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SPACE TUG AVIONICS DEFINITION STUDY

FINAL REPORT

VOLUME III ✦ AVIONICS BASELINE CONFIGURATION DEFINITION

GENERAL DYNAMICS
Convair Division

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April 1975

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Prepared for
National Aeronautics and Space Administration
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FOREWORD

This final report on the Space Tug Avionics Definition Study was prepared by General Dynamics, Convair Division for the National Aeronautics and Space Administration's George C. Marshall Space Flight Center in accordance with Contract NAS8-31010. The study was conducted under the direction of NASA Contracting Officer Representative, Mr. James I. Newcomb, and deputy COR, Mr. Maurice Singley.

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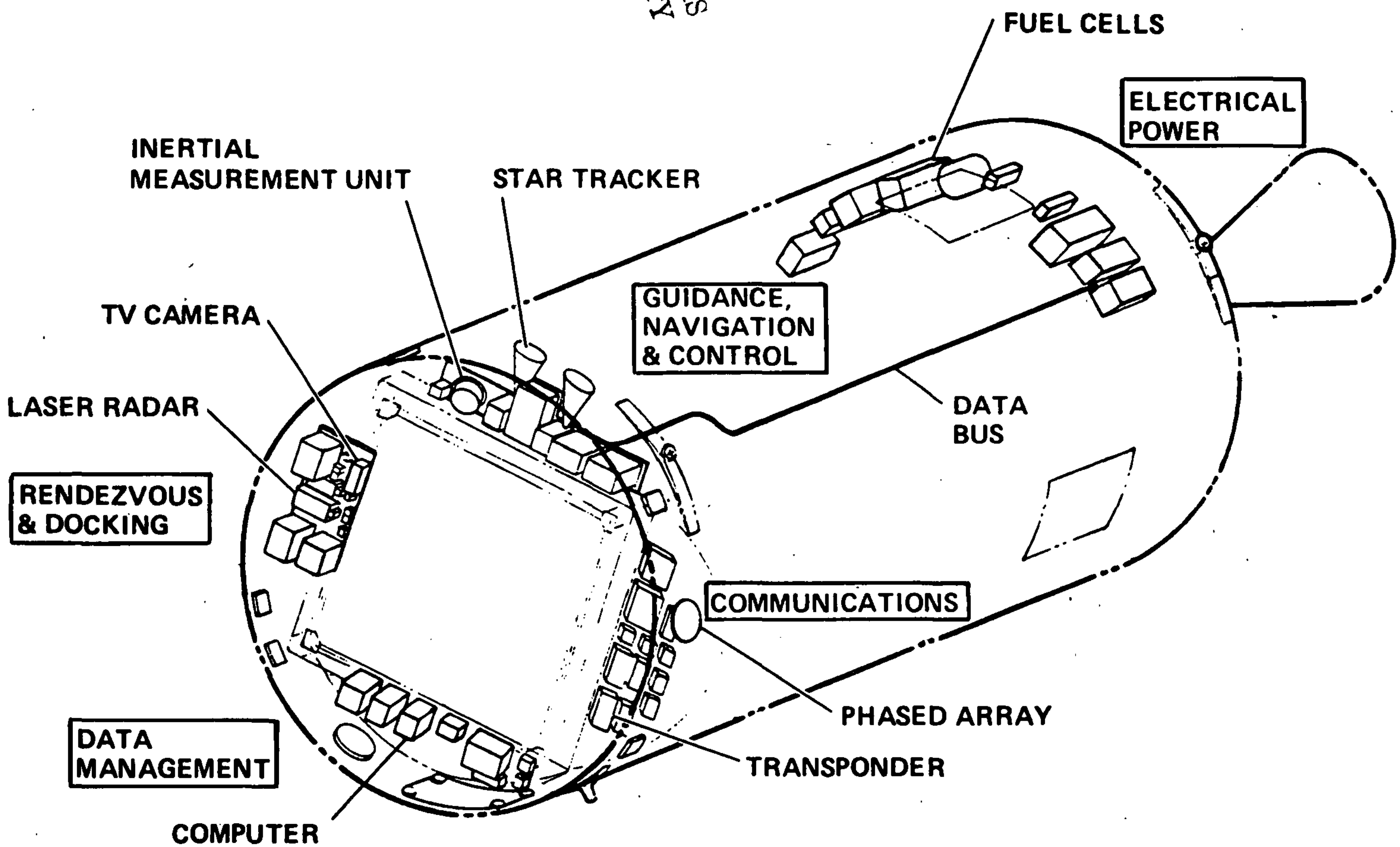
FUEL CELLS
ELECTRICAL POWER
INERTIAL MEASUREMENT UNIT
STAR TRACKER
GUIDANCE, NAVIGATION & CONTROL
TV CAMERA
LASER RADAR
DATA BUS
RENDEZVOUS & DOCKING
COMMUNICATIONS
PHASED ARRAY
TRANSPONDER
DATA MANAGEMENT
COMPUTER

SUMMARY

The Space Tug Avionics Definition Study was performed by General Dynamics Convair Division under Contract NAS8-31010 to the National Aeronautics and Space Administration, George C. Marshall Space Flight Center, Huntsville, Alabama. The objectives of this eight month study were to:

Define a baseline avionics system to meet Tug requirements using (circa) 1978 technology.

Define critical subsystems, functions and interfaces.

Identify areas requiring significant hardware technology advances.

The preferred baseline avionics system features a central digital computer that integrates the functions of all of the Tug's subsystems by means of a redundant digital data bus. The major subsystems of the avioncs system are:

Data Management Subsystem - The central computer consists of dual central processor units, dual input/output processors, and a fault tolerant memory utilizing internal redundancy and error checking. Control of the data bus is through a computer interface unit, which directs the bus traffic to and from the digital interface units that provide the interfaces to the various subsystems.

Communications Subsystem - Three electronically steerable phased arrays provide downlink transmission from any Tug attitude directly to ground or via TDRS.

Guidance, Navigation and Control Subsystem - Six laser gyros and six accelerometers in a dodecahedron configuration make up the Inertial Measurement Unit. Interferometric landmark tracking of ground based radars provides autonomous updates of position and velocity. Startrackers and sun sensors are onboard for attitude update.

Rendezvous and Docking - Both a scanning laser radar (LADAR) and a TV system employing strobe lamps are required as acquisition and docking sensors.

Electrical Power - Primary DC power at a nominal 28 volts is supplied from dual lightweight, thermally integrated fuel cells which operate from propellant grade reactants out of the main tanks.

Instrumentation - Provides conditioning and multiplexing of data from all Tug systems.

This volume contains the detailed definition of the baseline avionics system for the Space Tug resulting from system and subsystem trade studies. Included in the definition are Tug interfaces with the spacecraft, Orbiter and the ground, and the baseline philosophy and configuration for onboard checkout of the Tug.

Subsystem descriptions contain baseline configurations as well as the functional and operational features for each subsystem. Component details and characteristics and the supporting software are a major part of the subsystem descriptions.

SECTION 1

INTRODUCTION

The Space Tug is a propulsion stage that is carried into low earth orbit by the Space Shuttle to extend the Space Transportation System capability beyond the limits of the Shuttle. The Tug and its payload are carried in the Shuttle cargo bay and deployed on orbit (160 n. mi., 296 km nominal Shuttle orbit). Once the Tug mission is executed, the Tug returns to the Shuttle, is captured, placed back in the cargo bay, and returned to ground for refurbishment and reuse. The Tug will require an advanced avionics system with the capability to accomplish mission operations associated with payload delivery, retrieval and on-orbit servicing.

The Avionics System for the full-capability Space Tug to be developed by NASA for initial operations in late 1983 will be driven by requirements of 1) performance: to deliver 8000 pounds of payload into geosynchronous orbit and retrieve 3500 pounds, 2) mission duration up to 185 hours, 3) payload retrieval with potential for on-orbit servicing in the future, 4) autonomous flight operations, 5) Shuttle crew safety and mission success reliability (.97 for all missions), and 6) the 1983 IOC date for first operational flight. The 1978 Phase C/D timing will allow the Tug program to take maximum advantage of technology advances in the Avionics implementation of these requirements to attain low system weight, power system capacity, sensors and software for rendezvous and docking, navigation update, checkout, redundancy and its management.

This volume comprises a complete definition of the Space Tug Avionics System baseline. It contains the selected and optimized subsystems and components from the trade studies contained in Volume IV. As such it is a convenient replacement for the initial baseline avionics definition contained as a section in the MSFC Document 68M00039-2 "Baseline Space Tug Configuration Definition."

SECTION 2

SPACE TUG AVIONICS SYSTEM DESCRIPTION

2.1 BASELINE CONFIGURATION

The Space Tug Avionics System implements six functionally oriented subsystems and three peripheral interfaces. In addition it implements the electrical interface to the Tug mechanical systems.

The Avionics System features a central digital computer that integrates the functions of all the Tug's subsystems. This integrating subsystem is the Data Management Subsystem (DMS). Access to the DMS is accomplished at the user end of the DMS data bus through a terminal device designated a Digital Interface Unit (DIU).

There are four dual redundant DIU's in the Tug system. Three are located in the forward equipment area and accommodate the Guidance, Navigation, & Control (GN&C) subsystem, Rendezvous and Docking (R&D) subsystem, Communications subsystem, the spacecraft interface, and forward electrical power control and instrumentation. The fourth DIU set is mounted in the intertank area and accommodates the aft electrical power generation and control, and aft instrumentation as well as the mechanical system interface.

Except for safety and spacecraft signals, the principal Tug/Orbiter interface (both NASA and DOD) is accomplished through the Communications subsystem. This subsystem gains access to the DMS through a high speed Computer Interface Unit (CIU), which is the computer interface to the data bus. A third interface is implemented through the Tug deployment adapter to the ground.

The Tug avionics system block diagram appears in Figure 2-1.

Loading of the three instrumentation Signal Conditioner/Multiplexers is such as to balance both data rates and electrical connections. To minimize harness bulk and electromagnetic coupling, all analog data is digitized as close to the generating source as possible and all digital data is transmitted serially.

Subsystem sequencing and proportional control commands are generated at the DIU. Inter- and intrasubsystem interfaces are minimized so that the majority of component level devices communicate via the DIU/CIU/DIU data path, the only exceptions being instrumentation and electrical power interconnections.

2.2 SUBSYSTEMS SUMMARY

2.2.1 DATA MANAGEMENT SUBSYSTEM (DMS). The DMS provides the central processing and command functions on board the Tug. All Tug subsystems interface

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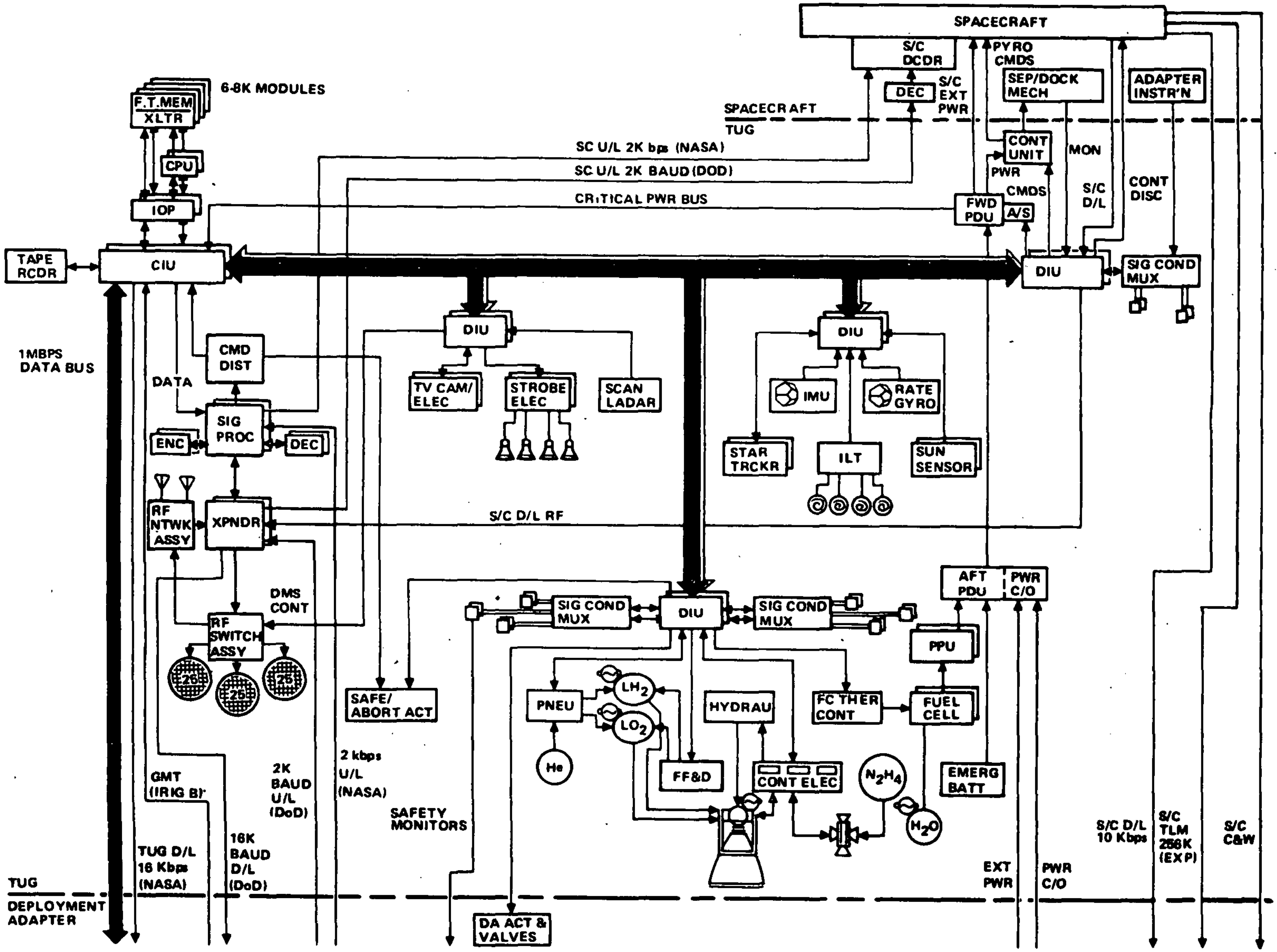

Figure 2-1. Tug Avionics System Block Diagram

with and are controlled by the DMS. DMS components account for approximately 11% of the avionics weight and consume an average of 107 watts.

Digital Computer: The computer is a modular unit adapted from the MSFC SUMC development. It utilizes a single 48K fault-tolerant memory with internal redundancy and an error detecting and correcting translator. Dual central processors (CPU's) and input/output processors (IOP's) employ error checking in a hardware/software self-test mode. The computer interfaces exclusively with the CIU.

Computer Interface Unit (CIU): The CIU is a dual redundant data preprocessing and control device. It performs buffering and formatting of systems data and as such functions as the prime interface with the Communications Subsystem. As the Data Bus controller and local terminal, it directs bus traffic to and from the DIU's.

Digital Interface Unit (DIU): The DIU's are dual redundant at each of the four data bus remote terminals. The device provides buffering between the data bus and user components. Control of user sequencing and limited preprocessing of user data are accomplished in the DIU, some under read only memory (ROM) control. The ROM control and processing store accommodates user peculiar requirements if any, the rest of the device having a standard input/output (I/O) complement.

Tape Recorder: The recorder is a NASA standard 10^8 bit unit interfacing strictly with the CIU. It stores status/maintenance data for playback when needed to augment the telemetry coverage window.

2.2.2 GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM (GN&C). The GN&C hardware is essentially composed of measuring devices that provide input to the GN&C processing software in the DMS. The single output device in the GN&C subsystem is the engine Control Electronics, which electrically interfaces the main engine steering and Auxiliary Propulsion System (APS) discrete control in the mechanical system. GN&C accounts for 20% of avionics weight and consumes an average of 382 watts.

Inertial Measurement Unit (IMU): The dodecahedron IMU achieves the equivalent of triple redundancy with only six laser gyros and six pulse rebalanced accelerometers. The IMU provides the prime vehicle state determining measurements.

Rate Gyros: Vehicle rate measurements are necessary to effect attitude stabilization. A dodecahedron rate gyro set provides the proper measurements. Rate data derived digitally in the DMS may negate the requirement for a separate rate gyro package. This will be determined from stability analyses during Tug development.

Star Tracker/Sun Sensor (ST/SS): The startracker and sun sensor work together to provide accurate vehicle attitude information. Both units are dual redundant. The sun sensor provides third axis measurements and thereby minimizes vehicle reorientation needed to accommodate using the startracker only.

Interferometric Landmark Tracker (ILT): Accurate vehicle position and velocity measurements required to update the guidance set are provided by the functionally redundant ILT. This, along with the attitude and time, provides a complete vehicle state update.

Control Electronics Unit (CEU): A single, internally redundant CEU driven by a dual redundant DIU pair provides both main engine (ME) thrust vector control, and discrete commands to the bi-state APS engines. ME position feedback loops are closed in the triply redundant CEU servoamplifiers. Propulsion system arm/safe functions are in the CEU.

2.2.3 RENDEZVOUS AND DOCKING SUBSYSTEM (R&D). The R&D subsystem is used to implement spacecraft retrieval and servicing. The subsystem has two functions: spacecraft post-deployment inspection and terminal docking. R&D accounts for approximately 7% of avionics weight and consumes an average of 50 watts of power for short durations during the Tug retrieval phase.

Scanning Laser Radar (SLR): The single SLR provides the prime range and range rate measurements for the R&D acquisition and docking maneuvers. Functional dual redundancy is provided by the TV system.

Low Light Level TV (LLLTV): The prime function of the dual redundant LLLTV and its electronics is spacecraft inspection and docking. A ground-based operator views the TV image and provides feedback data for control of the Tug all the way to "dock and latch."

Strobe Lamps/Electronics: A four lamp strobe system is employed to support the TV operation.

2.2.4 COMMUNICATIONS SUBSYSTEM (COMM). The COMM subsystem interfaces directly with the DMS CIU. When Tug is deployed it provides the uplink and downlink communication with the Shuttle and the ground. The COMM system functions as the main hardware interface to the Orbiter. Approximately 8% of avionics weight and 193 watts average power is allocated to the COMM subsystem.

Steerable Phased Array/Controller (AESPA): The AESPA consists of three 25-element electronically steerable phase array antennas and associated directivity phase controllers. The phase controllers are driven directly from the DMS. The 25-element arrays exhibit high performance margins in a degraded mode.

Hemispherical Antenna: Dual switched antennas provide local omnidirectional uplink/downlink coverage.

Transponder: The transponder is a dual redundant cross-strapped unit with network mode select. It provides for carrier generation and modulation, ranging turnaround, command data detection, demodulation, and signal preprocessing. Any of the three phased arrays can be driven from either of the redundant transponders.

Signal Processor/Distributor: This dual redundant unit is mode selectable to provide decoding of either NASA or DOD command formats. An internal data shunt provides access to the spacecraft engineering uplink. All uplink data is available at both the Tug signal processor and spacecraft decoder and is acknowledged/decoded when the proper vehicle is addressed.

SGLS Encrypter/Decrypter: To meet secure communications requirements on DOD missions, data encryption/decryption devices are provided. Their four-watt power requirement is not reflected, however, in the COMM subsystem power profile.

2.2.5 ELECTRICAL POWER SUBSYSTEM (EPS). Vehicle prime 28 Vdc power is supplied by the EPS. Prime power is generated by dual lightweight, thermally integrated fuel cells that operate from propellant grade reactants taken from the main propellant tanks. Power sequencing, distribtion, and safing mechanisms are included in the EPS. Including harnessing, the EPS comprises a little less than one-half of the avionics weight and consumes 130 watts.

Fuel Cells: The dual redundant fuel cells and support mechanisms are reactant supported by the main propellant tanks. The power generating subsystem waste heat is utilized to maintain both APS (N_2H_4) and hydraulic fluid temperature by integral recirculators and heat exchangers. Additional heat is jettisoned by way of four panel radiators mounted on the Tug intertank shell.

Emergency Battery: The battery functions as a contingency safety backup. This third backup function is sized around the worst case maximum loiter time upon Tug return to the Orbiter vicinity.

Solid State Controllers/Distributors: Prime power distribution is effected locally in both the forward and intertank areas. The intertank controller/distributor contains the Tug power changeover switch. The forward controller/distributor contains the arm/safe switch dedicated to spacecraft functions.

2.2.6 INSTRUMENTATION SUBSYSTEM. The Instrumentation subsystem is one of two that interface all other subsystems directly, including mechanical systems. This subsystem processes data not directly traversing the DMS data bus by first conditioning, multiplexing, and digitizing the data before serially injecting the data and time tag onto the bus. Functional backup measurements are provided for propellant control and safety measurements. The instrumentation subsystem accounts for approximately 7% of avionics weight and consumes an average of 50 watts.

Signal Conditioner/Multiplexer (SC/M): The SC/M employs internal dual redundancy and self-calibration. Three units are provided, one forward and two in the intertank area. Each unit provides for three levels of analog data along with bi-level discrete data.

Transducers: Each SC/M has associated with it a unique complement of transducers depending on the SC/M location. The forward SC/M instruments primarily the avionics and spacecraft interface areas. One of the intertank units instruments the LH_2 (forward) shell and intertank area and the other instruments of the LO_2 (aft) shell and thrust section areas. In general, only those tranducers providing safety or functional backup data are redundant. Maintenance peculiar transducers are all simplex.

The equipment list shown in Table 2-1 is a compilation of the total complement of avionics hardware for the Space Tug. The Avionics System weight is 898 pounds (408.2 kg).

2.3 INTERFACE SUMMARY

2.3.1 SPACECRAFT ELECTRICAL INTERFACE. The baseline spacecraft interface accommodates a single payload. The interface may be divided into two categories, the first of which is at the Tug/Spacecraft Adapter. Adapter functions include separation and docking mechanism command and feedback signals, and all adapter peculiar instrumentation.

Spacecraft functions include both those that actively interface Tug and those that are merely routed through the Tug. These originate in the spacecraft and are present at both the Tug/Adapter and Spacecraft/Adapter interfaces.

Spacecraft experiment data (256 Kbps), caution and warning, and an engineering downlink (10 Kbps) are hardwired through Tug with no electrical interface.

A second engineering downlink (10 Kbps) interfaces the Tug forward DIU where data is submultiplexed into the Tug downlink. In the DIU, selected spacecraft status data may also be shunted to the DMS computer.

The remainder of the spacecraft interface consists of external power and sequence discrete commands. These commands are issued directly from the DIU under DMS controlled response to uplink commands.

2.3.2 ORBITER ELECTRICAL INTERFACE. The prime Tug/Orbiter interface is defined at the Tug/Deployment Adapter union. Besides carrying across the hardwired signals from the spacecraft (those direct through Tug) this interface accommodates both the 2 Kbps NASA/2K baud DOD uplinks and 16 Kbps NASA/16K baud DOD downlinks originating in the Tug Communications subsystem. The 16 Kbps NASA downlink originates at the CIU. A bi-directional 1 Mbps data bus, also originating at the CIU, passes through the deployment adapter to ground.

Tug caution and warning signals are hardwired through the interface along with the power changeover command. Safety commands to the deployment adapter are hardwired from Tug to support the prime safety abort mode utilizing the DMS. Tug external power provisions and an IRIG-B format clock complete the interface.

2.3.3 GROUND ELECTRICAL INTERFACE. The Tug/ground electrical interface is implemented through the Tug deployment adapter and through the Orbiter launch umbilical (T-0) panel. Two Tug electrical functions accommodated by the interface are the 1 Mbps data bus and the 16 Kbps pulse code modulated (PCM) downlinks (NASA or DOD).

Table 2-1. Tug Avionics Equipment List

EQUIPMENT	NO. REQ	ENVELOPE DIMENSIONS			UNIT WEIGHT	UNIT POWER (WATT)	SUB SYSTEM WEIGHT (LB)
DATA MANAGEMENT							100
DIGITAL COMPUTER	(1)	10	14	9.5	34	60	
CIU	(2)	5	5	6.5	6.5	7	
DIU	(8)	5	5	6.5	5	5	
TAPE RECORDER	(1)	10	8	5	13	20	
GUIDANCE, NAVIGATION & CONTROL							190
INERTIAL MEAS UNIT	(1)	9 x 9 DIA			25	100	
IMU ELECTRONICS	(1)	10	20	5	30	100	
RATE GYROS	(1)	10	10	6	20	100	
STAR TRACKER	(2)	6	8	12	16	12	
SUN SENSOR	(2)	6.9	6.5	3	4.5	5	
CONTROL ELECTRONICS	(1)	12	12	18	50	50	
ILT–ANTS./RECEIVER	(1)	12	10	9	24	15	
RENDEZVOUS & DOCKING							63
SCANNING LADAR	(1)	6	8	20	28	10	
& ELECTRONICS	(1)	9	9	11	11	30	
TV CAMERA & ELECTRONICS	(2)	6	6	15	8/13	10	
TV STROBE LAMPS	(4)	3.5	3.5	3.5	0.25		
STROBE ELECTRONICS	(2)	2	3.5	2.5	1		
COMMUNICATIONS							149
ELEC STEERED PHASED ARRAY	(3)	3.5 x 15 IN. DIAM			16	93	
OMNI ANT/NETWORK/SWITCH	(1)	5	5	6	11.3	3	
TRANSPONDER	(2)	15	7	6	16.5	16	
SIGNAL PROCESSOR	(2)	13.5	6	6.5	11	18	
COMMAND DISTRIB	(1)	5	5	4	18	35	
SGLS ENCRYPTER	(2)	6	4	5	4.3	7	
SGLS DECRYPTER	(2)	6	4	5	4	2.5	
INSTRUMENTATION							74
TRANSDUCERS	(243)				20	–	
SIGNAL CONDITIONERS	(3)	12	10	6	18	22	
ELECTRICAL POWER, DIST & CONTR							322
FUEL CELLS POWERPLANT	(2)	12	6	15	42	20	
EMERGENCY BATTERY (150 AH)	(1)	8	11	7	36	–	
PWR DISTRIBUTION					46	–	
PWR PROCESSING	(2)	9	9	8	8	–	
HARNESSES/SWITCHES/MISC					140		
AVIONICS SYSTEM WEIGHT 898 LB							

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The high-speed bus is a preflight data path used to load, verify, and sequence the Tug DMS and subsystems. Its interface with LPS accommodates ground operations from immediately after the time of Tug maintenance to Orbiter lift-off.

A third and last function of the interface is the carrying through of the hardwired spacecraft 10 Kbps downlink to ground.

2.4 AVIONICS INSTALLATION

The Tug avionics hardware is installed on the vehicle in both forward and intertank locations. Three principle factors accommodated for in the installation are accessibility, proximity to related hardware, and thermal control. The equipment installation, shown in Figure 2-2 (GDC drawing IT-75-002), is based on currently available component data and as such is driven primarily by proximity and view orientation requirements and to a lesser degree by accessibility and thermal control considerations. Thermal isolation is provided between the equipment area and the LH_2 forward bulkhead/access hatch.

Integration of the thermal control subsystem and access flexibility in performing the maintenance/refurbishment task are consistent with the equipment layout.

The forward equipment area provides for canted shelf, right-angle shelf, and shell-mounted units distributed throughout the four vehicle quadrants. Each quadrant displays a certain functional dedication. Along with an interfacing DIU in three of the four quadrants (the CIU/computer is located in the fourth quadrant), the installation arrangement provides for mechanical isolation and/or easy implementation of a common mechanical reference as is required within the GN&C and R&D subsystems.

Shell-mounted equipment in the forward area is limited almost entirely to communications gear with the exception of the GN&C ILT antennas and other external view oriented devices.

The three phased array antennas are peripherally located at 120 degree (2.1 radian) intervals, the two hemispherical antennas at 180 degree (3.1 radian) intervals, and the ILT antennas clustered within a 40 inch (101.6 cm) square.

The intertank equipment area accommodates either right-angle shelf or shell-mounted units. Shell-mounted devices are limited to the four EPS fuel cell radiators. These are located at 90 degree (1.6 radian) intervals around the Tug and skewed 45 degrees (0.8 radian) with respect to the APS thrusters. The power distribution unit (aft PDU) in the intertank area interfaces the deployment adapter for external power functions. The interfacing DIU's and the engine control unit service the Tug thrust section (and deployment adapter as required for C&W) from their intertank locations.

FOLDOUT FRAME 1

FOLDOUT FRAME 2

FOLDOUT FRAME 3

Figure 2-2. Space Tug Avionics Equipment Installation

SECTION 3

DATA MANAGEMENT SUBSYSTEM DESCRIPTION

3.1 BASELINE CONFIGURATION

Equipment	No. Req	Dimensions, in. (cm) Length	Width	Height	Unit Op Power (Watts)	Unit Weight lb (kg)	Subsystem Weight lb (kg)
Data Management							100 (45.4)
Digital Computer	1	10 (25.4)	14 (35.6)	9.5 (24.1)	60	34 (15.4)	
Computer Interface Unit	2	5 (23.7)	5 (12.7)	6.5 (16.5)	7	6.5 (2.95)	
Digital Interface Unit	8	5 (12.7)	5 (12.7)	6.5 (16.5)	5	5 (2.3)	
Tape Recorder	1	10 (25.4)	8 (20.3)	5 (12.7)	20	13 (5.9)	

The data management subsystem (DMS) is dual redundant to meet both the reliability goal and the fail-operational requirement for system safety. It is composed of:

a. A data bus.

b. A fault tolerant computer (SUMC).

c. Two Computer Interface Units (CIU).

d. Eight Digital Interface Units (DIU).

e. A tape recorder.

The CIU's and DIU's are dual redundant and connected by a dual redundant data bus. The data busses are separate entities with cross-strapped connections at the computer and the line replaceable units (LRU) of the subsystem interfaces.

Each LRU can be addressed from either data bus. Since both busses are active, the data format must contain a code to designate which data bus is prime for a particular subsystem LRU.

As part of the redundancy management for error detection and designation of the controlling bus, hardware tests of format and parity will be accomplished in each CIU and DIU. The central computer will participate in the selection of the data bus configuration with hardware and software tests designed to detect failures.

The tape recorder is used to record data for maintenance purposes such as the information related to engine burns. Its capability of 320 Mbits will permit recording of the complete first engine burn. This information would then be telemetered to the ground as needed.

The buffer formatter incorporated in the CIU is identified as the PCM buffer. The PCM buffer gathers instrumentation data to be telemetered to the ground or the Orbiter. The similarity of the logic used in data bus control to that of the buffer formatter and the close relationship of the buffer formatter to the data bus in its operation makes it possible for these units to be integrated.

No auxiliary memory is provided. The amount of software involved in a typical Tug mission can be stored in the main memory of the central computer so that a separate storage device is not required.

The data management subsystem (DMS) is shown schematically in Figure 3-1.

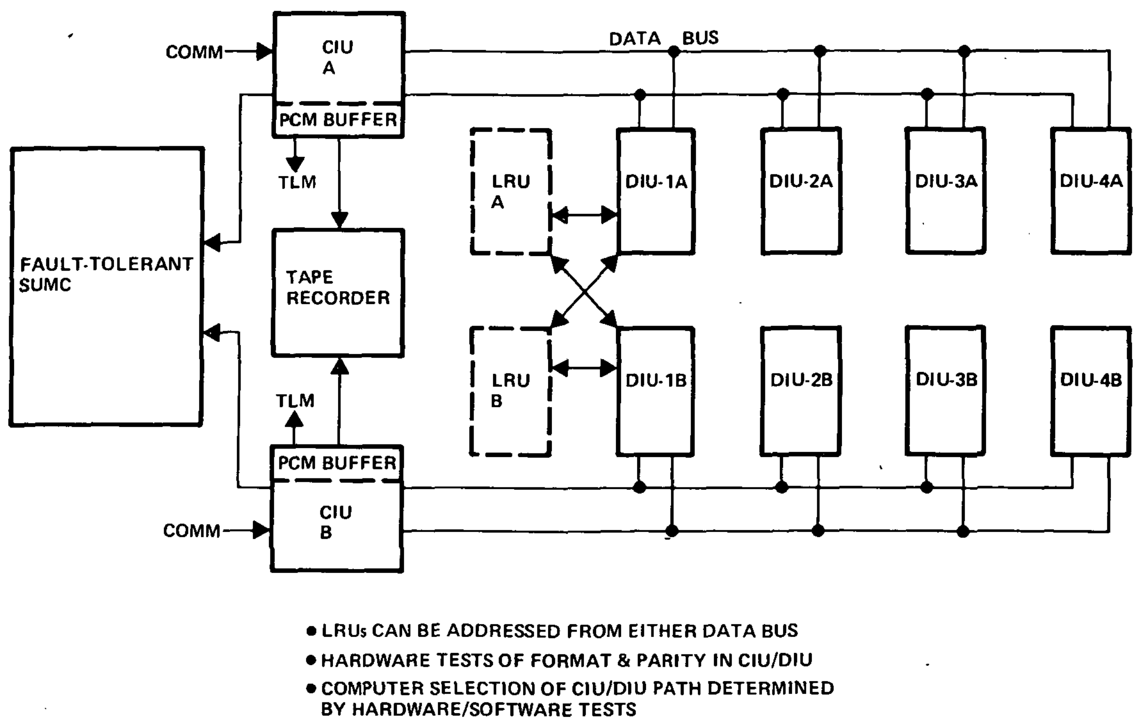


Figure 3-1. Recommended DMS Configuration

3.2 COMPUTER

The Tug computer is a 1978-1980 version of the Space Ultrareliable Modular Computer (SUMC) under development by MSFC. It is a microprogrammed, real-time, general purpose computer with a real-time multilevel interrupt structure. It is compatible with existing computers that operate with s/360 type instructions and data. A fault-tolerant memory and backup modules are provided for processing and input-output functions to meet the safety and reliability requirements. The word length is 32 bits so that the required number of double precision operations is reduced, the operation of the floating point instructions is improved, and the accuracy required for navigation and guidance is attained. Both full and half word data and instruction capability is provided. A 48K memory is capable of storing the instructions and data software extimated for the Tug mission with an adequate growth provision. The characteristics and capabilities of the computer are summarized in Table 3-1.

Table 3-1. Characteristics of Tug Computer

Computer Type	Stored program, parallel, general purpose with redundant modules and a fault-tolerant memory
Number System	Binary, fixed point, two's complement with floating point capability
Organization	Conventional, microprogrammed
Data Word Length	8, 16, or 32 bits including sign
Instruction Word Length	8, 16, or 32 bits based on s/360 type instructions
Memory	Random access, semiconductor
Memory Capacity	49, 152 x 32 bit words, main memory 16,384 x 32 bit words, spare memory to implement fault tolerance
Memory Speed	500 nsec access, 1.0 μsec cycle time
Input-Output	Direct memory access and program control transfer DMA rate is 1,000,000 words/sec Program rate is 250,000 words/sec.
Environment	Space. Conductive cooling to cold plate for thermal control
Reliability	To support 0.9953 probability of mission completion for data management subsystem

The computer is composed of Fault-Tolerant Memory, Central Processor Units (CPU), Input-Output Processors (IOP), and Fault Detector Switches (FDS). Translators associated with each memory module provide for use of error-correcting codes for fault tolerant storage of data and instructions in main memory. The fault detector switches provide for redundancy management of the CPU's and IOP's. Figure 3-2 is a reliability block diagram showing the redundancy relationships. Since the Tug mission is comparatively short in duration, the system need only correct for a single failure with the CPU and IOP modules and at least two failures in memory to meet the probability goal for mission success.

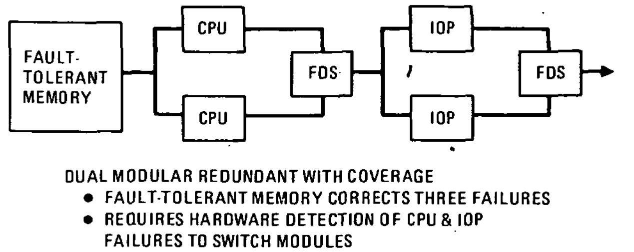


Figure 3-2. Tug Computer Redundancy Configuration

The main memory is a large-scale, integrated-circuit, semiconductor design with high speed, low weight, low cost and high reliability. Each memory word contains extra bits to permit error detection and correction techniques to be performed on each word transferred from memory. The volatility of the main memory is protected by providing an uninterruptible power source composed of dual-redundant fuel cells with a standby battery.

Computer memories are a major source of operational failures. The necessary improvement in memory reliability for the Tug computer is achieved with spare memory modules and error detection and correction logic supplied in a translator. The translator monitors all memory data transfers. The translator need only correct single errors and detect double errors to reach the reliability goal for Tug missions. Memory reliabilities very nearly equal to one are achievable with techniques that detect errors then load and switch spare modules into use. The spare module can be a bit plane or a standby block of memory. The bit plane switching is generally thought to be the most cost effective and weight effective but its operational feasibility is yet to be demonstrated. The extra control lines and data lines that have to be added with each spare bit may create a significant burden in circuit complexity and packaging for multiple failure systems.

The translator provides error detection and correction on instructions and data transferred from memory. A Hamming code is stored in the extra bits of each word in memory. The translator generates the code to be stored and adds it to computer words as they are entered into memory. Before a word from memory is transferred to a CPU or IOP, erroneous bits are detected and corrected.

The fault-tolerant memory approach recommended for the baseline DMS configuration is capable of correcting three errors and detecting four. This is accomplished by provision of a single error correction/double error detection (SEC/DED) translator for each 8K x 32 bit module and incorporating two spare modules in the memory design. Total memory capacity is 64K words, with six active modules and two spare. Memory addressing of 64K words is easily accomplished with the 32 bit instruction format of the SUMC computer.

Fault-tolerant memory management is easily accomplished by software recognition of a detected error flag from the translator. Data contents of the failed memory module are transferred to a spare under software control, and the error correction feature of the translator prevents the error from propagating into the newly activated module. Relative program addressing then allows the spare module to assume the role of the failed unit with no data path switching or execution time penalty.

The CPU fetches and executes instructions, interprets code, and performs operations for programs and executive routines not related to input-output operations and input-output housekeeping. The CPU is dedicated to serving the computing sections of program modules while the IOP is dedicated to execution of input-output sections of program modules and the inertial measurement unit processing. The CPU employs microprogram control and a multilevel interrupt structure.

The IOP transfers all data into and out of the computer and maintains supervisory control over all operations on the data bus. Although it is capable of fetching and executing executive program instructions related to input-output operations and input-output housekeeping, the IOP generally services the input-output requirements of the application program module running in the CPU. The IOP contains a processor dedicated to performing the dodecahedron IMU calculations. Outputs of each IOP are cross strapped to dual computer interface units (CIU). Channels for connection to peripheral equipment, including the data bus, are provided through the CIU. These channels are data paths to checkout, telemetry, ground link, and communication equipment.

3.3 DATA BUS COMPONENTS

The data bus is implemented with two lines, supervisory and reply, that connect computer interface units (CIU) and data interface units (DIU).

The supervisory and reply lines are twisted shielded pairs of cables that carry two megabit bi-phase L (Manchester type II) data signals. The supervisory line is synchronous while the reply line is asynchronous. The system is synchronized by continuous transmissions on the supervisory line. When there is no data bus activity, blank words consisting of a digital "1" followed by all zeroes will be transmitted on the supervisory line. There are no signals on the reply line when there is no activity.

A data bus word contains 20 bits. These are a word sync bit, two word code bits, 16 information bits, and one parity bit. Six word formats are provided for the supervisory bus as shown in Figure 3-3. The reply bus word formats are shown in Figure 3-4.

WORD TYPE	WORD SYNC BIT WS	WORD CODE BITS C1	C2	INFORMATION BITS 0	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	ODD PARITY P
COMMAND WORD	1	1	1	← DIU ADDRESS →						← OP CODE →					← CHANNEL ADDRESS →					P
WORD COUNT (WC)	1	1	1	← WORD COUNT →																P
WC END OF MESSAGE	1	0	1	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	P
DATA WORD (D)	1	1	0	← FIRST BYTE →								← SECOND BYTE →								P
D END OF MESSAGE	1	0	1	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	P
BLANK	1	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0

Figure 3-3. Supervisory Bus Word Formats

WORD TYPE	WORD SYNC BIT	WORD CODE BITS		INFORMATION BITS 0	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	ODD PARITY P
SYNC WORD	← NO SIGNAL →								1	1	1	1	0	1	← DIU ADDRESS →					P
ERROR STATUS WORD	1	1	1	← ERROR STATUS BITS →																P
DATA WORD	1	1	0	← FIRST BYTE →								← SECOND BYTE →								P
LAST DATA WORD	1	0	1	← FIRST BYTE →								← SECOND BYTE →								P
BLANK	1	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0

Figure 3-4. Reply Bus Word Formats

The data bus is controlled by the CIU. Only the CIU issues commands and transmits data on the supervisory line in accordance with instructions from the computer. The DIU transmits information to the CIU on the reply line. To control the data bus traffic, the CIU generates timing and sync signals for the data bus.

The CIU interface with the IOP is cross strapped to provide the capability of communicating with either of the dual IOP units. The self-test circuits of the CIU include tests for verification of the IOP and CIU interface.

The CIU contains the telemetry buffer formatter. This is the data table where instrumentation data is gathered to be telemetered to the ground or the Orbiter. The CIU has an interface with the communication system and the tape recorder to provide for the storage and transfer of this data.

Other elements of the communication subsystem operate with the Tug computer directly through the CIU and not through the data bus. These are the command distributor, command decoder, data link encoder, and transponder.

The DIU is the interface between the data bus and vehicle subsystem components external to the data management subsystem. Its input-output capabilities provide for discrete signals, analog signals, and digital information. The transfer of information from the DIU to the Tug computer is controlled by the computer with control instructions issued through the CIU for transfer over the data bus. Built-in-test capability is provided for test of data bus operation in cooperation with the CIU. Wraparound of signals transferred over the data bus is provided by the onboard checkout software.

The DIU's are dual redundant but each is a separate entity connected to a separate data bus. Cross strapping is provided from the DIU to each element of subsystem equipment so that the data from the subsystem can be transferred over either data bus. One data bus will be designated as the operating bus by a code in the supervisory bus command word. The other data bus will be operating as a standby unit with cross-strapped connection to either CIU, whichever is in control of data bus operation.

3.4 TAPE RECORDER

The recommended tape recorder unit is a NASA standard 10^8 spacecraft tape recorder with a digital data storage capability of 320 megabits. A constant information bit packing density of 5000 bits per inch (1968 bits/cm) per track is employed for all input bit rates. There are 23 tape speeds that provide a range of storage rates from 1.0 to 160.0 kilobits per second per track. Recording on four tracks can be accomplished sequentially or simultaneously. The maximum tape speed of 32 inches (81.3 cm) per second provides a recording time of 8.3 minutes while recording on four tracks simultaneously or 33.3 minutes while recording on four tracks in sequence. The minimum tape speed is 0.20 inch (0.51 cm) per second.

The tape recorder is a temporary digital data storage device for information that is normally telemetered to the Orbiter or to the ground. The use of the operational data and status data provided by telemetry is an integral part of the maintenance plan to enable quick turnaround for reuse of the Tug. When the Tug is outside the range of communications contact, the data can be stored on the tape recorder for later transmission.

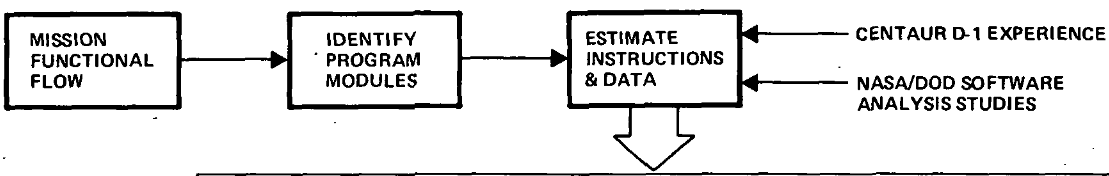
Operational data is of particular interest during an engine burn. An analysis of engine burn data showed that 403.2 megabits of data would be generated during the Tug mission. During the first (longest) engine burn of 1020 seconds, there would be 163.2 megabits of data generated.

The tape recorder is capable of storing all the data for the first engine burn. After this data has been transmitted, the tape recorder stores all of the data created during the remaining burns with a 33% reserve. The 160 kilobit per second rate at which data is generated is the same as the serial recording rate of the tape recorder at its maximum speed.

3.5 SOFTWARE

The estimate of software size was developed by comparing Tug mission requirements with actual experience in software development for Centaur missions. The software requirement is summarized in Figure 3-5.

Tug mission timelines were the basis for analyzing mission functional flow for functional requirements. Once the functions were established, program modules could be identified to relate to each function. The differences between the Tug mission and the Centaur mission were analyzed to determine the effect on the software module that was actually used for the Centaur mission. Estimates for new functions utilize the work of other contractors and Convair analyses of system interfaces and checkout requirements.



FUNCTION	CENTAUR D-1 ACTUALS	ESTIMATED Δ FOR TUG	TOTAL
EXECUTIVE	2,257	810	3,067
DATA POOL	1,530	1,150	2,680
NAVIGATION	471	3,900	4,371
GUIDANCE	2,245	2,350	4,595
FLIGHT CONTROL	1,950	390	2,340
TELEMETRY	1,767	2,250	4,017
SEQUENCING	776	1,040	1,816
RENDEZVOUS & DOCKING	–	3,500	3,500
TANK PRESS. CONTROL	724	70	794
PHASED ARRAY CONTROL	–	220	220
CHECKOUT (STATUS)	–	6,200	6,200
REDUNDANCY MANAGEMENT	–	3,126	3,126
UTILITY MODULES	–	3,900	3,900
TOTAL			40,626
INSTRUCTION MIX: ASSUME: 75% 16 BIT, 25% 32 BIT			25,391
HOL COMPILER OVERHEAD ASSUME 20%			5,078
ESTIMATED MEMORY REQUIRED			30,469

Figure 3-5. Development of Software Estimate (Words)

A Higher Order Language (HOL) and Floating Point Arithmetic are considered essential by software programmers to simplify and speed the coding and verification of software. HOL is historically characterized as adding up to 20% to the software size.

Software suppliers of compilers estimate an increase of 10 to 15%. The conservative value of 20% was used to derive the number of memory words required.

3.6 REDUNDANCY MANAGEMENT

The data management subsystem is essentially dual redundant to meet both the reliability goal and the fail-operational requirement for system safety in the vicinity of the Orbiter. A backup is required for the IMU and the main elements of the data management subsystem such as the central computer and its peripheral interconnections to guarantee stability during operations in the vicinity of the Orbiter. Flawless performance in this vicinity is necessary to ensure the safety of personnel. A high probability for success is required for the remainder of the mission because of the expenditures and resources invested.

The reliability analysis provided an apportionment of the unreliability of the total vehicle to each of its subsystems and to each of the avionics subsystems. The avionics system goal assigned in this way is 0.992.

An analysis of the avionics subsystem made with a model that adds the minimum weight for the greatest gain in reliability showed that the avionics system goal of 0.992 could be achieved with a dual redundant CPU, IOP, CIU, and buffer formatter in a system with a modular computer using a memory with error detection and correction. The results are shown in Figure 3-6. A similar analysis using a simplex computer with the memory, CPU, and I/O packaged to operate as a unit also showed a dual-redundant computer would be needed to achieve the reliability goal.

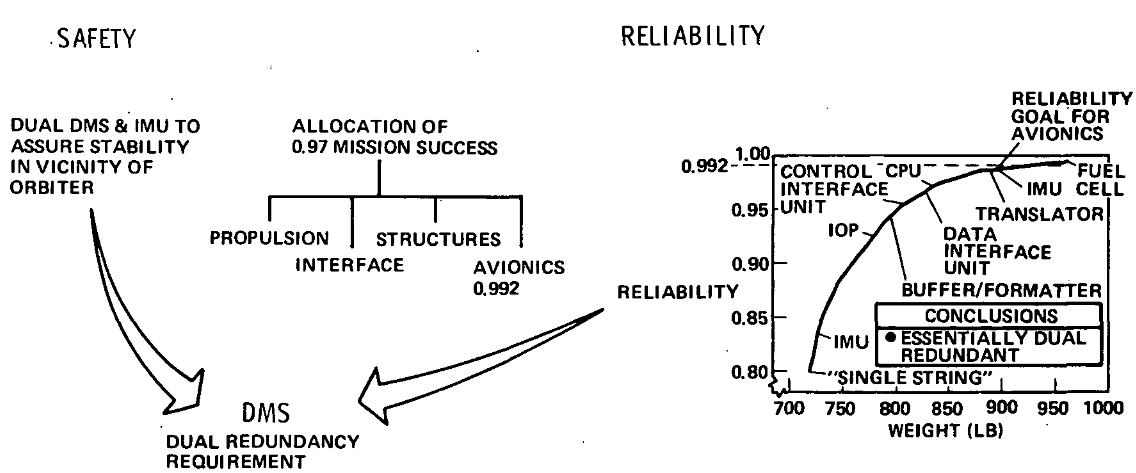


Figure 3-6. DMS Minimum Redundancy Requirements

Table 3-2 summarizes the types of redundancy utilized throughout the Tug avionics system.

Table 3-2. Summary of Redundancy Techniques Utilized

SUBSYSTEM	LEVEL REDUNDANCY	TYPE OF REDUNDANCY	REDUNDANCY MANAGEMENT APPROACH
DATA MANAGEMENT			
COMPUTER	DUAL (MODULAR)	PRIMARY + STANDBY	CPU/MEMORY HARDWARE CHECK AND SWITCH
DATA BUS	DUAL	INDEPENDENT CHANNELS	CIU CHANNEL CHECK WITH IOP SWITCH DIU CROSSTRAPPED TO LRUS
GN&C			
IMU	DODECAHEDRON	MULTIPLE SENSORS	DMS SOFTWARE PROVIDES: SENSOR DATA COMPARISON SELECTS SENSOR SET FOR COMPUTATION DETECTS SENSOR FAILURE & RESELECTS SENSOR SET
ILT (POS, VEL UPDATE)	FAULT TOLERANT	MULTIPLE CHANNELS	
ATTITUDE UPDATE	DUAL	ONE + SPARE	POWER UP/DOWN
FLT CONTROL	TRIPLE	MAJORITY VOTING	SELF-CORRECTING
RENDEZVOUS/DOCKING			
SENSORS	DUAL	PRIMARY + BACKUP	POWER UP/DOWN
COMMUNICATION			
PHASED ARRAY	FAULT TOLERANT	MULTIPLE-ELEMENT ANTENNA	GRADUAL DEGRADATION
SIGNAL PROCESSING	DUAL	INDEPENDENT CHANNELS	DMS SOFTWARE CHECK/ SWITCHING
ELECTRICAL POWER			
FUEL CELL	DUAL	ONE + SPARE	SELF-DETECTION & CORRECTION

Dual-redundancy implementation is common except for dodecahedron IMU sensors and triple-redundant flight control servos and amplifiers. Dodecahedron redundancy management is a software functional selection of the sensor set providing most reasonable data when compared to other combinations of sensors. This selection will be augmented by status data collected by the checkout software. Flight control servo failure is masked by a mechanical voting technique that is self-correcting for any single failure and requires no external redundancy management.

The dual-redundant subsystem elements are implemented in a primary plus standby configuration. Standby units are powered and operated during critical mission phases where repid reconfiguration is essential for mission success. Software controlled decision algorithms based on checkout status data will control reconfiguration for attitude sensors, rendezvous and docking sensors, and communications elements. Rapid fault recovery hardware is implemented in the redundancy management of the computer, data bus, and fuel cell.

Redundancy management of a dual-modular computer is a primary concern because of the need to detect errors and make corrections before significant harm to the vehicle can occur. There is no masking of failures. Figure 3-7 depicts the techniques used in management of the primary and backup units in the Tug computer.

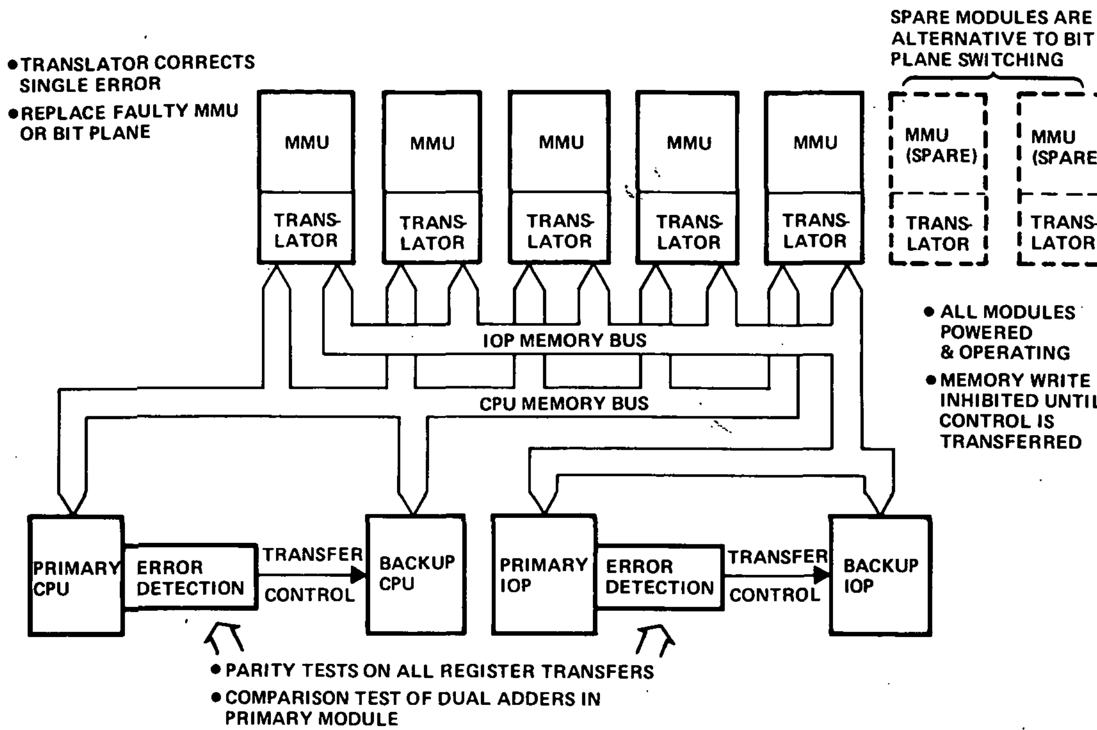


Figure 3-7. Fault-Tolerant Computer Redundancy Management

A basic premise of the Tug mission is that all modules have power applied and are in an operating mode throughout the mission. This permits a comparison of the operation of each module to improve the failure detection capability, although the backup modules are inhibited from writing into memory until they are given control by hardware error detection circuitry in the primary module.

Additional redundancy is implemented in the CPU logic to improve the probability of detecting failures. Dual adders will provide a comparison test of operability. Parity tests on all register transfers will be included. These techniques have been shown to be effective in achieving a coverage of 0.90 for dual configurations.

SECTION 4

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM DESCRIPTION

4.1 BASELINE CONFIGURATION

Equipment	No. Req	Dimensions, in. (cm) Length	Width	Height	Unit Op Power (Watts)	Unit Weight lb (kg)	Subsystem Weight lb (kg)
Guidance, Navigation & Control							190 (86.4)
Inertial Measurement Unit	1	9 x 9 (22.9 x 22.9) Diameter			100	25 (11.4)	
IMU Electronics	1	10 (25.4)	20 (50.8)	5 (12.7)	100	30 (13.6)	
Rate Gyro Package	1	10 (25.4)	10 (25.4)	6 (15.2)	100	20 (9.1)	
Star Tracker	2	6 (15.2)	8 (20.3)	12 (30.5)	12	16 (7.3)	
Sun Sensor	2	6.9 (17.5)	6.5 (16.5)	3 (7.6)	5	4.5 (2.0)	
Control Electronics	1	12 (30.5)	12 (30.5)	18 (45.7)	50	50 (22.7)	
Interferometric Landmark Tracker							
ILT Antenna	4	2 2 x 6 (5.1 x 15.2) Diameter			-	1 (0.45)	
ILT Receiver	1	12	10 (25.4)	9 (22.9)	15	20 (9.1)	

Navigation is the act of determining the relative position of a vehicle. The navigation system on the Tug is required to provide the position, velocity, attitude, or state, of the Tug. The guidance system then takes this information, compares it to the desired Tug state, and computes an optimum path to obtain the desired Tug state. The control system then implements the guidance requests to command the vehicle along the optimum path.

On the Tug, the navigation function is performed by the Inertial Measurement Unit (IMU) and the rate gyros, with the Interferometric Landmark Tracker (ILT), the star trackers, and the sun sensors providing updates, as shown in Figure 4-1. The IMU provides data to the Data Management Subsystem (DMS), which calculates the vehicle state. However, inherent imperfections in the IMU cause errors in this calculated vehicle state, and the update sensors are required to keep the calculated vehicle state from drifting too far from the actual vehicle state. The ILT system provides position and velocity update, and the star trackers and sun sensors provide attitude update.

The rate gyros provide vehicle angular rate measurement, which is used for vehicle attitude stabilization by the autopilot. The autopilot, which is implemented in the DMS computer, combines the vehicle attitude and rate data from the IMU and rate gyros and calculates commands to the vehicle attitude control system. Although vehicle angular rate is also available from the IMU, separate rate gyros allow the rate sensors to be remotely positioned relative to the IMU. The rate gyros are positioned on the Tug to provide optimum phased rate data. When detailed Tug spacecraft data becomes available, a stability analysis can be performed, which may indicate that the rate gyros are not required. That is, the analysis may show that with the rate gyros positioned at the IMU location, the vehicle is stable and therefore the rate data obtained from the IMU would be adequate. However, until this analysis can be performed it is assumed that rate gyros are required.

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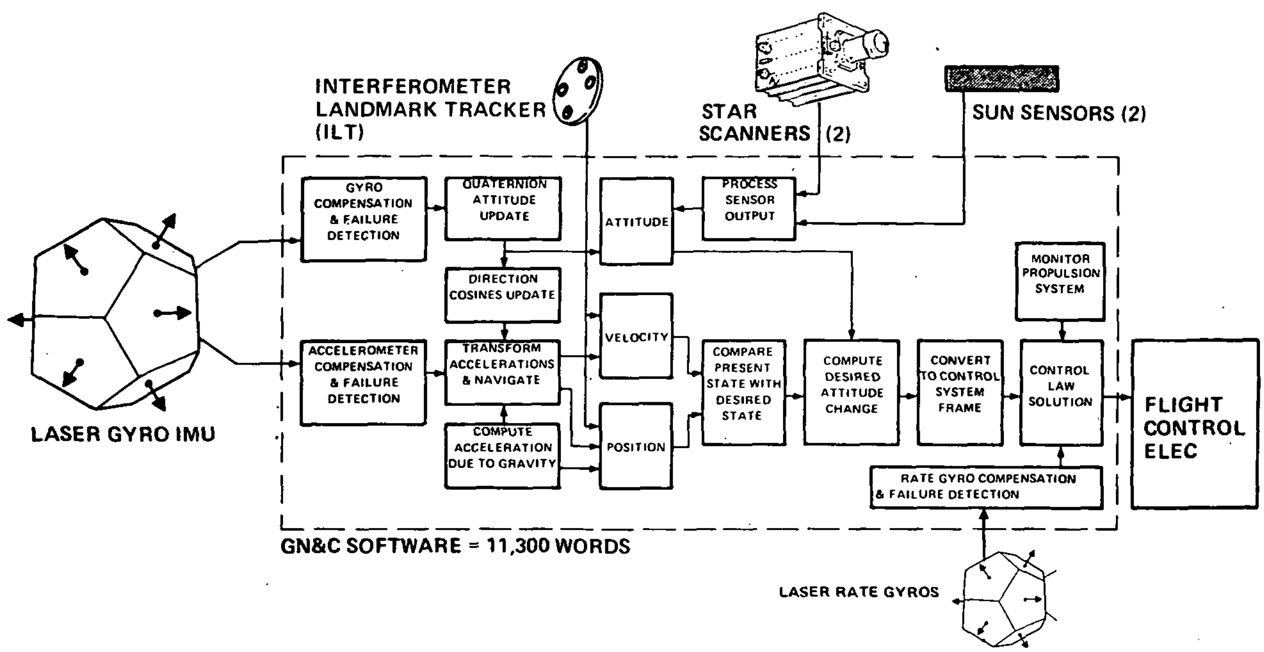


Figure 4-1. Baseline GN&C System

The flight control electronics package accepts the attitude control system commands from the DMS and appropriately commands either the Auxiliary Propulsion System (APS) or the main engine Thrust Vector Control (TVC) system, depending on whether the main engine is firing. During main engine firings, roll control is maintained by the APS.

Figure 4-1 shows how the DMS software integrates the functions of the individual GN&C packages. The GN&C packages all operate essentially independently of each other and therefore will be described separately in the following sections. Figure 4-1 also indicates the estimated software required for each function. The left side of the software box represents the basic IMU navigation function resulting in vehicle attitude, velocity, and position. The right side of the box indicates the guidance and control functions resulting in commands to the flight control electronics. The Kalman filters for the ILT update system are not shown in detail in this figure.

The update systems require more complex guidance computations than are usually required in an inertial guidance system. This is because the updates require that the Tug re-target its mission based on the new data. Also, changes in the Orbiter ephemerus can require the Tug to re-target its return.

4.2 NAVIGATION IMU

The Inertial Measurement Unit (IMU) for the Tug is a Sperry strapped-down laser gyro system in a dodecahedron configuration. As shown in Figure 4-2 the dodecahedron defines six vectors that are all at the same spherical angle of 63.4 degrees (1.1 radians) from each other. The input axes of the six accelerometers and the six gyros are then placed parallel to these vectors. This yields a redundant IMU configuration that can survive two failures.

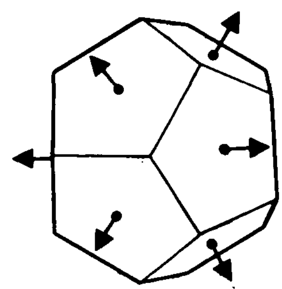


Figure 4-2. Dodecahedron IMU

The laser gyros employed are Sperry ASLG-15 laser gyros. The ring laser gyro consists of an optical cavity around which counter-rotating laser light beams travel. A laser tube emits beams that are confined to a closed triangular path defined by three mirrors located at the corners of the triangle. The gyro input axis is perpendicular to the plane of the light beams. A rotation of the gyro about this axis causes the light beam traveling in the same direction as the rotation to travel farther than the beam traveling in the other direction. This results in a motion of the interference pattern on one of the mirrors, which is detected by a photocell. This net output becomes a series of electrical pulses proportional to the angular rate applied to the gyro.

To obtain the required performance from the gyro, several control functions must be added. The path length of the gyro light beam must be accurately controlled to an integral number of wavelengths to sustain the light beam. This is accomplished by measuring the beam intensity and using this signal to position one of the mirrors to maximize beam intensity. This mirror is mounted on a piezoelectric actuator that dithers at a rapid rate. From this dither, the maximum point of beam intensity can be detected, and a dc bias is applied to servo the average mirror position to the point of maximum intensity. On turn on, to initiate laser oscillation, the piezoelectric actuator is commanded over a wide range until the loop locks in.

An additional control function is required to keep the gyro from latching-up at low input rates. This phenomenon occurs in laser gyros because of the coupling between the two counter-rotating beams. Latching-up is avoided in the Sperry gyro by inserting a bias cell in one of the reflecting mirrors. This cell effectively delays the one beam more than the other beam, which produces an apparent path length difference for the two beams. This results in a constant output from the gyro, even at zero input, which avoids the latching-up phenomenon at zero output. The data from the gyro is then biased by this constant offset, which is compensated for in the computer. The gyro characteristics are shown in Table 4-1.

Table 4-1. Sperry Model ASLG-15 Ring Laser Gyro Characteristics

Physical	
Configuration	Single axis with integral electronics and multi-layer magnetic shield
Size	Triangular: 7 in. base (17.8 cm), 4 in. high (10.2 cm), 2 in. thick (5.1 cm)
Weight	4 pounds (1.8 kg)
Power	4 watts
Optical Cavity	Equilateral triangle with 15 in. (38.4 cm) perimeter
Transition	1.15 microns
Nominal Scale Factor	3.3 arc seconds (16 μradians)
Bias Type	Optical
Discharge Tube	Separable metal/pyrex
Performance	
Random drift (1)	0.01 deg/hr (174 μrad/hr)
Day-to-day repeatability	0.5 deg/hr (8725 μrad/hr)
Scale factor (nominal)	3.3 arc-sec/pulse (16 μrad/pulse)
Scale factor repeatability	0.005%
Scale factor linearity	0.01%
"g" sensitivity	NIL
Threshold	0.01 deg/hr (174 μrad/hr)
Warm-up	30 minutes
Life	20,000 hours
Angular rate range	±400 deg/sec (±6.98 rad/sec)
Angular acceleration range	±1000 deg/sec^2 (±17.45 rad/sec^2)
Linear acceleration	10g

The accelerometer used is a Kearfott Model 2401. This is a standard, inertial accelerometer employing a pendulous proof mass supported by a flexure pivot. A capacitive pickoff is used to generate errors signals that are then used to generate signals to the magnetic torquer that rebalances the pendulum. The accelerometer performance parameters are listed in Table 4-2.

Table 4-2. Kearfott 2401 Accelerometer Performance

Zero-g bias	32,000 pulses/sec
Scale factor (nominal)	1,760 pulses/sec/g
Scale factor stability	60 ppm
Bias repeatability	±0.0004g
Bias stability	0.000015g
Temperature sensitivity	0.00001g/°F (0.0000055g/°K)
Warm-up	40 minutes
Nonlinearity (second order)	$0.00001g/g^2$
Pivot axis cross coupling	$0.00001g/g^2$
Acceleration range	±18g
Acceleration threshold	0.018 ft/sec (0.0055 m/sec)

The characteristics of the total IMU is presented in Table 4-3. Each of the accelerometer and gyro channels is completely independent of the other channels, with each channel having its own output register. The IMU data output will be parallel from the 12 registers to the Digital Interface Unit (DIU), under DIU control. In addition, the time reference signal, some limited built-in test discretes, and an on-off monitor will be provided. For instrumentation purposes, temperature data, accelerometer demodulator output, laser tube discharge current, and total IMU current will be provided to the DIU.

When operating, all IMU channels will be on and providing data to the DMS. The computer in the DMS will check the six gyro inputs for consistency. Because any gyro's output can be calculated from any three of the other gyro's output, each gyro can be functionally checked against the others in the set. In this manner, up to two channel failures can be isolated and the system survive without degradation. With two failures, the remaining four channels are sufficient to verify proper operation, but are not sufficient to isolate another failed channel. The accelerometer data is processed in a completely analogous manner, using many of the same software subroutines. Therefore, the dodecahedron IMU provides a fully redundant system capable of surviving two failures with no degradation.

Table 4-3. Dodecahedron IMU Characteristics

Dimensions	
Cluster	9 in. x 9 in. dia (22.9 cm x 22.9 cm dia.)
Power Supply and Electronics	1000 in^3 (16,390 cm^3)
Weight	
Cluster	25 lb (11.3 kg)
Power Supply and Electronics	30 lb (13.6 kg)
Functional Description	
Six Axes of Body Rate Sensing	
Six Axes of Body Acceleration Sensing	
Twelve Output Registers for Rate & Acceleration Words. I/O under Tug Computer Control	
Redundancy Management & Fault Determination under Tug Computer Control	
Power	200 watts dc
Reliability and Maintainability	
Ten Year Shelf, or Five Year Operational Life	
Wear-out Item, Discharge Tube	
Calibration, Alignment, Fault Detection, & Test; all Under Tug Computer Control	
Line Replaceable Units via Cluster Assembly and Power Supply/ Electronics Unit Assembly	
MTBF	
MTBF for 3 axis level of six axis cluster (gyro & accelerometer). (Single redundancy mode.)	
$\frac{1}{MTBF} = (76.10 + 33.0) \times 10^{-6}$	
MTBF = 10,000 hours	

4.3 POSITION AND VELOCITY UPDATE COMPONENTS

The Interferometric Landmark Tracker (ILT) is used to provide position and velocity update for the Tug. The ILT system uses a subset of the more than 8000 radars around the world as known landmarks. By taking repetitive readings of the directions to these radar stations, the orbital parameters of the Tug can be calculated.

In operation, about 300 radar stations will be used as the landmarks for the system. The longitude, latitude, altitude, and rf carrier frequency of each station will be stored in the DMS computer. As shown in Figure 4-3, as a radar station comes into view, the ILT will be commanded to the station's frequency. When a pitch and roll attitude reading on the station is obtained, it is checked against the expected coordinates of that station, and stored. If a reading on the station is not obtained, or if the attitude readings are grossly different than expected, the DMS computer will proceed to the next available radar station. In this manner, many attitude fixes to known landmarks will be obtained. If during the interval of the fixes, no propulsive burns have occurred, then the Tug orbit can be calculated. Since there will be many fixes, the orbit will be overdetermined and a Kalman filter will be employed to get the best estimate of the orbit.

The ILT hardware consists of a four antenna array and a four channel receiver as shown in Figure 4-4. The four planar spiral antennas are mounted along the vehicle yaw axis, 40 inches (102 cm) apart. The rf signal is received by all four antennas and sent to the four channel receiver. After amplification and processing, the phase relationships between the signals are output to the DIU. From the phase difference between the signal received by two antennas, the angle to the transmitting station in the plane of the two antennas can be calculated. There are several solutions for this angle, in general, but this ambiguity can be resolved using prior knowledge of the approximate angle and an offset antenna, as shown in Figure 4-4. The same process is repeated for the other two antennas.

To perform an accurate update, the inertial attitude of the ILT array must be known at all times. This is provided by the IMU with the attitude update system bounding the long term IMU attitude errors. The mechanical alignment of the IMU to the antenna array will be controlled during assembly and will be updated in the flight environment by adding three states to the Kalman filter.

Another major error source of the ILT system is the electrical phase shift uncertainty in the signal processing in the receiver. This is compensated for by injecting a calibration signal of the same frequency and amplitude as the radar signal immediately following receipt of the radar signal. This signal is processed in the same manner as the radar signals, and the results are used to compensate for the receiver phase errors.

The ILT system provides inherent redundancy by the use of four antennas and receivers. If one of the four channels should fail, which would be detected by the calibration test previously described, the system could still operate, although somewhat degraded,

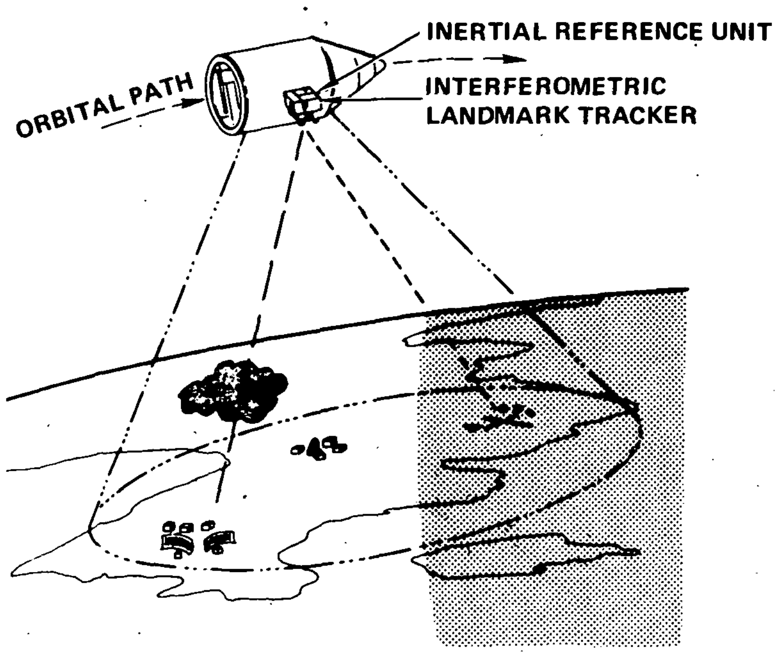


Figure 4-3. ILT Operation

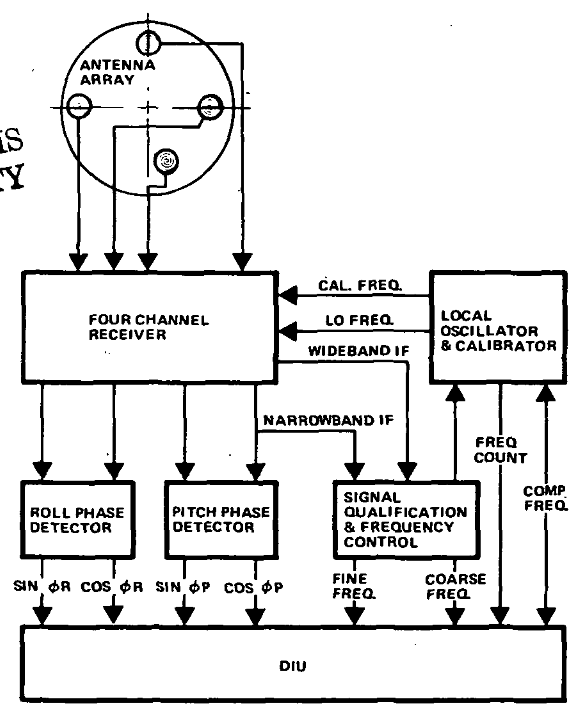


Figure 4-4. ILT System Hardware

with three channels. In addition, if a total failure should occur in the ILT system, ground radar tracking data can provide back-up position and velocity update. As is true of all the update systems, the effects of not performing the update results in a gradual degradation of performance.

The estimated weight of the ILT system is 20 pounds (9.1 kg) with an average power consumption of 15 watts.

4.4 ATTITUDE UPDATE COMPONENTS

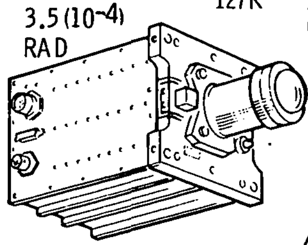
The attitude update components for the Tug are a pair of star trackers and a pair of sun sensors. The star tracker is a Bendix fixed head, image disector type and the sun sensor is an Adcole digital device, as shown in Figure 4-5..

The star tracker has an eight degree (0.14 radian) field of view while in search mode. This eight by eight degree (0.14 by 0.14 radian) square is raster scanned until the brightest star in the field is found. The tracker then switches to the acquired mode, scanning a small cruciform about the acquired star. The X and Y outputs of the tracker now represent the position of the star relative to the boresight of the star tracker.

The Tug will point the star tracker so that the desired star is the brightest star in its field of view. When the star is acquired, the Tug attitude control system will null

REQUIREMENT – 0.04 DEG 6.9 (10^{-4}) RAD ACCURACY NEEDED TO MEET INJECTION VELOCITY

SYSTEM	PERFORMANCE (1σ)	UNIT COST ($)	WEIGHT	REDUNDANCY	FEATURES	MANUFACTURER
STARTRACKER	0.02 DEG 3.5 (10^{-4}) RAD	127K	16 LB (7.3 KG)	DUAL	FIXED HEAD, IMAGE DISECTOR TYPE	BENDIX
SUN SENSOR	0.03 DEG 5.2 (10^{-4}) RAD	65K	4.5 LB (2.0 KG)	DUAL	64 DEG (1.1 RAD) FIELD OF VIEW, DIGITAL OUTPUT	ADCOLE



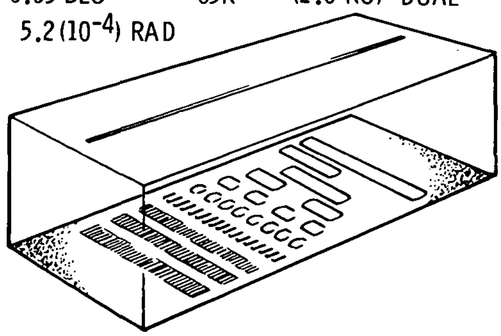


Figure 4-5. Attitude Update System

the star along the star tracker's boresight. When the star is nulled, the inertial to body conversion matrix in the DMS computer will be corrected.

Two star trackers will be implemented to provide redundancy. A small light emitting diode will be used as a test source to test the star tracker. This test, along with reasonableness tests on the tracker data, will allow determination of a failed tracker.

The star trackers have a sun shutter that is automatically actuated if the sun approaches the tracker field of view and also an overcurrent monitor that disables the high voltage if there is too much light falling on the tracker. These features allow the Tug to be fully an all-attitude vehicle without constraints from the star tracker.

The sun sensors complement the star trackers. Although a three axis update could be obtained using only the star trackers, this would require rotating the vehicle through approximately 90 degrees (1.6 radians) to a second star. By proper star selection, a three-axis update using the star tracker and sun sensor can be obtained with little disturbance to the vehicle attitude. The sun sensor also provides wide angle capture with no ambiguity. Therefore, the sun sensor, with the star tracker, can provide Tug attitude determination even if the IMU has lost inertial reference.

The selected sun sensor is an Adcole digital sun sensor with a 64 degree (1.1 radian) field of view. It is a solid state device with direct digital output. The X and Y outputs would be implemented using a separate sensor for each axis. The sun sensors would also be dual redundant, using a light source and reasonableness tests to eliminate the failed unit in the same manner as the star trackers.

4.5 FLIGHT CONTROL COMPONENTS

The flight control components consist of the rate gyros and the flight control electronics package. The rate gyros are a dodecahedron configuration of laser gyros very similar to the IMU cluster, except the gyros are smaller. The details of the rate gyro cluster are shown in Table 4-4. The laser rate gyros operate in an identical manner to the IMU gyros described in Section 4.2. The output of the rate gyros is sent to the DMS computer as shown in Figure 4-1.

In the DMS computer, the data from the six rate gyros is compared to determine if any of the channels has failed. This is done in the same manner and using much of the same software as used for the IMU gyro failure detection. After the rate gyro data is checked for failures, it is combined with the attitude command from the guidance system to obtain a vehicle engine command. If the vehicle is in coast phase, the proper APS engines will be commanded on to null the errors. If the main engine is firing, the TVC system will command the engine gimbals to the correct angle to null the pitch and yaw errors. During main engine burns, roll control will be maintained by the APS system.

Table 4-4. Rate Gyro Characteristics

Dimensions	
Cluster	4-1/2 in x 4-1/2 in dia. (11.4 x 11.4 cm dia)
Power Supply and Electronics	500 in^3 (8190 cm^3)
Weight	
Cluster	5 lb (2.3 kg)
Power Supply and Electronics	15 lb (6.8 kg)
Functional Description	
Six Axes of Body Rate Sensing	
Six Output Registers for Rate I/O under Tug Computer Control	
Redundancy Management & Fault Determination under Tug Computer Control	
Power	100 watts dc
Reliability and Maintainability	
Ten Year Shelf, or Five Year Operational Life	
Wear-out Item, Discharge Tube	
Calibration, Alignment, Fault Detection, & Test; all under Tug Computer Control	
Line Replaceable Units via Cluster Assembly and Power Supply/ Electronic Unit Assembly	
MTBF	
MTBF for three-axis level of six-axis cluster. (Single redundancy mode.)	
MTBF = 13,140 hours	

The translation of the DMS computer commands to the actual TVC servovalve commands and APS engines commands is accomplished by the flight control electronics. Figure 4-6 shows one of the two TVC channels in the flight control electronics. The TVC engine position command from the DMS computer is received by the flight control electronics from the redundant DIU's. The engine command is differenced with the engine position feedback in three identical channels. The difference between the commanded position and actual position results in a command to three identical servovalves. Through a force summing actuator, these three outputs are voted, with the majority command being applied to the engine. In this manner, a single-failure-tolerant system is implemented with no voting logic required in the electronics.

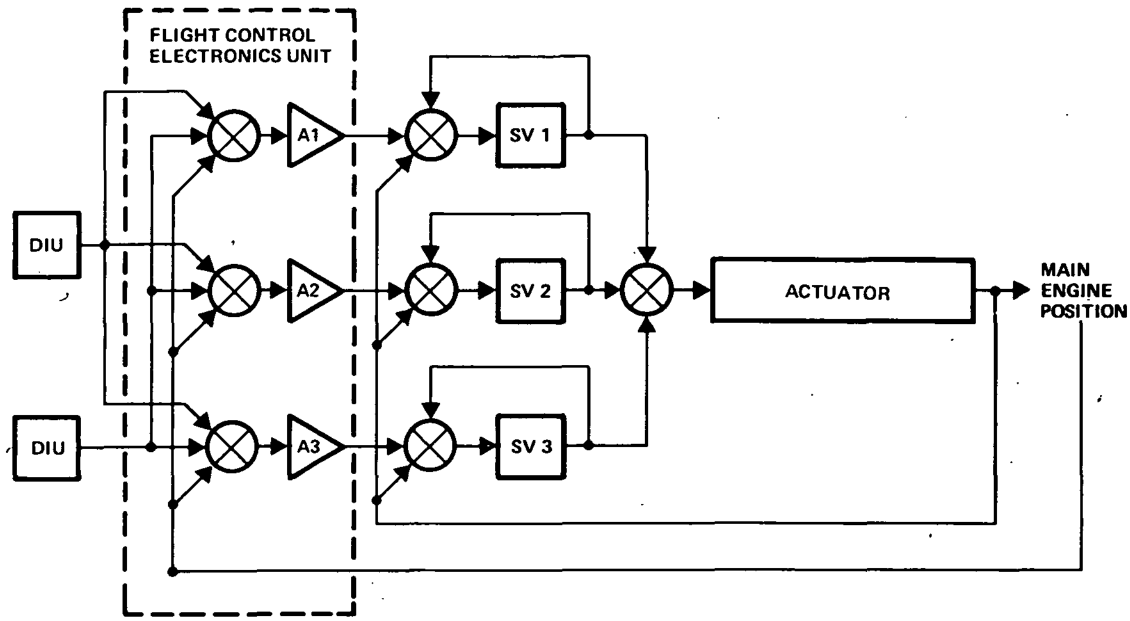


Figure 4-6. Thrust Vector Control

The APS engine commands originate in the DMS computer, which commands each engine on using discretes. These discretes are accepted by the flight control electronics into a dual-redundant amplifier. One of the dual-amplifier channels, channel A, is connected to APS engine valves. If the DMS computer detects that an engine is not coming on as commanded, it will switch the APS engines to the redundant channel, channel B. The computer detects a channel failure by observing vehicle rates obtained from the IMU. If the engine still does not respond, it will be commanded off and an alternative engine will be used to control the vehicle. Since there are redundant engines as well as redundant electronics to drive the engines, two levels of redundancy are obtained, which results in the system being capable of surviving multiple failures.

The flight control electronics unit, containing both the TVC servoamplifiers and the APS drivers, weighs 50 lb (22.8 kg) and consumes 50 watts.

4.6 SOFTWARE

Figure 4-1 summarizes the software required for the Guidance, Navigation, and Control subsystem. The navigation software, after the failure detection and compensation is accomplished, requires fairly standard strapdown navigation equations. Since the vehicle angular rates are low and the vibration environment is benign except during the relatively short engine burns, the strapdown integration algorithms need not be iterated at a high rate. After the strapdown equations calculate the body to inertial conversion matrix, the accelerometer data, after failure detection and compensation, is transformed to the inertial coordinate frame and integrated to provide vehicle position and velocity.

The ILT system requires considerable software to perform the position and velocity update. First the longitude, latitude, and frequency of the 300 radar stations must be stored. Then the logic required to select the next appropriate radar station and tuning the ILT to its frequency must be implemented. And finally, a multistate Kalman filter is required to compute the position and velocity data from the ILT data.

The attitude update system requires only the command and control software to sequence the update and select the appropriate stars and the inertial position and intensity of the approximately 50 stars required. The control logic will have to select stars that are not too near the sun or earth and are close to the present attitude of the star tracker. Also, the software to command the attitude update sensors to perform a self test and evaluate the data is required.

The guidance software must compute the optimum trajectory to the target and calculate the optimum steering law to inject the Tug onto the optimum trajectory. From the steering software the desired vehicle thrust vector is calculated. The autopilot takes this attitude reference, combines it with the vehicle rates from the rate gyros, and calculates the TVC engine commands that are output to the flight control electronics. During coast phase the APS engine commands are derived in a similar manner except that the vehicle attitude command would not come from the guidance software and engine command logic is required to select the proper engine to fire. The TVC system requires no assistance from the DMS for redundancy management, since its output is voted by the hydraulic system. The APS system does require the DMS computer to observe vehicle rates to isolate flight control electronics and APS engine failures.

The total memory requirement for the GN&C software is summarized below:

Navigation	4,371 words
Guidance	4,595
Flight Control	2,340
	11,306 words

SECTION 5

RENDEZVOUS AND DOCKING SUBSYSTEM DESCRIPTION

5.1 BASELINE CONFIGURATION

Equipment	No. Req	Dimensions, in. (cm) Length	Width	Height	Unit Op Power (Watts)	Unit Weight lb (kg)	Subsystem Weight lb (kg)
Rendezvous & Docking							63 (28.6)
Scanning Ladar	1	6 (15.2)	8 (20.3)	17 (43.2)	10	28 (12.7)	
Ladar Electronics	1	9 (22.9)	9 (22.9)	11 (27.9)	30	11 (5.0)	
TV Camera* & Electronics	2	3.5 (8.9)	6 (15.2)	15 (38.1)	10	8/13 (3.6/6.0)	
TV Strobe Lamps	4	3.5 (8.9)	3.5 (8.9)	3.5 (8.9)	-	0.25 (0.1)	
Strobe Electronics	2	2 (5.1)	3.5 (8.9)	2.5 (6.3)	-	1 (0.5)	

*One with 10:1 zoom lens

The Rendezvous and Docking Subsystem baseline is a hybrid configuration combining the best merits of an autonomous sensor subsystem (Scanning Ladar), principally for rendezvous, and a remote-manned subsystem (slow-scan low light level TV), principally for docking.

There are six functional elements associated with rendezvous and docking with a spacecraft (SC), as shown in Figure 5-1.

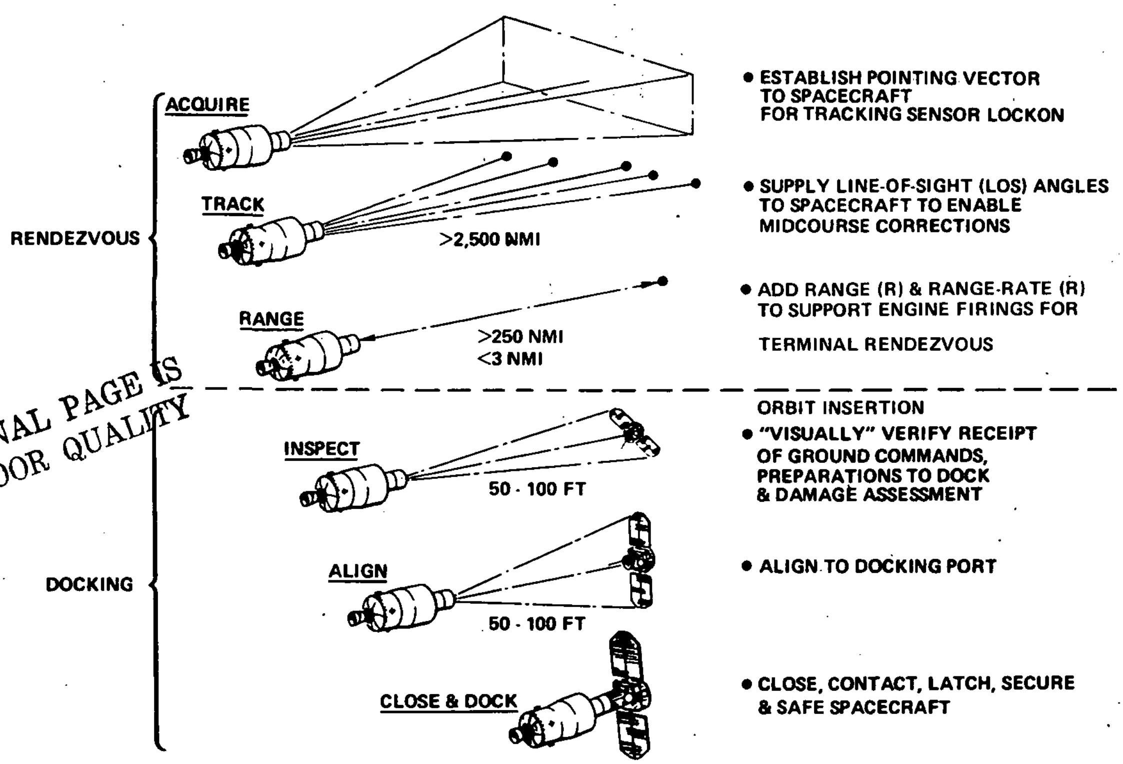


Figure 5-1. Functional Elements of Rendezvous and Docking

Acquisition entails either the searching of the dispersion volume produced by Tug's uncertainty in its position or an accurate knowledge of the pointing vector from Tug to SC. When the SC pointing vector is established, the subsystem locks the tracking sensor into the tracking mode.

Tracking entails measuring the line-of-sight (LOS) angles to the SC. Midcourse corrections become necessary to ensure that the desired target point for the initiation of the insertion (main engine) burn is achieved. The addition of LOS information to the navigation subsystem's Kalman filter reduces the relative ephemeris errors between Tug and SC and provides for much finer guidance. The degree that this can be achieved is a function of the maximum range of the tracking sensor as well as the basic navigation accuracy of Tug.

Ranging is an element of both the rendezvous and the docking functions. If direct ascent rendezvous at geosynchronous orbit is being employed, the approach is for Tug to insert into co-orbit in the near vicinity of the SC. Range is a necessary input to the navigation update from LOS tracking if accurate insertion of <3 n. mi. (5.5 km) is desired. The range measurement must occur at LOS distances gr3ater than 250 n. mi. (550 km) to allow sufficient time for the update computation, Tug orientation maneuver prior to burn, and the main engine firing. The range capability of a sensor would have to exceed 250 n. mi. (550 km). Post-injection ranging must also be accomplished as an element of docking to gain early control of the relative velocity between Tug and SC.

Inspection is accomplished at a standoff of typically 50 to 100 feet (15 to 30 m) and requires only gross range information.

Alignment on the docking port involves the Tug performing an orbit maneuver about the SC to achieve a gross alignment for docking sensor lockon.

Closure and docking is accomplished by closing at a controlled range rate and minimizing rotations of the LOS. Relative velocities between the Tug and the SC must be carefully controlled to ensure that there is no possibility of collision and that the auxiliary propulsion system exhaust impingement be minimized. Target-relative attitude information (pitch, yaw, and roll) is required as well as range, range rate, and LOS angles information to accomplish docking.

The division between the rendezvous and docking elements is somewhat arbitrary but is intended to divide the mission into phases that have requirements unique from each other. Thus "docking" was extended to include the terminal rendezvous (small trim) burns that eventually match the SC position and velocity.

Performance of the rendezvous and docking function is dependent on the sensors of the rendezvous and docking subsystem, the navigation and guidance capabilities of the Guidance, Navigation, and Control subsystem, the computational support provided by the Data Management subsystem, and the all-attitude communication link to the ground.

Figure 5-2 depicts all of these components although the Tug's baseline rendezvous and docking subsystem consists only of the scanning ladar, the low light level TV, their associated electronics, strobe lights, and the computer memory dedicated to rendezvous and docking software.

The role of each sensor as it relates to the six phases is presented in Table 5-1.

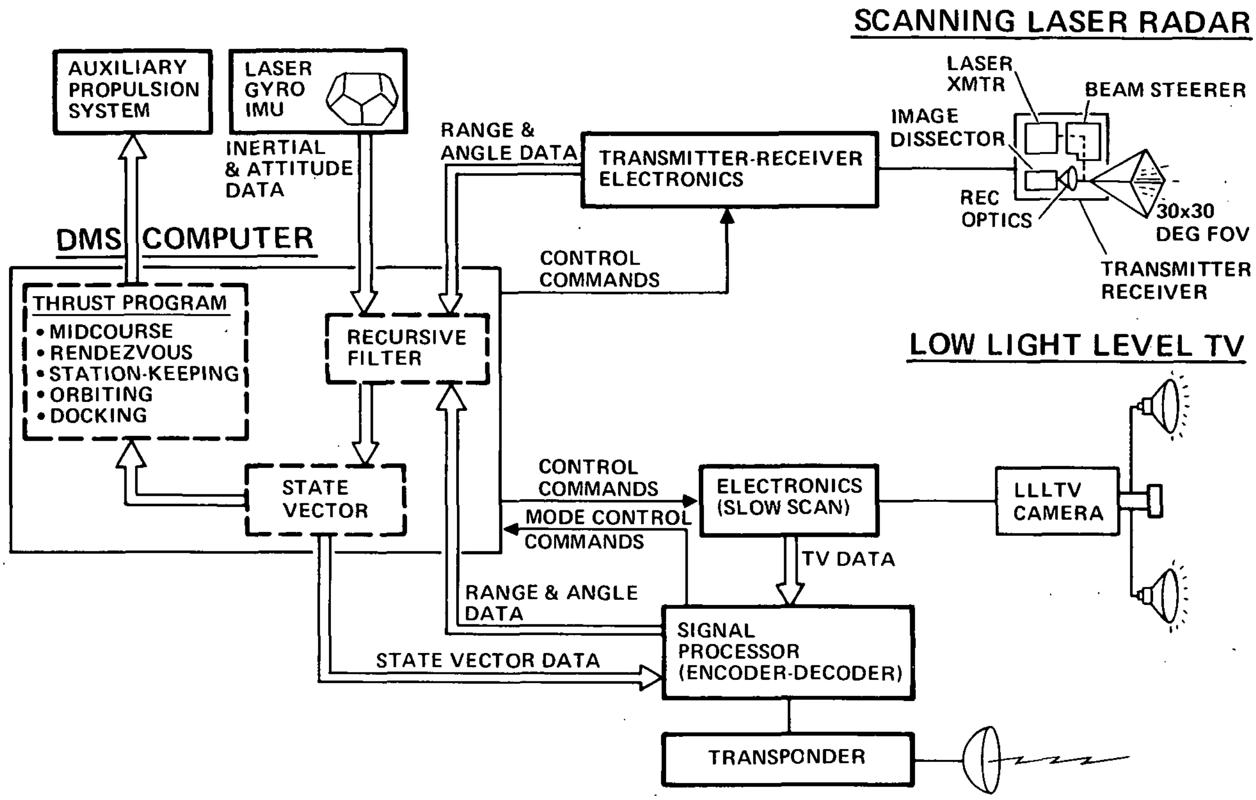


Figure 5-2. Baseline Rendezvous and Docking System

Table 5-1. Sensor Role in Rendezvous and Docking Phases

Function	Scanning Ladar	Slow-Scan LLLTV
Acquisition	Primary	Backup
Tracking	Primary	Backup
Ranging		
Preinjection	Primary	
Postinjection	Primary	Backup
Inspection		Primary
Alignment to Axes		Primary
Closure & Docking		
Initial Operational Capability (ICC)		Primary
Fully Operational	Primary	Backup

5.2 SUBSYSTEM OPERATION

The subsystem operation is described in the following sections by mission phase and explains the primary/backup role for each sensor.

5.2.1 SPACECRAFT ACQUISITION. Figure 5-3 illustrates the degradation of onboard knowledge of the line-of-sight (LOS) as a function of target range. The autonomous navigation subsystem's knowledge degrades inversely with range due to the uncertainty in both Tug and SC positions. At long ranges, no sensor can compete with this knowledge. This means that the SC is, in effect, acquired throughout the mission with an accuracy that is inversely proportional to target range.

Acquisition confirmation of a passive target has been limited to sun illuminated target conditions. The GaAs SLR in a passive mode, that is, using the image disector and the sun illuminated target, will serve as the primary acquisition sensor. (Commanding the GaAs SLR into the passive acquisition mode will necessitate the removal of the 0.9μ bandpass filter fromthe receiver optics of the Gen-3 prototype.) The information required is simply line-of-sight (LOS) angles to the target and can optionally be supplied by the LLLTV camera. LOS will be obtained by scanning the complete detected pattern into the computer. After "viewing" a number of frames, the computer will make an actual target determination. The laser and beam steerer will be inactive during this phase.

Having made an actual target decision, the angular coordinates of this target will be sent by the computer to the SLR to initiate the track mode. Frame time during the acquisition phase is 140 seconds.

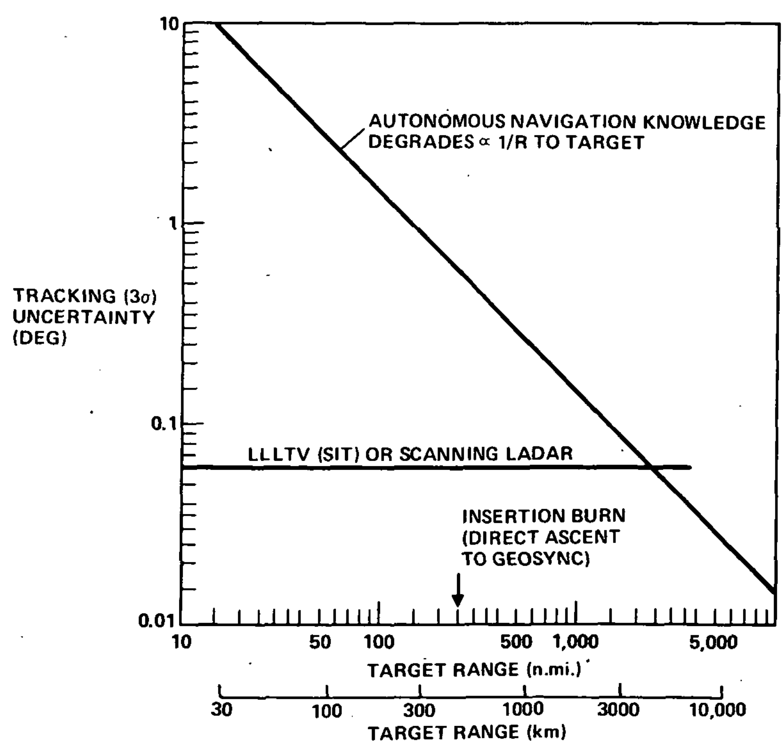


Figure 5-3. Tracking Uncertainty Versus Range for Navigation System and R&D Sensors

5.2.2 SPACECRAFT TRACKING. Below 2500 n. mi. (4650 km), as shown in Figure 5-3, either of the subsystem sensors can provide LOS measurements to an accuracy better than that calculated onboard using navigation knowledge. These LOS measurements, when added to the navigation subsystem's Kalman filter, provide a further refinement in knowledge of the relative state between the Tug and SC as well as one order of magnitude improvement in the knowledge of LOS prior to the injection burn (direct ascent to geosynchronous orbit). This can minimize both the time and impulse necessary to rendezvous and dock with a SC in a significantly different orbit.

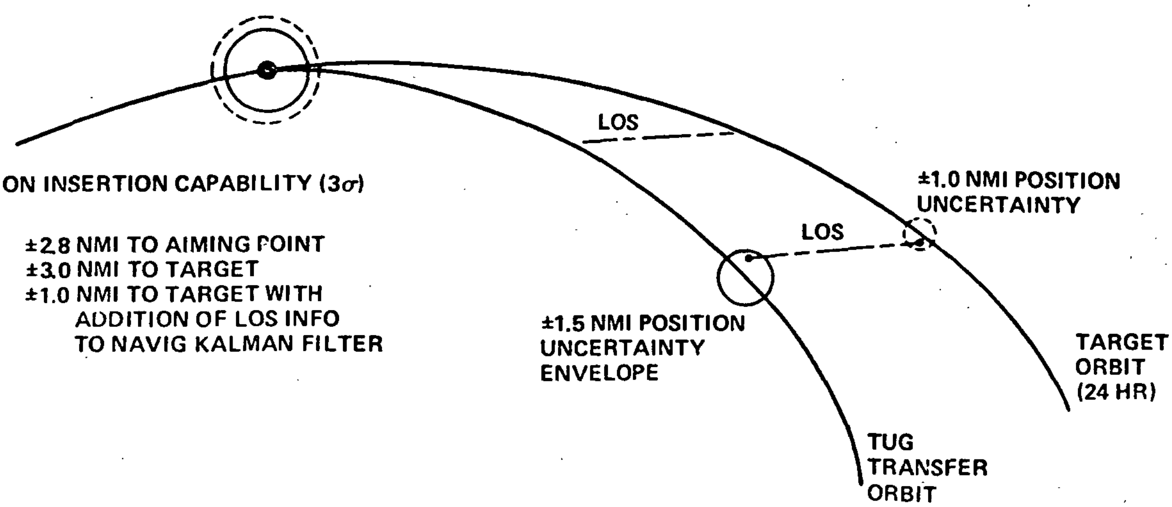
Tracking the SC at these long ranges requires that the SC be solar illuminated, with a cone angle of less than about 20 degrees about the LOS. This represents an added mission constraint that must be traded against the improved insertion accuracy available from long-range tracking. Thus this technique may be selectively employed using either sensor. If not employed, however, the targeting point must be additionally offset to limit the LOS uncertainty cone as well as to provide time to reacquire the spacecraft by a systematic searching of the uncertainty cone postinjection. With either sensor, SC acquisition is repeatedly confirmed by comparing the sighting with those of background stars of visual magnitude equal to or greater than the minimum calculated for the SC at that range. It is this registration that ensures the 0.06-degree tracking accuracy for these sensors. The required star catalog is quite manageable, about 70 words (representing a three-degree field of view (FOV).

SLR is designated the primary sensor for this mission phase; in the event of its failure, LLLTV provides a performance-equivalent backup.

5.2.3 SPACECRAFT RANGING, PREINJECTION. The range at which preinjection ranging information must become available (if it is to be used) varies with the rendezvous technique and the required main engine burn duration. A frequent application might be a direct ascent rendezvous with the spacecraft at geosynchronous orbit, such as would be the case for a servicing sortie. This would require approximately a four-minute insertion burn, with a reorientation prior to burn some 250 n. mi. (465 km) from the target. (Lower SC orbits and reinsertion from phasing orbits would require reorientation at considerably shorter ranges.)

Additional studies are required to finalize this trade; however, it is clear that the navigation subsystem accuracy is sufficient to ensure insertion in close proximity to the SC, thus obviating all requirements for rendezvous sensors (in distinct contrast to extended docking sensors, see Section 5.2.4), as illustrated in Figure 5-4. This means that long-range tracking and (particularly) preinjection ranging should be justified upon the improvements gained beyond that available from the navigation subsystem.

Note that LLLTV has no preinjection ranging capability. Thus, in event of a failure of the primary sensor in this phase, the navigation subsystem will effect rendezvous from onboard knowledge.



- POSITION INSERTION CAPABILITY (3σ)

  ±2.8 NMI TO AIMING POINT
  ±3.0 NMI TO TARGET
  ±1.0 NMI TO TARGET WITH ADDITION OF LOS INFO TO NAVIG KALMAN FILTER

- INSERTION VELOCITY RESIDUALS EASILY REMOVED BY NULLING LOS RATES WHILE CLOSING WITH TARGET

Figure 5-4. Autonomous Navigation Rendezvous Capability

5.2.4 SPACECRAFT RANGING, POSTINJECTION. The range at which postinjection ranging information must become available is only a function of the navigation, guidance and control capability of Tug. Although time and impulse performance to close and dock would suffer, the autonomous navigation subsystem can easily place Tug within 3 n.mi. (5.5 km) of the target SC, thus placing an upper limit on the maximum range for postinjection ranging. However, there is no way to close and dock with the spacecraft without deriving target-relative range and LOS data. That is, the autonomous navigation subsystem capability is sufficient for rendezvous, but postinjection ranging is required for docking.

On postinjection reacquisition, SLR easily provides range and LOS measurements to an accuracy more than sufficient for this purpose. LLLTV (in a functional backup mode) cannot easily provide range (being only a detector) but provides LOS to an accuracy commensurate with SLR – providing it remains on the sunlit side of the SC (a targeting consideration). To range with LLLTV, the target's cross section must first be measured (linear or area measure) and then compared with a cross-section reference. Since most spacecraft projections vary with orientation, an assessment must be made as to which SC feature is to be measured and how the measurement reference is to be selected. This jugemental process is more appropriately accomplished with man-in-the-loop methods since only man can provide the discretionary judgement in situ. (Pattern recognition schemes require an a priori determination of each decision variable and each allowable combination of these, even though that combination might be adaptive.)

Simulation studies have shown (Volume IV, Section 5.4) that insertion position and velocity knowledge possessed by the navigation subsystem postinsertion, augmented by LOS data from LLLTV, is sufficient to enable gradual closure with the spacecraft until a range-lock can be secured. Range-lock with LLLTV occurs before 0.5 n.mi. (0.9 km) but range accuracy is an exponential function of range, that is, a function of discernible target detail. Once range-lock has been achieved, controlled closure to within inspection distances is easily accomplished with LLLTV and man-in-the-loop.

GaAs SLR will serve as the primary sensor during the postinjection ranging phase, supplying LOS initially and subsequently range. Using the angular coordinates generated during the track phase, a narrow field of view active track (laser operating) will be repeatedly attempted by the SLR. Thus, the range of the SLR will be greatly extended by the field of view restriction (an important consideration during orbital phasing operations). If the SLR fails to obtain range lock in active track around the directed angle, the system will return to the passive tracking phase and await a later attempt at ranging.

The SLR will repeatedly attempt to obtain active track. Range should be obtained by 85 n.mi. (160 km), and the SLR will remain in the ranging mode unless a computer-generated command causes it to track passively with a new set of direction coordinates. This capability allows the computer to make a false target decision after passive or active track has been obtained and break track lock on the false target. Frame time during the track phase is to be 14 seconds.

A failure of SLR in this mission phase requires the use of LLLTV and remote, manned closure with the attendant substantial reduction of the closure rate and an unavoidable increase in the APS propellant consumption. The LLLTV additionally requires a variable field of view from 2 to 10 degrees.

5.2.5 SPACECRAFT INSPECTION. Visual inspection of the SC was a requirement originally met by a continuous TV allocated to the communication subsystem. This capability can more than adequately be met by the slow-scan LLLTV and the visual inspector seated at the remotely situated docking console. Further, predocking inspection is a necessary functional prelude to docking. This being the case, SC inspection was transferred to the Rendezvous and Docking Subsystem. The strobe lamps are sufficient to obviate the requirement for solar illumination during the inspection, alignment, and final docking mission phases.

Inspection could also be accomplished using the SLR if the intensity of the return signal were output throughout a full (acquisition) scan. Since this is inherent in a TV scan and would unavoidably complicate the SLR design, it was decided to provide a redundant LLLTV in the event of failure of the primary unit. The redundant unit can use a fixed, 30-degree FOV with "pan," "tilt," and "zoom" accomplished electronically either on the original image (via a scan converter at the ground station) or within the image section of spaceborne LLLTV cameras. The weight penalty for this redundant unit is 8 lb (3.6 kg).

The optional encryption device (communication subsystem) can easily ensure security of the transmitted image since it is serially slow-scanned and digitally encoded prior to transmission.

5.2.6 ALIGNMENT TO SPACECRAFT DOCKING AXES. Simulation studies have shown that alignment to the docking port axes is easily accomplished with LLLTV and is a natural adjunct to the inspection function. Conversely, alignment via SLR in a totally autonomous fashion is a complex task involving pattern recognition of a deterministic placement of retroreflectors on the SC. Subsystem simplicity results in designating LLLTV as both primary and backup (redundant unit) sensors for the alignment as well as inspection functions.

The port search/alignment phase is to consist of Tug slowly orbiting the target vehicle at a radius of approximately 75 ft (25 m). Location of the docking port using the TV camera (used for inspection) would terminate this phase. The SLR is to be used to provide range and LOS angles to the target vehicle. This information will be used to maintain the desired orbit characteristics.

In this mode, the SLR will be in the active track mode and should continue to track a single target; however, it is likely that the target will be lost during this phase. From a distance of 75 ft (25 m) a retroreflector will not necessarily remain in the field of view when Tug is not aligned with the docking port. Essentially this phase is a continuation of the track phase as far as the SLR is concerned, as the frame time and field of view are constant. If a retroreflector drops out of the field of view or the intensity drops substantially (off-axis performance), the SLR will be commanded to revert to the acquisition mode supplying range, LOS angles, and target intensity for each target in the field of view with intensity within a factor of (TBD) of the reference intensity (established at the beginning of the port search mode). This will require additional gain discrimination logic. The computer will discriminate another retroreflector in the field of view and command active track mode (supplying the LOS), thus once again establishing active track. This operation may be repeated many times before the docking port is located (via TV) and confirmed (via SLR).

5.2.7 CLOSURE AND DOCKING. The primary subsystem for final docking is LLLTV. Performance assessment will be conducted on SLR during docking to validate its capabilities. It is intended that SLR subsequently become the primary sensor and that docking be accomplished autonomously, with LLLTV in the assessment/backup role. Such an approach facilitates an orderly development of an autonomous subsystem without incurring unnecessary risk, cost, or schedule impacts.

Operationally, SLR would be the primary sensor and provide precise range and angular information for docking. Docking phase commences with a ±5-degree (coarse) alignment to the docking port and range equal to approximately 75 ft (25 m).

Prior to initiation of the docking phase, the computer will have commanded an acquisition mode and discriminated the four retroreflectors within the field of view (as before) while aligned to ±5 degrees of the docking port. (This determination could optionally be accomplished via TV.) The LOS angles are then supplied for each of up to four retroreflectors on initiation of the docking mode with the SLR establishing active track on all four. If the computer detects that the SLR has failed to establish track or that it is tracking something other than a retroreflector, the docking mode can be reinitiated with revised LOS angles, or the acquisition mode reinitiated to redetect the retroreflectors.

During the docking phase, the frame time for the SLR is to be 1.4 seconds.

5.3 SCANNING LADAR

The Scanning LADAR (LAser Detection And Ranging) sensor employs a scanning laser beam to detect and subsequently range-lock on the target. The detector synchronously scans with the laser beam or operates independently to passively detect a solar illuminated target and to provide line-of-sight (LOS) measurements to enhance navigation accuracy.

The selected autonomous baseline sensor component remains the GaAs scanning ladar (SLR), prototypes of which have been developed for NASA/MSFC by ITT. The Generation-3 SLR development (Figure 5-5) incorporates a docking capability by discriminating among four retroreflectors in a skewed-T configuration; the fourth retroreflector is used only to identify those lying on a line and for roll indexing. The SLR is located on Tug's front end for a forward view.

Generation-3 was developed primarily for the purpose of determining the relative orientation between Tug and the spacecraft's docking axes. It additionally incorporates improved components, e.g., high speed digital logic and gimballess beam steerers. A description of this sensor system follows.

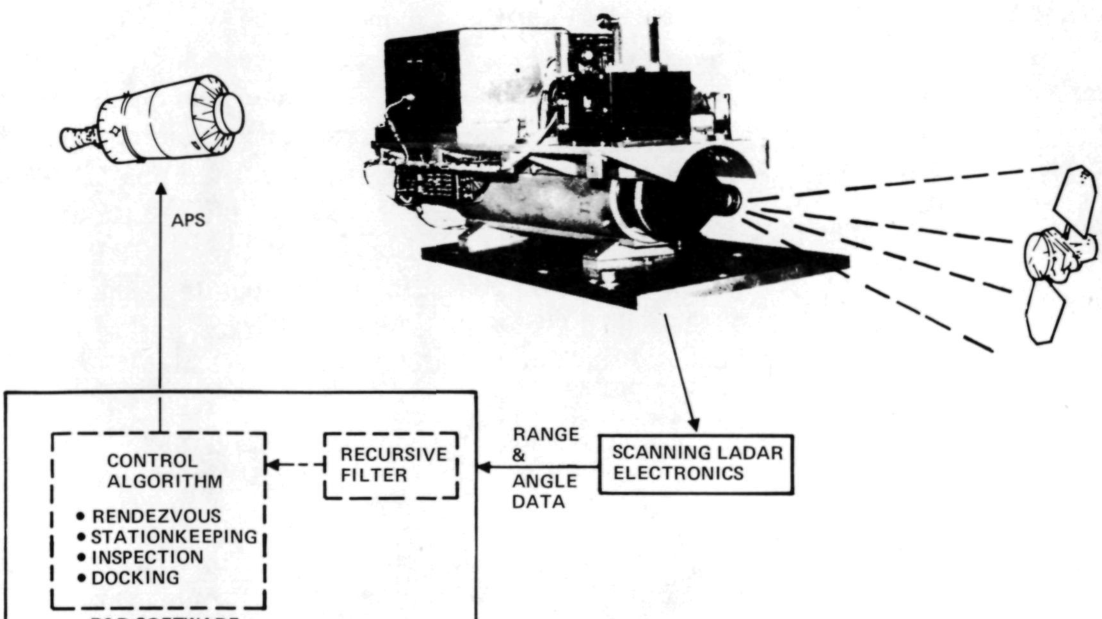
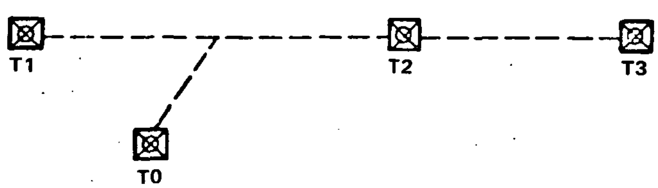


Figure 5-5. GaAs Scanning Ladar Generation-3

5.3.1 FUNCTIONAL DESCRIPTION. At long ranges (to 85 n. mi. or 160 km), the Gen-3 GaAs SLR signal outputs consist of two LOS angles, their rates, range, and range rate. Rates are pseudo and are obtained within the sensor by differencing. LOS angles are updated and output every 64 ms with horizontal and vertical outputs alternating at 32 ms. Range is accumulated, smoothed, and output every 512 ms. LOS angular rate is arbitrarily updated every 512 ms, together with range rate. All output is parallel digital. LOS angles and angular rates are 14 bit, range and range rate are 32 bit.

In addition, during the docking (multitarget) mode, LOS angles to two other strategically placed retroreflectors (T1 and T2) are also output in addition to the central docking retroreflector (T0) (see sketch). Currently, the retroreflector arrangement on the docking face is a skewed T, with T0 on the docking centerline and out of the paper. For spacecraft applications, it is proposed that T0 be located within the docking port (into the paper) and T1, T2, and T3 located on the docking face. T3 is used only to identify the T1-T2 baseline and for roll polarity, and can be subsequently dropped from the scan.



Distinguishing T0 from (T1, T2, and T3) and T2 from T3 is currently done internal to the Gen-3 SLR; the baseline SLR assumes this task is done by the data management subsystem computer.

5.3.2 PHYSICAL CHARACTERISTICS. The Gen-3 system employs an IBM-developed, single-mode GaAs laser carefully aligned within an optical resonator. It lazes at 0.9040 microns with a bandwidth of 0.0030 microns (30 Å), a rise time of 10 ns and outputs 0.5 watts peak power at a pulse repetition frequency (PRF) of 1 kHz.

Table 5-2 presents the characteristics required of the baseline GaAs Scanning Ladar. The Gen-3 prototype would require rework to meet these specifications. The principal modifications assumed are 1) an improvement in the laser by eliminating the requirement for critical, precise alignment within an optical cavity, 2) the elimination of computational functions that can be readily (and more reliably) done by the DMS computer, 3) the addition of redundant circuitry, 4) the application of composite technology to the system's optical bench, 5) repackaging for flat-pack installation on Tug's forward equipment shelf, and 6) the addition of self-test features.

Table 5-2 is based on the target characteristics presented in Table 5-3.

5.3.3 PACKAGE CONFIGURATION. The SLR is to be configured into two line replaceable units (LRUs): a transmitter/receiver and an electronics LRU. The transmitter/receiver is required to transmit and receive signals normal to its mounting plane. The packages should easily conform to the size and weight specified in Table 5-2. A failure in the transmitter/receiver or the electronics package should require replacement of that package only; it is required that they not be matched pairs. It is acknowledged that this will require selected electronics to be placed within the transmitter/receiver.

Table 5-2. GaAs Scanning Ladar Characteristics

1. LRU Size (L × W × H)	
Receiver/Transmitter	20 × 6 × 8 in. (51 × 15 × 20 cm)
Electronics	11 × 9 × 9 in. (28 × 23 × 23 cm)
2. Weight	
Receiver/Transmitter	28 lb (13 kg)
Electronics	11 lb (5 kg)
3. Power (watts)	40 Active 30 Passive
4. Duty Cycle	
Acquisition	100% Passive
Tracking	100% Passive/Active
Port Search	Same as Track Mode
Docking	100% Active
5. Resolution	
Track Range	±1 meter
Port Search/Docking Range	±10 cm
Angular Resolution (LOS)	±0.03 degree
Docking Mode Relative Attitude	±1 degree at 25 meters
6. Required Bandwidth (output)	≈183 Hz
7. Commands (4)	On-Off Active (Filter In)/Passive (Filter Out) Acquisition-Track-Dock Track AZ-Track EL (up to four targets)
8. Laser	1. Wafer Type Using Transfer Optics Rather Than a Cavity 2. 0.5 W Peak Power 3. 1 kHz Rep Rate
9. Optics	Up to 10 cm Entrance Aperture
10. Processing	
Passive Mode	LOS Angles Only
Active Mode:	
Acquisition & Track	LOS Angles, Range* and Target Intensity
Docking	LOS Angles and Range for Up to Four Targets
11. MTBF	(TBD) Hours

*Track only

Table 5-3. Assumed Target Characteristics

Retroreflector Diameter	10 cm
Projected Target Area	10 m^2
Projected Diffuse Target Area	0.5 m^2
Visible Bidirectional Reflectivity	1.6×10^{-2} sr^{-1}
Retroreflector Bidirectional Reflectivity	$\frac{0.2\ D^2}{\lambda^2}$ sr^{-1}
Retroreflector Off-Axis Efficiency	0.2 at 30 degrees
Number of Acquisition Retros	12 16 Total
Number of Docking Retros	
Target CEP	1 n. mi. (1.85 km)
Target Intensity 0.4-0.8 μ	108 w/sr
Target Cross Section Cooperative	1.93×10^{11} cm^2/sr
Target Cross Section Uncooperative	1.6×10^3 cm^2/sr

5.3.4 SELF TEST. It is required that the GaAs SLR be capable of self testing. The type of test required is quantitative as well as qualitative, e.g., internal testing of laser power output and beam steering alignment are required. Additionally, external targets will be used in the Shuttle bay and on the spacecraft (if attached) to determine end-to-end operation. The standard targets will be displaced from the Tug at a known distance and angle that is stored in the computer. SLR measured range and angle will be compared to the stored value to determine end-to-end operation.

A conceptual practical design for these testing features is a laser power output sensor at the entrance to the beam steerer, a pinhole deflection sensor at the exit of the beam steerer (slightly outside of the field of view but within the deflection range), and LED emitter within the receiver optics. The computer will be the recipient of the laser power output signal.

5.3.5 SOFTWARE SUPPORT. Reliability is increased and power/weight reduced by delegating to the onboard computer all those functions that it can accomplish, with due consideration to its speed and memory limitations. The SLR will supply range and LOS angles (elevation and azimuth) for each of up to four targets within its field of view (docking mode). Target discrimination, docking range, and spacecraft orientation (3 DOF) will then be accomplished by the computer.

In the passive acquisition mode, LOS angles and intensity will be supplied for each detected target. In the active acquisition mode, range is additionally supplied and detection of numerous targets is inhibited by discrimination against targets (TBD) below an established reference intensity. In both cases, target rejection can be accomplished by forcing tracking mode (active or passive) and supplying LOS angles to the desired target (or targets) for docking mode.

5.3.6 SLR/TUG INTERFACE. The functional interface is summarized in the preceding two tables. It should be noted that _28 Vdc, +4.5 Vdc, -4 Vdc is the single power source available on Tug. Additional requirements on this power source were not investigated. Ground commands are also available in addition to the established computer interface and are principally used for ground checkout (Table 5-2). These commands and all telemetry requirements will be interfaced via a digital interface unit (DIU) for input to the DMS computer.

5.4 SLOW-SCAN, LOW LIGHT LEVEL TELEVISION

The low light level television (LLLTV) sensor uses a conventional low light level TV camera operated in a slow-scan mode. Single television frames are taken once every 16 seconds and transmitted to a ground station where a console supervisor evaluates the displayed data. Strobe lamps provide illumination for each frame's exposure when in close proximity to the spacecraft; the sun provides illumination when the spacecraft is at a distance.

A manned, remote simulation has demonstrated the feasibility of LLLTV during the spacecraft acquisition, tracking, late terminal rendezvous, inspection (orbiting spacecraft), docking-axis alignment, and docking phases of the mission. Its feasibility as an early terminal rendezvous sensor has not been demonstrated by simulation.

5.4.1 FUNCTIONAL DESCRIPTION. The LLLTV sensor system performs two primary functions: manned, remote docking and visual inspection of the spacecraft. The requirement for visual inspection of the spacecraft after deployment or prior to docking can be met using a "snapshot" TV approach as depicted in Figure 5-6. The system consists of a fixed-mount TV camera with an electronic shutter, wide angle lens (30-degree (0.52 rad) field of view) and a silicon intensified target (SIT) vidicon. A snapshot of the spacecraft is taken by momentary exposure of the SIT vidicon. The vidicon retains the image until read out by a scanning electron beam. A slow scan rate and 4 bit gray level encoding result in a digital data rate of 50 Kbps as compared to the 2.5 MHz bandwidth of general-purpose television.

The image is transmitted to a ground-based console for viewing by an operator. A scan converter at the ground station reconstructs the image where it is stored in a video disc file for operator retrieval and examination.

This "snapshot" system has a significant weight advantage since the gimbals and drive mechanisms for camera pan and tilt are eliminated. The 30-degree field of view lens allows visual coverage over a wide range of separation distances. The operator can select any section of the image frame (pan, tilt capability) and electronically enhance

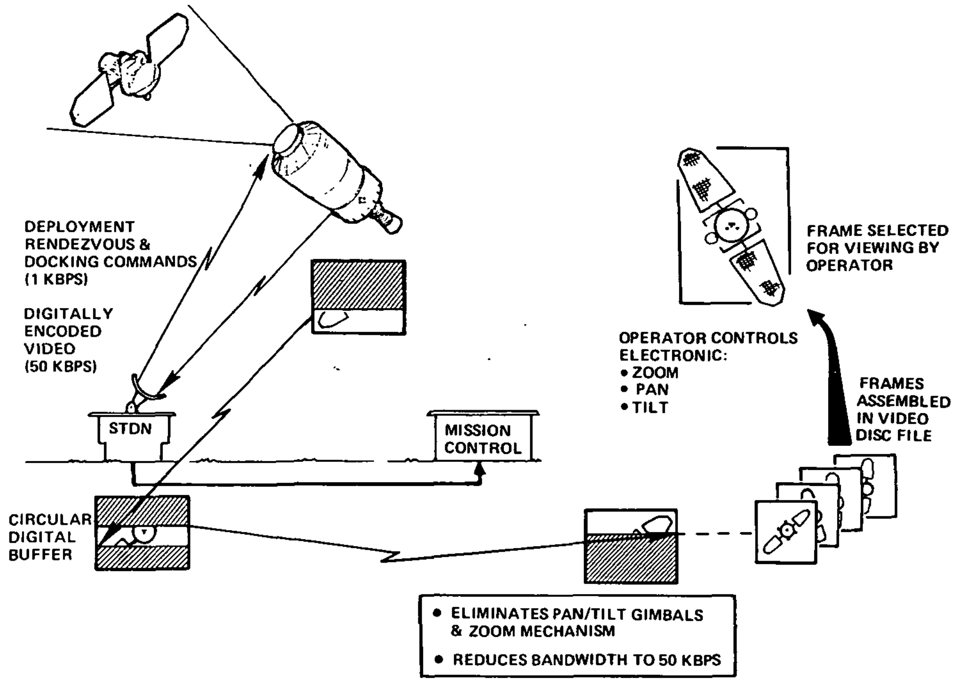


Figure 5-6. Slow-scan LLLTV Operation

(zoom capability) that section for close inspection at some sacrifice in resolution. Resolution could be subsequently recovered on the next frame by an electronic pan, tilt, and zoom within the image section of the spaceborne vidicon should this become a requirement.

The snapshot system of providing a single image to the operator every 16 seconds for his evaluation and control has been demonstrated as a successful technique for accomplishing Tug rendezvous and docking with a spacecraft. The elements of the operator's console are shown in Figure 5-7. The spacecraft image as taken by the Tug's TV camera is processed by the scan converter, displayed on the TV screen, and stored in the video disc recorder or video tape for future operator retrieval. The operator's console contains the controls for positioning, sizing, and orienting a reticle by which range and attitude correction commands are generated. The ground-based computer processes the Tug's state vector information with the operator's reticle adjustments and provides the range and angle correction data to the Tug's flight computer for execution. Tug mode controls provide on-off discretes and override commands. The data being transferred to and from the Tug are separated for clarity.

The docking strategy for the remote-manned subsystem is to place the remote operator in a supervisor's role rather than a controller's role. This means that he can operate at a much reduced task load, delegating much of the operation to the spaceborne and ground computers. In essence, Tug provides task continuity and the basic docking operation, whereas the supervisor operates as a feedback sensor (via positioning the reticle) removing accumulated biases, and accomplishes overall operation evaluation/decision making.

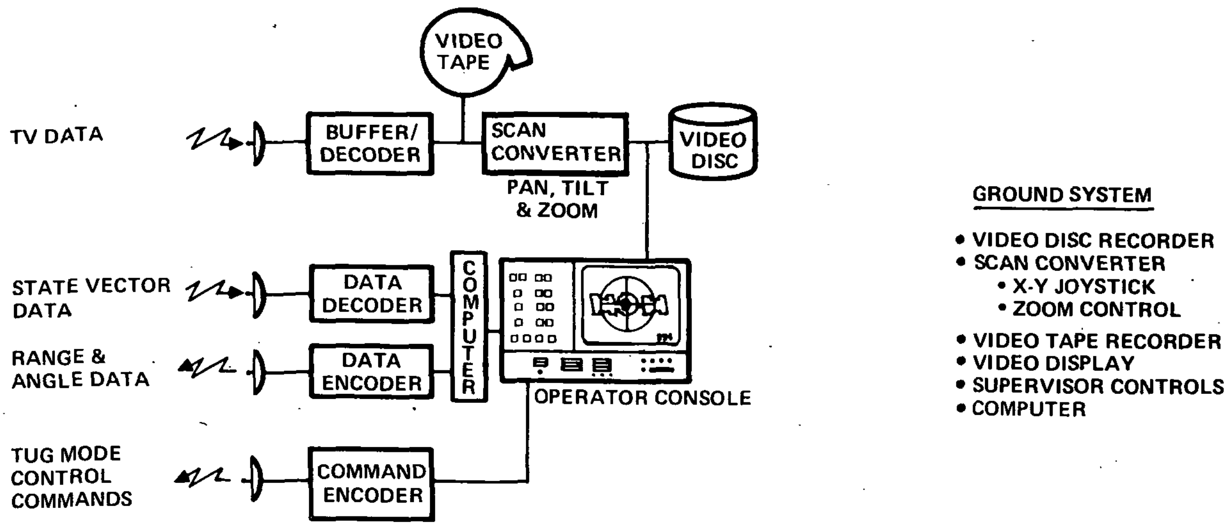


Figure 5-7. Elements of the Ground-based Operator's Console System

Each image or frame is received approximately on 16-second centers (Figure 5-8). The reticle is ground computer driven based on information known to Tug at exposure time. If the supervisor detects a discrepancy, he takes control of the reticle by pressing the mode SELECT switch putting it in LOCAL control. He then positions (joystick), sizes (rotary pot), and orients (large rotary control) the reticle to the spacecraft docking port (presuming this had shown the discrepancy). With the CROSS SECTION switch on PORT, a return to REMOTE control activates the computer, which interprets the measurements as PORT measurements and computes Tug pitch and yaw (from reticle location), relative roll (from orientation), and range (from size).

If the discrepancy is on the target "T" (shown within the port), the crosshair need only be positioned on the "T" (joystick) to enable a computation of spacecraft "pitch" (about the horizontal) and "yaw" (about the verticle). To indicate TARGET measurements, it is necessary to hold the springloaded CROSS SECTION switch in this position while returning to REMOTE control.

The supervisor's console for Convair's Manned-Remote Rendezvous and Docking Simulation study is representative of what would be required at a ground installation (Figure 5-9). In addition to the digital displays — to the left of the TV monitor — are status, caution, and warning lights on the facade below the monitor. Controls for placing, sizing, and orienting the range reticle — shown on the screen — are contained on the central console panel. It is the reticle that provides the principal feedback from the ground-based supervisor. In this sense, the supervisor is not a controller or pilot.

The panel immediately to the right of the reticle controls (detailed in Figure 5-10) commands the flight mode and closure velocity (within limits). On the far right are the video disc controls. On the far left are the spacecraft controls that are operative only if the spacecraft happens to be active cooperative (photo details not available).

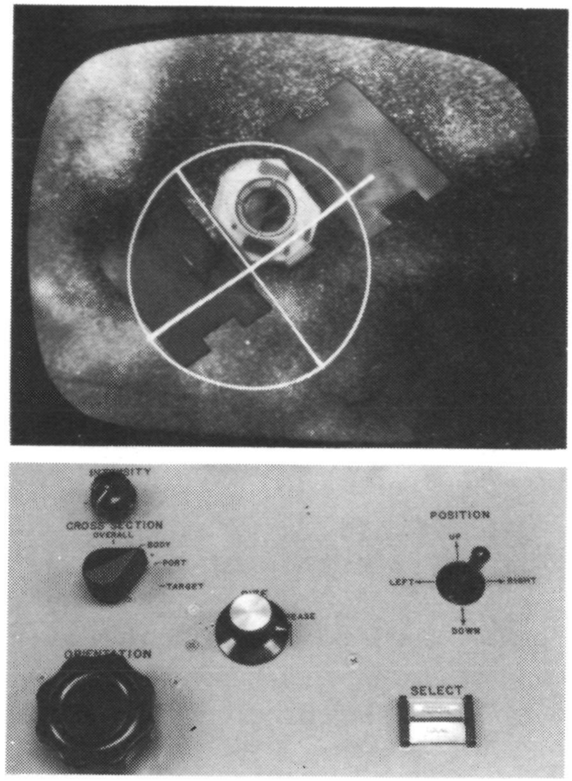


Figure 5-8. Remote, Manned Docking Procedure

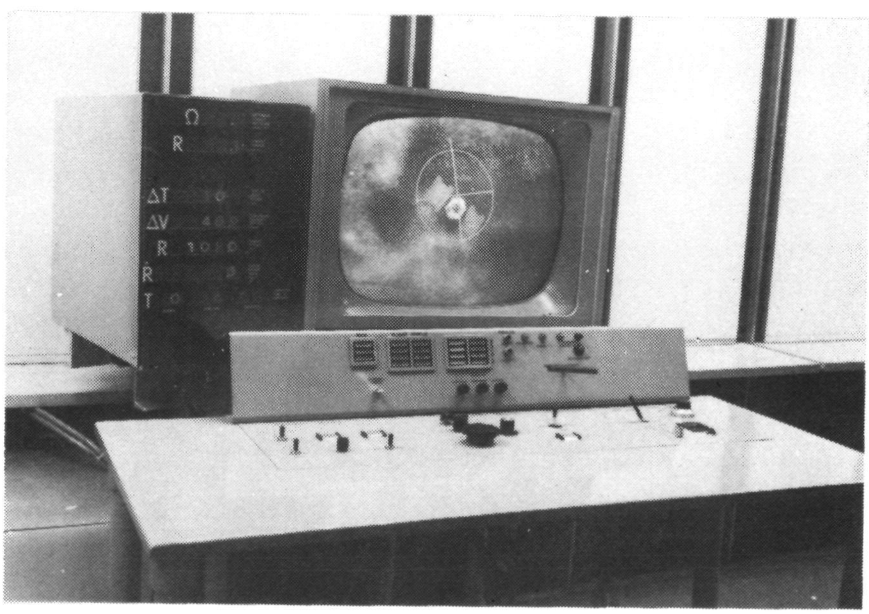


Figure 5-9. Typical Remote Supervisor's Console
(Convair's Simulation Study)

Figure 5-11 is more representative of the supervisor's console that would be available in 1983. The principal display would be a large screen diode matrix rather than the conventional rear-projection TV available today. Digital displays and controls are integrated into the console below the large display. A fully separate, small screen display and console would be located nearby for backup in event of a failure.

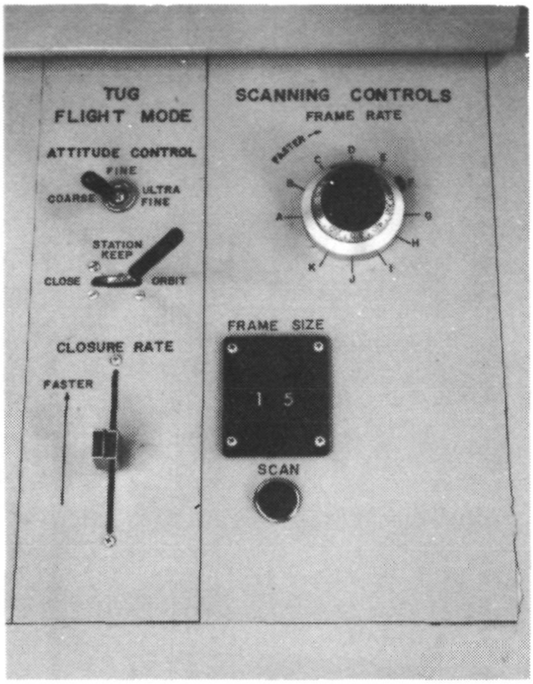


Figure 5-10. Tug Flight Mode and Video Disc Recorder

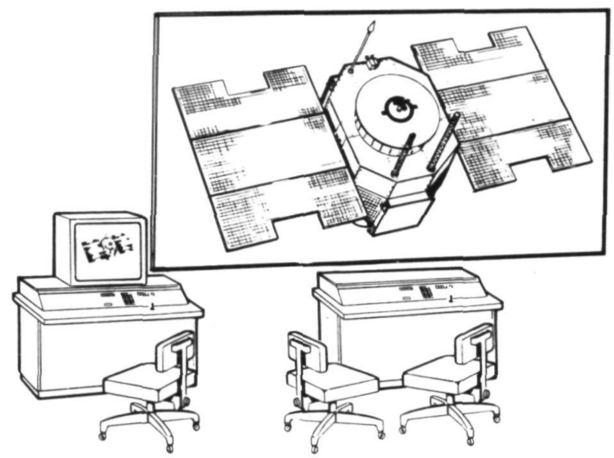


Figure 5-11. Envisioned Remote Supervisor's Console, Circa 1983

5.4.2 PHYSICAL CHARACTERISTICS. Based on the requirements for acquisition, tracking, port search, docking, and visual inspection, the required LLLTV sensor characteristics are as contained in Table 5-4. The principal requirements are high MTBF, low weight, and low cost (in that order). Innovations (e.g., charged coupled device imaging techniques) that better meet these requirements have not been investigated. Four strobe lamp assemblies — similar to camera flash attachments — are cross-strapped to two capacitance-discharge electronic units as a highly reliable, lightweight, negligible power illumination source for visual inspection/docking.

To back up the primary system (GaAs SLR) during terminal rendezvous, one of the two redundant cameras must employ a variable field of view (FOV) lens to enhance image detail at long distances. A zoom lens was chosen since its failure consequence is much less serious than the equivalent turret lens. A compromise of a wide-open 20-degree versus a desired 30-degree FOV for docking makes it a suitable backup to the primary docking alignment sensor (the 30-degree FOV LLLTV).

5.4.3 PACKAGE CONFIGURATION. The television camera system is to be configured into a single line replaceable unit (LRU) roughly conforming to the dimensions presented in Section 5.1. The image input (receiving) axis is to be normal to its mounting plane to facilitate installation on Tug's forward equipment shelf.

Table 5-4. Slow-scan LLLTV Characteristics

Sensitivity (S/N -1)	1.6×10^{-11} f. c. (point source irradiance)
Fields of view	30 degrees (fixed) and 2 to 20 degrees (zoom)
Focus range	6 ft to infinity
Resolution	500 X 500 lines
Frame times	140 and 14 seconds
Exposure time	0.2 seconds (with electronic shuttering)
Scan rate	One frame every 140 and 14 seconds
Image persistance	20 seconds (after shuttered)
Output bandwidth	3 kHz and 30 kHz
AGC	10^4
Dynamic range	10^7 (including AGC)
Camera survivability	Direct look at sun, any FOV
Approximate size	$15 \times 3.5 \times 6.0$ inches
Viewing direction	Normal to mounting plane
Approximate weight	8 pounds
Approximate power	8 W operational, 1.5 W standby $\}+28^{+4.5}_{-4.0}$ Vdc
Commands	On-standby-off AGC override Frame/scan rate Shutter open/closed (electronic) Test pattern
Mean time between failures (MTBF)	TBD hours
Self test	Internally generated test pattern

5.4.4 SELF TEST. It is required that the television system be capable of generating an internal test pattern for use in status verification and health checking (in conjunction with the onboard computer). Sixteen gray shades are sufficient for this purpose.

For maximum weight reduction, the test pattern may bypass the camera lens optics. Testing may be requested at any time during operation, but generally only for a single frame. All test mechanisms must be designed to fail operational; that is, the test pattern should fail rather than normal camera operation.

Reseau marks should also be emplaced on the faceplate to facilitate measurement of display distortion during actual operation.

5.4.5 SOFTWARE SUPPORT. The DMS (onboard) and ground computers are available to support the television system in both operation and test. It is the ground computer that provides line-of-sight angle to the target from the supervisor's operation on the television picture and determines status and health from the internal test pattern.

Simplification of the hardware in favor of a software operation is desirable from a cost and reliability point of view. Recommendations of potential functions to be accomplished by computer command are contained in Table 5-4.

5.4.6 LLLTV/TUG INTERFACE. The 28, +4.5, -4.0 Vdc power source is the single power supply available. Additional requirements on this supply were not investigated.

The commands listed in Table 5-4 will be computer supplied and/or ground supplied through a command decoder. Slow scan video output will be interfaced with a digital interface unit (DIU) that will digitally encode the signal for processing to the computer and/or to the ground. No buffering of this signal is anticipated. All telemetry requirements will be similarly interfaced via a DIU.

SECTION 6

COMMUNICATIONS SUBSYSTEM

6.1 BASELINE CONFIGURATION

Equipment	No. Req	Dimensions, in. (cm) Length	Width	Height	Unit Op Power (Watts)	Unit Weight lb (kg)	Subsystem Weight lb (kg)
Communications							149 (67.7)
Phased Array Antenna	3	3.5 x 15 (8.9 x 38.1) Diameter			93	16 (7.3)	
Hemispherical Antenna	2	4.6 (11.7)	6 (15.2)	2 (5.1)	-	1 (0.45)	
RF Network	1	3.3 (8.4)	3.8 (9.6)	1 (2.5)	-	2 (0.9)	
RF Switch	1	5 (12.7)	5 (12.7)	6.3 (16.0)	3	7.3 (3.3)	
Transponder	2	15 (38.1)	7 (17.8)	6 (15.2)	16	16.5 (7.5)	
Signal Processor	2	13.5 (34.3)	6 (15.2)	5.6 (14.2)	18	11 (5.0)	
Command Distribution Unit	1	5 (12.7)	5 (12.7)	4 (10.2)	35	18 (8.2)	
Encrypter	2	5.8 (14.7)	4.3 (10.9)	5.3 (13.5)	7	4.3 (1.95)	
Decrypter	2	6 (15.2)	3.6 (9.1)	5.8 (14.7)	2.4	4.1 (1.9)	

The Communications subsystem provides telemetry, tracking, and control capability at S-band frequencies either directly to STDN or AFSCF ground stations, or by TDRS satellite relay. Major elements of the selected baseline configuration, as shown in Figure 6-1, are a steerable phased array antenna system for transmit directivity to TDRS satellites, omnidirectional transmit and receive antennas for minimal attitude restrictions when communicating near the Orbiter, Shuttle-era transponder and signal processor hardware to minimize development cost for Tug peculiar units, and GFE

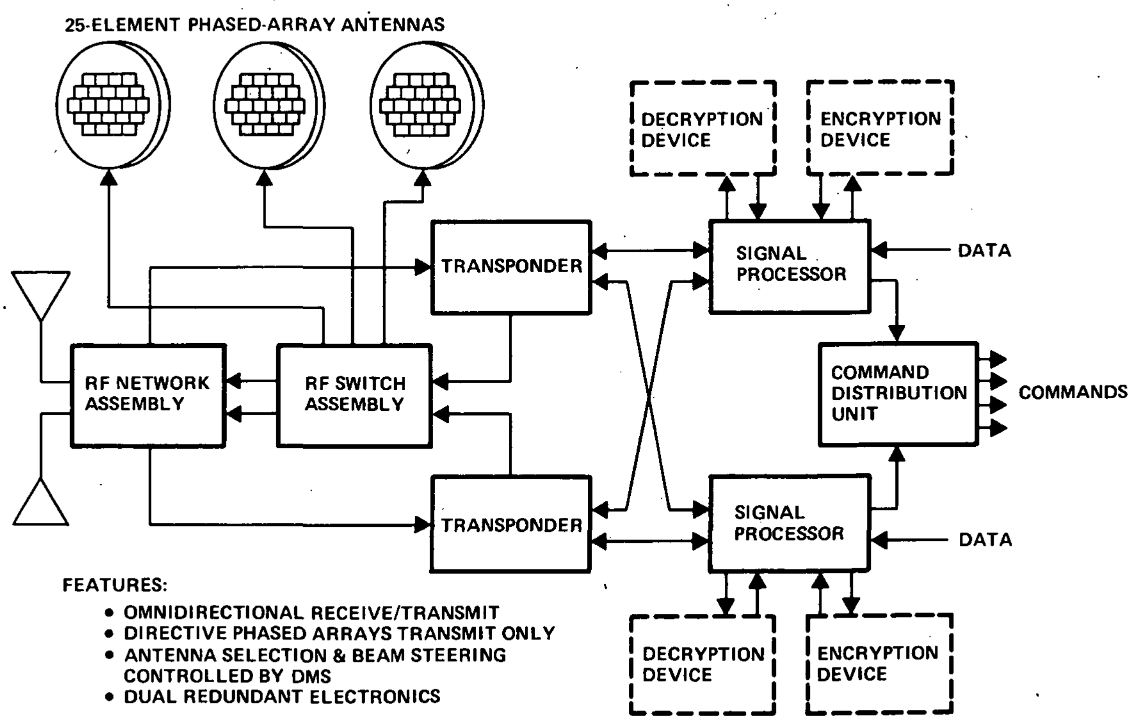


Figure 6-1. Communications Subsystem

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encryption/decryption communication security devices that are installed and active for DOD missions only. Table 6-1 summarizes the Communications subsystem power requirements.

The baseline Communications subsystem features electronically steerable phased array antennas with phase shift logic controlled by the onboard DMS computer. Each antenna array forms a directive transmit lobe that can be steered over a full plus or minus 60 degree (1.1 radian) scan angle. Three arrays are located near the vehicle forward end at 120 degree (2.1 radian) intervals around the vehicle circumference providing adequate communication coverage without restricting Tug maneuvers required for spacecraft thermal control and orbital placement.

An RF Switch Assembly routes low power rf to a selected antenna array. In addition to directive transmit antennas, an omnidirectional receive and low power transmit antenna system is formed by two hemispherical coverage antennas, a ring coupler, and two diplexers. Receive signals are routed directly to the transponder so that antenna selection is not necessary for Tug command access. Transponder transmit rf is routed to either of the omnidirectional antenna diplexers by the RF Switch Assembly.

Either of the redundant transponder units can transmit through any of the directive or omni antennas; however, the transponder receivers are coupled directly to only the

Table 6-1. Communications Subsystem Power Requirements

Unit (LRU)	Unit Power (Watts)	Mode		
		STDN Direct	SGLS Direct	TDRS Network
Network Signal Processor				
Direct	10.0	X	X	
TDRS	18.0			X
Network Transponder				
Direct	14.7	X	X	
TDRS	16.0			X
Phased Array	93	X	X	X
Command Distribution	35	X	X	X
RF Switch Assembly	2.8	X	X	X
RF Network Assembly	—	X	X	X
Decryption Device	2.4		X	
Encryption Device	7.0		X	
Total Power (Watts)		155.5	164.9	164.8

omni-directional antenna system. Both transponder receivers are active at all times with the outputs selected by external switching. Only one transmitter is active at a time. Each transponder provides the capability of receiving and transmitting S-band TT&C signals on STDN, TDRS, and AFSCF networks in either coherent or noncoherent modes, and each has provision for turnaround of the range tracking signal. Umbilical ports at each transponder unit transfer and receive DOD encrypted commands or data originating in the attached spacecraft or Orbiter. Secure command signals from the Orbiter are received at AM/FSK baseband over a redundant coax. An isolation coupler in the transponder extends this command link to the Tug/SC disconnect. Address codes within the data differentiate the commanded vehicle. A similar command link is provided for NASA missions interfacing directly with the command decoder located in the signal processor and these command signals are transferred at digital baseband (BIØ-L). Tug data originates in the PCM formatter section of the CIU.

The redundant signal processors condition and encode data prior to transmitter carrier modulation, decode command signals, and provide bypass switching of the encryption/decryption devices so that information can be transferred in clear channel on DOD missions. The command distribution unit is a collection of dual redundant latching relays for critical control of the safe/abort actuators. These commands can be initiated either by the DMS computer or remotely over the Orbiter umbilical or rf links. A serial digital channel is also provided to the CIU for remote access to data bus peripherals and computer memory.

Dual redundancy characterizes the communications subsystem, since crew safety requires a fail-operational configuration. Safety margins and reliability have been enhanced in several areas by functional cross strapping and multiple elements such as in the phased array antenna system. Selection and management of redundant components is controlled by DMS software evaluation of equipment condition monitors on a periodic basis. Failed units are eliminated by power supply switching or signal path selection.

Expected performance characteristics of the baseline system are shown in Table 6-2. Adequate margin is available for TT&C communication on each of the cooperating networks, STDN, TDRS, and AFSCF for command bit rates of 2 Kbps and telemetry data rate up to 256 Kbps.

Figure 6-2 summarizes the operational aspects of communication subsystem. When the Tug is in the vicinity of the Orbiter, the omnidirectional antennas will be used for transmitting safety information to the Orbiter and receiving commands from the Orbiter. During the Tug mission (transfer orbit to injection and during payload retrieval operations), Tug engineering data (telemetry) and the docking information, such as from the TV, are transmitted to the ground directly or via TDRS using the directive phased array antennas. Tug attitude constraints from spacecraft thermal requirements during transfer and while performing the docking function require all attitude communication depicted by the three phased arrays. Reception of commands etc., are via omnidirectional antennas at all times in the mission.

Table 6-2. Communications Subsystem Performance Summary

Performance Feature	Tug Capability
Network Compatibility	STDN, TDRS, SGLS compatible
Command Bit Rate	NASA 2K bps DOD 2K baud
Telemetry Bit Rate	16K bps, 64K bps, 256K bps selectable
Communications Security (DOD)	Data encryption command decryption
Communications Impact on Tug Attitude	Minor – 3 phased array antennas for directive transmit; omni-directional receive and low power OMNI transmit
Reliability/Safety	Dual redundancy components with functional cross-strapping

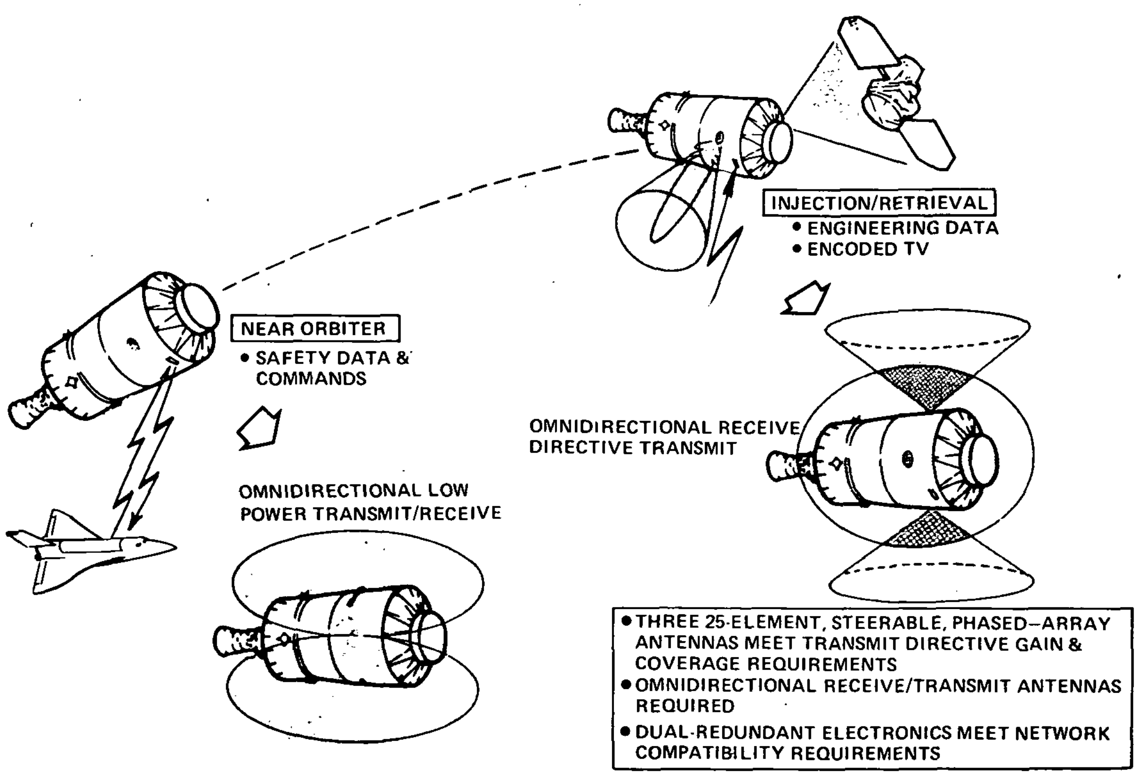

Figure 6-2. Tug Communications Coverage Versus Mission Phase

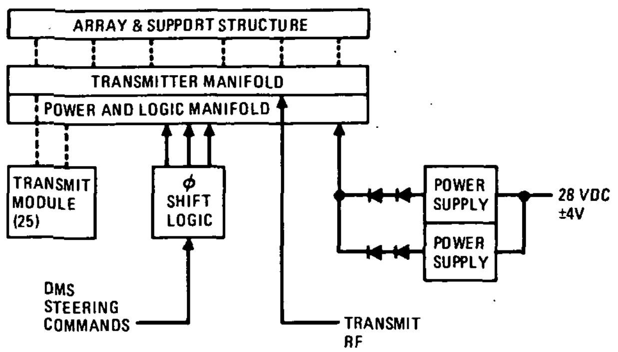

Figure 6-3. Phased Array Antenna Functional Block Diagram

6.2 ANTENNA COMPONENTS

Four major elements make up the antenna component group. These elements are the phased array antennas, omnidirectional antennas, rf network assembly, and rf switch assembly.

A functional block diagram of the phased array antenna unit is shown in Figure 6-3. Twenty-five transmit modules with internal phase shifting networks provide a directive gain with sufficient margin to allow up to seven module failures. Dual power supplies complement the inherent reliability of the multiple element transmit array to form an extremely reliable antenna/transmitter unit. Antenna steering and selection is controlled by commands generated within the DMS computer.

Initial acquisition of the directive antenna system is directed by ground station control. Ground antenna coordinate data is uplinked to the Tug where it is received on the omnidirectional antenna system. The DMS processes this data, selects the appropriate antenna array, and steers the transmit beam of that array to the required angle. An S-band carrier signal is transmitted until the ground station acquires and uplinks a command to begin transmitting data. Steering commands are periodically updated by the DMS to maintain satisfactory link-gain margin.

An omnidirectional receive and transmit system is coupled to the transponder units by the RF Network Assembly made up of two diplexers and a ring coupler as shown in Figure 6-4. This arrangement allows either transponder to receive and transmit on the omnidirectional antenna system. The ring coupler is a totally passive element and therefore has not been duplicated due to an inherent high reliability similar to that of the antenna elements.

An RF Switch Assembly consisting of five coaxial switches interconnected as shown in Figure 6-5 allows either transponder to transmit on any of the phased array antennas or the omnidirectional antenna system. No single point failure can prevent transmission on the omnidirectional antennas, and at least two of the phased array antenna are already accessible.

6.3 TRANSPONDER UNIT

The transponder unit consists of five major sections as shown in Figure 6-6: receiver, baseband modulator, data demodulator, and baseband coupler. The unit will transpond or receive in any of three modes determined by external mode selection. In any mode, the transponder can operate on either of two sets of frequencies, one at each end of

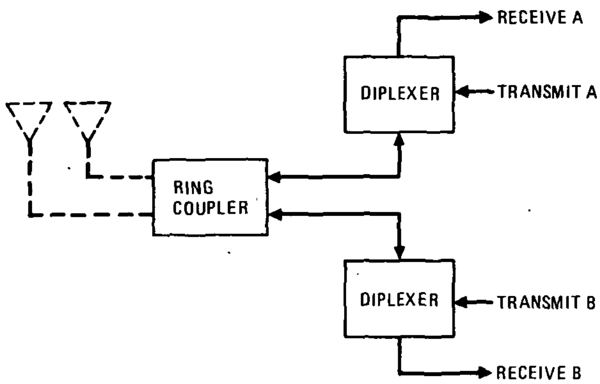

Figure 6-4. RF Network Assembly Functional Block Diagram

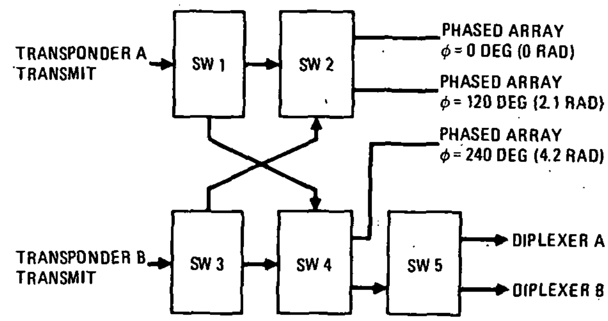

Figure 6-5. RF Switch Assembly Functional Block Diagram

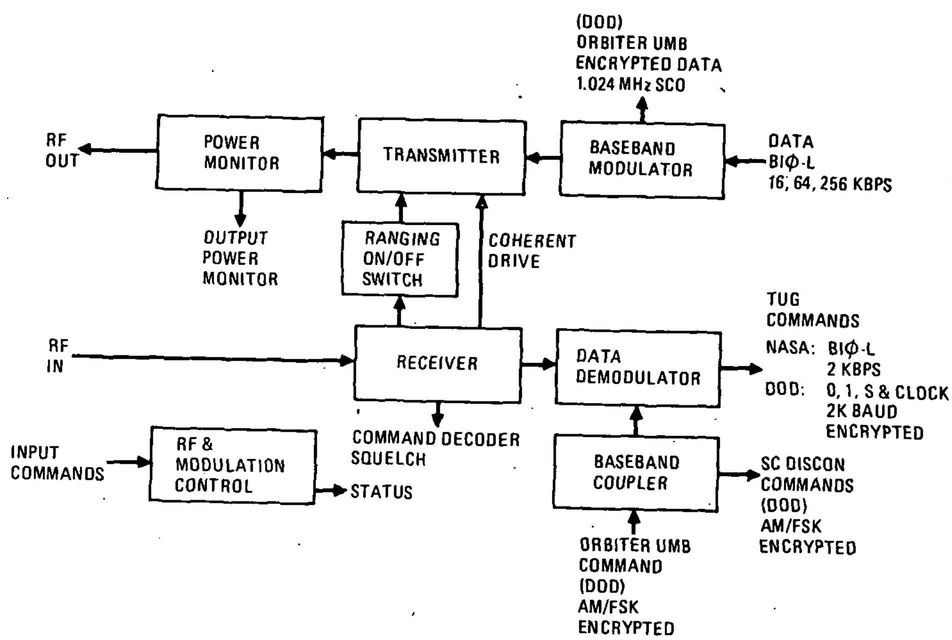

Figure 6-6. Transponder Functional Block Diagram

the standard STDN or SGLS band. The receiver can accept and detect S-band phase shift keyed (PSK), Bi-phase-L (BIØ-L) modulated or composite phase modulated (PM), BIØ-L and tone ranging data at either of two frequencies from STDN or AFSCF ground stations or pseudo-noise spread spectrum PSK, BIØ-L at either of two frequencies from TDRS satellites. Synchronous detection is provided for the TDRS spread spectrum signal and PSK or PM detection for the STDN and AFSCF signals modulated at a 2 Kbps data rate. A command decoder squelch signal is developed from receiver AGC to prevent decoding data with a high probability of error due to low receive signal strength.

The data demodulator accepts detected signals from the receiver and DOD AM/FSK baseband signals from the Orbiter command umbilical through a baseband coupler that isolates the Tug and spacecraft command systems. The command signals are demodulated to arrive at a four wire 1, 0, S, and clock, 2K baud, DOD command interface and a single wire 2 Kbps, BIØ-L, NASA command format.

BIØ-L data at bit rates of 16 Kbps, 64 Kbps or 256 Kbps are accepted by the baseband modulator. The serial data modulates a 1.024 MHz subcarrier oscillator, which then phase modulates the S-band carrier generated within the transmitter. An isolated interface port is provided for hardwire communication of baseband modulated data to the Orbiter. This hardware link is required for DOD encrypted data signals.

The transmitter accepts the baseband signal from the baseband modulator, coherent carrier drive, and tone ranging signals from the receiver. The transponder has simultaneous receive and transmit capability permitting the development of a coherent transmit carrier at a 240/221 turnaround ratio for STDN and TDRS modes and a 256/205 ratio for SGLS modes. Tone ranging signals are mixed with the baseband signals and phase modulate the carrier in the STDN and SGLS modes. The transmitter provides for PSK (suppressed carrier) modulation of the S-band carrier with the BIØ-L data in the TDRS mode. In the absence of a received signal, the transmitter is capable of providing noncoherent rf transmission of data. Output power of the transmitter is four watts nominal, which is sufficient to drive the electronic modules of the phased array antennas and provide adequate drive to the omnidirectional antenna system.

External control from the DMS will select the appropriate transponder mode (STDN, TDRS, SGLS) and control on/off switching of ranging, modulation, and rf output. Selection of redundant transponders will be accomplished by DMS control of power to each unit. Rf power monitor, receiver AGC, and mode status monitors will be accessible to the DMS for determining transponder health and network state.

6.4 SIGNAL PROCESSING COMPONENTS

There are four elements to the signal processing components: the signal processor unit, data encryption device, command decryption device, and command distribution unit. Functional block diagrams of the signal processor and command distribution units are shown in Figures 6-7 and 6-8.

The signal processor unit accepts NRZ-L data and clock from the PCM formatter section of the computer interface unit (CIU). The mode selector under DMS control routes data to a rate 1/2, constraint length of 7, convolutional encoder for the TDRS mode or to the data encryption device for DOD secure SGLS transmission. No bit encoding is required for the NASA STDN mode. After encoding, the data is converted to BIØ-L serial digital format and conditioned for acceptance by the transponder baseband modulator. Transponder command link data in either ternary (0, 1, S, clock) or BIØ-L is accepted by the signal processor. A DMS controlled mode selector switches these signals to either a Viterbi decoder for TDRS mode or to a command decryption device in the secure SGLS mode. The command decoder accepts BIØ-L or ternary

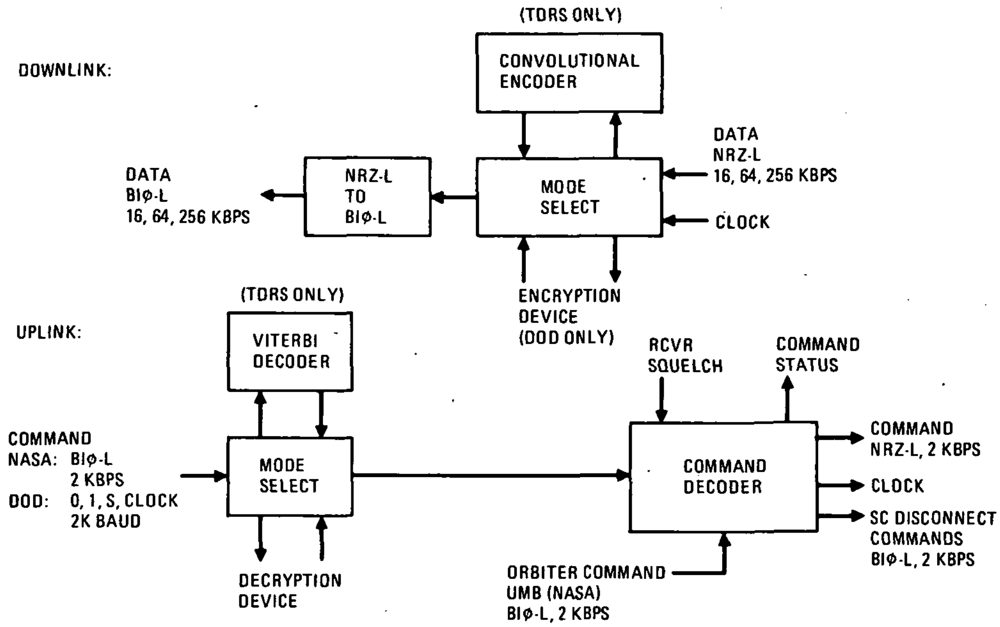

Figure 6-7. Signal Processor Functional Block Diagram

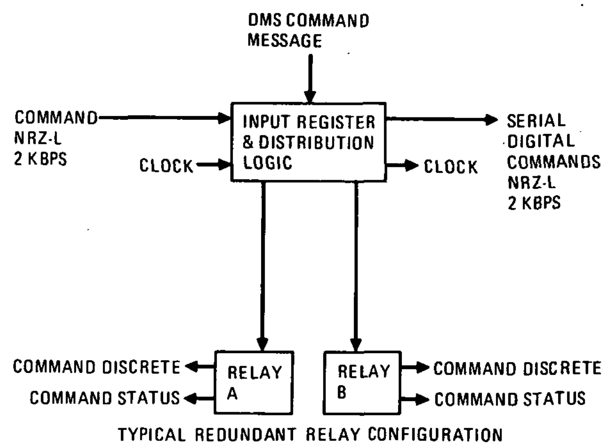

Figure 6-8. Command Distribution Unit Functional Block Diagram

data and decoders address, parity, and data bits from the command message. Data is transmitted in NRZ-L serial format to the command distribution unit.

The command distribution unit accepts the serial NRZ command data from the signal processor, decodes distribution bits, and routes data either to discrete redundant relays controlling safing or abort functions or develops a serial digital command message formatted for input to the CIU of the data management subsystem. Control of the command discrete relays can also be initiated by DMS computer software through a data bus DIU. In the event of command conflicts for safing or abort, the rf command link has override authority. Status discretes indicating the state of all relays are monitored by the DMS software.

SECTION 7

ELECTRICAL POWER SUBSYSTEM DESCRIPTION

7.1 BASELINE CONFIGURATION

Equipment	No. Req	Dimensions, in. (cm) Length	Width	Height	Unit Op Power (Watts)	Unit Weight lb (kg)	Subsystem Weight lb (kg)
Electrical Power							120 (54.5)
Fuel Cell Power Plant	2	12 (30.5)	6 (15.2)	15 (38.1)	20	42 (19.1)	
Emergency Battery	1	8 (20.3)	11 (27.9)	7 (17.8)	8	36 (16.4)	

The electrical power subsystem (EPS) provides +28 Vdc power in support of Tug and spacecraft functions. The primary power is generated by two fuel cells, which utilize hydrogen and oxygen reactants. The fuel-cell power plant is an evolution of the lightweight cell technology being developed by Power Systems Division of United Technologies Corporation. The technology development includes cell operation at low reactant pressures, allowing acquisition of the hydrogen and oxygen from the main engine propellant tanks.

The other elements of the power subsystem are waste heat rejection components and product water storage components. These components are an important part of the power subsystem but are not included in the avionics system.

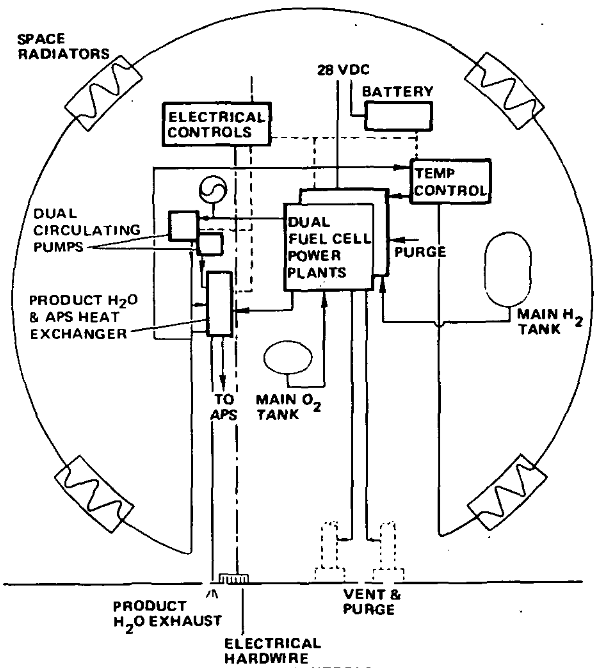

Figure 7-1. Electrical Power Subsystem

The baseline configuration, shown in Figure 7-1, consists of two independent power plants, which have separate reactant feedlines from the main engine propellant tanks. The electrical output is shown at the top of the power plants. An emergency battery is included in the subsystem as a double-failure backup power source to those functions and instrumentation signals required by the Orbiter to assure the safety status of the Tug.

A unique feature of the EPS is the product water and auxiliary propulsion system's hydrazine heat exchanger in which waste heat from the fuel cells is used to maintain the circulating hydrazine fluid within desired temperature limits. The resulting water condensate is stored in the heat exchanger for venting during periods when the Tug is under acceleration.

Waste heat is also rejected by means of a space radiator assembly consisting of four separate radiator panels located around the outer shell structure. Each power plant has its own hydrogen backup cooling system, which is an integral part of the heat exchanger, that becomes active only if the space radiator cooling system fails.

The detailed weights for the total EPS are listed in Table 7-1, and the power requirements are listed in Table 7-2 by mission phase.

Table 7-1. Baseline Tug Power System Flight Weights

Item			Weight lb	Weight (kg)
Dual Thermally Integrated Power Plants			84.8	(38.2)
Reactant Supply Lines & Isolation Valves			8	(3.6)
Waste Heat Rejection			58.5	(26.3)
Space Radiators (3.5 kW) & Lines	[42.5	(19.2)]		
Line Dryer	[2	(0.9)]		
APS Circu. Pumps, +4 lb (+1.8 kg)				
APS Heaters Removed, -22 lb (-10 kg)				
FC-40 Cooling Fluid	[14	(6.3)]		
Product Water/APS Heat Exchanger/ Dump Valves			18.5	(8.3)
Purge & Vent Lines			3	(1.4)
Electrical Controls External of Power Plants			3	(1.4)
Primary Power Supply			175.8	(79.1)
Emergency Battery (+20 Minute Reserve)			36	(16.2)
		Total	211.8	(95.3)

Table 7-2. Typical Tug Power Requirements Per Flight Phase (Watts)

	ASCENT	PREDEPLOY C/O	DEPLOY TUG	ON-ORBIT PL C/O	COAST	ENG BURN	GUID UPDATE	R&D	RETRIEVAL NORMAL	RETRIEVAL EMERG	DESCENT	RTLS ABORT
AVIONICS												
DATA MGT	99	114	114	114	114	134	134	134	114	114	99	114
GN&C	–	382	382	382	382	382	382	382	382	340	–	–
R&D	–	50	–	–	–	–	–	50	–	–	–	–
COMMUNICATIONS	10	10	72	72	95	165	165	165	72	72	10	10
INSTRUMENTATION	66	66	66	66	66	66	66	66	66	66	66	66
POWER SYS	115	140	130	130	130	140	130	140	130	49	115	115
AVG HEATERS	30	230	37	14	21	17	9	41	14	7	4	30
AVIONICS TOTAL	320	992	801	778	808	904	886	978	778	648	294	335
OTHER TUG REQUIREMENTS												
MAIN ENG CIR PUMPS	–	40	40	40	40	–	40	40	40	5	40	40
CONTROL V's & "O" g VENT	225	256	281	225	201	536	218	261	281	174	364	728
APS MOTOR HEATERS	–	–	–	–	64	50	30	50	50	20	–	–
OTHER SYS TOTALS	(225)	(296)	(321)	(265)	(305)	(586)	(288)	(351)	(371)	(199)	(404)	(768)
TOTAL TUG REQUIREMENTS	545	1,288	1,122	1,043	1,113	1,490	1,174	1,329	1,149	847	698	1,103
SINGLE PL REQUIREMENTS	600	650	700	700	200	200	200	–	–	–	40	–
TUG POWER REQUIREMENT	1,145	1,938	1,822	1,743	1,313	1,690	1,374	1,329	1,149	847	738	1,103

Both fuel-cell power plants are in operation simultaneously, sharing the electrical load. Each power plant is capable of supplying the full rated output (2 kW) in the event one power plant should become inoperative

The emergency battery is sized to supply full Tug power during the last 3000 feet (914 meters) of Tug retrieval by the Orbiter. This is currently time lined to take a nominal 0.28 hour, with an added 50% contingency.

7.2 THERMALLY INTEGRATED POWER PLANTS

Dual-fuel-cell power plants are an evolution of the lightweight/passive water removal technology development by the Power Systems Division of United Technologies Corp. Fuel-cell stack construction consists of 68 cells with two cells per plaque. Each cell has an area of 0.114 ft^2 (106 cm^2), for a total stack area of 7.75 ft^2 (7200 cm^2). Each power plant also contains the necessary sensing instrumentation and microprocessor for power plant control and implementation of the power systems redundancy management autonomous reconfiguration capability (Figure 7-2).

The power plant accepts propellant grade reactants directly from the main propellant tanks. These reactants may be supplied in liquid, gas, or mixed phase condition, at

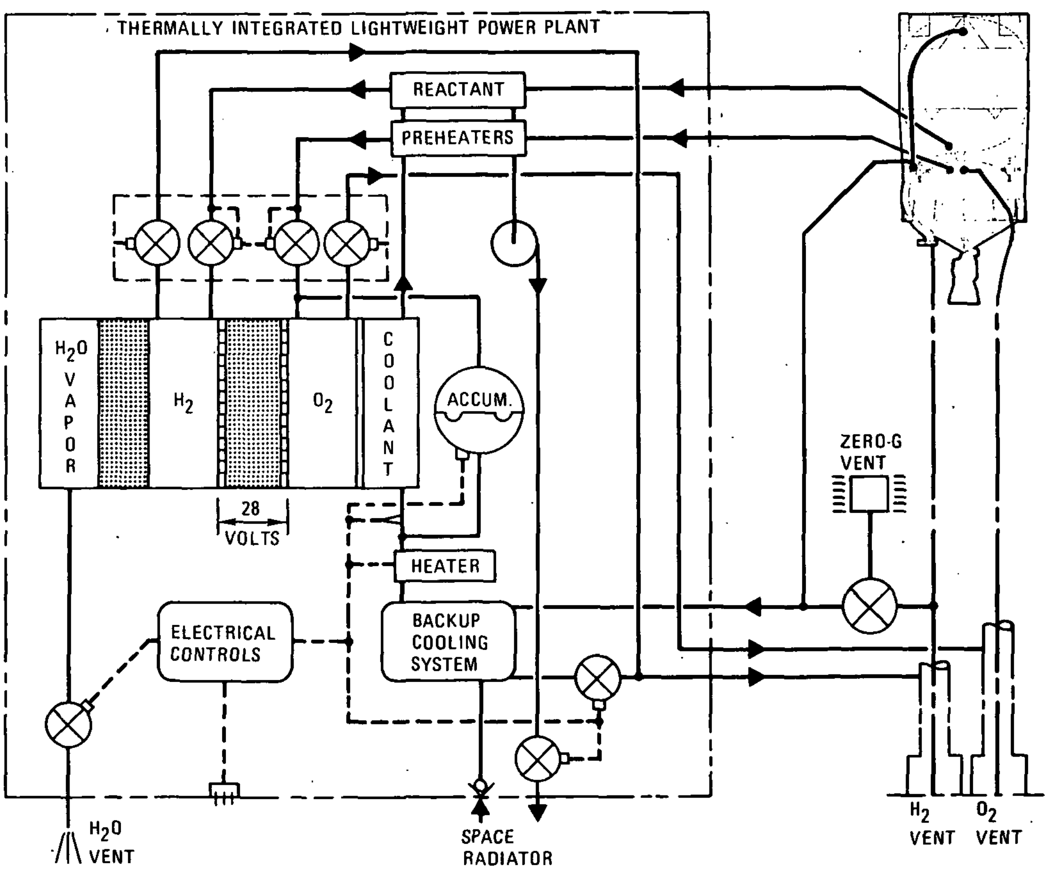

Figure 7-2. Thermally Integrated Power System

liquifying temperatures up to 120°F (322°K). The power plant conditions the reactant temperature and pressure for fuel cell stack use. As the reactants may also include inerts that mask the catalyst and cause a flight voltage drop, the power plant controls will autonomously initiate venting. The venting will flush and dilute the inerts in the reactant chambers until reduced voltage from masking has been returned to an acceptable limit.

Product water is exhausted from the power plant at 4 psia (27.6 kN/m^2). Since the power plant normal operating temperature is above 160°F (344°K) and 4 psia (27.6 kN/m^2) and water vapor becomes steam above 153°F (340°K), product water is exhausted as steam. The water vapor concentration within the power plant is monitored, and product water venting is autonomously controlled by the power plant's controls. Power plant characteristics and insensitivity to operating variables are shown in Figure 7-3.

7.3 PERIPHERAL COMPONENTS

7.3.1 <u>WASTE HEAT REJECTION</u>. Redundant techniques are employed for power system waste heat rejection. Each power plant has a dual dedicated heat rejection capability; that is, a primary space radiator assembly and a backup H_2 heat exchanger within the power plant (Figure 7-4).

The space radiator assembly consists of four separate radiators in series. They are located at 90 degree (1.6 radian) increments around the intertank outer structure. Each radiator has a single radiating surface to service the two separate, but parallel, power plant coolant systems. The power plant FC-40 coolant medium is used, with circulating pumps, filters, and temperature controls all within the individual power plants. Leakage of coolant is sensed by the power plant coolant accumulator low volume position switch. Actuation of the low volume indicator shuts down that power plant's space radiator coolant system and switches to the backup H_2 heat exchanger within the power plant. That power plant is switched out of the power sharing mode and placed on standby. Both the space radiator assembly and the H_2 heat exchanger are sized for the maximum power plant power output heat dissipation requirement.

7.3.2 <u>PRODUCT WATER STORAGE AND APS HEATING</u>. A product water condenser and storage accumulator provide the capability to retain product water during the longest payload delivery coast phase. The circulating APS fluid is used to condense the product water (steam) into water at APS fluid temperatures of approximately 90°F (306°K). Heat absorbed by condensing product water is utilized to make up for heat losses in the APS fluids.

The unit contains a fuel cell coolant loop. Waste heat from the fuel cell raises the circulating APS hydrazine fluid within operating temperature limits when the APS heat losses exceed the condensing heat gain. Fluid temperature control is accomplished by bimetal "bypass" type thermostats. APS fluid additionally provides the thermal heat sink necessary to absorb power plant waste heat during ascent and abort.

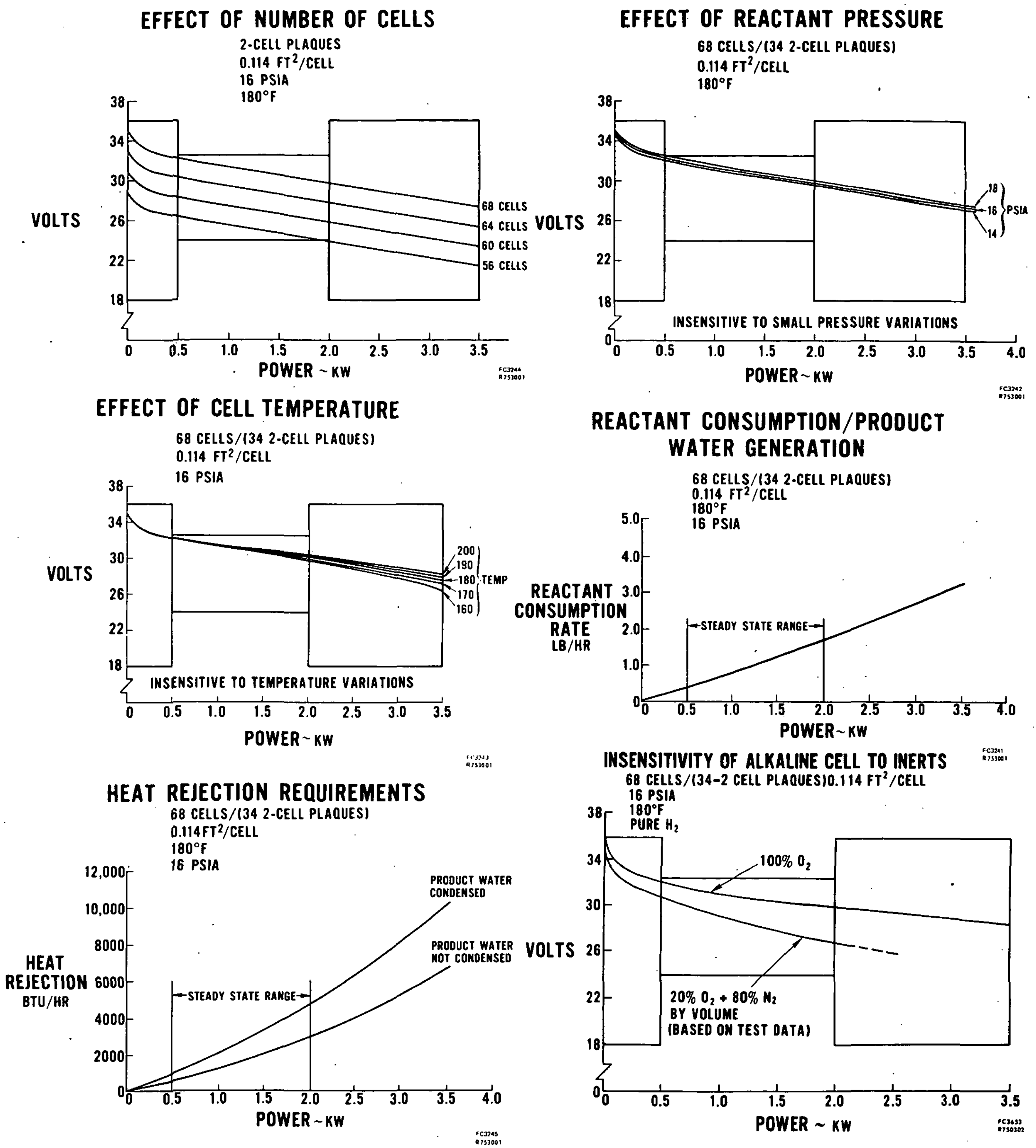

Figure 7-3. Power Plant Characteristics and Insensitivy to Operating Variables

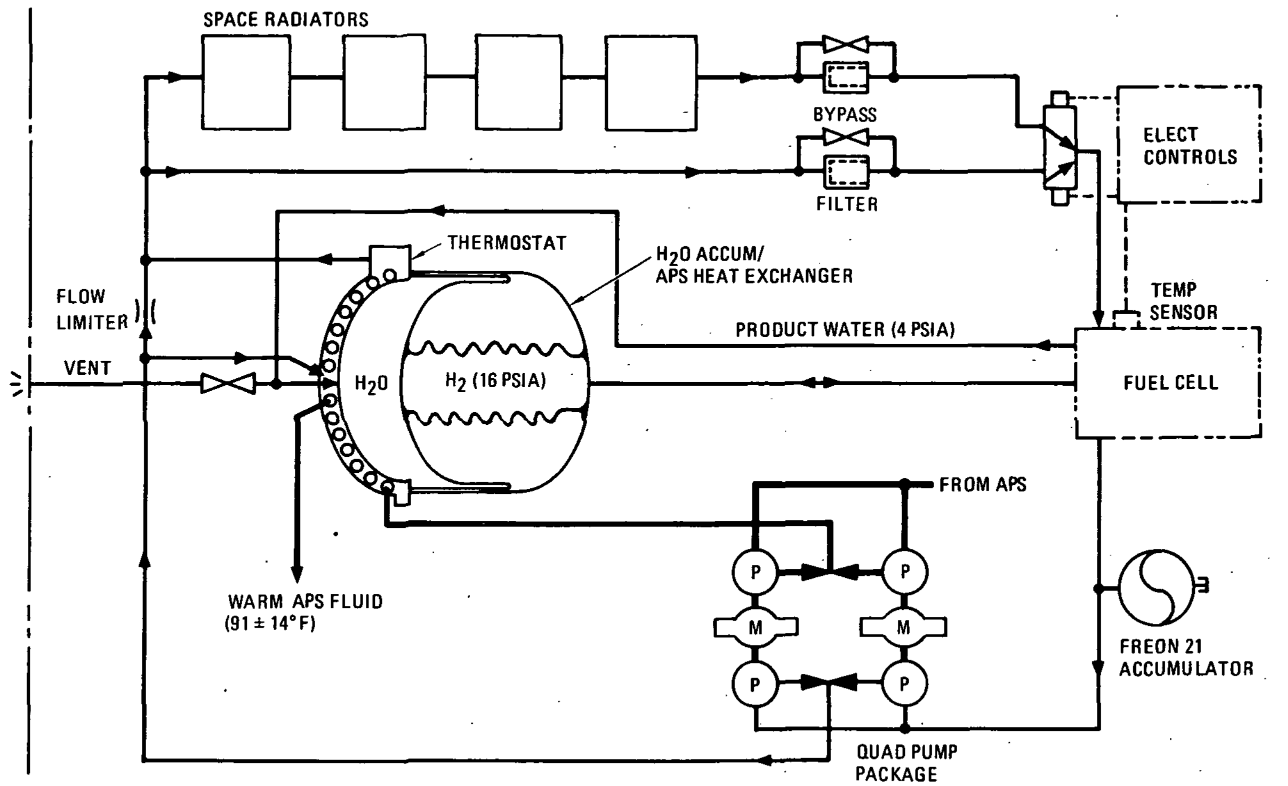

Figure 7-4. Waste Heat Rejection Components

7.3.3 REACTANT STORAGE AND SUPPLY. Hydrogen and oxygen reactants are drawn from the main propellant tanks through shared lines to each power plant. Each feedline has an isolation valve at the propellant tank outlet. In addition, each insulated feedline is sized to hold a liquid volume of reactants to sustain power plant operation during Tug retrieval or abort. The volume is sufficient to enable power plant operation during the time frame from the last main engine burn to retrieval by the Orbiter.

7.3.4 ELECTRICAL CONTROL AND INSTRUMENTATION. The majority of electrical controls and all the instrumentation are an integral part of the dual power plants. A separate redundancy management microprocessor controls the voting and autonomous reconfiguration decisions. Commands will be returned to the power plants for reconfiguration implementation thereby providing control consistent with prime operational objectives; i.e., safety in or within 3000 feet (914 m) of the Orbiter and uninterrupted power during the mission phase.

7.3.5 PURGE, VENT, AND SAFING. Feedlines to the power plants are encapsulated and vented into the main propellant tanks leakage containment membranes. The helium purge supply is connected into, pressurized, and controlled by the main tank purge lines.

Primary safety solutions require Orbiter safety while the power plants are producing electrical power during ascent and abort. After the Orbiter has landed and has been transferred to ground power, the power plant reactants are vented. Venting is accomplished by the power plant's vent valves, exhausting unused reactants into the main propellant tank vent lines. The isolation valves are opened, thereby purging the power plants and supply lines with the main propellant tanks purge helium. The purge continues to exhaust through the main propellant tank's vent lines.

Closing the power plants vent valves maintains the main tank inert gas ground pressurization within the power subsystem's reactant cavities.

7.3.6 EMERGENCY BATTERY. An emergency battery is provided for a separate backup, short-term, Tug power supply during the last 3000 feet (914 m) of Tug retrieval. The emergency battery is used only in the event of primary power multiple failures and safe shutdown. In this emergency retrieval mode, all nonessential Tug/payload power will be powered down or switched off.

The current success timeline for the last 3000 feet (914 m) of Orbiter approach, retrieval and reconnect is 0.28 hour. An additional Orbiter crew decision time will be provided. Current battery sizing, based on the average maximum current of 34.1 amperes, will supply power for 0.67 hour. This provides an additional Orbiter crew safety decision time of 23 minutes.

SECTION 8

POWER DISTRIBUTION AND CONTROL SUBSYSTEM DESCRIPTION

8.1 BASELINE CONFIGURATION

Equipment	No. Req	Dimensions, in. (cm) Length	Width	Height	(Watts)	Unit Weight lb (kg)	Subsystem Weight lb (kg)
Power Distribution and Control					82 (avg)		202 (91.8)
Forward Power Distribution Unit	1	10 (25.4)	6 (15.2)	8 (20.3)		10 (4.5)	
Aft Power Distribution Unit	1	12 (30.5)	15 (38.1)	8 (20.3)		24 (10.9)	
Power Processing Unit	2	9 (22.9)	9 (22.9)	8 (20.3)		8 (3.6)	
Remote Power Controller	59	0.5 (1.3)	1 (2.5)	1 (2.5)		0.2 (0.01)	
Harnesses/Connectors						130 (59.0) Total	
Arm/Safe Switches	2					5 (2.3)	

The power distribution and control subsystem provides the switching control and distribution equipment by which dc power from the fuel cells (and the emergency battery) is applied to the equipment loads.

Power is distributed to the individual components from either the aft or forward power distribution units. Power control and overcurrent/overvoltage protection are accomplished by a remote power controller (RPC) at each load. The Data Management Subsystem (DMS), via data bus through a Digital Interface Unit (DIU) to the RPC at that load, controls power at the load (Figure 8-1).

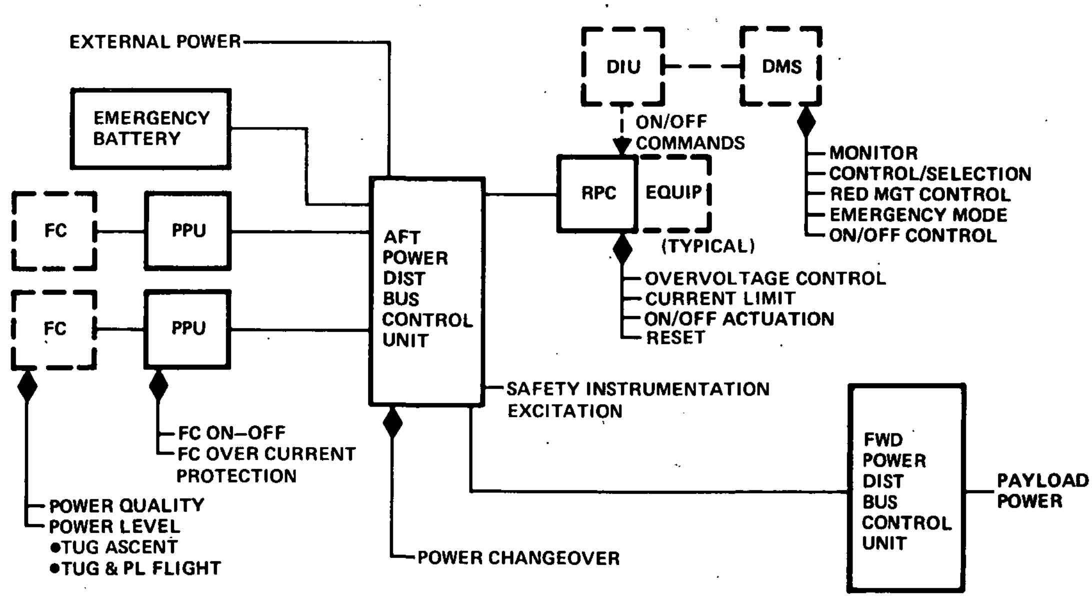

Figure 8-1. Electrical Power Distribution and Control Functional Division and Control Hierarchy

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8.2 REMOTE POWER CONTROLLER

Solid-state switching will be used throughout Tug's systems to control the application of power to individual equipment. The remote power controller is the polarized, bi-stable, solid-state switching device that has an output circuit that can be actuated by a low-level polarized signal to the input circuit. The output circuit presents a high impedance to the flow of current from the power source in its OFF or tripped condition and low impedance in its ON condition. Each equipment or device requiring ON/OFF power control has a RPC assigned to it that will be co-located with that equipment. The multiplexed ON/OFF commands from the DMS computer are decoded and converted to the low-level signal that control the RPC's input circuit. Status signals from the RPC, indicating its ON/OFF position, are returned to the computer via the DIU and data bus. Figure 8-2 shows the DIU-RPC interface.

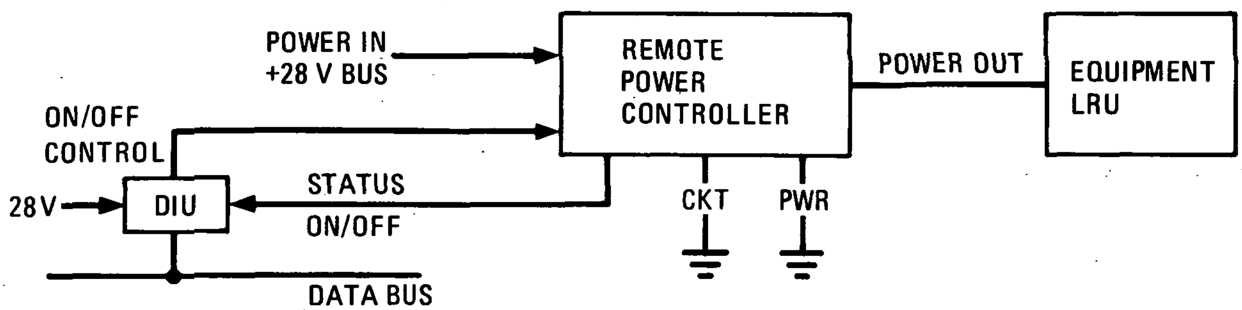

Figure 8-2. Remote Power Controller Interface

8.3 POWER DISTRIBUTION COMPONENTS

Power distribution consists of the aft power distribution unit, which routes the fuel-cell power to equipment located in the intertank area between the hydrogen and oxygen propulsion tanks, and the power processing units, which protect the power system from an overcurrent condition. The aft power distribution unit also distributes the main power to the forward section of the Tug, including the payload, by way of the forward power distribution unit.

Figure 8-3 shows the aft section power and distribution components. A power processing unit (PPU) is dedicated to each fuel cell and consists of an RPC for ON/OFF control to be activated automatically by circuitry in the PPU should an overcurrent condition arise from a short circuit in the system.

The aft power distribution unit contains a motor-driven power changeover switch for selecting the power source desired. During prelaunch phases ground power sources would be utilized. Prior to launch the changeover switch would be commanded to internal power (Tug fuel cells) by a ground initiated command.

The two fuel cells are paralleled to share the electrical load. The equivalent internal resistance of a fuel cell determines that fuel cell's share of the load. The apparent fuel-cell resistance is affected by:

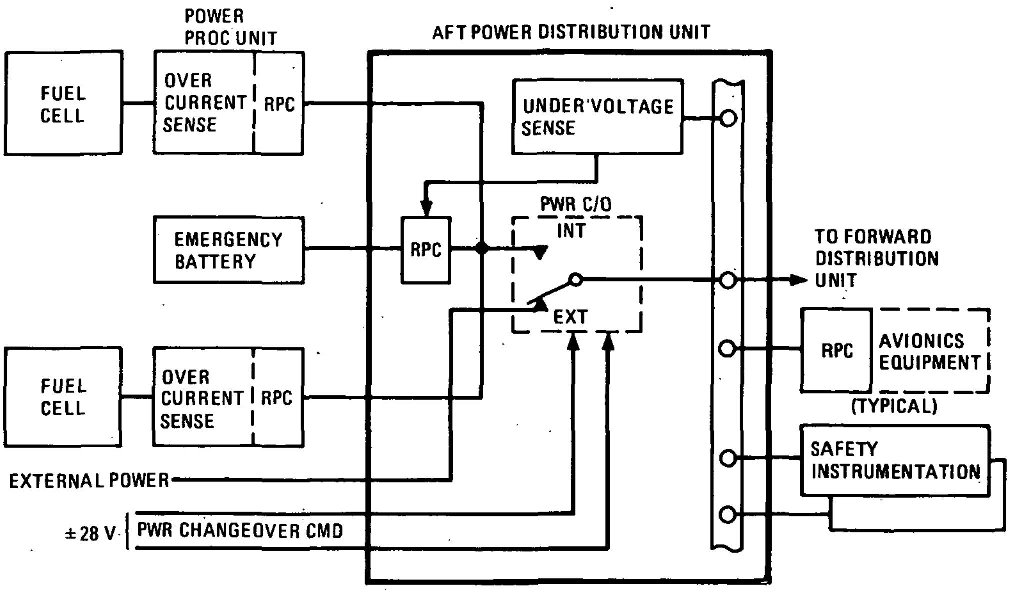

Figure 8-3. Aft Section of the Power Distribution and Control Subsystem

a. Pressure: Increasing pressure results in lower internal resistance.

b. Temperature: Increasing temperature results in lower internal resistance.

c. Cell Area: Increasing cell area results in lower internal resistance.

d. Cells per Stack: Increasing the number of cells per stack results in higher internal resistance.

Factors that influence internal resistance have a direct effect on voltage regulation and therefore on load sharing ability. The fuel cells tend to be stable in a load sharing situation, since the higher temperature unit, having a lower effective resistance tends to run cooler with time, and thereby increase its internal resistance, hence sharing less load. The lower temperature unit, on the other hand, with higher internal losses, tends to run warmer, lowering its internal resistance and hence sharing a greater portion of the common load current.

Load sharing realizes the advantage of eliminating Tug switching transients from a primary fuel cell to the backup. The load can be shifted to only one fuel cell without interruption of the total power. The response time to the change in load demand is on the order of 100 microseconds.

The emergency battery is applied to the load only if the bus voltage were to go below an allowable threshold. An undervoltage sensing circuit within the aft power distribution unit automatically activates the remote power controller connecting the battery to the bus. This assures power to those instrumentation sensors measuring critical parameters that could influence the safety of the Shuttle and crew.

Remote power controllers are attached directly to each individual load to accomplish overvoltage and current control and individual load ON-OFF and reset actuation.

Figure 8-4 shows the forward power distribution unit, which provides that function to all of the equipment in the forward section of the Tug, including power being supplied to the payload.

The forward unit contains an arm/safe switch to prevent an inadvertent payload separation command from activating the separation/docking mechanism by allowing application of power to the control unit only after the switch has been armed.

Redundant equipment will have separate remote power controllers for ON/OFF control. Not all equipment will have RPCs as shown in the figure. A critical power bus applies power to the computer interface units (CIU) whenever there is power applied to the Tug. The CIU receives the main digital uplink from the ground launch processing system. Power on the CIU whenever Tug power is applied allows ground control to activate power-on functions to other units and emergency control over the whole system.

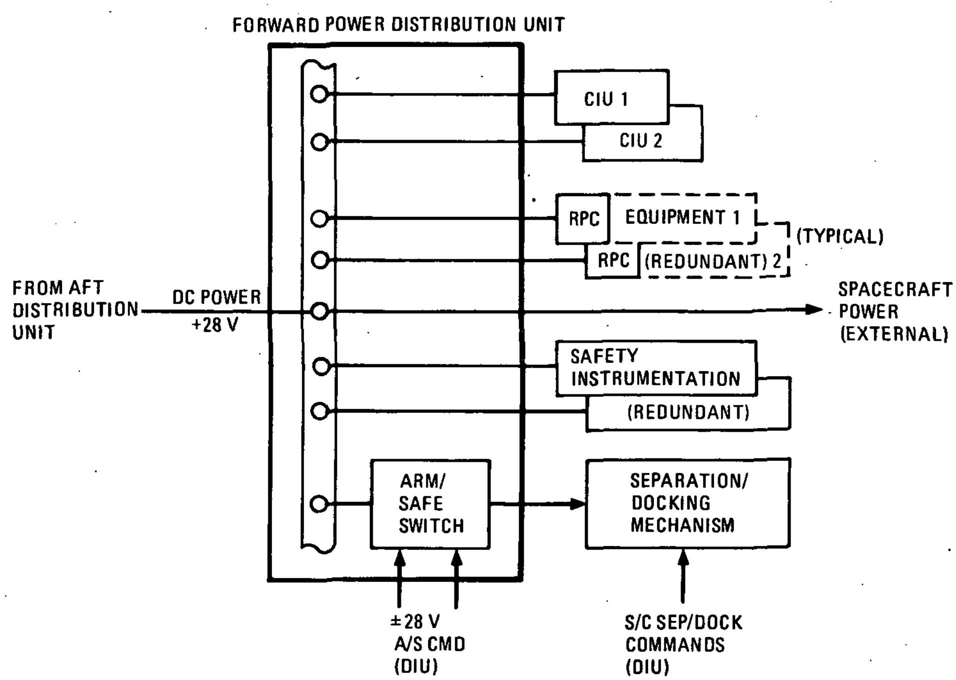

Figure 8-4. Forward Section of the Power Distribution and Control System

SECTION 9

INSTRUMENTATION SUBSYSTEM

9.1 BASELINE CONFIGURATION

Equipment	No. Req	Dimensions, in. (cm) Length	Width	Height	Unit Op Power (Watts)	Unit Weight lb (kg)	Subsystem Weight lb (kg)
Instrumentation							74 (33.6)
Transducers	243					20 (9.1) Total	
Signal Conditioner/MUX	3	12 (30.5)	10 (25.4)	6 (15.2)	22	18 (8.2)	

The Tug instrumentation subsystem functionally interfaces with the vehicle flight, checkout, and communication systems. The subsystem consists of three Signal Conditioner/Multiplex (SC/M) devices and a full complement of both active and passive transducers. One SC/M is mounted forward and two are mounted in the intertank area.

The transducer set is composed of units supporting both flight and maintenance efforts. Inflight instrumentation is used to supply both transducer derived functional data for various subsystems and data for status monitoring. This data is interactive with other vehicle operations during flight.

Maintenance peculiar instrumentation is used strictly for post-mission assessment of component condition.

Instrumentation requirements are established by other subsystem and operational requirements. An example of the detailed Tug measurement list format, reflecting individual component requirements, is provided in Figure 9-1. The requirements are such that, when combined with system functional data, the measurements provide data sufficient to determine the state of a device.

The total number of transducers required to support the measurement list is defined by category in Figure 9-2 along with a representation of the integration of the instrumentation system measurement vector with system functional data.

9.2 SENSORS

Tug instrumentation system sensors (transducers) are classified as being active or passive. Active sensors are electrically interfaced and in general require conditioning, analog-to-digital conversion, and multiplexing into the DMS. Active sensors support both the Tug inflight and maintenance functions.

REVISION=BASIC 01 FEB 1975

TS-000/STRUCTURE TH-000/HYDRAULICS
TP-000/PROPULSION TA-000/AVIONICS
TG-000/PNEUMATICS TI-000/INTERFACE

F=FORWARD A=AFT R=REDUNDANT

ID	MEASUREMENT	CAT	RANGE	UNITS	WORD LENGTH (BITS)	FREQ (HZ)	RATE
	TS-000/STRUCTURE						
	TS-000/PRIMARY SHELLS						
	TS-000/FWD SKIRT						
TS-001T	Q1 TEMP (AESPA Q1)	FI	M20/120	DEGF	8	0.1	0.8
TS-002T-MP	Q1 TEMP/PAINT	FIP	LT120	DEGF			
TS-003T	Q2 TEMP (AESPA Q2)	FI	M20/120	DEGF	8	0.1	0.8
TS-004T-MP	Q2 TEMP/PAINT	FIP	LT120	DEGF			
TS-005T	Q3 TEMP (AESPA Q3)	FI	M20/120	DEGF	8	0.1	0.8
TS-006T-MP	Q3 TEMP/PAINT	FIP	LT120	DEGF			
TS-007T	Q4 TEMP (AESPA Q4)	FI	M20/120	DEGF	8	0.1	0.8
TS-008T-MP	Q4 TEMP/PAINT	FIP	LT120	DEGF			
TS-009S-M	Q1 S/C LOAD PT(S)	FI	TBD	LB/I2	8	1.0	8.0
TS-010S-M	Q2 S/C LOAD PT(S)	FI	TBD	LB/I2	8	1.0	8.0
TS-011S-M	Q3 S/C LOAD PT(S)	FI	TBD	LB/I2	8	1.0	8.0
TS-012S-M	Q4 S/C LOAD PT(S)	FI	TBD	LB/I2	8	1.0	8.0
TS-013X	FWD FLUID UMB COMPL	FO	0/1	TOE	1	0.1	0.1
TS-014X	FWD ELEC UMB COMPL	FO	0/1	TOE	1	0.1	0.1
	TS-100/INTERTANK						
TS-101S-M	LH2 Q1 IB STRUT LD	AI	TBD	LB/I2	8	1.0	8.0
TS-102S-M	LH2 Q1 OB STRUT LD	AI	TBD	LB/I2	4	1.0	4.0
TS-103S-M	LH2 Q2 IB STRUT LD	AI	TBD	LB/I2	8	1.0	8.0
TS-104S-M	LH2 Q2 OB STRUT LD	AI	TBD	LB/I2	4	1.0	4.0
TS-105S-M	LH2 Q3 IB STRUT LD	AI	TBD	LB/I2	8	1.0	8.0
TS-106S-M	LH2 Q3 OB STRUT LD	AI	TBD	LB/I2	4	1.0	4.0
TS-107S-M	LH2 Q4 IB STRUT LD	AI	TBD	LB/I2	8	1.0	8.0
TS-108S-M	LH2 Q4 OB STRUT LD	AI	TBD	LB/I2	4	1.0	4.0
TS-109S-M	LO2 Q1 IB STRUT LD	AI	TBD	LB/I2	8	1.0	8.0
TS-110S-M	LO2 Q1 OB STRUT LD	AI	TBD	LB/I2	4	1.0	4.0
TS-111S-M	LO2 Q2 IB STRUT LD	AI	TBD	LB/I2	8	1.0	8.0
TS-112S-M	LO2 Q2 OB STRUT LD	AI	TBD	LB/I2	4	1.0	4.0
TS-113S-M	LO2 Q3 IB STRUT LD	AI	TBD	LB/I2	8	1.0	8.0
TS-114S-M	LO2 Q3 OB STRUT LD	AI	TBD	LB/I2	4	1.0	4.0
TS-115S-M	LO2 Q4 IB STRUT LD	AI	TBD	LB/I2	8	1.0	8.0
TS-116S-M	LO2 Q4 OB STRUT LD	AI	TBD	LB/I2	4	1.0	4.0
TS-117S-MP	BAY ATTACH PT1 LD	AIP	TBD	LB/I2			
TS-118S-MP	BAY ATTACH PT2 LD	AIP	TBD	LB/I2			
TS-119S-MP	BAY ATTACH PT3 LD	AIP	TBD	LB/I2			
TS-120S-MP	BAY ATTACH PT4 LD	AIP	TBD	LB/I2			

Figure 9-1. Tug Measurement List

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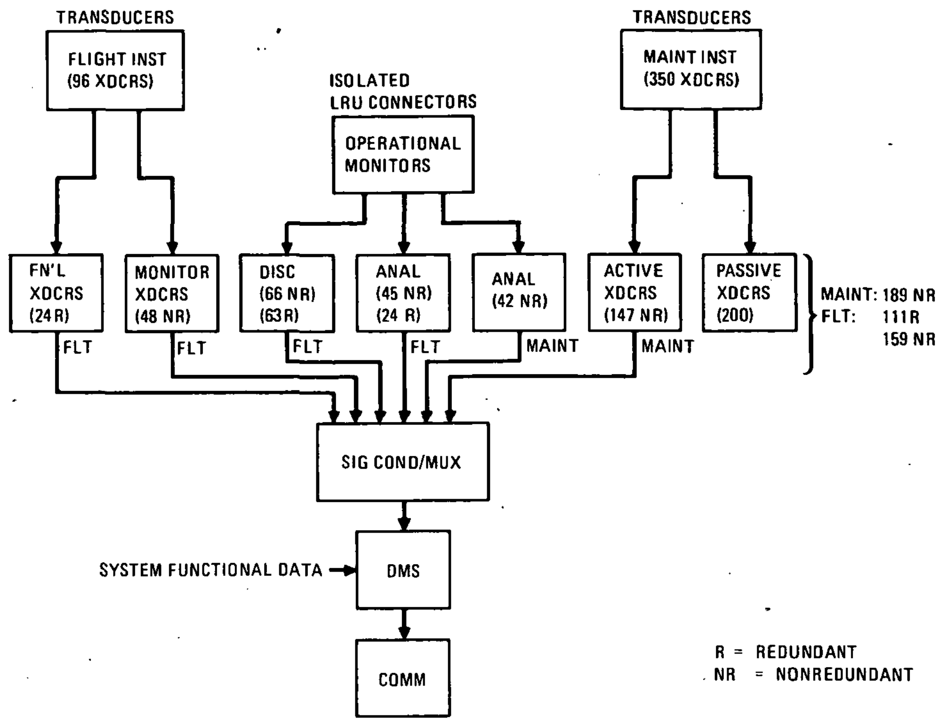

Figure 9-2. Measurement Source Summary

Passive sensor measurements are derived from limit detecting, bi-state, nonreverting devices with no electrical interface. These devices are visually read on the ground after exposure to mission environment and as such are strictly in support of the maintenance task.

A summary of the Tug maintenance instrumentation requirements is given in Figure 9-3. These measurements, which account for approximately one-half of the total active measurements, will not be redundant.

The physical parameters to be represented by the instrumentation system transducers are:

- Current/C (A)
- Voltage/V (A)
- Displacement/D (A, P)
- Pressure/P (A, P)
- Temperature/T (A, P)
- Wear/W (P)
- Discrete/X (A)
- Strain/S (A, P)
- Leak/L (A, P)
- Acceleration/A (A, P)
- Rate, Flow/R (A)

Along with the transducer derived parameters, the instrumentation system provides for accommodating LRU operational data. These signals are generated internally to an LRU and are available to the instrumentation system via special purpose connectors and/or harnessing. In general, these signals are preconditioned/isolated by the target LRU and represent internal voltages, currents, and states.

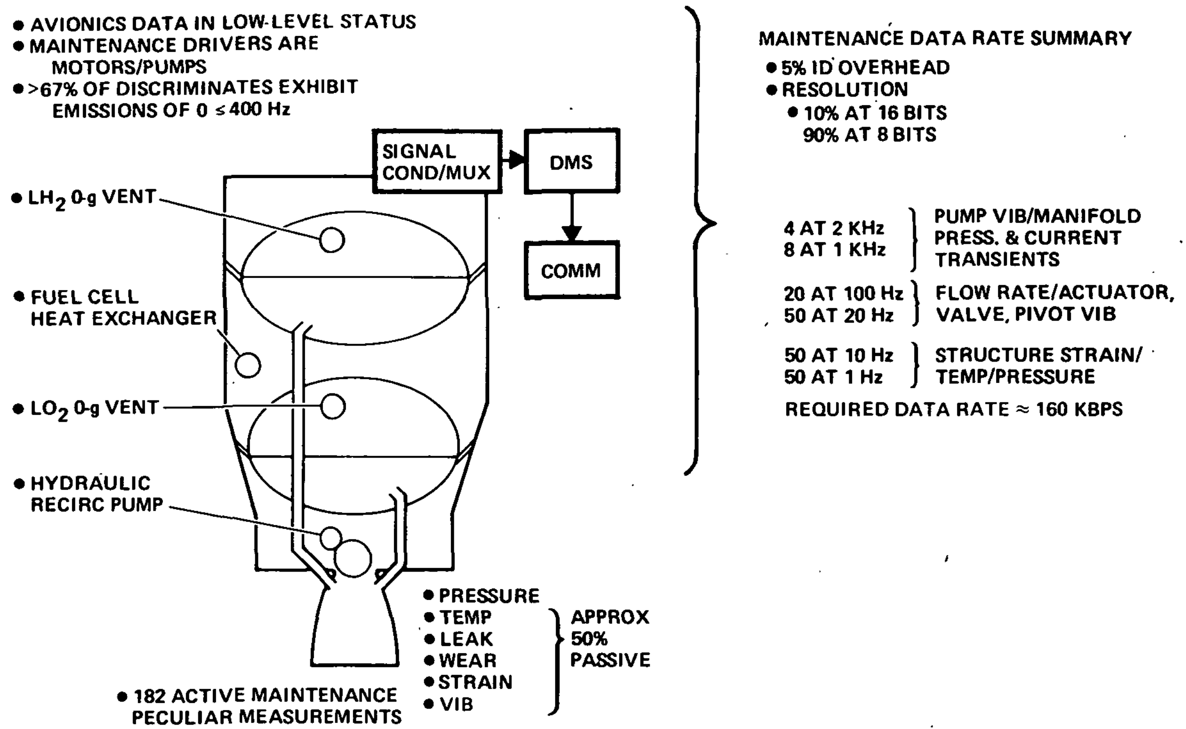

Figure 9-3. Maintenance Required Data

9.3 SIGNAL CONDITIONING/MULTIPLEXING

The second of the two categories of hardware comprising the instrumentation system is the Signal Conditioner/Multiplexer. As indicated in Figure 9-4, this unit is a three-stage device consisting of a passive conditioner, multiplexer, and analog-to-digital (A/D) converter with an integral programmable gain amplifier. Signal conditioning is accomplished in the first and third stage.

The first-stage signal conditioner is passive. As such, all analog signals input to the SC/M must be scaled down consistent with available A/D levels. A minimum of three levels of analog input is accommodated by the A/D converter. Basic conversion will be to 12-bits with 8-bit truncation utilized by 90% of the measurements. In addition, channels of bi-level discrete data will be available for switch monitoring. Discrete data is input in groups of eight. Input capability of the SC/Ms is:

a. Low-level, 0 to 30 mV, 110μV/bit (8 bit), differential 78 channels, plus two wrap around self-check.

b. Mid-range, ±0.5V, 4 mV/bit (8 bit), differential 32 channels.

c. High-level, 0-5V, 20 mV/bit (8 bit), single-ended 16 channels.

d. Bi-level discretes, 0 or 28V, single-ended 64 channels, (eight 8 bit words).

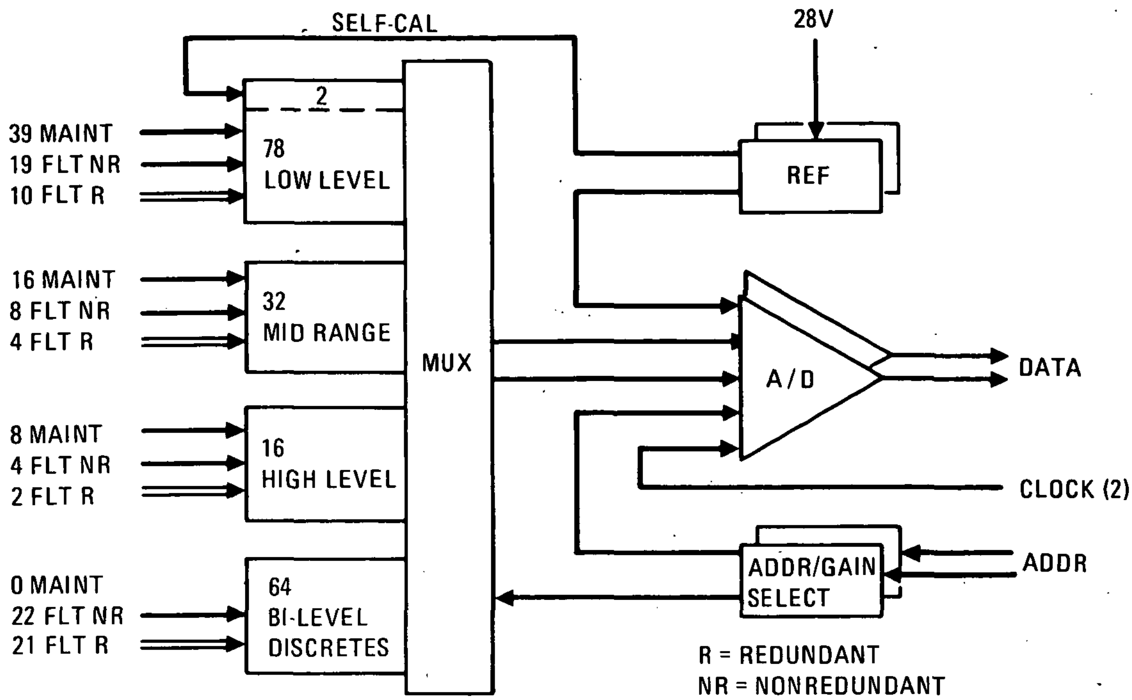

Figure 9-4. SC/M Block Diagram

Each of the two intertank SC/Ms, which primarily process maintenance data, demand maximum sample rates as high as 10,000 per second. Sampling demand on the forward SC/M is less than half of that of the intertank units. Sampling will be asynchronously driven by a data request from the CIU control ROM and DIU to manage the data submultiplexing. The SC/M convert time is approximately one-half of the sample time to acknowledge the requested address, gate in the data, and convert within the allocated sample time. With 12-bit convert times approaching 20 μs the SC/M meets the required convert capability of 25 μs. If necessary, relief on the convert time to almost 50 μs may be gained by mechanizing a one or two word lag across the interface. The A/D converter will be implemented using a programmable gain amplifier and successive approximation algorithm, to accommodate both 8 and 12 bit requirements. Three-level gain selection will be phase concurrent with the multiplexer address select. Data will be clocked out bit by bit throughout the conversion, resulting in a three-wire interface with the DIU of a clock, address, and data.

SECTION 10

TUG CHECKOUT

10.1 DEFINITION AND PHILOSOPHY

Onboard checkout is a critical function of the Space Tug. The results of this special emphasis task (to establish a checkout philosophy for the Tug and to determine the test requirements associated with that philosophy) are presented here because their implementation affects the hardware and software definition of several subsystems.

Checkout is a very broad term. Almost everyone has his own concept of what checkout means. However, most would agree that checkout in general means measuring the response from a unit under test that has stimuli applied to it, and evaluating that response against some criteria to decide if the unit is performing as expected. Responses to be evaluated can also be obtained by monitoring the unit's normal functional output without specific test stimuli. Measuring a temperature or pressure transducer output is an example of monitoring-only response. Gathering this kind of response data, as well as response to stimuli for realtime evaluation or postflight evaluation is the basis for including within the meaning of checkout the following six different kinds of test activities. These six test types provide a complete definition of checkout for the Space Tug. They are:

a. Safety — represents the minimum checkout requirement where a minimum number of critical parameters are monitored to provide safety assurance while the Tug is attached to or in the vicinity of the Orbiter.

b. Status Verification — provides an indication of the operational condition of the subsystem. Accomplished by monitoring checks of vital parameters such as input power, software flags, pressures, and self-test sequences.

c. Initialization — pertains to those input stimuli required to cycle and prepare a subsystem for initiation into its operational mode, such as assuring that proper flight constants are in the computer.

d. Calibration — performed by accurate application of measurement stimuli and determining from responses those compensating parameters that will be used in flight to minimize known error sources. Alignment and/or update of the inertial reference is included at this level.

e. Functional Tests — actively exercise a subsystem (or subsystem component) to verify its ability to perform its intended function.

f. Maintenance Support — system performance is monitored during the mission to detect data trends toward parametric limits. This data, combined with an integrated systems preflight checkout, provides the basis for committing the Tug to its next mission. Maintenance support imposes a quantity of measurements representing the upper bound on checkout instrumentation.

These six kinds of tests, defined herein as Tug checkout, cover all levels of test activities that the Tug systems will experience.

The general elements of a "test" are shown in Figure 10-1. The generalized test consists of 1) a checkout philosophy where requirements for testing, extent of testing, and time for testing are established by the exercise of judgement, 2) unit under test receiving some applied stimuli and producing responses to those stimuli from three data sources: internal operational signals, functional output signals, and special instrumentation, and 3) decision elements involving response data processing, evaluation, and decisioning. Various combinations and degrees of these elements are required by the six tests that were previously defined.

Figure 10-2 shows those elements of the generalized test that are associated with each of the six test types. The shading means that that element is active in that particular test. Examples of the kinds of stimuli and the kinds of processing and evaluations of the responses are shown, as well as the data sources for obtaining the responses. The specific responses are delineated by the test parameters associated with the particular unit under test. The characteristics of those test parameters and the number of test parameters measured for each test type determine the necessary test support requirements.

The magnitude of the test parameters measured in each test level varies with the kind of checkout philosophy. Associated with each philosophy is the attendant software/hardware required to support the stimulus/response elements of each test.

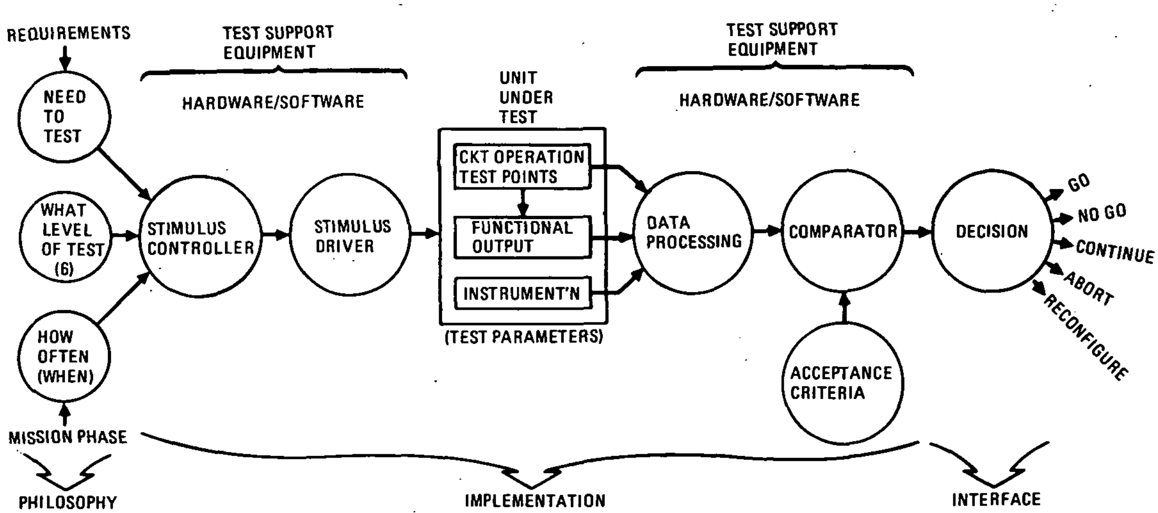

Figure 10-1. Generalized Test Model

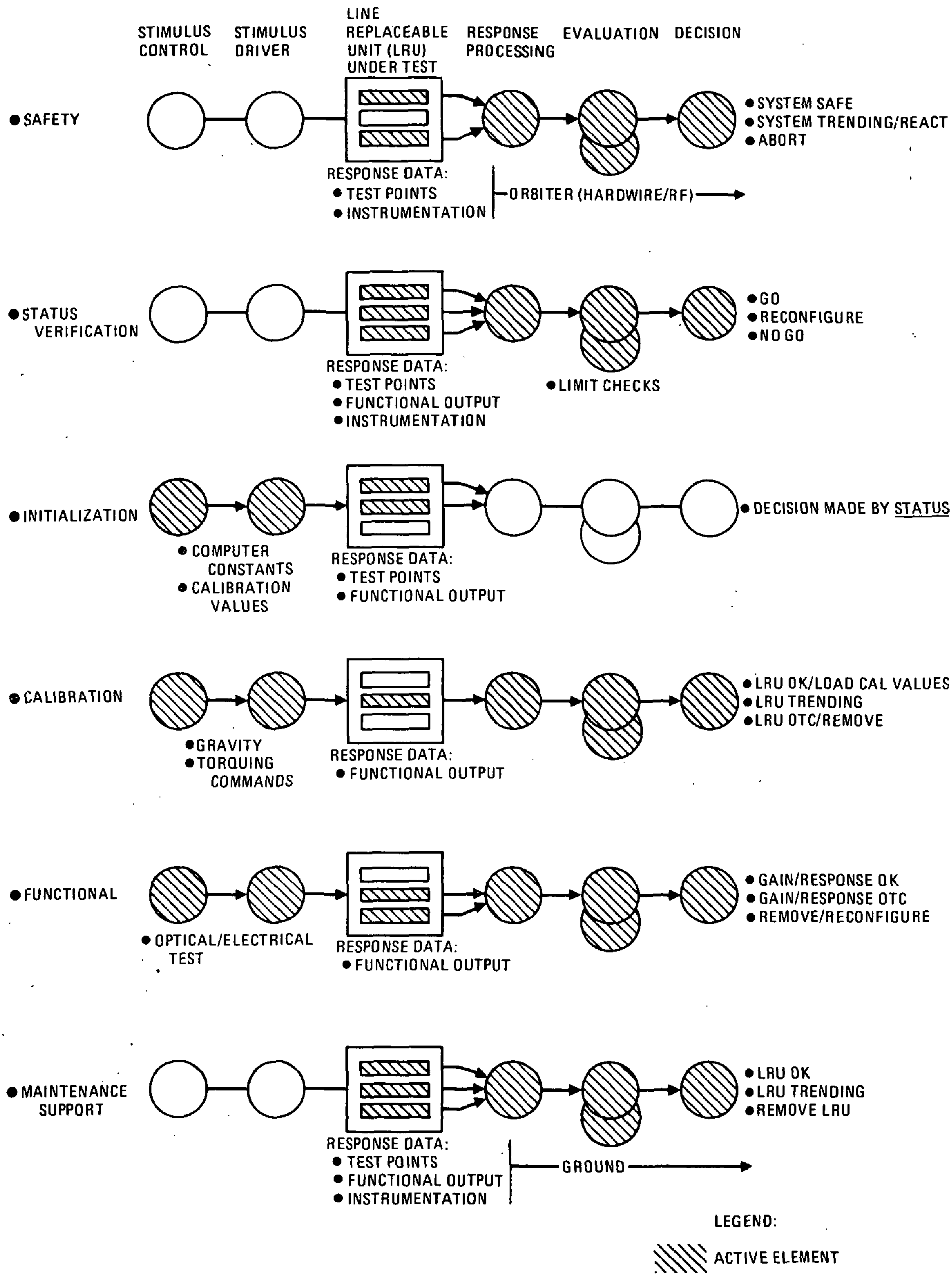

Figure 10-2. Elements of Each Test Type

Checkout philosophies cover the spectrum from no testing to extensive testing, and associated with each is an upper limit on confidence achievable with that philosophy. A set of six philosophies and their characteristics were defined to cover the philosophy spectrum:

a. Hands off (HO) — use to failure.

b. Hard time-remove/replace (HT_{RR}) — remove/replace every (T, event, cycle)

c. Hard time-test (HT_T) — test every (T, event)

d. CMM – no preflight test ($CMM_{\overline{PF}}$) — full trend analysis

e. CMM – preflight test (CMM_{PF}) — limited trend analysis

f. Test and retest (T&RT) — repeated preflight tests

The dominant philosophy chosen for Tug is Condition Monitored Maintenance with preflight testing (CMM_{PF}). This philosophy provides the best opportunity for implementing a low cost checkout system while still maximizing the return on 1978 state-of-the-art technology and minimizing the potential for excessive cost. The CMM_{PF} checkout philosophy employs the judgement that the last flight is the best test of the system. Data is gathered during the Tug flight to this end, and maintenance action is dictated by this inflight data. Completion of the Tug maintenance/refurbishment activities is marked by a complete functional electrical test. Verification is accomplished for components, subsystems, systems, and interfaces. From that time on, status verification and safety are the principal test activities through the remaining ground operations, launch, and return.

10.2 TEST REQUIREMENTS

All of the component level units were evaluated to determine the applicability of each test type to each of the components during each flight and ground operational phase. The test requirements matrix in Table 10-1 summarizes the total number of components undergoing the different tests during the 10 mission phases identified. This matrix represents the CMM_{PF} philosophy, which guided the judgement as to what units get tested, when, and by what test type. The exceptions to this philosophy represent the functional test of the computer and the computer interface units during shuttle ascent, and the optical sensors and rf system on-orbit where the operational condition is best for their functional checkout. The matrix distribution leads to a sensible allocation of where the responsibility for performing the test should be placed based on the following criteria: recurring test demands (status test, maintenance support), phase-peculiar testing (safety monitoring, functional tests, calibration, initialization), and the requirement for high support software storage used in few mission phases (functional test).

Table 10-1. Test Requirements Summary Matrix

Mission Phases	No. of Components Undergoing Test					
	Safety	Status	Calib.	Funct. Test	Initial.	Maint. Support
Prelaunch	2	8	10	25	5	0
Shuttle Ascent	8	22	1	3	14	2
On Orbit	9	30	1	12	12	2
Tug Deploy	9	28	0	1	2	8
Tug Ascent	0	20	0	0	0	6
Payload Deploy	0	24	0	0	3	7
Tug Descent	8	24	0	0	1	7
Orbiter Capture	9	26	0	0	0	0
Shuttle Descent	2	12	0	0	0	0
Gnd Ops	1	4	10	35	20	11

The allocation of the test responsibilities is shown in Table 10-2. Those tests under "Tug Allocation" will be implemented with software residing in the central computer. The prime elements of this software are status verification and inflight maintenance support data acquisition. The other tests will also be implemented with test support software residing in 1) the Launch Processing System at KSC for the majority of functional tests, calibration and the software associated with postflight maintenance data processing, and 2) the Orbiter for evaluation of the safety monitoring data. The test support software memory storage requirements are also shown in Table 10-2.

Table 10-2. Checkout Allocation

Ground Allocation	Orbiter Allocation	Tug Allocation
Functional Test Calibration Maintenance Processing	Safety (monitor) (No decom or display formatting included)	Safety (reaction sequence) Status Initialization Partial Functional Critical control loops Critical Functions and Components Maintenance Acquisition
88K Words Total	1.5K Words Total	8.9K Words Total

10.3 BASELINE CONFIGURATION

Figure 10-3 is an overall view of the Tug onboard checkout system. Checkout has its major impact on the Tug avionics system in the area of computer memory storage for software instruction programs and data. The capacity of the Data Management Subsystem was sized with the checkout tasks considered. The instrumentation subsystem (right-hand side of the figure) depicts the following response data sources:

Line Replaceable Unit (LRU) is the component level in the Avionics System. The LRUs may contain varying degrees of built-in test equipment (BITE), from no BITE where many test parameter response data are provided to evaluate the health of the unit, to total BITE within the unit where one parameter indicates the go/no-go status of the unit. Special LRU instrumentation measurements are conditioned and multiplexed by means of the signal conditioner unit. Additional instrumentation is provided to acquire data relating to unit performance in flight in support of the ground maintenance function. The central computer has the capability of formatting any or all of the acquired data for transmission to the ground via telemetry. The maintenance data can also be stored in the tape recorder for later transmission or post-flight read-out.

The prime test activities of the onboard checkout system are safety monitoring, status verification, and maintenance support data acquisition.

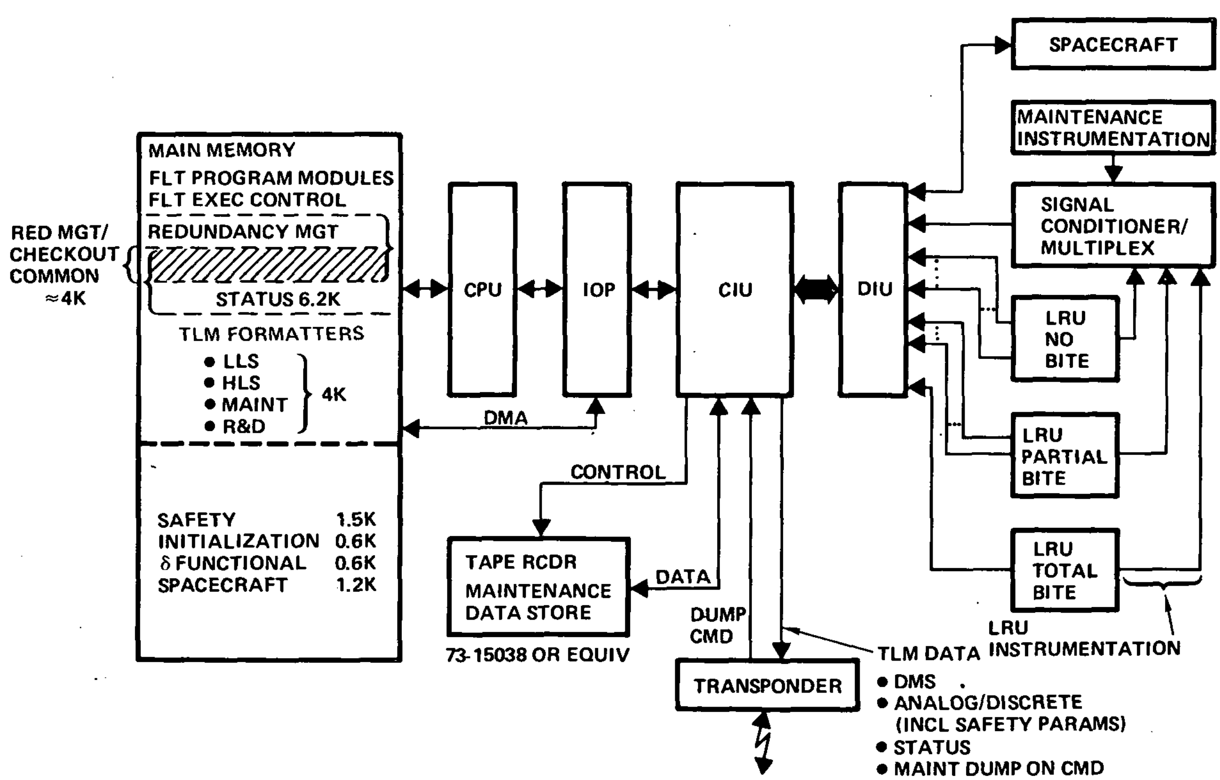

Figure 10-3. Tug Checkout System Block Diagram

10.4 SAFETY

Safety of the Shuttle and its crew is ensured during both Tug attached and detached phases by providing appropriate crew Tug Caution and Warning (C&W) indicators. During operational phases when the Tug is attached to the Orbiter, two operationally redundant communication paths are used to transmit the 10 Tug caution and warning signals to the Orbiter.

For warning signals the primary path is implemented by hardwiring the output of the three warning sensors (parameters) directly to the Orbiter C&W electronics unit. Prime safety restore sequencing is initiated via the dual redundant 2Kbps command uplink through the Tug DMS.

The secondary path for transmitting Tug warning signals to the crew is implemented by multiplexing the data from separate sensors onto the Tug telemetry data stream and transmitting it to the crew via the Orbiter payload signal processor (PSP). The safety restore uplink is again provided via the 2Kbps uplink through the CIU.

During detached operations, both deployment and capture, continuous monitoring of Tug and spacecraft C&W functions are maintained via the Tug/Orbiter rf telemetry link. For deployment, the rf link will be established and verified after Tug rotation but before retraction of the Tug deployment adapter umbilicals.

Redundant paths associated with the Tug to PDI ensure back-up modes of operation during all Tug/Orbiter launch phases including prelaunch activities.

10.5 STATUS VERIFICATION

Upon receiving a full complement of hardware, the Tug progresses into the integration phase. All subsystem/component calibrations requiring peripheral support devices (such as optical beds) have been completed at this time.

10.5.1 PREOPERATIONS FUNCTIONAL TESTING/STATUSING. Completion of the Tug maintenance/refurbishment activities signifies the beginning of a new launch cycle for the Tug. Effort at this point has involved verification at the unit (LRU) level. Now the integrity of the Tug itself is verified. Verification is accomplished by performing unit, subsystem, and system level tests on the Tug. These tests are primarily electrical. Control of Tug level tests is partially under control of functional routines loaded into the DMS by the LPS. Similarly, the Tug will be exercised through a complete electrical test wherein the DMS is fully integrated with the full-up Tug avionics. For this test the DMS will not require loading of the flight (or simulated flight) program. All interfaces are verified using interface simulators. These units allow verification of onboard tests required to assure integrity of both adapter and spacecraft unions. All software controllable calibrations that do not require peripheral support hardware have been completed at this time.

After mating with the deployment adapter, the Tug is given a complete functional integrated test including a flight (or simulated/replica flight) program demonstration. Using only the spacecraft simulator, this will include verification of all adapter/vehicle/separation (retrieval) sequencing, and concludes the terminal integrated test, which is the only and final integrated test planned. Upon completion of this test, an end of cycle status (including calibration) log is generated for the Tug to be input to the LPS active history files. The log will be retrieved when the Tug reenters the operations cycle and is assigned a launch.

The status and safety portions of the flight checkout routines will be loaded into the DMS by procedure any time the Tug is powered up after the terminal integration test. This is the first step after warehousing (if any) and commences progression into the flight operations cycle. The status report generated at this point is compared to that obtained at completion of the terminal integrated test, thereby analytically verifying and validating the Tug flight readiness.

After mating the spacecraft and the loading of hazardous fluids (under peripheral support equipment control), status and safety are the principal DMS resident routines. These are under control of a resident preflight executive.

From the time the Tug is loaded into the Orbiter cargo bay, various preflight functional and control routines are cycling in the DMS. A T-2 day status report is again executed. Upon verification of status, the flight program is loaded and validated under LPS control. Provision is made in the flight verified memory for approximately 1K of non-dedicated storage wherein special-purpose troubleshooting or detailed statusing routines can be uplinked. Propellant loading is also accomplished under LPS control with the loop closed through Tug onboard propellant system instrumentation. No special purpose routines are required on board Tug for propellant loading.

Tug flight/launch readiness is monitored by LPS by interrogating the hardwire Tug 16K PCM link, which includes interleaved status data. Tug 16K, 64K, and 256K bit PCM (rf) is benign at launch. Entry of Tug readiness into the Shuttle launch ladder is under LPS control.

10.5.2 <u>INFLIGHT STATUSING</u>. Continuous status checks of Tug components will be provided as a part of the Tug telemetry. This compacted processed status data provides isolation to the component (LRU) level at a minimum and, in most instances, to a functional element within the device.

Inflight statusing is accomplished by a combination of DMS processed functional/operational/instrumentation data and built-in-test-equipment (BITE). BITE is not implemented across the board at the LRU level but is rather optimized as to LRU criticality, development history, and operations requirements. BITE is executed on either a continuous (internally driven) or DMS sequence/command basis. Continuous BITE components are:

C-2

a. Fuel cells (fast detection/recovery)

b. DMS (full self-check — software/hardware implementation)

Commanded BITE components are:

a. Scanning ladar (full-DMS assist).

b. TV camera/electronics (full-DMS assist).

c. Signal conditioners (limited - A/D (PGA) converter check).

d. Engine control electronics (full-end-to-end checks).

e. IMU (partial - quick look technique).

f. Star tracker (limited - functional).

g. Sun sensor (limited - functional).

h. ILT (limited - functional/gain).

10.6 MAINTENANCE SUPPORT DATA ACQUISITION

Tug maintenance will be performed each time the vehicle returns from a mission. The maintenance/refurbishment process commences with receipt of the vehicle at the Tug Processing Facility (TPF) and proceeds through to successful integration of the vehicle and deployment adapter, which signifies the end of the maintenance effort for that flight cycle.

The Tug onboard maintenance support is indicated in the functional flow diagram of Figure 10-4.

Maintenance is performed based on several data sources. Primary among these is data gathered during the operational phase of the just completed mission.

Data gathered during the operational, or flight, phases is composed of real-time telemetry and special purpose maintenance measurements. Maintenance measurements take two forms — active and passive.

Active measurements are those integral to, or electrically interfacing with, the Tug instrumentation system. These in general require conditioning, digitizing, time tagging, and multiplexing. They are transmitted to the Tug DMS via the data bus to be formatted for transmission to the ground in real time on the 256 Kbps link. If ground coverage is unavailable at that point in the mission profile, the data will be recorded on tape for later playback on command from the ground. A limited amount of maintenance data will also be generated during Orbiter ascent while the Tug is in the cargo bay and limited to the 16 Kbps Orbiter link. The Tug tape store will again be used to record the maintenance data until rf is established with the ground upon deployment. On-orbit Tug checkout/statusing procedures will include a commanded dump of the ascent data. Mission coverage anticipated by the described operations is predicated on the availability of TDRS.

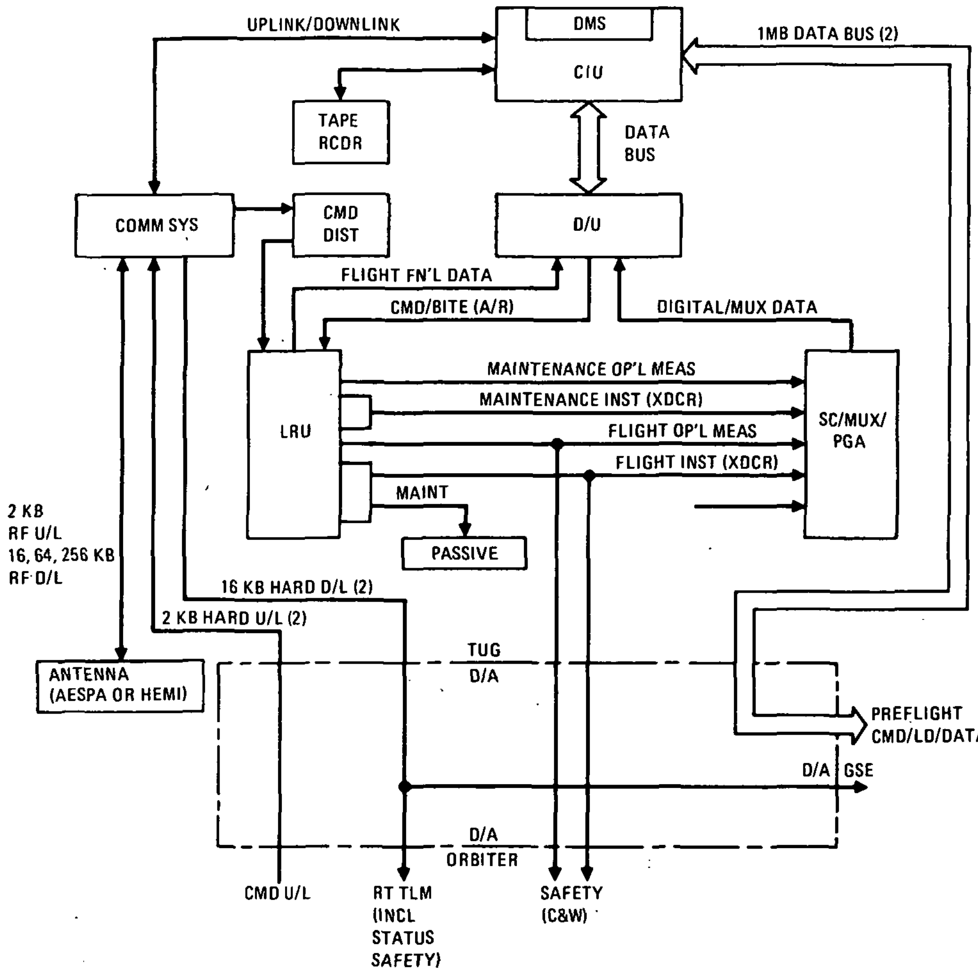

Figure 10-4. Integrated Tug Checkout System Functional Flow

Passive measurements are derived from limit detecting bi-state nonreverting devices with no electrical interfaces. After exposure to environment the devices will be visually read on the ground during the maintenance/refurbishment operations. Any device that has been tripped will be visually prominent and indicate that the unit to which it is applied has exceeded some physical parameter threshold and that the unit is to be removed and replaced. This activity accordingly will be entered into the LPS Tug component history logs.

Active measurements will be recovered either from the real-time or commanded maintenance dump and/or from the tape itself as it is recovered upon return to the landing site. LPS data processing services will reduce the tapes (either ground recordings of the dump or the flight tape) and, along with the Tug component history files, perform a trend/prediction analysis to identify potential failures and flag the components (LRUs) for replacement during the current maintenance cycle.

The entire maintenance effort will be accomplished by 1) removing and replacing those units approaching end-of-life based on time or cycles (the DMS will keep track of cycles and output the updated total as a part of status), 2) removing and replacing those units found to have exceeded some preestablished physical threshold (this activity is driven by inspection of the Tug complement of passive detectors and covers the range from local heating to plumbing/insulation leaks to mechanical wear), or 3) removing and replacing those units flagged by the trend/prediction analysis algorithm as either having failed or approaching failure within the next mission cycle (this activity is driven by active maintenance measurements and the Tug real-time telemetry).

SECTION 11

INTERFACE DESCRIPTION

The three Tug electrical interfaces reflect implementation of spacecraft support (single spacecraft), Orbiter safety, Orbiter Tug support capability, and Tug support requirements of the Orbiter. The interfaces shown in Figure 11-1 are compliant with this impelementation and represent the functionally consistent flow across the interfaces.

The Orbiter electrical interface with Tug includes a NASA 2 Kbps uplink, a DOD 2K baud uplink and a 1 Mbps bi-directional, high-speed bus. The 1 Mbps bus is used primarily on the ground and provides a responsive wideband path accommodating software loading/verification, updating, and safety reaction requirements. Each uplink (2 Kbps/2K baud) is shared between Tug and the spacecraft with each doing its own address recognition/decoding. These links are driven by Tug rf when detached from the Orbiter.

Downlinks are accommodated by a dual 10 Kbps spacecraft link, a 16 Kbps NASA link, and a 16K baud DOD link. One spacecraft downlink is passed straight through the Tug with no electrical interface, along with wideband 256 Kbps spacecraft experiment data and spacecraft caution and warning (S/C C&W) data. A second replica spacecraft downlink is passed through a Tug forward DIU where data may be stripped but by the DMS if necessary to support spacecraft requirements. Either way the full data complement is submultiplexed into the Tug downlink. Provision is made for spacecraft ground activity without having to power up Tug for support.

Prime abort commands for both Tug and deployment adapter are executed by the Tug DMS and require a six-wire command interface to the adapter.

Tug signal flow from top to bottom is presented in Figure 11-2.

11.1 BASELINE TUG/SPACECRAFT CONFIGURATION

The Tug/spacecraft interface is defined at the spacecraft adapter/Tug junction since all points at forward stations are in support of spacecraft functions. The baseline interface is configured to a single spacecraft. Each additional spacecraft will reflect approximately a 90% increase in traffic and hardware across the interface.

11.1.1 UPLINKS

a. 2 Kbps dual redundant (NASA) — interfaces through Tug Communications subsystem from either a hardwire or rf source. Receiver is spacecraft communication subsystem.

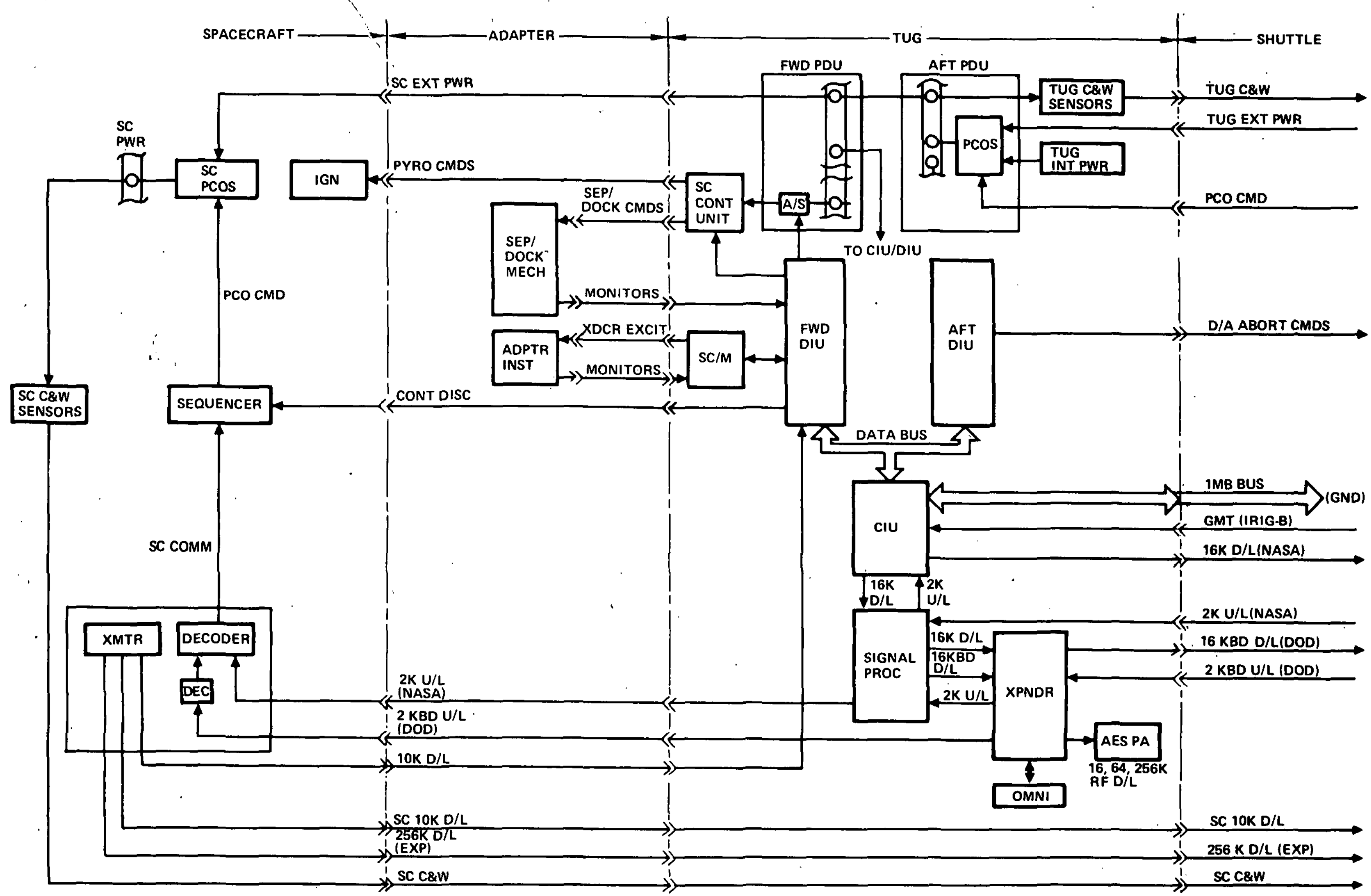

Figure 11-1. Tug Functional Interfaces

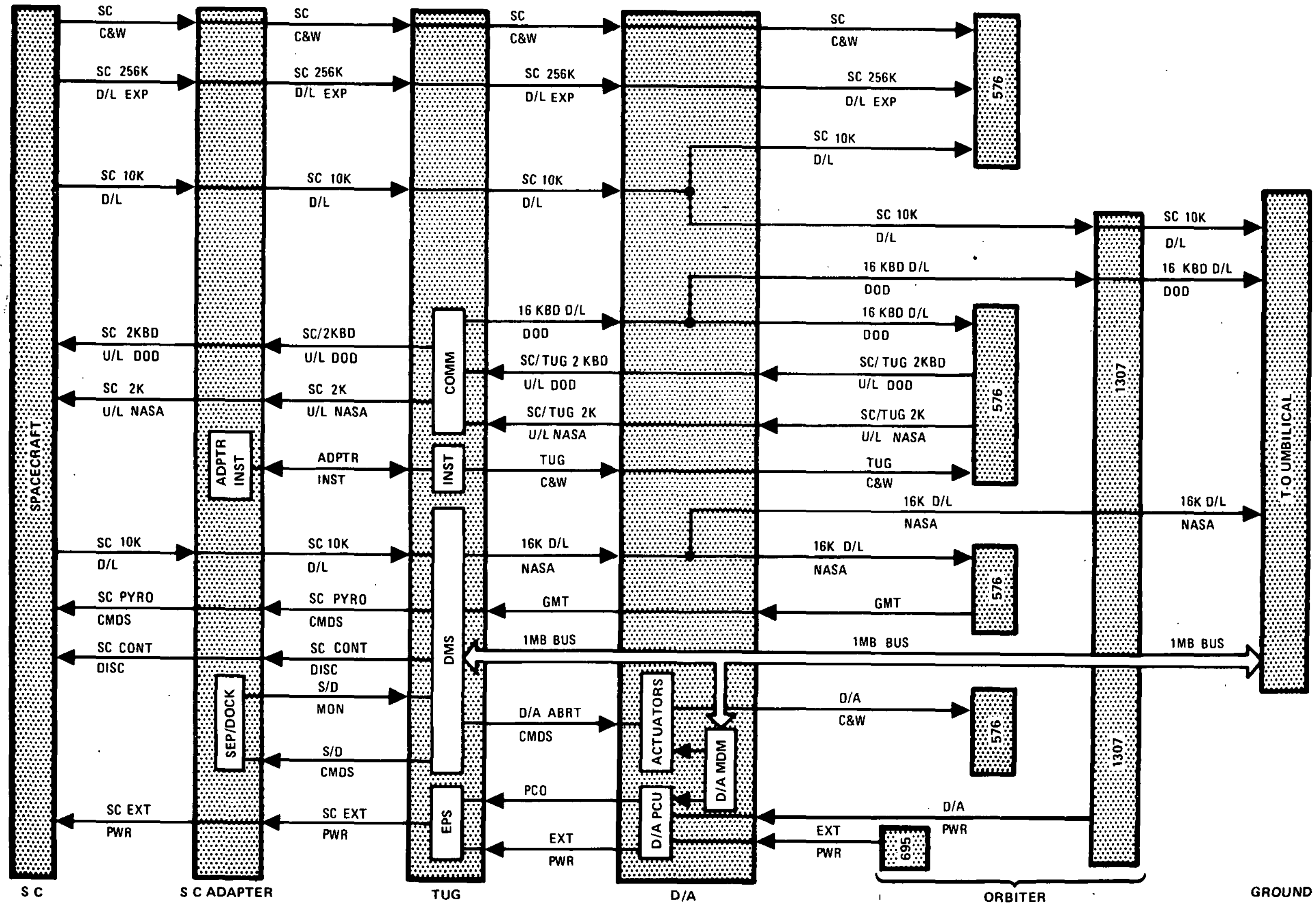

Figure 11-2. Interface Signal Flow Diagram

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b. 2K baud dual redundant encrypted (DOD) — interfaces through Tug Communications subsystem from either a hardwire or rf source. Receiver is decrypter in spacecraft communications subsystem.

All Tug spacecraft data is available at either the Tug or spacecraft communications subsystem. The uplinks enter Tug on a single channel and are effectively separated in the Communications subsystem for distribution to the Tug DMS or spacecraft. Address acknowledged data is responded to and gated in by the proper receiving device.

11.1.2 DOWNLINKS

a. 10 Kbps (maximum) nonredundant — interfaces with Tug forward DIU. Spacecraft status/response data may be monitored as it moves through the DIU on to the Tug Communications subsystem where it is submultiplexed into the Tug 16 Kbps (NASA) downlink (hardwired or rf). DOD spacecraft data flow is identical except that encryption takes place in the Tug Communications subsystem signal processor for transmission down the 16 K-baud link (DOD).

b. 10 Kbps (maximum) nonredundant — hardwire link straight through Tug to Orbiter Payload Data Interleaver (PDI). Tee in D/A to ground through T-0 umbilical.

c. 256 Kbps nonredundant — hardwire link straight through Tug to Orbiter wideband signal processor. Link content is spacecraft experiment data as opposed to engineering (support) data on the 10 Kbps links.

11.1.3 FUNCTIONS

a. Spacecraft caution and warning — 30 dual-redundant hardwired signals straight through Tug provide safety status data to Orbiter.

b. Control discretes — 10 dual-redundant signals driven by the Tug forward DIU. Used to primarily for preprogrammed spacecraft sequencing such as power application mode control and appendage deployment.

c. External power — 28V nonredundant bus.

d. Pyro commands — five dual-redundant paths used to implement spacecraft ordnance functions as required. Driven through arm/safe outputs to Tug mounted control unit.

e. Separation/docking commands (adapter interface only) — six dual-redundant paths originating from arm/safe outputs in Tug control unit. Sequence controlled by forward DIU.

f. Separation/docking monitors (adapter interface only) — 25 nonredundant plus four dual-redundant monitors used to verify completion of either maneuver. Real-time command loop uses forward DIU for feedback.

g. Adapter environment instrumentation (adapter interface only) — nine analog measurements plus excitation interfacing with Tug forward signal conditioner (SC/M).

11.2 BASELINE TUG/ORBITER CONFIGURATION

The Tug/Orbiter interface is defined at the Tug Deployment Adapter junction. The implemented interface reflects a minimum Orbiter interface both electrically and softwarewise. Minimum electrical interface minimizes harness bulk and connector size and simulates smoothness in the return docking maneuver. Minimum software dependence improves Orbiter mission peculiar adaptivity and ensures a visible, highly responsive safety data/command link.

11.2.1 UPLINKS

a. 2 Kbps dual redundant (NASA) — command uplink containing either Tug or spacecraft address decoded data. Interfaces with Communications subsystem signal processor.

b. 2K baud dual-redundant (DOD) — command uplink containing either Tug or spacecraft address decoded data. Tug data is decrypted in the Communication subsystem subsequent to passing the replica encrypted bit stream to the spacecraft.

The 2K command uplinks from the Orbiter's payload signal processor provide backup access to safety functions via the signal processor command distributor. These commands override those generated at the DIU's as a result of executing the prime DMS safety/abort program.

11.2.2 DOWNLINKS

a. 16 Kbps dual redundant (NASA) — downlinks PCM telemetry replica data to Orbiter Payload Interrogator and ground facilities (preflight).

b. 16K baud dual redundant encrypted (DOD) — secure form of the 16 Kbps link.

11.2.3 FUNCTIONS. During operational phases when the Tug is attached to the Orbiter, two operationally redundant communication paths are used to transmit the seven Tug caution and three Tug warning measurements to the Orbiter.

For warning signals the primary path is implemented by hardwiring the output of the three Tug warning sensors directly to the Orbiter caution and warning electronics unit.

In the secondary path for transmitting Tug C&W signals to the crew, the data from separate sensors is multiplexed onto the Tug telemetry data stream and transmitted to the crew via the PSP/GPC/MSS CRT display equipment. Tug caution and warning parameters are:

CAUTION	WARNING
APS ISO VLV OPEN	N_2H_4 TANK OVERPRESSURE
ME ISO VLV OPEN	LO_2 TANK OVERPRESSURE
APS CLUSTER FAILED	LH_2 TANK OVERPRESSURE
APS PRI/ELEC FAILED	LO_2 TANK UNDERPRESSURE
APS PROP LOW	LH_2 TANK UNDERPRESSURE
ME ARM/SAFE ARMED	
APS ARM/SAFE ARMED	

a. Power changeover command — executes Orbiter controlled Tug prime power mode change from Orbiter supply to Tug internal (fuel cell) supply.

b. Tug external — Orbiter supplied redundant 28V prime power bus. When external power is applied to the Tug, the Tug CIU/DIU communications system is energized by the critical power bus allowing selective power control of other Tug systems. The CIU/DIU/Critical Power Bus provides backup safety control. It also enables application of power to the spacecraft, thereby allowing prelaunch and maintenance operations without activation of the full avionics system.

c. GMT — IRIG-B format time code provided to Tug DMS.

d. Spacecraft functions — feedthrough of hardwired spacecraft functions as defined in Section 11.1.

11.3 BASELINE TUG/GROUND CONFIGURATION

All Tug/Ground interfaces are by way of the Deployment Adapter (D/A). When mated to the Orbiter, the combination Tug and D/A have strictly a communications link to the ground. In the unmated state a ground power plug is also provided to the D/A Power Control Unit (PCU).

11.3.1 UPLINKS. (BI-DIRECTIONAL BUS) 1 Mbps dual-redundant (feedthrough from Tug/Orbiter interface).

11.3.2 DOWNLINKS

a. Tug dual-redundant PCM telemetry in both 16 Kbps (NASA) and 16K baud (DOD) formats.

b. Spacecraft nonredundant 10 Kbps telemetry downlink (feedthrough from Tug/Orbiter and Tug/spacecraft interfaces).

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