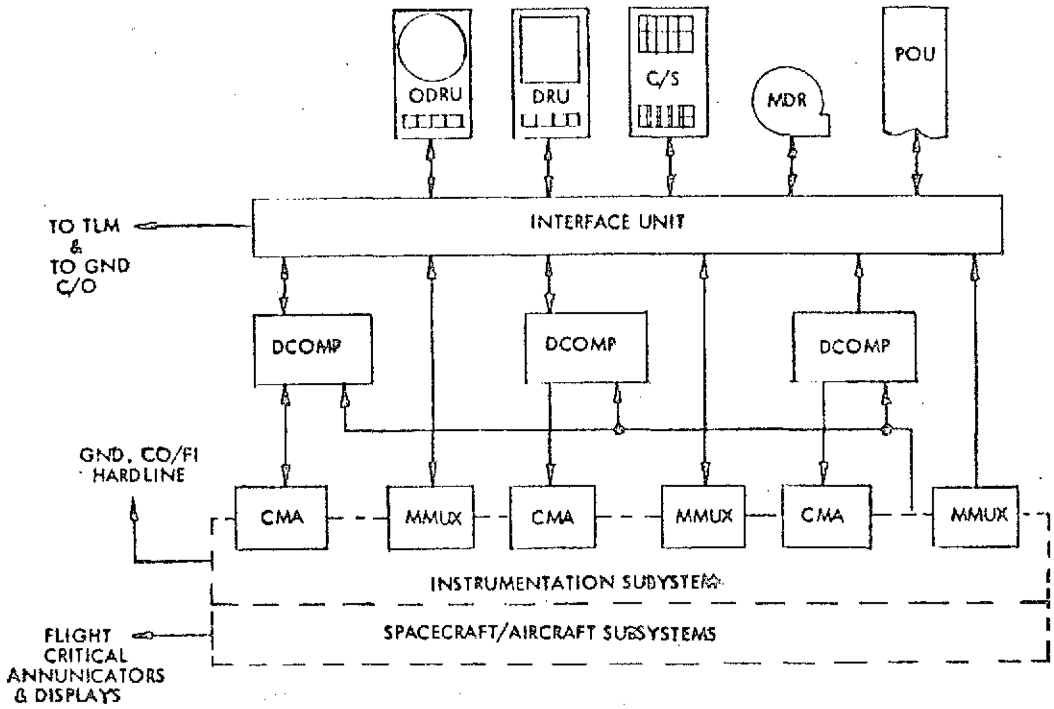


Fig. C-3 Data Management – Alternative B Orbiter

a general purpose digital computer (DCOMP), continuously scans system parameters through its central multiplex adapter (CMA) and automatic remote signal acquisition units (SAR-A). The BITE outputs are sensed and signal measurements are compared to limits and exceptions are recorded on magnetic tape by a maintenance data recorder (MDR) and/or on a printer (POU). In addition, a data acquisition system under manual control of the manual multiplexer (MMUX) and manual signal acquisition units (SAR-M) can select parameters for display on an oscilloscope or digital voltmeter. Controls and displays are contained in the MADAR manual display/control unit (C/D group).

The C/D group also contains a data retrieval unit (DRU) which optically projects on a rear projection screen a selected (one of 10,000) individual frame, utilizing 16 mm film as a source. Aside from the displayed information, each film frame includes frame selection codes and data point selection codes to aid in presequenced trouble-shooting data under manual control.

The system is programmed to test all subsystems (without interrupting operation), down to a line replaceable unit (LRU) level, in normal functioning modes. The manual displays and controls allow an operator, under guidance of the projection display, to isolate problems in subsystems to LRUs and to print and magnetically record this information.

A new design interface unit is required to control the digital inputs from the MADAR Digital Computers (DCOMP), wide-band analog from SAR-manual multiplex units which provide line analog data to the oscilloscope/voltmeter for onboard-manual troubleshooting, and for GSE/TLM interface control.

C.1.6 Control Display (C/D) Summary Comparison

A correlation in the C/D requirements for all alternatives is provided in Table C-1.

C.2 WEIGHT STATEMENT

The Alternate B weight statement relative to the baseline is provided in Table C-2.

Table C-1

PILOTAGE C&D PANEL AREA AND WEIGHT COMPARISON

	AIRCRAFT			SPACECRAFT			SPACE SHUTTLE			
CREW SIZE	4 MEN	3 MEN	2 MEN	2 MEN	2 MEN	3 MEN	3 MEN	3 MEN	3 MEN	2 MEN
VEHICLE	C-5A	1011	S-3A	GEMINI	LM	CM	ALT A	ALT B	ALT C (TEST)	ALT C (OPS)
TOTAL PANEL AREA* - SQ FT	57	26.75	9.3	19.4	16	27.1	28.1	23.9	21.8	13.5
TOTAL C AND D WEIGHT - LBS	790	313	400	260	290	344	878	810	705/988	717

	PILOT/COPILOT STA		SYS ENGR STA		CRT'S		SUMMARY ·	
	WEIGHT (LB)	AREA (SQ FT)	WEIGHT (LB)	AREA (SQ FT)	NO	WEIGHT (LB)	WEIGHT (LB)	AREA (SQ FT)
ALTERNATE A	602	13.5	276	14.6	0	0	878	28.1
ALTERNATE B	602	13.5	208	10.4	1	63	810	23.9
ALTERNATE C_T (HORZ/VERT TEST)	514/797	13.5	189	8.3	1	63	705/988	21.8
ALTERNATE C_O (OPERATIONS)	528	13.5	0	0	3	189	717	13.5

D05781 (1)

Table C-2

ALTERNATE B DELTA WEIGHTS

	Baseline Reference						
	Additions			Deletions			
Subsystem	Equip	Quan	Total Wt (lb)	Equip	Quan	Total Wt (lb)	Δ Wt
Guidance, Navigation, and Control	Digital Computer GNC Signal Condr	1 2	126 160	Navig Data Repeater Converter	2	110	
Subtotal			286			110	+176
Communications							0
Electrical Power							0
Displays and Controls	All	--	810	All	--	717	+ 93
Data Mgmt (MADAR)	MADAR [your list]		326	All	--	665	-339
Instrumentation							0
Net Total							- 70

APPENDIX D

DETAILED EQUIPMENT LIST

GUID/NAV/CONTROL EQUIPMENT ITEM	USER: MIL (M) COM'L (C)	PROGRAM APPLICATION	ITEM TAG DESIGNATOR	MANUFACTURER	CHARACTERISTICS MTBF (KHR)	POWER (WATT)	WEIGHT (LB)	VOLUME (IN^3)	REMARKS
Digital Computer	M	S3A	1832	Univac	2	625	126		32K Memory
Inertial Meas. Unit	C	747	Carousel IV	Delco	4	400	53		Modify for space use or use Carousel IV B (TIIIC System)
Star Tracker		Skylab	ATM	Bendix	6	25	77		
Horizon Sensor	M	Agena	13-166	Barnes	12	25	25		
Navig. Data Repeater/ Converter	M	S3A	059A	Bendix	3.71	251	55		
TVC Electronics	M	Agena			>5	20W	30		Same circuits and packaging techniques. New package design.
ACPS Electronics	M	AF-P467			>5	30W	50		
Main Eng. Gimbal Actuator	NASA	Saturn					50		
Compass Coupler	C	L1011	2591201	Sperry	25		17.2		
Compass Controller	C		2594911		6		2.0		
Flux Valves	C		2575570		50		4.0		
Magnetic Compensator	C		2591200		50		1.8		
Directional Gyros	C		2594401		2.5		29		
Vertical Gyros	C	L1011	2593742	Sperry	2.5		29		

GUID/NAV/CONTROL

EQUIPMENT ITEM	USER: MIL (M) COM'L (C)	PROGRAM APPLICATION	ITEM TAG DESIGNATOR	MANUFACTURER	CHARACTERISTICS MTBF (KHR)	POWER (WATT)	WEIGHT (LB)	VOLUME (IN^3)	REMARKS
Rudder Servos	M	C5A	4Y91577	Bertea	20		95.2		20% Mod.
Elevon Servos	M	C5A	4Y91013	Bertea	5		376		20% Mod.
PFCS Servos	NASA	STCL	697660	GELAC	4		50		New
Trim/Backup Servos	M	C141	544268	Air Research	10		10		
Speed Brake Servos	C	New		New			10		
Elevon PFCS Computer	C	L1011	672293	Collins	0.9		92.4		50% Mod.
Rudder PFCS Computer	C	L1011	672293	Collins	1.7		46.2		50% Mod
Central Air Data Computer	M	S3A	A/N-5	Bendix			58.6	1026	C.P., Bite, VAST/GSE 9 x 19 x 6
Air Data Sensor Assy	M	YF-12		Rosemount					
Pitch Rate Gyros	C	L1011	672300	Collins	1.25		7.4	37.1	
Roll Rate Gyros	C	L1011	672300	Collins	1.25		7.4	37.1	
Yaw Rate Gyros	C	L1011	672300	Collins	1.25		7.4	37.1	
Normal Accelerometer	C	L1011	672302	Collins	25		4.1	71.3	
Lateral Accelerometer	C	L1011	672301	Collins	25		4.5	85.2	
Longitudinal Accel	C	L1011			25				
Pitch AFFDS	C	L1011	672314	Collins	2		55.4		
Roll AFFDS	C	L1011	672315	Collins	2.2		55.8		
Engine Speed Controls	M	AH56	C1033	NASH	1.25		60		
Speed Control Computer	C	L1011	672294	Collins	4		26.8		
Anti-Skid Control) Touchdown Switch) Wheel Speed Sensor)	M	New		New			30		

ELECTRICAL POWER SYSTEM EQUIPMENT ITEM	USER: MIL (M) COM'L (C)	PROGRAM APPLICATION	ITEM TAG DESIGNATOR	MANUFACTURER	CHARACTERISTICS MTBF (KHR)	POWER (WATT)	WEIGHT (LB)	VOLUME (IN^3)	REMARKS
Fuel Cell (MK I)	M	Space Shuttle	New	Pratt & Whitney	8.9		298	8100	C.P., GSE, New 15 x 15 x 36
Fuel Cell (MK II)	M	"	New	"	8.9		298	8100	C.P., GSE, New
O_2 Cryogenic Tank & CTL*	M	AAP	AAP,33"Dia	Bendix	200	120	138		Heat, GSE
H_2 Cryogenic Tank & CTL	M	AAP	AAP,39"Dia	Bendix	200	200	97		Heat, GSE
Ni Cd Battery (Emerg)	M	Agena	Type XI	Eagle Picher	45		62	931	GSE 19 x 7 x 7
AC Generator	C	S3A	4QN					490	GSE, 5% Mod. 10 x 7 x 7
Static Inverter 3Ø 400 Hz	M	Apollo	28V5200Y 914F53-1	Wagner Elect.	61		40	396	C.P., GSE 11 x 6 x 6
Transformer/Rectifier	M	P3C		*	110		18	250	F.A., VAST, 5% Mod.
Generator Control	M	S3A	AVZ-86	Westinghouse	100		11.5	336	F.A., GSE, 5% Mod.
DC Bus							595		
AC Bus							290		
D.C. Distribution Unit									
A.C. Distribution Unit									

* Includes Remote Power Controllers and Remote Controlled Circuit Breakers

COMMUNICATIONS AND TRACKING EQUIPMENT ITEM	USER: MIL (M) COM'L (C)	PROGRAM APPLICATION	ITEM TAG DESIGNATOR	MANUFACTURER	CHARACTERISTICS MTBF (KHR)	POWER (WATT)	WEIGHT (LB)	VOLUME (IN^3)	REMARKS
Audio Panel	M	S3A		Instr. System		790	77	300	C.P., or FA, VAST/GSE
S-Band Pwr Amplifier	M	S3A	Apollo Block II	Collins	.9993	180	32	729	C.P.
Unified S-Band Equip.	M	S3A	↓	Collins	.9979	37	38	1260	C.P.
Pre-Mod Processor	M	S3A	↓	Collins	.9997	12.5	11.3	300	C.P.
Up-Data Link	M	Apollo	Apollo Block II	Motor-ola	.996	10	224	1080	C.P.
S-Band Antenna Switch		Apollo					3		
S-Band Antenna		Apollo					5		
UHF Transceiver	C	S3A	4K90008	Collins	2	30	32.5	1026	F.A., Bite, VAST/GSE
UHF Antenna Selector	C	S3A		Collins			1.4		
UHF Antenna	C	S3A		Collins			2.3		
ATC Transponder	M	C5A	621A-6	Collins			15		
ATC Transponder Ant.	M	C5A	5-65-5366-11L	Sensor			3		
VHF Recovery Beacon	M	Apollo	Apollo Block II	RCA			13		
VHF Recovery Antenna	M	Apollo	Apollo Block II				1		

COMMUNICATIONS AND TRACKING EQUIPMENT ITEM	USER: MIL (M) COM'L (C)	PROGRAM APPLICATION	ITEM TAG DESIGNATOR	MANUFACTURER	CHARACTERISTICS				REMARKS
					MTBF (KHR)	POWER (WATT)	WEIGHT (LB)	VOLUME (IN^3)	
Tacan Transceiver	M	S3A					37		
ILS Receiver	C	C5A	ILS-70	Collins	1.4		10		
Radar Altimeter	M	S3A	APN-201	Hoffman	1.3		9.9	228	Bite, VAST/GSE 10 x 6 x 3.8
AILS Receiver	M	C-SCAN							Scanning beam
Orbit Altimeter		SKYLAB				125W	45		
Precision Ranging Sys		CIRIS				120W	25		

DATA MANAGEMENT EQUIPMENT ITEM	USER: MIL (M) COM'L (C)	PROGRAM APPLICATION	ITEM TAG DESIGNATOR	MANUFACTURER	CHARACTERISTICS MTBF (KHR)	 POWER (WATT)	 WEIGHT (LB)	 VOLUME (IN^3)	REMARKS
Gen. Purp. Dig.Computer	M	S3A	1832	Univac	2	1800	255	18144	C.P., or F.A, Bite, VAST/GSE
Drum Storage Unit	M	S3A	MU576/AYS	IBM ECHO	1.2	390	70	2765	C.P., or F.A, Bite, VAST/GSE
Dig. Mag. Tape Unit	M	S3A	RD 348/Ash	Science	4.8	60	20	933	C.P., or F.A, Bite, VAST/GSE
Communication SIU	M	S3A	TBD			750	40	864	C.P., or F.A. 50% Mod.
Instrumentation SIU	M	S3A	TBD			50	10	346	C.P., 50% Mod.
G&N SIU	M	S3A	TBD			750	80	864	C.P., or F.A. 50% Mod.
ECL's SIU	M	S3A	TPD			200	15	346	C.P., 50% Mod.
Elect. Pwr SIU	M	S3A	TPD			200	15	346	C.P.
Booster SIU	M	S3A	TPD			500	30	864	C.P., Bite, VAST/GSE
GSE/LCC SIU	M	S3A	TPD				30		
Display Gen Unit	M	S3A	AN/ASA 82	Loral	.2	1200	80		C.P.

CONTROLS & DISPLAYS EQUIPMENT ITEM	USER: MIL (M) COM'L (C)	PROGRAM APPLICATION	ITEM TAG DESIGNATOR	MANUFACTURER	CHARACTERISTICS MTBF (KHR)	 POWER (WATT)	 WEIGHT (LB)	 VOLUME (IN^3)	REMARKS
MAIN INSTRUMENT PANEL									
Flight Attitude Indicator	M	C-3A	OD-59/A	Bendix			6.94	201.0	
Horizontal Situation Indicator	M	S-3A	OD-59/A	Bendix			8.0	202.5	
Aero Surface Indicator	C	L-1011	672445	Collins/LS			2.25	80.0	
AFCS Modes	C	L-1011	672299	"			2.75	85.75	
AFCS Warning	C	L-1011	672295	"			1.88	56.0	
Instr Warning	C	L-1011	672297	"			1.88	56.0	
Autopilot/Land	C	L-1011	672309-13	"			20.2	228.0	
Meter-Airspeed/Mach/	M	C-5A	2594466	Sperry			8.2	112.0	
Meter-Altitude/Vertical Speed	M	C-5A	2594463	"			8.2	112.0	
Altimeter	M	S-3A	AN/APN-201	Hoffman			3.0	38.0	
True Airspeed Indicator	M	S-3A	AN/AYN-5	Bendix			1.5	43.7	
Altitude Indicator	M	S-3A	"				1.5	43.7	
Multi-Purpose Keyboard	M	S-3A	AN/ASQ-147	Hartman			22	1072.5	
Engine Gimbal Override	NASA	M	--	N.A.R.			1.95	73.5	
RCS Control Override	C	--	Off-Shelf Parts	LMSC			2.75	128.0	
Main/OMS Override	C	--	" "	"			1.20	64.0	
Tank Jettison Override	C	--	" "	"			1.90	40.0	

CONTROLS & DISPLAYS EQUIPMENT ITEM	USER: MIL (M) COM'L (C)	PROGRAM APPLICATION	ITEM TAG DESIGNATOR	MANUFACTURER	CHARACTERISTICS MTBF (KHR)	POWER (WATT)	WEIGHT (LB)	VOLUME (IN^3)	REMARKS
Abort	C	--	Off-Shelf Parts	LMSC			1.10	32.75	
Instr Brightness Control	C	--	" "	"			0.75	6.25	
Caution and Warning Test Panel	C	--	" "	"			1.20	22.5	
Booster Status Panel	C	--	" "	"			2.90	45.0	
Master Sys Caution and Warning	C	--	" "	"			2.97	48.6	
Multi-Function CRT (Flight Mgnt)	M	S-3A	AN/ASA-82	Loral			63.0	5669.5	
Flight Mode Indicator	C	--	Off-Shelf Parts	LMSC			2.80	144.0	
Multi-Function CRT (Subsystems)	M	S-3A	AN/ASA-82	Loral			63.0	5669.5	
Engine/Propulsion Displays	C	--	105030	Astron-autics			8.5, 6.2, 1.2	120.4	
Engine/Propulsion Displays	C	--	105028	"			6.2	112.2	
Mode Select (Main/OMS/ABES)	C	--	Off-Shelf Parts	LMSC			1.1	25.0	
Mode Select (RCS/APU)	C		" "	"			1.1	25.0	
Area Nav-Growth	C	L-1011	144000, 142000	Astron-autics			(63)**	(1147.5)	
Throttle Quads, Speed Brake, and Rudder Trim Control	M	C-5A	C1033, Et.Al	Nash, Et Al			38	5230.0	

** Not Included

CONTROLS & DISPLAYS

EQUIPMENT ITEM	USER: MIL (M) COM'L (C)	PROGRAM APPLICATION	ITEM TAG DESIGNATOR	MANUFACTURER	CHARACTERISTICS: MTBF (KHR)	POWER (WATT)	WEIGHT (LB)	VOLUME (IN^3)	REMARKS
ABES Controls	M	C-5A	--	--			1.1	16.0	
G&N Panel	C	--	Off-Shelf Parts	LMSC			2.6	64.0	
Landing Gear Controls	M	C-5A	--	--			4.2	128.0	
Emer Landing Gear Extension Controls	M	C-5A	--	--			1.5	76.0	
ATC Panel	M	C-5A	Arinc 572	Collins			2.0	48.0	
Communications Panel	M	S-3A	LS-601/ AI	Instr. Systems			14	3456	
EC/LS Panel	C	--	Off-Shelf Parts	LMSC			3.9	64.0	
Engine Start	M	C-5A	--	--			2.9	64.0	
Attitude Hand Controller	NASA	CM	--	N.A.R.			8	32.5	
Translation Controller	C	--	--	--			8.2	34.1	
C & W Annunicators	C	--	OFF-Shelf Parts	LMSC			5.6	576	

CONTROLS & DISPLAYS EQUIPMENT ITEM	USER: MIL (M) COM'L (C)	PROGRAM APPLICATION	ITEM TAG DESIGNATOR	MANUFACTURER	CHARACTERISTICS MTBF (KHR)	POWER (WATT)	WEIGHT (LB)	VOLUME (IN^3)	REMARKS
EYEBROW/OVERHEAD PANEL									
Engine Fire Control Panels	C	L-1011	--	CAL-LAC			8.0	380.0	
EC/LS Gas Supply Override Valves	C	--	Off-Shelf Parts	LMSC			22.3	1710.0	
Elect. Pwr Generation and Dist	C	--	" "	"			6.9	1140.0	
Elevon Disable	C	--	Off-Shelf Parts	LMSC			2.5	40.5	
Rudder Disable	C	--	Off-Shelf Parts	LMSC			2.8	40.5	
SAS, Pitch, ATS, and Trim Emer. Controls	C	L-1011	--	CAL-LAC			3.9	54.0	
Antiskid Controls	C	L-1011	--	CAL-LAC			2.3	27.0	
Sensor Heat Controls	C	L-1011	--	CAL-LAC			1.9	20.3	
PFCS Mon., Rudder & Elevon Emer. Controls	C	L-1011	--	CAL-LAC			3.2	54.0	
Rudder Limiter	C	L-1011		CAL-LAC			2.1	45.0	
Engine Start	C	L-1011	--	CAL-LAC			2.4	54.0	
APU Engine Controls	C	--	Off-Shelf Parts	LMSC			4.7	216.4	
Cabin Lights	C	L-1011	--	CAL-LAC			2.0	15.0	
Mission Timer	NASA	CM or LM	--	N.A.R			2.5	22.5	
Event Timer	NASA	CM or LM	--	N.A.R			1.9	18.0	
Exterior Lights	C	L-1011	--	CAL-LAC			2.8	45.0	

INSTRUMENTATION EQUIPMENT ITEM	USER: MIL (M) COM'L (C)	PROGRAM APPLICATION	ITEM TAG DESIGNATOR	MANUFACTURER	CHARACTERISTICS MTBF (KHR)	POWER (WATT)	WEIGHT (LB)	VOLUME (IN^3)	REMARKS
Transducers *	M								
Signal Conditioners	M								
FM Wideband Recorder	M	P3V, L1011	417	LEC	4	200	39	2660	F.A., Bite, MADAR/AIDS
Flight Data Recorder	M	C5A	CDPIR				48		
Time Code Generator	C	A/C	8521	Systron Donner	5	28	15		F.A., Bite
Film Camera	M	A/C					7		
P.C.M. Telemeter Equip	M	Apollo	Apollo Block II	Collins	.997	24	44	1274	F.A.

*See following pages

INSTRUMENTATION EQUIPMENT ITEM	DEV. STATUS*	TYPE	ITEM TAG DESIGNATOR	MANUFACTURER	CHARACTERISTICS WORST CASE ENVIR. (FIG.	ACCURACY (%)	WEIGHT (OZ)	VOLUME (IN^3)	COST	REQD. SUPPORT EQUIPMENT	
TEMPERATURE											
0 to +200 Deg. F	F	Thermo-couple	1A59741	Douglas	5	1.2	5	4	$1,000		
0 to +160			1A66215		4	1.2	5	4	1,000	DC Ampl Bridge	1A82395 1A82274
-400 to +500			1A67862		2	4.7	5	4	1,000		
-400 to +900			1A67863		2	4.7	6	5	1,500		
- 60 to +320			1B34472		2	1.2	5	4	1,000		
-400 to +100			1B34473		1	4.7	5	4	1,000		
-400 to -200			1B37878		2	4.7	5	4	1,000		
0 to 2000			1B64968		1	4.7	6	5	1,500		
-400 to +200			NA527315	N A	1	4.7	5	4	1,000	Bridge	V7750463
0 to +1800			NA527323		2	4.7	5	4	1,000		
-400 to +100			NA527441		1	4.7	5	4	1,000		
-100 to +2000			60B67223	Boeing	1	1.2	6	5	1,500	DC Ampl Zone Box	60B73113 60B67608
-100 to +500			60B67240		1	4.7	6	5	1,500		
-100 to +3200			60B67536		5	4.7	6	5	1,500		
-200 to +700			60B67609		3	4.7	6	5	1,500		
-300 to +200			60B70768		3	4.7	6	5	1,500		
-200 to +3200			60B71141		1	4.7	6	5	1,500		
-300 to +500			60B72067		2	4.7	5	4	1,000		
-400 to +100			60B72099		2	4.7	5	4	1,000		
0 to +4000	D										
PRESSURE											
0 to 400 } PSIA 1500 to 4500	F	Strain Gage	1B31356	Douglas	2	4.75	10	3	1,000	—	

*F - Flight Proven
M - Modification Required
D - Development Required

INSTRUMENTATION EQUIPMENT ITEM		DEV. STATUS*	TYPE	ITEM TAG DESIGNATOR	MANUFACTURER	CHARACTERISTICS WORST CASE ENVIR. (FIG.	ACCURACY (%)	WEIGHT (OZ)	VOLUME (IN^3)	COST	REQD. SUPPORT EQUIPMENT	
PRESSURE (Continued)												
0 to +400	PSIA	F	Strain Gage	1B31413	Douglas	3	4.75	10	3	$1,000	—	
0 to +3500				1B40242		3	4.75	10	3	1,000	—	
0 to +50				1B43320		3	2	5	2	500	--	
0 to +2000				NA527412	N A	2	4	10	3	1,000	—	
0 to +2500				60B72075	Boeing	1	4.75	10	3	1,000	DC Ampl	60B73112
0 to +3000				60B72080		2	4.75	10	3	1,000		
0 to +500				60B72091		2	4.75	10	3	1,000		
0 to +3500				60B72178		2	4.75	10	3	1,000		
0 to +45				60B72199		1	2	10	3	1,000		
0 to +2000				60B72200		2	4.7	10	3	1,000		
0 to +10				6063252-10	Poseidon	4	2	8	2	1,000		
0 to +15				6063252-15		4	2	.8	2	1,000		
0 to +0.5			Pizo-electric	PS2-12162		5	1	10	3	700		
0 to +700			Strain Gage	703682	Boeing	2	4.75	10	3			
0 to +2	PSIG		Pizo-electric	HF0(6)	Bytrex	5	2	3	3	500	CD5-626	
-6000 to +6000	PSID		Strain Gage	1A72914	Douglas	1	4.75	10	5	1,000	--	
-25 to +25				60B72077	Boeing	1	2	10	5	1,000	DC Ampl	60B73112
FORCE												
0 to +50K	LBS	M	Strain Gage	WCR-36	W. C. Research	2	3	5	8	500	AFDS 2	

INSTRUMENTATION EQUIPMENT ITEM		DEV. STATUS*	TYPE	ITEM TAG DESIGNATOR	MANUFACTURER	CHARACTERISTICS WORST CASE ENVIR. (FIG.	ACCURACY (%)	WEIGHT (OZ)	VOLUME (IN^3)	COST	REQD. SUPPORT EQUIPMENT	
VIBRATION												
-70 to 70	G	F	Magnetic Pick up	1A68707	Douglas	2	8	8	4	$1,000	Instl.	1B58286
-5 to 7			"	60B72192	Boeing	4	5	8	4	1,000	—	
RATES												
-25 to 25	Deg/S	F		50Z12400	SIU	2	4.75	15	8	1,500	DC Ampl	50Z12400
0 to 7K	RPM		Tach. Pulse Converter	60B73156	Boeing	2	4.7	10	10	1,500	—	
0 to 30K			Tach.	5-0163	N A	1	4.7	15	10	1,500	S/C	V7-750453
0 to 5K		M	Magnetic	.410	Westburg	2	0.1	8	9	1,000	S/C BAC	474
0 to 10K	PPPS	M	Pick up	Pulse Rate to D.C. Conv.	Honey-well	4	1.	3	1.5	800	—	
DIMENSION												
-7.5 to 7.5	Deg.	F	Pot	1A66248	Douglas	1	4.7	10	10	1,500	—	
0 to 110		M	RVDT	41590	GELAC	1	3.	10	10	1,000	S/C	41590-2
0 to 360		M	Resolver Syncro	L1184	LAS	2	.1	16	8	1,500	S/C	41590-2
0 to 100	%	F	Position Pot	NA527285	N A	1	4.7	10	8	1,000	—	
0 to 100			Angular	NA527306		2	4.7	10	8	1,000	—	
0 to 100				NA527307		2	4.7	10	8	1,000	—	
0 to 100				SX2633		1	4.7	10	8	1,000	—	
0 to 100				1A78153	Douglas	5	4.7	10	8	1,000	—	
0 to 100			Position Pot	2001612001	N A	1	4.7	10	8	1,000	—	

INSTRUMENTATION EQUIPMENT ITEM	DEV. STATUS*	TYPE	ITEM TAG DESIGNATOR	MANUFACTURER	CHARACTERISTICS WORST CASE ENVIR. (FIG.	ACCURACY (%)	WEIGHT (OZ)	VOLUME (IN^3)	COST	REQD. SUPPORT EQUIPMENT
DIMENSION (Continued)										
0 to 340 INCH	F	Extensiometer	1A68709	Douglas	3	4.7	32	20	$1,500	S/C 1B39437
0 to 12			1B27508		4	4.7	24	20	1,500	
0 to 6	M	Pot	176	Bournes	2	1.5	10	7	1,000	S/C Aids 8
0 to 3		LVDT	41590	GELAC	2	1.	4	4	1,000	S/C 41590-1
VDC EVENT										
0/5 VDC	M	Compare	616184	Emerson	4	.2	1	.3	1,000	–
0/5		CKT	10T	Optimized Devices	2	0.1			1,000	AIDS 3
0/20		Compare	CATS 25	Naval Elec Lab	4	.2	1	.3	1,000	–
0/28		Detector	1310	Hi-G Inc	2	2.5		3	1,000	AIDS 3
0/28	F	Micro S/N	1B44241	Douglas	3	4.7	3	3	1,000	–
0/28		Differentiator	V7-750310	N A	1	4.7	3	3	1,000	
VDC - ANALOG										
0 to 7.5 VDC	F	Ampl	50Z12400	IBM	5	4.7	4	3	1,000	
50 to 60										
24 to 32			60B73113	Boeing	3	4.7	3	3	1,000	
0 to 300	M	Compare	616184	Emerson	5	4.7	3	3	1,000	
0 to 5										
3 to 5		CKT	CATS 20	Naval Elec Lab	4	4.7	3	3	1,000	
0.1 to 0.2			CATS 35		5	1.	3	3	1,000	

INSTRUMENTATION EQUIPMENT ITEM	DEV. STATUS*	TYPE	ITEM TAG DESIGNATOR	MANUFACTURER	CHARACTERISTICS WORST CASE ENVIR. (FIG.	ACCURACY (%)	WEIGHT (OZ)	VOLUME (IN^3)	COST	REQD. SUPPORT EQUIPMENT
VDC ANALOG (Continued)										
0 to ±50 VDC		CKT	CDC 10	Naval Elec Lab	5	1.	3	3	$1,000	—
0 to 5	F		1A68710	Douglas	2				1,000	Control Unit 1A68710-5
VAC										
120RMS VAC	M	CKT	K104-01	Eon Instru.	2	0.1	16	80	1,000	—
0 to 4		Differen-tiator	—	Honey-well	4	0.1	1	1	1,000	—
CURRENT										
0 to 200 AMP	F	Mag Picker Det.	1A59741	Douglas	5	1.	5	3	1,000	—
0 to 20			1A68316		5	1.	5	3	1,000	
0 to 500		CKT	ME4310019	N A	5	1.	5	3	1,000	
0 to 100		Network	50Z12400	Boeing	5	1.	5	3	1,000	
0 to 3	M	CKT	CS1	Electra-matic	1	1.	6	3	1,000	
0 to 5	D		—							
FREQUENCY										
380 to 420 HZ	F		50Z12399	IBM	1	0.1	5	3	1,000	—
POWER										
0 to 4 WATT	M	CKT	AV-LD		4	1.0	1	.2	800	—

INSTRUMENTATION EQUIPMENT ITEM		DEV. STATUS*	TYPE	ITEM TAG DESIGNATOR	MANUFACTURER	CHARACTERISTICS WORST CASE ENVIR. (FIG.	ACCURACY (%)	WEIGHT (OZ)	VOLUME (IN^3)	COST	REQD. SUPPORT EQUIPMENT	
RESISTANCE	OHMS			C5I	Electra matic							
FLOW 0 to 300	LB/SEC	F	Mag Pick up	6-0163	N A	3	4.7	10	6	$1,500	S/C Calib	V7-750459 V7-750-467
6 to 60	LB/H	M	Thermal	60	Thermal Instru	2	1.	4	4	1,500	AIDS 2	
0.1 to 1	↓	↓	↓	FMS-339	Tylan	2	1.	12	14	1,500	AIDS 2	
0 to 10	↓	D										

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