REPORT NO. CASD-NAS75-012
CONTRACT NAS 8-31010

(NASA-CR-120658) SPACE TUG AVIONICS DEFINITION STUDY. VOLUME 4: SUPPORTING TRADE STUDIES AND ANALYSES Final Report, Jul. 1974 - Mar. 1975 (General Dynamics/Convair) 188 p HC $7.00 CSCL 22B G3/19

N75-27060

Unclas
29795

SPACE TUG AVIONICS DEFINITION STUDY

FINAL REPORT

VOLUME IV ✦ SUPPORTING TRADE STUDIES & ANALYSES

GENERAL DYNAMICS
Convair Division

REPORT NO. CASD-NAS75-012
CONTRACT NAS 8-31010

SPACE TUG AVIONICS DEFINITION STUDY

FINAL REPORT

VOLUME IV + SUPPORTING TRADE STUDIES & ANALYSES

April 1975

Prepared by
GENERAL DYNAMICS CONVAIR DIVISION
P.O. Box 80847
San Diego, California 92138

Prepared for
National Aeronautics and Space Administration
GEORGE C. MARSHALL SPACE FLIGHT CENTER
Huntsville, Alabama

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.

The study results were developed during the period from July 1974 to March 1975. The final presentation was made at NASA/MSFC on 3 April 1975. Principal Convair contributors to the study were:

Maurice T. Raaberg	Study Manager
Carl E. Grunsky	System Task A Leader
Richard A. Shaw William A. Robison	Guidance, Navigation, & Control
Edward J. Beveridge	Communications
Chuck R. Botts	Electric Power
Ron N. Roth	Power Distribution
Billy R. Lutes	Interfaces
Michael J. Hurley	Rendezvous & Docking Task B Leader
Lou G. Tramonti	Flight Mechanics
Edward J. Beveridge	Data Management Task C Leader
Bruce A. Gurney	DMS
Lou A. Saye	Checkout Task D Leader
James A. Burkhardt	Instrumentation
Frank E. Jarlett	Simulation/Demonstration Task E Leader
Lee E. Bolt	Cost/Programmatics
Norman E. Tipton Eric Makela	Safety & Reliability

Requests for additional information should be addressed to:

Mr. James I. Newcomb
Space Tug Task Team, PF02
Marshall Space Flight Center, AL 35812

TABLE OF CONTENTS

TABLE OF CONTENTS, Contd

TABLE OF CONTENTS, Contd

LIST OF FIGURES

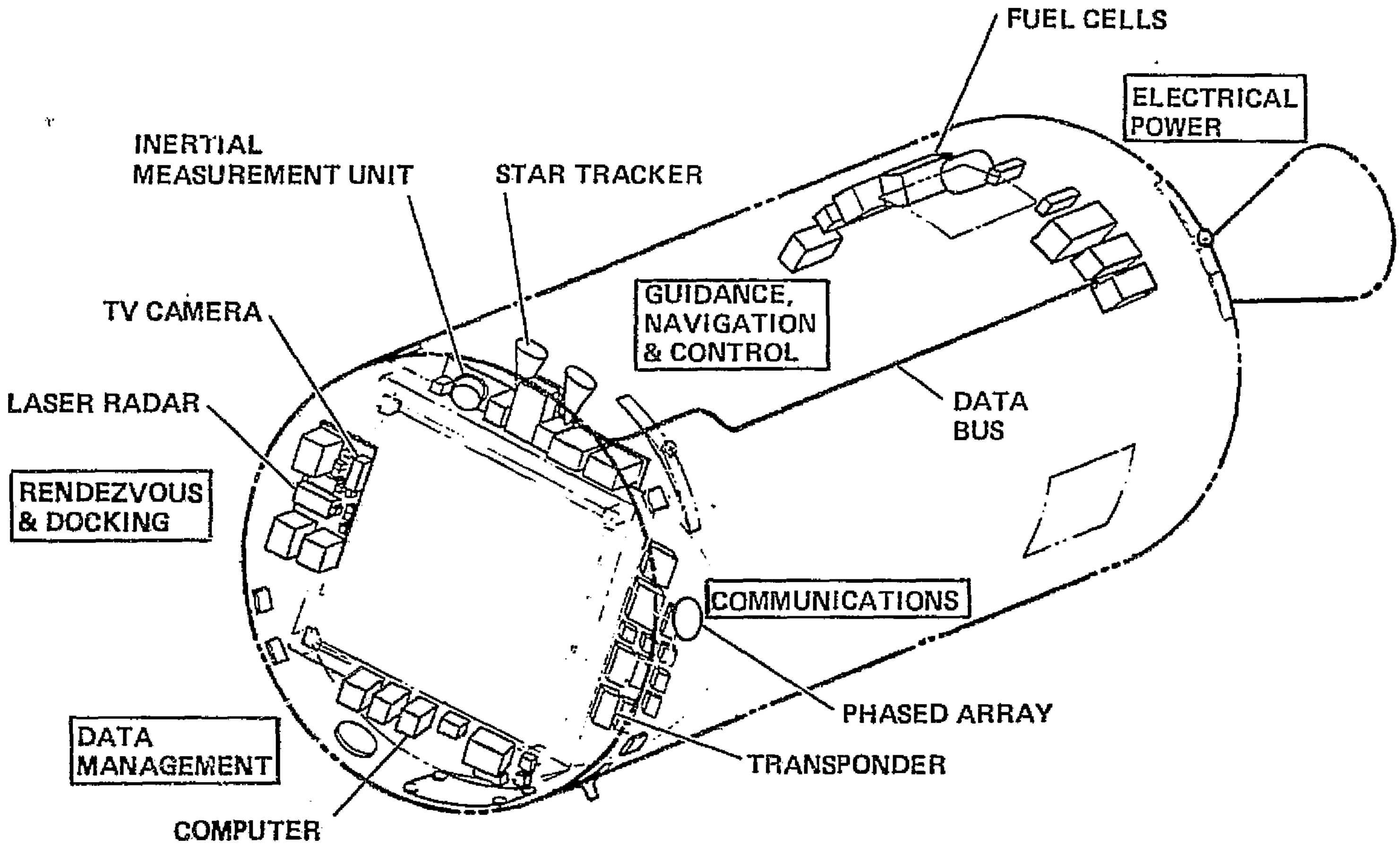
PRECEDING PAGE BLANK NOT FILMED

LIST OF FIGURES, Contd

LIST OF FIGURES, Contd

LIST OF TABLES

LIST OF TABLES, Contd


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

SECTION 1

SUMMARY AND INTRODUCTION

In the course of the Avionics Definition Study, many analyses and trade studies were performed for the evaluation of the most desirable solutions to subsystem requirements. These were accomplished at system, subsystem, and at component levels within the major task groupings of the Study Plan. The purpose of this volume is to record and to explain the criteria, the candidate options evaluated, the selection process, and the recommended solutions that have been integrated together in the configuration descriptions of Volume III of this Final Report. For purposes of clarity and to relate the trades properly to the baseline Avionics System, this Volume IV has been organized by subsystem encompassing in each section the pertinent trades/analyses for that technical area.

Significant trade recommendations and conclusions that are discussed in the following sections include:

Guidance, Navigation and Control

Navigation update is essential for the Tug irrespective of IMU choice; and, a moderate accuracy IMU is adequate.

The laser gyro dodecahedron IMU is a low cost, low risk selection for meeting the requirements.

Communications

Link demands are greatest for the downlink telemetry and involve high data rates up to 160 Kbps, assuming maintenance data and/or remote TV are to be sent to ground stations.

A lower risk and flexible solution for Communications includes three transmit-only steerable phased arrays, two hemispherical antennas, and dual Orbiter type transponders.

Electrical Power

Dual lightweight fuel cells, thermally integrated with the other Tug subsystems, are superior.

An emergency battery to furnish 40 minutes of Tug operation is a backup to the redundant fuel cell, providing additional safety.

Rendezvous and Docking

LLTV has been demonstrated to be a feasible docking sensor for remote manned operation. Present study results lack sufficient depth to make a final selection of the Rendezvous and Docking Subsystem.

Data Management Subsystem

The 1978 technology SUMC internally redundant compute is recommended in the face of evolving Tug requirements. The essential flexibility can be provided.

Software development and maintenance can benefit materially from the adoption of a higher order language of the HAL type.

Tug Checkout

The general Tug checkout philosophy is Condition Monitored Maintenance with preflight testing.

Extensive Tug onboard checkout capability is recommended.

Avionics Installation

Avionics system weight, 898 pounds (404.1 kilograms), has survived the many internal subsystem changes from the MSFC baseline, and is within less than 10 pounds (4.5 kilograms) of the original value.

A preliminary layout of the avionics equipment around the periphery of the payload support structure has been generated.

Sensitivity Analysis

Multiple payloads can be handled with only very minor adjustments to the accommodations for single Tug payloads.

Operational autonomy will be driven primarily by the support requirements for ground support of rendezvous and docking. The baseline Tug avionics can be operated either at Level II or Level III with minor deletions from the baseline. Recommended autonomy continues to be Level II minus.

Outputs of the trades have been utilized to update the original MSFC configuration of MSFC 68M00039-2 progressively throughout the Avionics Definition Study. Volume III presents the latest updated configuration representing the trade conclusions of the total study.

Generally, the continued recommendation of 1978 technology solutions appears to be cost effective against the Tug mission requirements. This should permit further evolution of subsystem designs as the Tug vehicle development progresses, and the real world problems are more fully understood. NASA has been making significant progress in the Tug related technology areas with existing laboratory projects, and the continuity of these is essential to ensuring Tug program success and improvements for the future.

SECTION 2

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

2.1 UPDATE ACCURACY REQUIREMENTS ANALYSIS

Before the update accuracy requirements could be ascertained, it was necessary to analyze the basic Inertial Measurement Unit (IMU) performance on a reference mission. The reference chosen was the synchronous mission, as this is one of more difficult for the guidance system because of its long duration. In addition, the synchronous equatorial mission represents a large percentage of the anticipated Tug missions and is one of the best defined. Therefore, it was used for all guidance system error analysis.

Figure 2-1 shows this baseline mission to spacecraft deployment. After separation from the Orbiter, the Tug performs a main engine burn to inject it into the phasing orbit. The maximum Tug phasing orbit, 2.9 hours, was used for the baseline. After one orbit in the phasing orbit, another main engine burn is executed to inject the Tug into the transfer orbit to synchronous altitude. After the five hour coast in the transfer orbit, synchronous altitude is obtained, the Tug main engine circularizes the orbit, and the spacecraft is deployed.

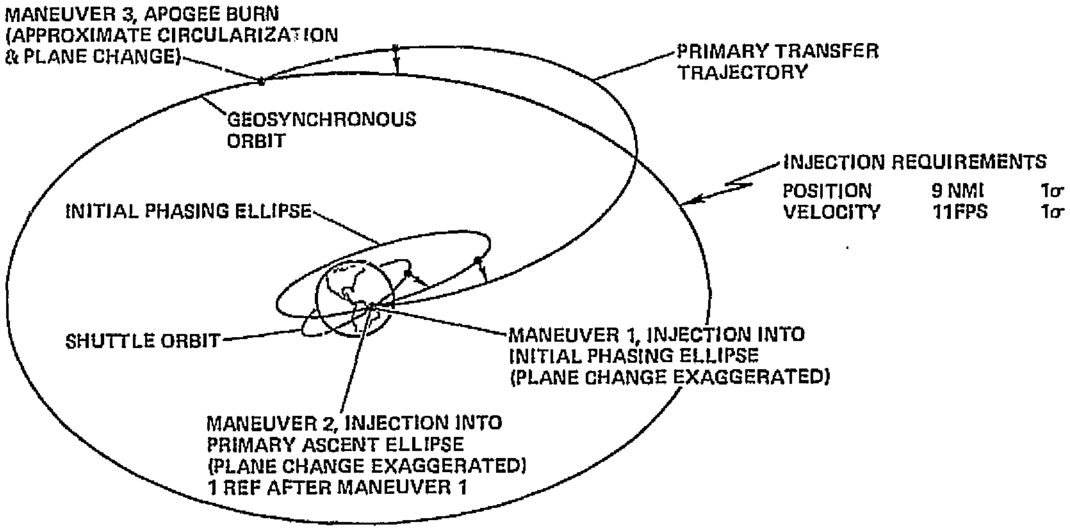


Figure 2-1. Baseline Synchronous Mission — Payload Placement

To determine the best way to meet the 3σ Tug synchronous injection requirements of 27 n. mi. (50 km) and 33 ft/sec (10 m/s), an error analysis was performed with errors only in initial attitude, position, and velocity. The results, which are equivalent to using a perfect IMU, are shown in Table 2-1. Note that even with good initial conditions and a perfect IMU, the Tug position requirement cannot be met. Therefore, a position and velocity update is required just to meet the Tug injection requirements.

Table 2-1. 3σ Errors at Synchronous Orbit Due to Initial Conditions

Item	Magnitude	Synchronous Position Error n. mi. (km)	Synchronous Velocity Error ft/sec (m/s)
Initial Alignment	72 arc sec (349 μ radians)	6.5 (12)	2.8 (0.85)
Initial Position	0.3 n. mi. (0.56 km)	22.6 (42)	1.9 (0.58)
Initial Velocity	3.75 ft/sec (1.14 m/s)	46.5 (86)	4.1 (1.25)
Tug Requirement		27 (50)	33 (10)

Although initial attitude errors do not cause too large an error during ascent, an attitude update system is required to bound the attitude error caused by gyro drift. To meet the pointing accuracy requirement of 0.2 degree (0.00349 radian) throughout a 160 hour mission without an attitude update, the gyro drift must not exceed 0.001 degree/hour (17.5 μ radians/hour). This performance can only be approached by using a large, complex electrostatic gyro system. And, of course, the navigation system requires better attitude than 0.2 degree (0.00349 radian) to perform accurate navigation. Therefore, an attitude update system is also required to meet Tug requirements with reasonable IMU gyro performance.

Having established the requirement for update systems, the next problem is to determine how accurate an update system and IMU are required. For a medium accuracy IMU and typical update system performance, it was observed that the performance of the position and velocity update system is the dominant error source. This is shown in Table 2-2. Sensitivity of the navigation update to the time of the update is given in Figure 2-2 for several update system accuracies. As shown, it is desirable to perform the update as close to the synchronous apogee as possible. This allows the

Table 2-2. IMU Accuracy Requirements (1σ)

Error Source	Value	Position Error* n.mi. (km)	Velocity Error* ft/sec (m/s)
Axis Misalignment	144 arc sec (698 μ radian)	0.07 (0.13)	5.1 (1.6)
Accelerometer Bias	100 μg	0.05 (0.09)	1.7 (0.53)
Accelerometer Scale Factor	60 ppm	0.01 (0.02)	0.26 (0.08)
Gyro Fixed Drift	0.1 deg/hr (0.0017 rad/hr)	0.09 (0.17)	3.0 (0.93)
Gyro Scale Factor	55 ppm	0.01 (0.02)	0.24 (0.07)
Update Velocity Accuracy	3.75 ft/sec (1.14 m/s)	7.6 (14.1)	5.0 (1.6)
Update Position Accuracy	1.4 n.mi. (2.6 km)	1.9 (3.5)	0.8 (0.25)
Total (RSS)		7.8 (14.5)	7.9 (2.41)

*Assumes position, velocity update 3 hours before injection and attitude update 15 minutes before injection.

**Meets injection accuracy requirements of 9 n.mi., 11 ft/sec (17 km, 3.3 m/s).

sensors to accumulate the maximum amount of data with which to calculate the Tug position and velocity. Also, the later the update is performed, the less sensitive the injection errors are to IMU performance. However, an early update is desired to allow an efficient midcourse correction to be performed. An update time of three hours before the apogee burn with an update accuracy 3.75 ft/sec (1.14 m/s) and 1.4 n.mi. (2.6 km) can meet Tug injection requirements. These are the position and velocity updates accuracies used in Table 2-2.

The following sections, 2.2 and 2.3, describe the navigation update systems and IMU's that can meet these requirements. It should be noted that in the IMU area the requirements are met by almost any of the systems presently available. Therefore, adding

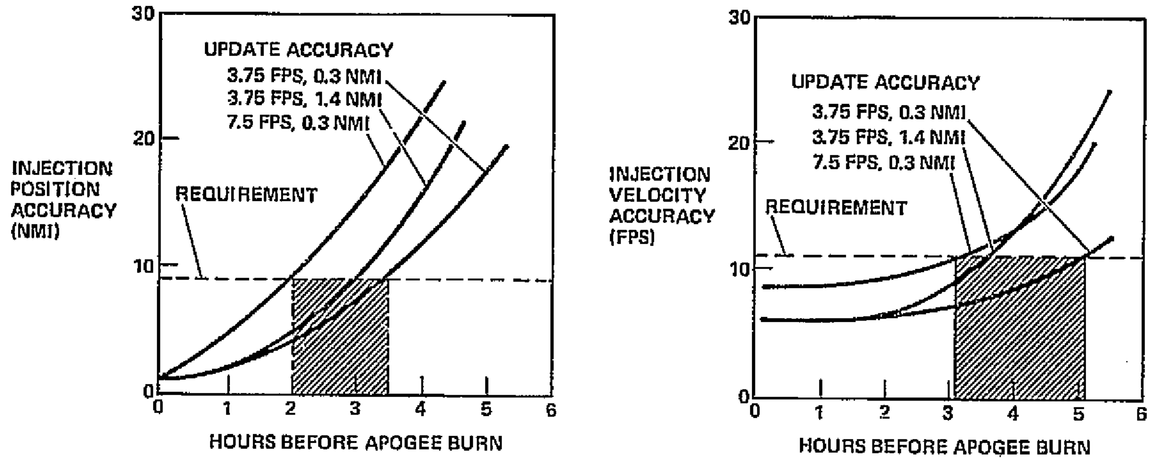


Figure 2-2. Position, Velocity Update Requirements (1σ)

the update systems not only made it possible to meet Tug mission requirements, but also allowed the relaxation of the IMU requirements so that highly reliable, low cost IMU's with moderate accuracy could be considered. The laser gyro IMU is in this category.

2.2 UPDATE TECHNIQUES TRADE

As discussed, a navigation update is needed to meet Tug requirements. The update techniques can be categorized as either direct or indirect. Direct systems provide position and velocity information with a minimum of Tug onboard computations. Ground tracking and navigation satellites are examples of direct navigation update systems (see Figure 2-3). The ground tracking system performs all the required measurements and calculations on the ground and sends the Tug its position and velocity. Navigation satellites can provide data from which the Tug can easily calculate its position and velocity.

Indirect systems require a considerable Tug onboard computation on many sensor readings. One-way Doppler can provide one component of Tug velocity immediately, but to obtain all components, many sitings must be used. After many readings are obtained, a Kalman filter can be used to determine position and velocity. The Interferometric Landmark Tracker (ILT) and horizon scanner are similar in that they both provide earth attitude information. The ILT measures the direction to a radar station while the horizon scanner effectively measures the direction to the center of the earth. If the vehicle inertial attitude is known, this data can be input to a Kalman filter and, in time, the complete vehicle orbit can be calculated. This can perhaps be best visualized in low orbit, where the horizon scanner gives the direction to the center of the earth. As indicated in Figure 2-3, this vector allows the vehicle

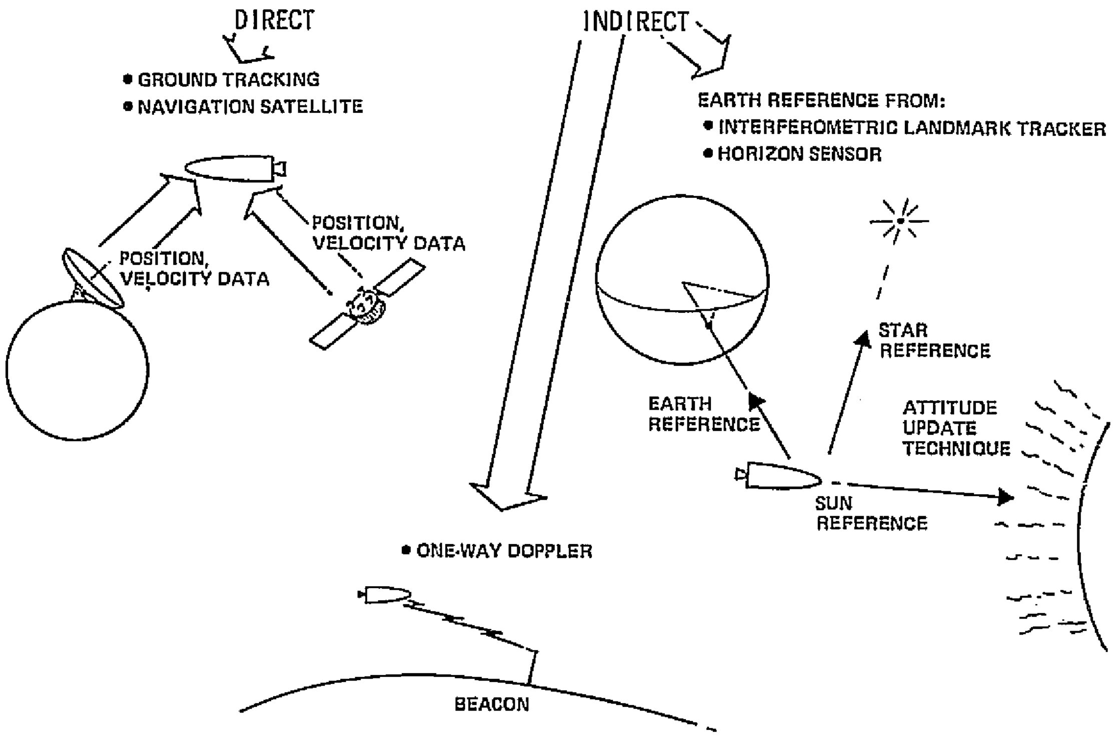


Figure 2-3. Position, Velocity Update Techniques

longitude and latitude to be calculated. After several of these position updates, the complete orbital parameters can be calculated. The ILT system, which provides vectors to known landmarks, similarly allows the Tug orbital parameters to be calculated. Although this technique is best visualized in low orbits, it can be used on any orbit.

Except for the ILT system, all of the indirect position and velocity update systems require an accurate inertial reference. On Tug, this is provided by the star tracker and sun sensor working with the IMU. The ILT system can also provide attitude information. However, a good initial estimate of the Tug attitude is required or the ILT system will not converge. To maintain all attitude capture and minimize the complexity of the ILT system implementation, Tug attitude update from the ILT was considered as a backup capability only.

The direct navigation update systems are ground tracking and navigation satellites. Ground tracking requires no additional equipment onboard, since the communications system provides the transponder required to track the Tug. The ground station calculates the Tug orbit from the tracking data and transmits the orbit to the Tug. A navigation satellite update would require the addition of a four channel L-band correlation receiver. Output of this receiver would go to the Data Management System (DMS) computer where the Tug position would be calculated. By taking several readings of the position data from the satellite, the Tug velocity can be calculated.

Indirect navigation update systems studied were ILT, horizon scanners, and one-way Doppler. The ILT requires an array of four antennas and a four channel receiver, as shown in Figure 2-4. Four antennas receive signals from the same, uncooperative, ground based radar. By comparing the phase (time) difference between the four signals, the angular direction of the radar station can be determined. Since any three channels are sufficient to determine the pitch and yaw components, a fourth channel provides improved accuracy and redundancy.

2.2.1 ILT SYSTEM. The ILT system is planned for the use of a selected set of about 400 S-band radars. The frequency and location of these radars will be stored in computer memory. As the Tug passes over a station, the computer will tune the ILT to the station frequency. ILT output will then be used to improve the computer's present position estimate. If the ILT data is grossly different than expected, or if several signals are received, the data will be rejected. If no signal is received, the computer will initialize the ILT to look for the next station. There is adequate rf power in these radars to update from backlobe signals in low orbits. At higher altitudes, mainlobe and sidelobe signals may be required. With mainlobe signals, the system is usable beyond 100,000 n.mi. (185,200 km). Due to station maintenance or on high altitude missions, data may not be received from all stations. Analysis has shown that over half of the stations can be down without significant performance degradation.

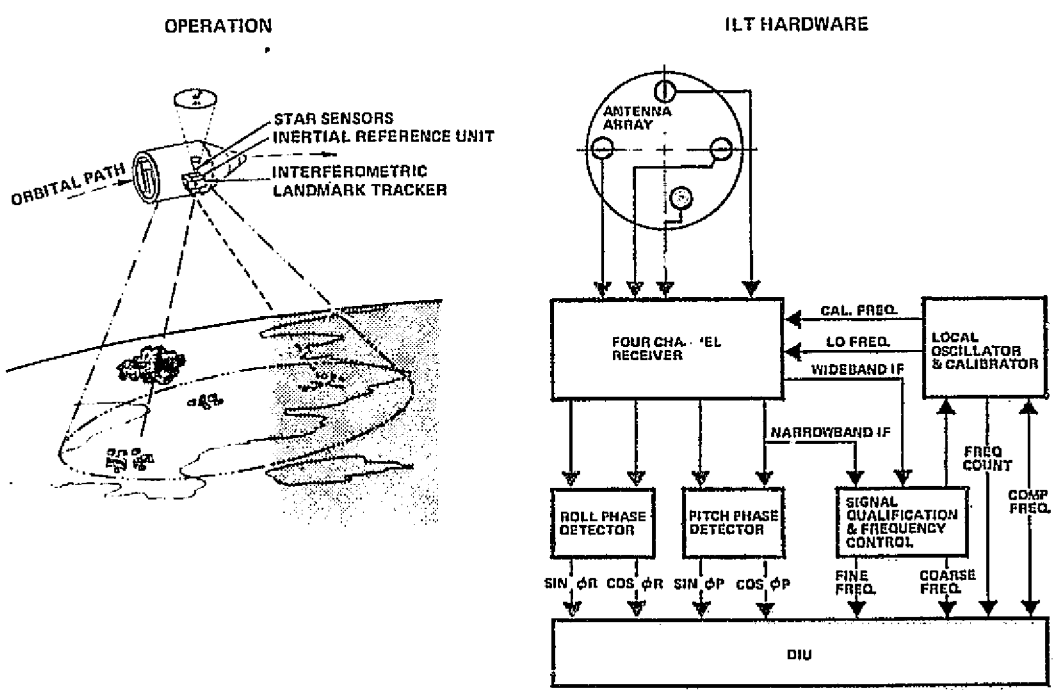


Figure 2-4. Interferometric Landmark Tracker (ILT)

2.2.2 HORIZON SENSOR. Horizon scanners use a mechanical scan of infrared sensors to detect the earth's horizon. By scanning in two axes, the direction of the earth's horizon in both pitch and yaw can be obtained. By bisecting the angle between the horizons, the direction of the earth's center can be determined. The pitch and yaw components of this earth vector are then outputs to the DMS.

Only the edge tracker type of horizon scanner is capable of operating at synchronous altitude. The edge tracker has four sensors equiangularly spaced and with small fields of view. The detector fields are moved in and out until each sees equal amounts of earth and space during oscillation. This edge discontinuity is then tracked by each sensor.

2.2.3 ONE-WAY DOPPLER. One-way Doppler methods require a Doppler receiver that extracts the rf carrier received by the communications subsystem from a ground station. As shown in Figure 2-5, by comparing the rf carrier to a very precise clock, the Doppler receiver can determine the Tug velocity relative to the ground station. To meet the accuracy requirements, both the Tug and the ground station must have a very accurate time reference. The relative velocity obtained from the Doppler receiver will be sent to the DMS computer, which will have a catalog of the location of the properly equipped ground stations. As the Tug passes over each one, a velocity measurement will be obtained. After several measurements, an accurate description of the Tug orbit can be calculated.

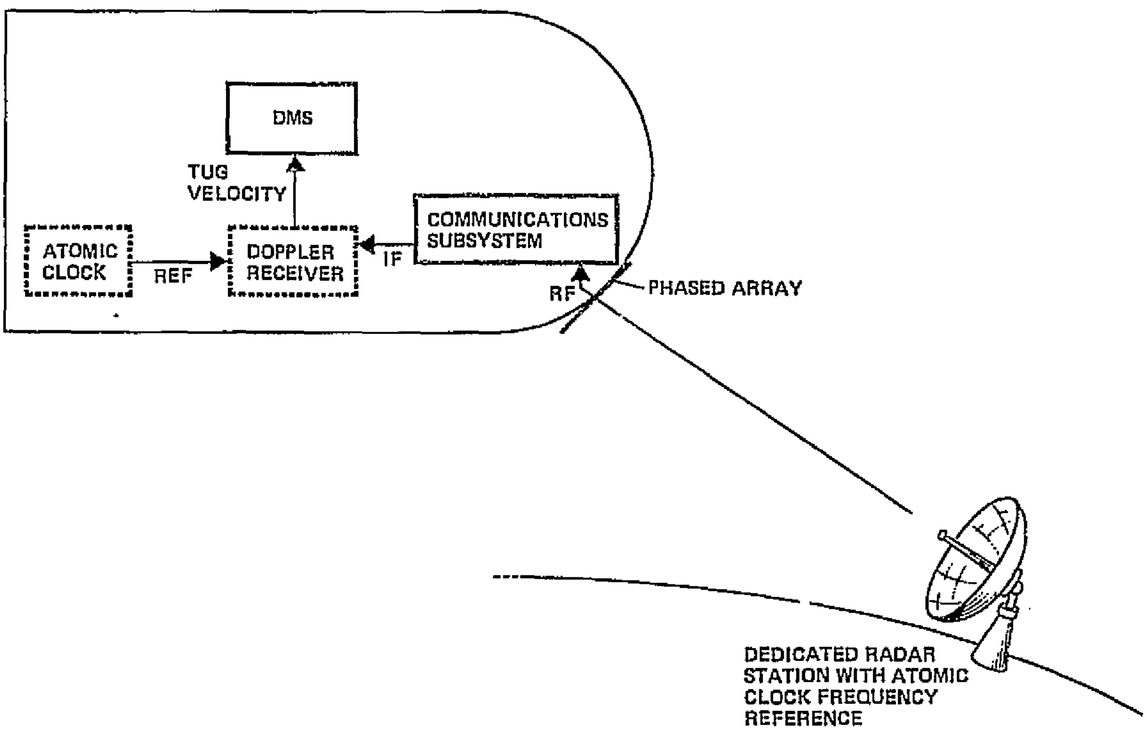


Figure 2-5. One-Way Doppler

2.2.4 COMPARISON OF METHODS. Table 2-3 compares the anticipated performance of different position/velocity update systems in 1978. Ground tracking offers good position data and adequate velocity data with minimal airborne equipment cost, since the baseline communications system includes all the equipment required for ground tracking. However, ground tracking provides limited coverage and entails considerable operational complexity. The restricted coverage on escape missions or on high inclination orbits may prevent an update when needed before the critical return engine burn. Further, a wide range of Tug missions is difficult to support with the slow turnaround time presently available from ground tracking systems. Since the ability to support ground tracking is inherent in the Tug communications subsystem, this update system will not be traded against the others but will be considered a backup capability.

Horizon scanners have been proven in space applications as attitude reference systems. Since greater accuracy is required for position/velocity update than for attitude update, horizon scanners have difficulty in meeting this requirement, especially at high altitudes. They also have difficulty in operating over the widely varying altitudes the Tug will encounter. In addition, horizon sensors are expensive, have moving parts, and can lose earth reference at sunrise and moonrise.

Table 2-3. Position, Velocity Update System

TECHNIQUE	PERFORMANCE (1σ)	AIRBORNE EQUIPMENT COST (DEVEL/UNIT) ($M)	SOFTWARE ESTIMATE	AUTONOMY	WEIGHT/ POWER (LB/WATTS)	EQUIPMENT	OPERATIONAL COMPLEXITY
GROUND TRACKING	0.1 NMI/2 FPS	0	0	III	0/0	COMMUNICATIONS SUBSYSTEM	CONSIDERABLE GROUND SUPPORT REQUIRED
HORIZON SCANNER	5 NMI/6 FPS	0.759/0.163	3,000	I	20/15	2 HORIZON SCANNERS	ERRONEOUS DATA AT SUNSET AND SUNRISE–POOR HIGH ALTITUDE PERFORMANCE
PREFERRED INTERFEROMETRIC LANDMARK TRACKER	0.5 NMI/1 FPS	1.55/0.066	3,000	I	20/15	4 SPIRAL ANTENNAS, 4-CHANNEL RECEIVER & DUAL-PHASE DETECTORS	POSSIBLE MULTIPLE STATION CONFUSION AT HIGH ALTITUDE. OPERATING AT SYNC ALTITUDE ON ATS-F
SATELLITE (GPS)	0.3 NMI/2 FPS	0.97/0.110	1,000	II	10/15	ANTENNA, CORRELATION RECEIVER	ONLY USABLE AT LOW ALTITUDES
ACCEPTABLE ONE-WAY DOPPLER	0.6 NMI/1 FPS	1.11/0.121	3,000	II	15/25	DOPPLER RECEIVER, ATOMIC CLOCK	LIMITED NUMBER OF STATIONS

The Interferometric Landmark Tracker (ILT) can provide accurate angular tracking of non-cooperative radar stations. An ILT system is presently providing attitude information at synchronous altitude on ATS-F. At high altitudes, the ILT may have some difficulty resolving closely spaced radars. Nevertheless, because of the high accuracy and the inherent reliability of the system, this is the preferred update system.

Satellite update is very attractive, but impractical because the GPS navigation satellites are planned for an 11,000 n.mi. (20,284 km) orbit and updates would be available only to vehicles below that altitude. Since the Tug mission extends to synchronous altitude and beyond, this system is not acceptable.

One-way Doppler can provide very accurate velocity data if a very accurate clock is available both on the Tug and the ground stations. Only a limited number of ground stations are presently equipped with the required atomic clocks and, therefore, the system accuracy degrades between updates. The one-way Doppler system was given an autonomy rating of Level II since a limited number of support stations are available. However, since this system is being developed for the Orbiter and meets the Tug requirements, it is considered an acceptable update system.

2.2.5 CONCLUSIONS. From these comparisons, the preferred system for position and velocity update is the Interferometric Landmark Tracker, which has high accuracy and inherent reliability. Onboard costs are slightly greater, but the ground operating costs are negligible for the Tug program.

An acceptable alternative is the one-way Doppler technique, which has a somewhat greater dependency on special provisions at ground stations, additional ground operating costs, and some coverage limitations due to lack of desired station locations.

2.3 IMU CONFIGURATIONS OPTIONS

A medium accuracy Inertial Measurement Unit (IMU) is adequate to meet the Tug requirements with an update system. Therefore, the main considerations in IMU selection are cost, weight, and reliability. IMU performance requirements that were derived to meet the Tug synchronous injection requirements are as shown previously in Table 2-1. For the planned update sequence, which entails a navigation update three hours before injection and an attitude update 15 minutes before injection, the injection accuracy is relatively insensitive to IMU performance. The injection errors are dominated by the navigation update accuracy, and IMU requirements could be further relaxed, and still meet injection requirements.

IMU's can be divided into two classes, strapdown IMU's and gimbaled IMU's. The classical gimbaled IMU has the inertial instruments mounted on a non-rotating platform isolated from vehicle rotations by a set of gimbals. Strapdown IMU's have the inertial instruments hard mounted to the vehicle structure, and tend to be less accurate. Attitude update of strapdown IMU's is easier, since the update sensors can be mounted to the same reference plane as the IMU sensors. Since Tug attitude update is essential and a medium accuracy IMU is adequate, and because redundancy is more easily implemented, strapdown systems are the preferred approach for Tug.

Four classes of IMU's were identified that would be available in 1980: 1) laser gyro strapdown IMU, 2) Electrically Suspended Gyro (ESG) strapdown IMU, 3) conventional gyro strapdown IMU, and 4) conventional gimbaled IMU.

The 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.

The selected laser gyro is the Sperry ASLG-15 or equivalent. This gyro, integrated with Kearfott 2401 accelerometers, has been tested extensively at Marshall Space Flight Center. The proposed configuration for Tug would be a dodecahedron configuration with six gyros and six accelerometers integrated to provide a strapdown IMU that can tolerate two failures.

A typical ESG system applicable for the Tug is the Autonetics MICRON, which utilizes a one centimeter beryllium rotor electrostatically suspended in an evacuated housing. By constructing this rotor with a small mass unbalance, the rotor axis of rotation can be detected, resulting in a free rotor, two axis gyro. Both the ESG and the laser gyro are ideal for strapdown application because they do not require precision rebalancing. Two ESG's and three accelerometers are combined with the appropriate electronics to create the MICRON system. For the Tug application, three systems would be combined, mounted in a skewed configuration.

The Hamilton-Standard DIGS IMU was used as the conventional strapdown IMU example. This IMU uses conventional, single-degree-of-freedom gyros that are pulse rebalanced. The DIGS system is presently flying on the McDonnell Douglas Delta launch vehicle. It was chosen because it is typical of the strapdown technology flying today. For the Tug application, two DIGS systems would be needed, with the systems skewed such that no two instrument axes are parallel.

The Kearfott KT-70 was identified as a representative gimbaled system because it is the Space Shuttle IMU. The KT-70 is a conventional, four gimbal system with two speed resolvers, which allow a fairly accurate attitude update to be accomplished. For the Space Shuttle, the KT-70 was utilized in a three-system configuration because of safety and reliability considerations. For the same reasons, this same three unit configuration was selected for this trade.

Table 2-4 compares the characteristics of the four candidate IMU's evaluated for Tug. All units meet the basic performance requirements, with little to be gained from increased performance, as described earlier.

Table 2-4. IMU Characteristics

	PERFORMANCE (3σ GYRO DRIFT) (0.1 DEG/HR REQUIRED)	COST DEV/UNIT ($M)	SOFTWARE ESTIMATE (WORDS)	RELIABILITY (0.9997 REQUIRED)	WEIGHT	OTHER CONSIDERATIONS
LASER GYRO (DODECAHEDRON)	0.1 DEG/HR	7.5/0.496	4,500	0.9999	55	MULTIPLE SENSOR OUTPUTS RUGGED-SOLID STATE '78 TECHNOLOGY
ESG (3 MICRON SYS)	0.03 DEG/HR	8.5/0.526 (0.175 EA)	4,000	0.9974	51	COMPLEX CALIBRATION PROCEDURE, LOW SHOCK TOLERANCE
STRAPDOWN (DUAL DIGS)	0.1 DEG/HR	3.7/0.783 (0.391 EA)	4,500	0.9998	78	THERMAL REDESIGN REQUIRED
GIMBALLED (3 KT-70)	<0.1 DEG/HR	3.6/0.732 (0.244 EA)	3,500	0.9985	180	DIFFICULT TO ISOLATE FAILURES

The MICRON system should be operational by 1978, although it does not appear to be as far along as the laser gyro. At present, the MICRON system in normal configuration does not meet the reliability requirement, not being designed for redundant operation. The somewhat superior performance anticipated is of little value for the Tug mission.

The DIGS strapdown and Kearfott KT-70 are representative systems that are currently operational. The KT-70 is being used on the Space Shuttle Orbiter. Both systems could possibly meet the Tug requirements, but both are heavier, represent comparable development costs, and cost considerably more in production.

The laser gyro dodecahedron IMU is the preferred IMU for Tug. It is the lowest cost system and offers the highest reliability. This system is being developed and should be operational by 1978. It offers superior reliability due to the solid state gyros and the dodecahedron configuration that remains completely operable after two failures.

2.4 TVC CONFIGURATION

The Thrust Vector Control (TVC) system converts main engine position commands from the Data Management System (DMS) into a flow rate command to the hydraulic servovalve. This is accomplished by differencing the DMS engine position command with the actual engine position. This net position command is amplified and filtered and applied to the engine hydraulic servovalve.

Since the TVC system is highly integrated with the final design of the Tug hydraulic and actuator systems, this trade is of necessity preliminary in nature. Assumptions had to be made concerning the approximate configuration of the actuators and hydraulic systems due to lack of available definition.

A single string TVC servoamplifier consists of simply an operational amplifier that differences the command and position signals, provides the appropriate filtering, and drives the servovalve. This simple analog configuration has been proved in many applications and is recommended for the Tug. However, to meet the reliability requirements, redundancy must be provided in the TVC servoamplifier.

Three techniques were considered to provide the required redundancy: 1) DMS monitoring with dual servoamplifiers, 2) triple redundant voting servoamplifiers, and 3) triple redundant servoamplifiers driving triply redundant, voting servoactuators. DMS monitoring of the servoamplifier performance is done by comparing the actual engine position with the commanded engine position, with appropriate filtering to compensate for the lags in the system. This comparison would be accomplished in the DMS computer. If the engine position and command do not agree to within a predetermined tolerance, the computer assumes the servoamplifier has failed and enables the backup servoamplifier. Although this system requires the least hardware, it also is the most demanding on the DMS system, requiring continuous monitoring of the TVC system during engine burns. It is estimated that this monitoring would have to be performed at a minimum 50-hertz rate to perform reconfiguration without degradation.

Triple redundant servoamplifiers, with their outputs compared and voted electronically, eliminate the requirement for high frequency monitoring by the DMS. This would be similar to the servoamplifier implementation used in the Saturn S-IVB stage where the three servoamplifiers are wired in a "pair and a spare" configuration. The "pair" servoamplifiers consist of a control servoamplifier, which commands the engines, and a reference servoamplifier, which is connected to a dummy load. If the control and reference amplifiers disagree, the "spare" servoamplifier is enabled to command the engines.

Both of the above configurations do not consider the redundancy required in the hydraulic actuator system. By taking into account the fact that redundancy will almost certainly be required in the actuator system, the redundant servoamplifier can be implemented in a simpler manner with improved reliability. This is accomplished by utilizing three servoamplifiers driving three servovalves. as shown in Figure 2-6. The required voting is accomplished by force summing in the actuators, which eliminates the need for any failure detection circuitry in the servoamplifiers. By integrating the servoamplifiers with the servoactuator system, one voting system can reconfigure to survive a failure in either system. Since this system requires no DMS assistance, as required in method 1, and is more reliable than method 2 (no electronic voter circuitry is required), this is the recommended TVC configuration.

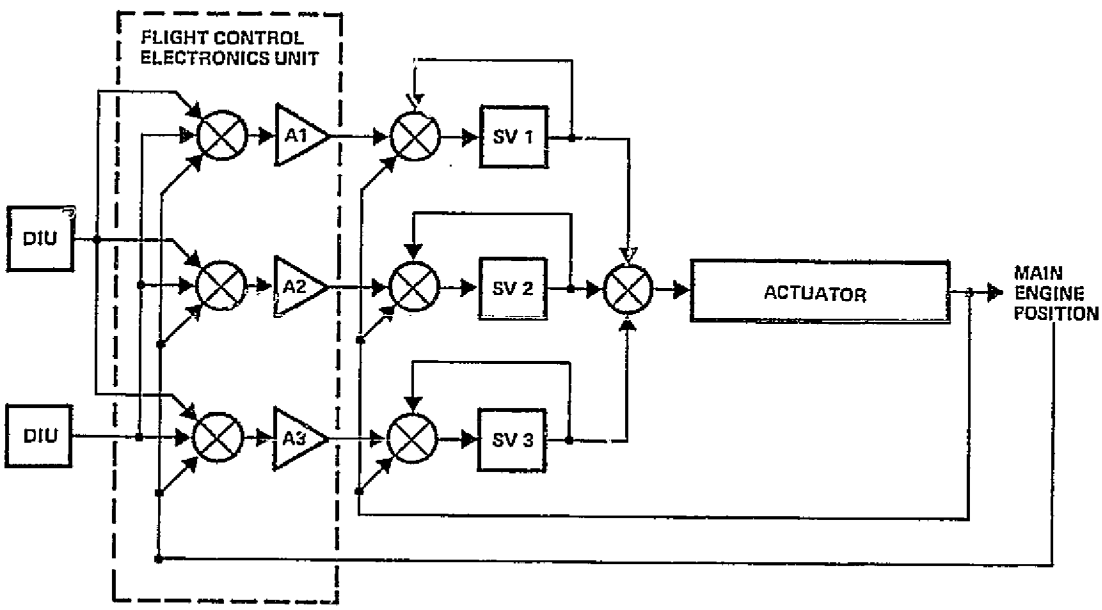


Figure 2-6. Thrust Vector Control Diagram

SECTION 3

COMMUNICATIONS SUBSYSTEM

3.1 LINK REQUIREMENTS ANALYSIS

A primary driver of system performance is the data bit rate that must be supported on the telemetry downlink. Uplink command bit rates are low and do not represent a great system impact. Safety considerations demand that ground and Orbiter command and control facilities have the capability of safing, aborting, and determining status of the Tug regardless of vehicle orientation when near the Orbiter. Another requirement driving the selection of a communications subsystem configuration is compatibility with STDN, TDRS, and AFSCF telemetry tracking and control networks. Most important is that Orbiter crew safety absolutely must not be compromised by Tug communications failure, and this drives the configuration selection to a minimum of dual redundant components.

Telemetry bit rate requirements are established by the quantity of measurements, high sample rates during operation of the main propulsion system, available transmit power, and the processing limitations of the Orbiter payload interrogator communication system. In addition to PCM data, a digitally encoded television frame must be transmitted every 15 seconds during rendezvous and docking phases. Television gray levels are encoded to three bit resolution, assembled into a PCM format with appropriate synchronization and vehicle state data, and transmitted at a serial bit rate of approximately 64 Kbps.

When the Tug is in the vicinity of the Orbiter, safety critical vehicle and spacecraft data must be available to the Orbiter crew. The Orbiter payload interrogator communication system provides a limited range, up to 20 n. mi. (36.5 km), 16 Kbps telemetry link for this safety critical data. The data included in this 16 Kbps link would represent a small subset of the total data collected onboard the Tug for status monitoring, maintenance logging, and redundancy management. This larger set of data must either be recorded onboard or transmitted to ground monitoring stations. During main propulsion system operation, high bit rates are encountered to support the dynamics of the monitored engine parameters. A 160 Kbps telemetry link is required to support real time transmission of vehicle data during main engine burn phases. Consideration of telemetry bit rates up to 256 Kbps permits more flexibility in the use of onboard recording of high sample rate data during engine burns. Data can be dumped in real time with onboard recording as backup, or a high speed dump of recorded data can be initiated by ground command. This high speed dump could be accomplished in 3 minutes compared to a typical 48 minute dump for a 16 Kbps telemetry link. Candidate alternatives to be considered for telemetry bit rates are:

a. 64 Kbps telemetry link with a maintenance tape recorder. Store and dump first engine burn on telemetry link. Store remainder of data.

b. 256 Kbps telemetry link with no maintenance tape recorder. All burn data will be transmitted in real time to the ground.

c. 256 Kbps telemetry link with a maintenance tape recorder. Transmit burn data and store on recorder. Dump recorder to ground on demand.

Another link that can be considered as an alternative to onboard storage of first burn data is to transmit data to the Orbiter during the burn, where it would be stored on the Orbiter tape recorders. This link requires telemetry communications to the payload interrogator system for a range up to approximately 250 n. mi. (463 km) and is presently limited to a 16 Kbps transmission rate. Analysis of Tug data requirements during burn phases indicates a 160 Kbps data rate is needed for adequately sampling all engine and avionics parameters. This rate far exceeds the present Orbiter payload data processing capability and is not recommended for further consideration unless Orbiter capability is revised to accept higher bit rate data.

Tug communication requirements are shown in Figure 3-1. The recommended configuration is to provide a telemetry link with selectable bit rates of 16K, 64K and 256 Kbps, depending on the mission phase. A 16 Kbps link would be used in communicating Tug/spacecraft status and engineering data to the Orbiter. Rendezvous and docking operations would utilize the 64 Kbps telemetry rate for transmission of encoded TV and vehicle data. The high bit rate, 256 Kbps, would be for real time transmission of parameters during main propulsion system operation or for a high speed dump of the onboard tape recorder.

Downlink power requirements were next analyzed to determine the worst case effective isotropic radiated power (EIRP) required to support the three selectable bit rates (16, 64, and 256 Kbps) on each of the ground communications networks. The link analysis is summarized in Figure 3-2, which shows the TDRS link requiring the greatest EIRP for a given bit rate. A substantial decrease in Tug EIRP can be obtained on the TDRS return link by using a forward error control (FEC) code, such as a rate 1/2 or rate 1/3 convolutional code. However even with FEC coding, the communications subsystem must develop a transmit EIRP of 23 DB at the high data rate. Two alternatives must be traded:

a. A low gain (omni) antenna system driven by a power amplifier.

b. A directive gain antenna system.

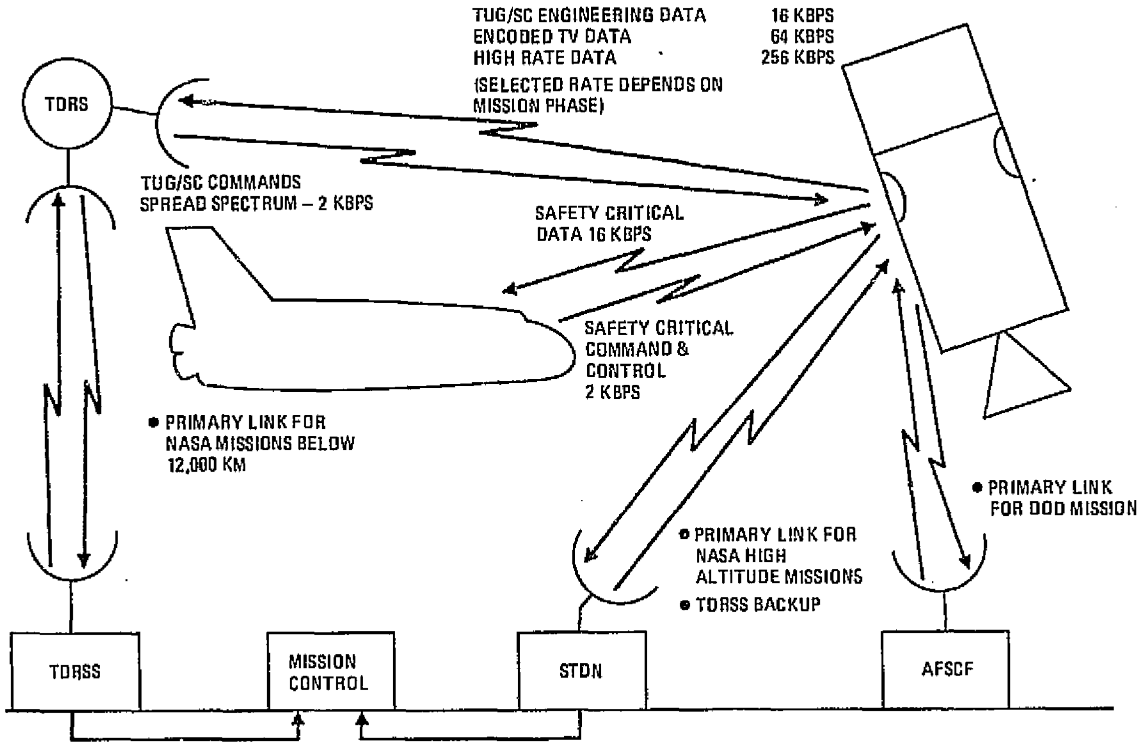


Figure 3-1. Communication Requirements for Tug

The Tug/Orbiter data link must operate reliably with limited adjustment of Tug attitude when near the Orbiter. The selected antenna system must have a low power omnidirectional transmit capability to meet this requirement.

Communications uplink requirements are presented in Figure 3-3. TDRS forward link is the only link analyzed here since adequate margin exists on the direct networks at 2 Kbps. As shown in Figure 3-3, directive gain is not required for the Tug receive antenna system. Gain margin can be enhanced by an additional 5 dB by employing FEC coding on the TDRS forward link. A low gain (near isotropic) receive antenna system configuration has been selected that allows the forward link to be established regardless of Tug attitude and assures Tug response to safety critical commands when near the Orbiter. These are the major constraints on the communications subsystem and the antenna system requirements as summarized in Figure 3-4.

3.2 ANTENNA SYSTEM EVALUATION

An omnidirectional antenna system initially appears attractive since no complex antenna selection is required and omni antennas are low cost, highly reliable devices. However, analysis of transmit power needed with 0 dB antenna gain indicates a requirement for 200 watts of rf power to be developed in the power amplifier. Power levels of this large magnitude can not be achieved with present day conventional

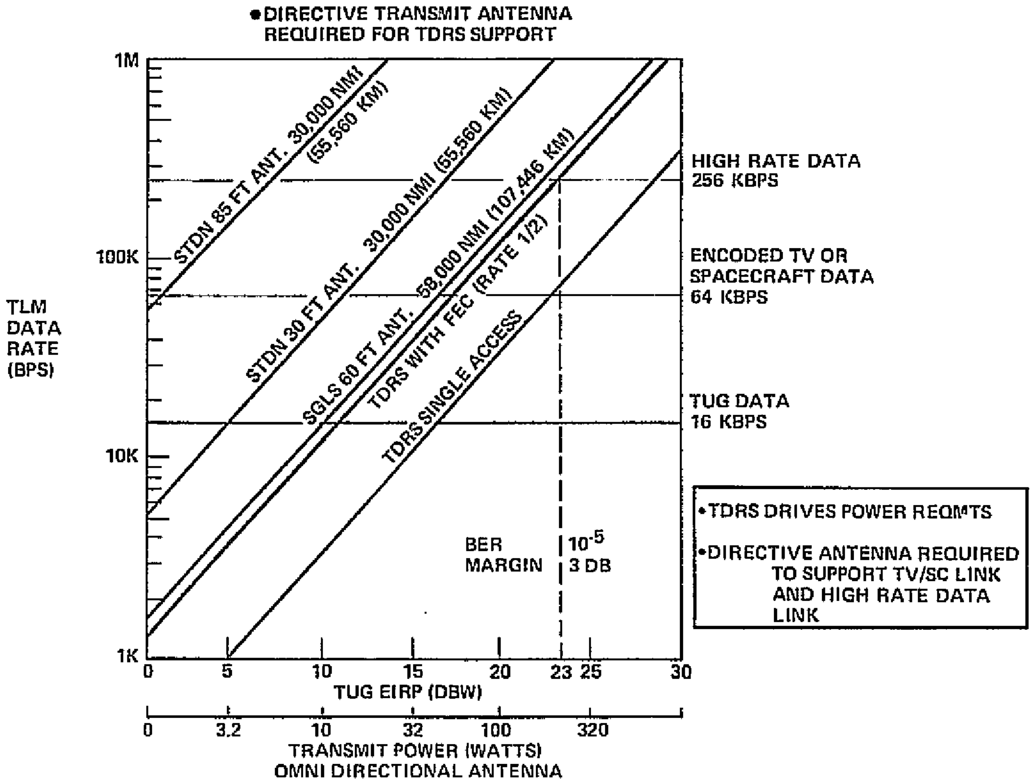
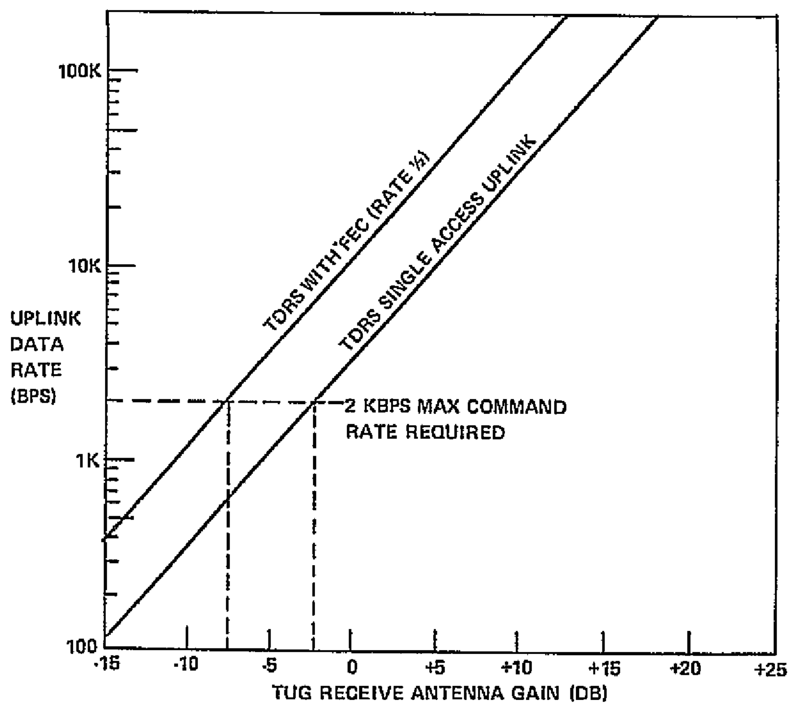


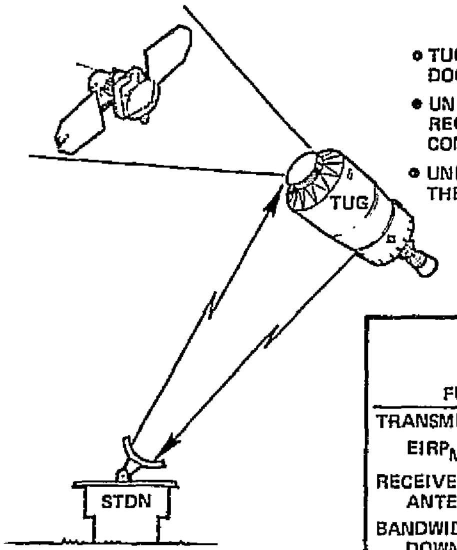
Figure 3-2. Communications Downlink Requirements



NETWORK	OMNI RECEIVE TOTAL GAIN MARGIN (DB)
STDN	
85 FT ANT.	+14.2
60 FT ANT.	+6.2
SGLS	
60 FT ANT.	+4.5
TDRS	
SINGLE ACCESS	+2.0
SINGLE ACCESS W/FEC	+7.0

DIRECTIVE GAIN NOT REQUIRED FOR TUG RECEIVE ANTENNA

Figure 3-3. Communications Uplink Requirements



- TUG ATTITUDE CONSTRAINED DURING DOCKING MISSION PHASE & NEAR ORBITER
- UNINTERRUPTED COMMUNICATION REQUIRED FOR TV, DATA & COMMAND LINKS
- UNINTERRUPTED COMMUNICATION DURING THERMAL CONTROL MANEUVERS

REQUIREMENTS
COMMUNICATION SUBSYSTEM

FUNCTION	REQUIREMENT
TRANSMIT	
$EIRP_{MAX}$	23 dBW
RECEIVE	
ANTENNA GAIN	0 dB
BANDWIDTH	
DOWNLINK	
TUG ENGINEERING DATA	16 kbps
TUG MAINTENANCE DATA	256 kbps
ENCODED TV	50 kbps
SPACECRAFT TLM	10 kbps
UPLINK	
NASA	2 kbps
DOD	2K BAUD

ANTENNA REQUIREMENT

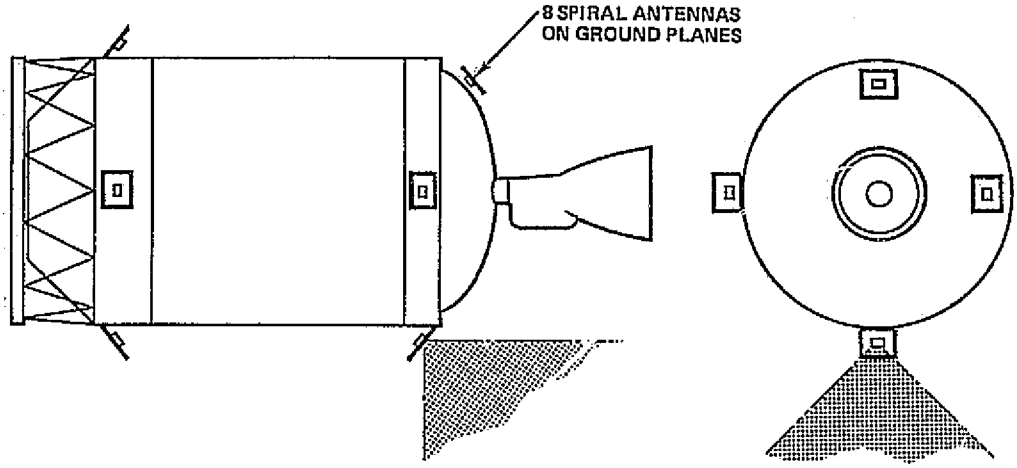
- OMNI RECEIVE
- STEERABLE DIRECTIVE TRANSMIT
- OMNI TRANSMIT (NEAR SHUTTLE)

Figure 3-4. Antenna Evaluation

semiconductor amplifiers; consequently, a traveling wave tube amplifier (TWTA) is considered the most likely amplifier candidate. TWT amplifiers are currently being used on several communications satellites and the Orbiter communications system incorporates a 150 watt TWTA for transmitting to TDRS. However, a TWTA has several disadvantages, foremost being its low reliability, especially in severe environments such as sustained in the Orbiter payload bay during launch and return. High thermal densities would also seriously degrade the reliability of this amplifier type.

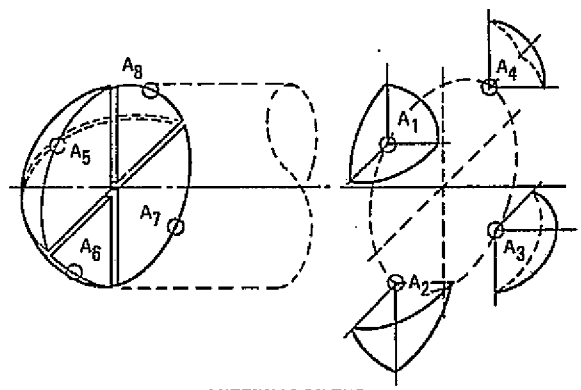
A directive gain antenna system eliminates the need for a high transmit power amplifier. However, the directivity of the antenna pattern allows only limited coverage of the total radiation sphere. Multiple directive antennas are required with antenna selection by a coaxial switch matrix to prevent major restriction of vehicle orientation with respect to ground stations, Orbiter, and TDRS satellites. Antenna selection must be under control of the Data Management Subsystem (DMS) since knowledge of Tug position and receiving station position is required to determine antenna selection.

Eight medium gain, 5.5 dB at 60 degrees (1.05 radians), directive antennas with a beam width of 60 degrees (1.05 radians) can be arranged on the vehicle, as shown in Figure 3-5, to afford essentially all-attitude coverage, Figure 3-6. Antenna selection is controlled by the DMS driving a coax switching matrix. Power amplifier

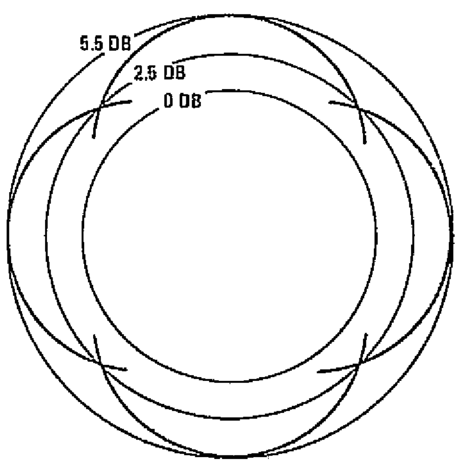


- ALL-ATTITUDE COVERAGE PROVIDED BY 8 RCP SPIRAL ANTENNAS
- APPROPRIATE ANTENNA SELECTED BY DMS COMPUTATION & COMMAND

Figure 3-5. Directive Antenna System Candidate



ANTENNAS ON TUG



PATTERN CROSSOVER
(ANGLES NOT TO SCALE)

Figure 3-6. Pattern Coverage for Fixed Directive Antennas requirements for this arrangement are 50 to 150 watts depending on bit rate selection. This approach offers a very low cost solution for transmit directivity with minimum vehicle attitude restrictions. A TWT amplifier would be required if transmission of the higher bit rate data becomes mandatory.

A third alternative antenna configuration to be considered is an arrangement of three electronically steerable phased array antennas located at 120 degree (2.1 radian) intervals around the vehicle circumference. A phased array antenna system consists of multiple electronic modules with three bit digital phase shift control. Each module can develop one watt of rf power. A 25-element phased array can support downlink bit rates up to 256 Kbps with a calculated 8 dB margin at the 60 degree (1.05 radian) scan limit when FEC coding at rate 1/2 is used on the downlink. Without FEC coding, the phased array system still shows a 3 dB gain margin. Table 3-1 shows the performance specification for a 25-element phased array.

Table 3-1. Phased Array Antenna System Specifications

Transmit elements per system	25
Transmit gain	
Boresight	19 dB
60 degree (1.05 radian) scan	14 dB
Transmit Power	25 watts
Transmit EIRP	
Boresight	33 dBW
60 degree (1.05 radian) scan	28 dBW
Noise Figure	4.5 dB
DC Power	93 watts
Diameter	15 in. (38.1 cm)
Thickness	3.4 in. (8.6 cm)
Coverage	120 degree (2.1 radian) Cone
Weight	16 pounds (7.3 kg)

Selection of a phased array antenna is again controlled by the DMS and a coaxial switch matrix. Beam steering commands are generated by the DMS.

The choice of three phased arrays, rather than two or four, is based on the 120 degree (2.1 radian) coverage cones of the steerable beam. Placing the arrays at 120 degree (2.1 radian) intervals around the vehicle gives reasonable assurance that at least one array can be steered to either a ground station or TDRS satellite located roughly perpendicular to the Tug roll axis. Coverage blind spots exist at the nose and tail; however, minor vehicle attitude adjustments can eliminate these non-covered areas. Four arrays will not significantly improve the coverage at nose and tail and result in considerable coverage overlap over the remainder of the sphere. Two antenna arrays located 180 degrees (3.2 radians) from one another leave significant coverage gaps perpendicular to the roll axis. Approximately one to three minute dropouts in data will occur during thermal roll maneuvers. Relative spherical coverage of these three configuration options is shown in Figure 3-7 (A, B, and C).

Antenna beam steering can be accomplished autonomously by the phased array system if receive amplifiers, diplexers, and a monopulse tracking receiver are included in the design. Since link calculations for Tug command links indicate receive antenna gain is not a requirement, a simpler and more reliable tracking approach was selected using the DMS to generate beam steering commands. Elimination of the auto tracking capability relieves the impact of TDRS spread spectrum bandwidth on duplexer and tracking receiver designs and eliminates several single point failure modes. DMS processing requirements are not significantly increased since the limited Tug roll rate of 1 deg/sec (0.018 rad/sec) will require update of the beam steering command only about 10 times per second.

Antenna system trade options are shown in Table 3-2, and an evaluation of each option against the traded parameters indicates no standout parameter separates the three options except reliability. The phased array system has inherent high reliability due to its solid state construction and tolerance to multiple module failures without seriously degrading performance. Seven failures are required in a 25 element array to sustain 3 dB gain degradation. The phased array system also operates at a power level of 95 watts or about one-third that of the other two systems, causing less impact on thermal dissipation design.

The recommended configuration to achieve the required transmit EIRP is the phased array system based on its higher reliability and increased safety margin for operation near the manned Orbiter.

3.3 TRANSPONDER OPTIONS TRADE

Three candidate transponder configurations were evaluated as shown in Table 3-3. The candidate selected for recommendation is a modified Orbiter unit with network

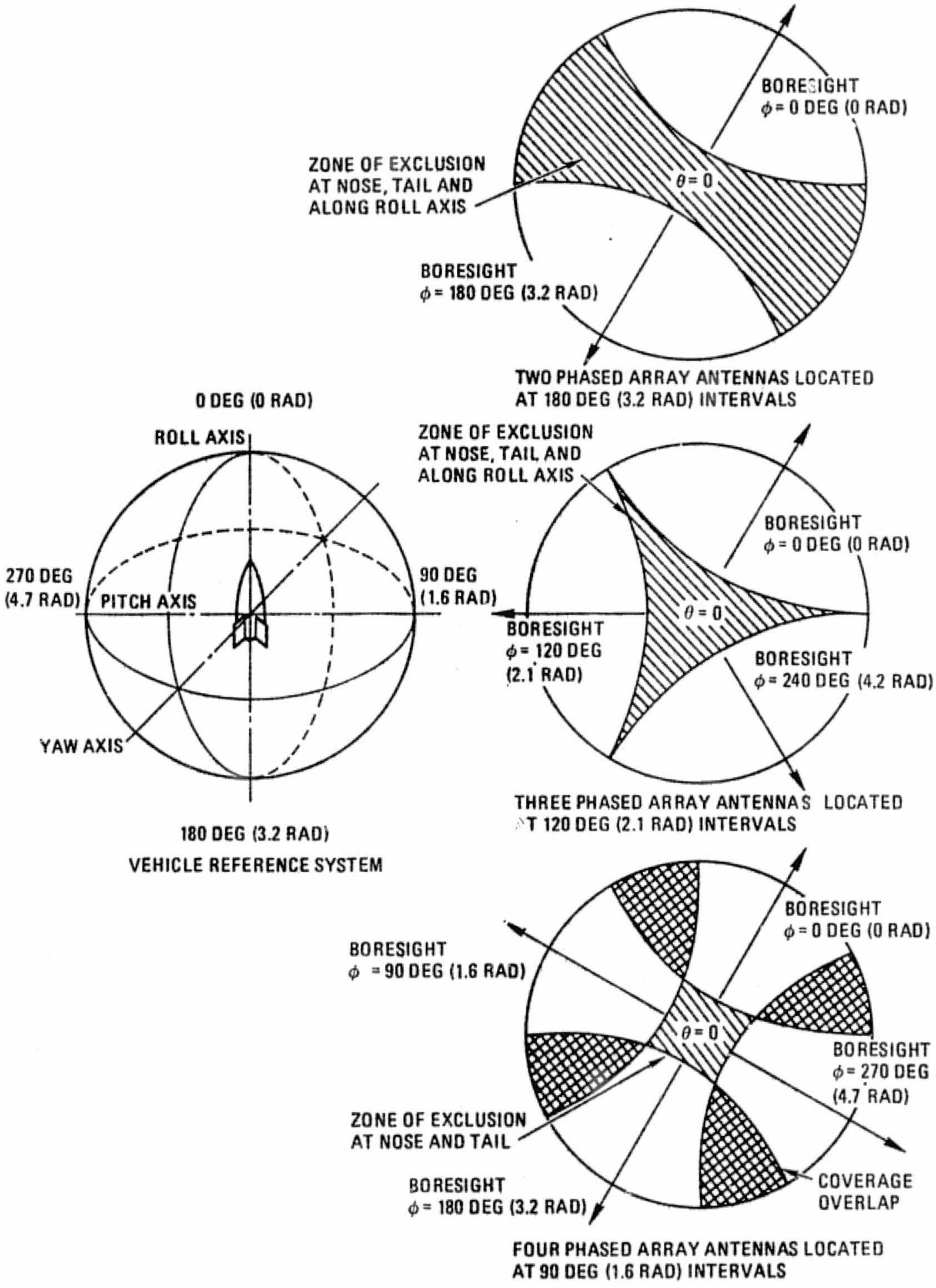


Figure 3-7. Spherical Coverage for Multiple Phased Array Antennas

Table 3-2. Antenna System Trade

Antenna System Trade Options	System Components	Weight lb (kg)	DC Power (watts)	Relative Reliability	Relative Develop Cost	Relative Recurring Cost	Comments
• Omnidirectional Receive/Transmit with Low Power Transmit Mode	(2) Hemi. Ant. (1) Ant. Coupler (2) TWT Power Ampl.	2.0 (0.9) 1.0 (0.5) 27.5 (12.5) 30.5 (13.8)	— — 360 360	High Power TWTA Req'd. Low Rel.	Low	Low	Simple low cost approach; low transmit reliability
• Multiple Directive Transmit; Omni-Receive, Low Power Omni-Transmit	(8) Directive Gain Ant. (2) Hemi. Ant. (1) Ant. Coupler (1) Ant. Switch Assy. (2) TWT Power Ampl.	8.0 (3.6) 2.0 (0.9) 1.0 (0.5) 7.3 (3.3) 27.5 (12.5) 45.8 (20.8)	— — — 5 250 255	TWTA Req'd. Low Rel.	Low	Low	Good spherical coverage on transmit; low transmit reliability
• Multiple Phased Array Transmit; Omni Receive; Low Power Omni-Transmit	(3) Phased Array Ant. System (2) Hemi. Ant. (1) Ant. Coupler (1) Ant. Switch Assy.	48.0 (21.8) 2.0 (0.9) 1.0 (0.5) 7.3 (3.3) 58.3 (26.4)	93 — — 5 98	Solid State Multiple Elements - Very High Rel.	Med.	Low	Best reliability and safety margin. Recommended

Table 3-3. Transponder Trade

Transponder	Weight lb (kg)	Modifications Required	Cost		Selection
			Non-Recurring (Dollars)	Recurring (Dollars)	
ERTS (NASA) FLTSATCOM (DOD)	30 (13.6) 10 (4.5) 40 (18.1)	TDRS Compatibility	650K 796K 1446K	197K 167K 364K	Eliminate - high development cost
Modified shuttle	16.5 (7.5)	Tug Data Rates and RF Output	470K	140K	Selected candidate
Integrated with phased array electronics	5 (2.3)	New Design	1750K	150K	Eliminate - phased array tracking receiver not required, integrated approach not feasible

compatibility achieved by module selection within the transponder. Recommendation is based on the moderate development cost and unit weight for this transponder configuration. No high risk development is required since only minor modifications are needed to accommodate Tug data rates and rf output levels.

Modification of ERTS and FLTSATCOM transponders to achieve TDRS compatibility incurs a higher risk and development cost penalty, as well as a cumbersome system design using technology of the mid-1960's. For these reasons a transponder configuration based on these units as elements of the design has been eliminated.

A phased array antenna system with monopulse tracking and autonomous beam steering could obtain attitude and position information directly by tracking landmark or satellite beacons, eliminating the requirements for a coherent transponder, ground track and relay of attitude/position updates. This capability was examined to determine feasibility of the phased array antenna for Tug position and attitude update support. Conclusions reached were that the phased array beamwidth and steering resolution eliminated further consideration of this approach for attitude/position update, since a very large costly array (> 128 elements) is required for the necessary beamwidth, and module redesign to a minimum of four bit phase shift resolution is required. High recurring cost and increased development risk eliminate the phased array landmark tracking concept for Tug.

Another candidate configuration based on the phased array is integration of transponder capability, coherent rf and tone ranging turnaround, with the phased array transmitter and tracking receiver electronics. An attractive feature of this approach is that a lightweight implementation can be obtained by sharing frequency synthesis and power sources of the existing electronics. Disadvantages do exist, however; multiple phased arrays are required for adequate directive coverage with each array requiring dual transmit and tracking receiver electronics to meet safety requirements. Also, auto track and directive receive gain are not implemented in the recommended baseline for the phased array antennas, which eliminates a large part of the shared electronics in an integrated approach. Unnecessary complexity and redundancy led to elimination of the integrated transponder candidate.

Table 3-3 shows a summary of the transponder trade comparisons and the cost and weight basis on which the modified Shuttle transponder design was selected as the recommended candidate.

SECTION 4

ELECTRICAL POWER TRADE STUDIES

Adequate vehicle performance to satisfy requirements through cost effective solutions has been the goal of the electrical trade studies and analyses. The best implementation has been sought consistent with low program development risk. Key issues addressed during the studies were:

a. Power levels by mission phase — Tug and payload.

b. Fuel cells versus batteries.

c. Dedicated reactants at high pressure versus low pressure propellant grade from main tanks.

d. Peripheral systems associated with the candidate power source.

e. Thermal integration of the fuel cell and peripherals with the vehicle system.

f. Interface requirements.

g. Emergency battery considerations.

h. Safety, reliability, and redundancy implementation.

i. Candidate fuel cell options and selection of the recommended power source.

j. Power distribution and control requirements analysis.

4.1 POWER REQUIREMENTS ANALYSIS

Potential Tug missions cover a spectrum of orbits, sequencing instructions, and mission durations. Convair has generated four reference mission timelines covering geosynchronous placement, geosynchronous placement and payload retrieval, low earth orbit, and planetary missions. For the nominal Tug geosynchronous spacecraft placement mission, the electrical power level requirements by mission phase were determined and are presented in Table 4-1. Two high power needs impact the baseline design:

a. The Tug and the Deployment Adapter solenoid valves were changed to a latching type to eliminate large holding currents (200 watts dissipation).

b. Available Orbiter power during ascent was recently reduced by NASA (JSC-07700) to 1000 watts, which is below Tug and payload requirements.

In examining where the extra power should be obtained above that available from the Orbiter, (b) above, and in consideration of the potential switching complexity, the best solution was determined to be operation of the Tug power plant in the Orbiter payload

Table 4-1. Electrical Power Requirements per Flight Phase

2 KW TUG POWER	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	[illegible]	[illegible]	114	114	99	114
GN&C	–	382	382	382	382	382	[illegible]	[illegible]	382	340	–	–
R&D	–	50	–	–	–	–		[illegible]	–	–	–	–
COMMUNICATIONS	10	10	72	72	95	165	161	[illegible]	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	[illegible]	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)	[illegible]	(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

bay. Both the possibilities of batteries or fuel cells were studied, and operation of either is practical depending on the final choice of power source type.

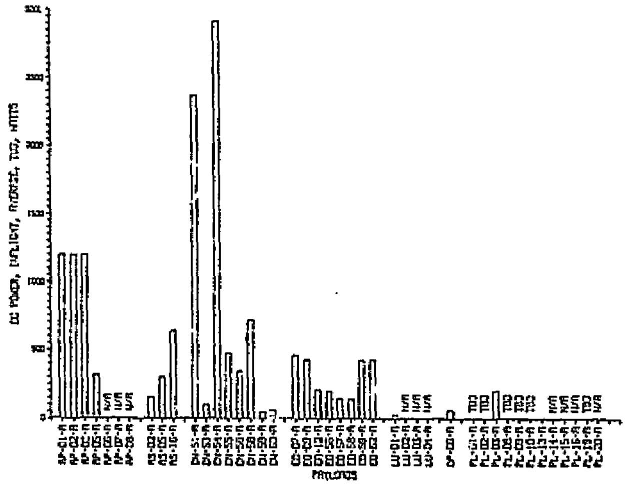
A large percentage of the Tug power demand is for non-avionic loads, such as main engine circulating pumps, venting control, and APS motor/fuel heaters. These loads vary somewhat with the particular design of the heating and thermal integration aspects of the power system with other vehicle functions. During ascent, descent, abort, and emergency conditions, the power loads are at the lowest values where some Tug function are shut down. Peak power requirements are realized for the Tug predeployment checkout, during main engine burns, and for the rendezvous and docking with orbiting payloads.

Peak spacecraft checkout power requirements to be accommodated are sufficiently high. Projected nominal values for single payloads were 700 watts and for multiple payloads as 1.15 kW; therefore, these become important in the selection of the Tug power source.

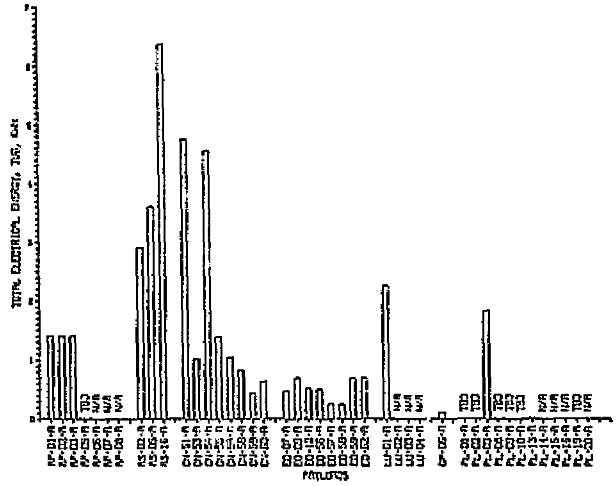
4.2 PAYLOAD POWER AVAILABILITY ANALYSIS

Spacecraft power requirements have been analyzed from two principal sources: the "Space Transportation System Payload Data and Analysis (SPDA) Study," NAS 8-29462, a continuing Convair study contract, and the Tug study just performed by McDonnell Douglas, "IUS/TUG Payload Requirements Compatibility Study," NAS 8-31013.

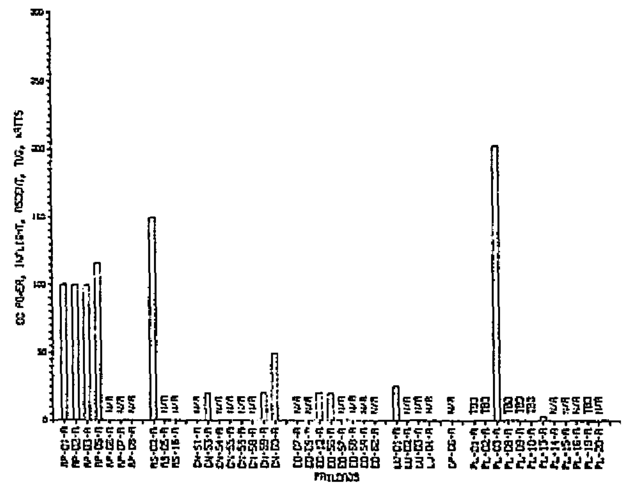
4.2.1 SPDA REQUIREMENTS. A search was made through the SPDA data bank, and 43 potential Tug payloads were identified and analyzed for the electrical power interface requirements. This effort produced the computerized plots shown in Figure 4-1 (a, b, c, d, and e) for this payload spectrum, which give the projected values of average inflight power, peak inflight power, and power during ascent while attached to Tug. Also illustrated are the corresponding peak power durations for the spacecraft while attached to the Tug, both in and out of the Orbiter, and the total payload/Tug energy



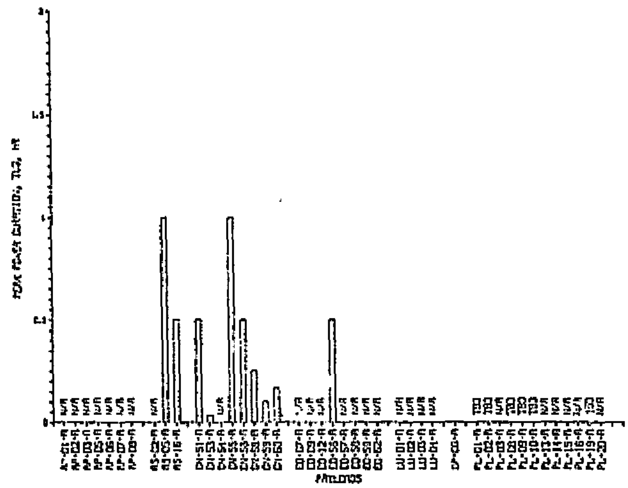
a. DC Power, Inflight, Average



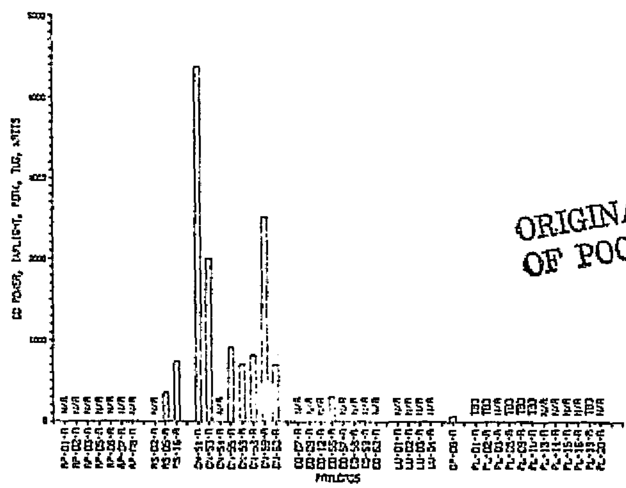
d. Total Electrical Energy



b. DC Power, Inflight, Ascent



e. Peak Power Duration



c. DC Power, Inflight, Peak

Figure 4-1. Payload/Tug Interface Requirements

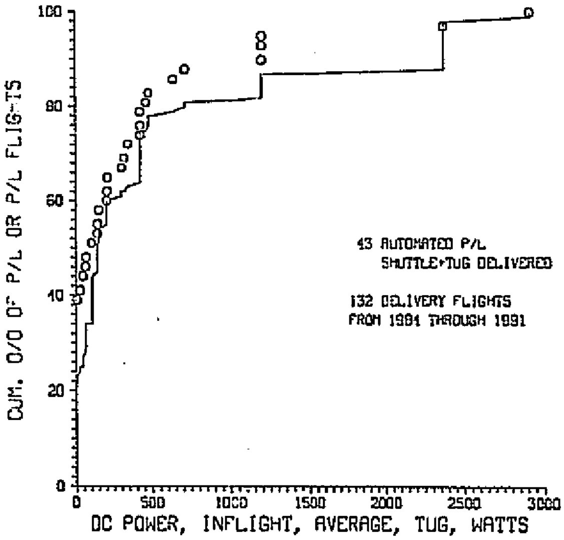
requirements. Only two or three payloads create unusual requirements; these "tall poles in the tent" for the present data are: CN-51-A and CN-54-A for power level, and AS-16-A, CN-51-A, and CN-54-A for total energy. These three payloads represent less than 10% of the 132 delivery flights in the period from 1984 through 1991. The cumulative summaries are shown in Figure 4-2 (a, b, and c).

4.2.2 IUS/TUG INTERFACE COMPATIBILITY STUDY DATA. During the progress of the Avionics Definition Study, a parallel examination of Tug payload accommodation requirements was being performed by the McDonnell Douglas Company (MDAC) as supported by the General Electric Company, NAS 8-31013, which developed comparable avionics interface information for both single and multiple payloads. From the MDAC study, Table 4-2 shows the anticipated Tug payload assignments and lists 56 potential spacecraft. The avionics power requirements summary data from the same study is shown in Table 4-3.

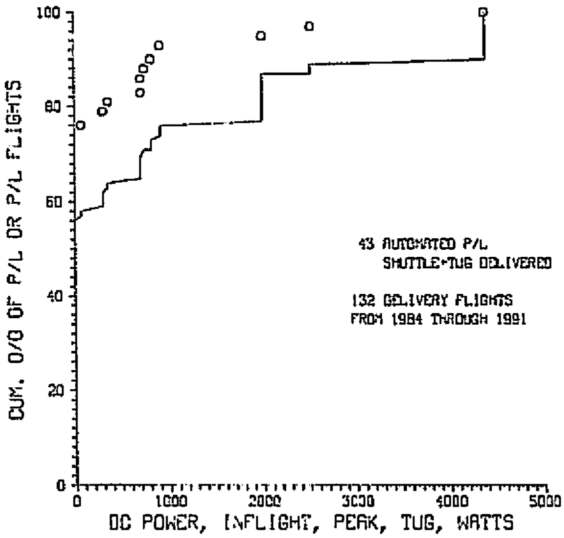
The few exceptional payloads that require power levels and energy capacity in excess of the nominal values can be handled by sharing the spacecraft loads between the Orbiter and the Tug power sources, or by peaking batteries carried on board the particular spacecraft. Some of the fuel cell options that have been examined in the trade studies have sufficient flexibility to meet the additional requirements, if the power plant design is properly thermally integrated with the vehicle to allow removal of the extra heat and water internally generated in the fuel cells.

Table 4-2. Payload Assignments

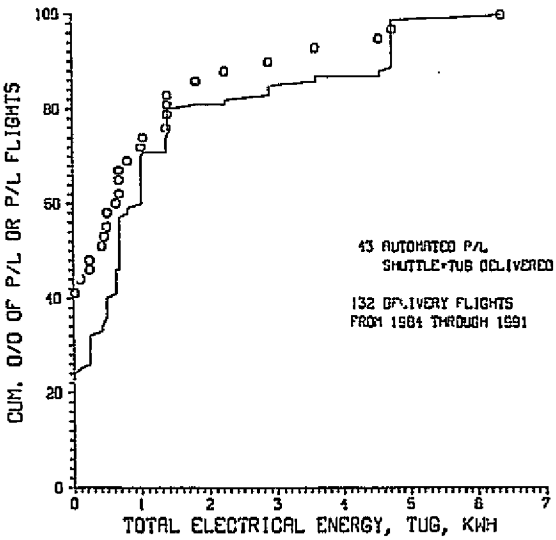
IUS ONLY			TUG ONLY			IUS AND TUG		
SINGLE	MULTIPLE	KICKSTAGE	SINGLE	MULTIPLE	KICKSTAGE	SINGLE	MUTIPLE	KICKSTAGE
OP-01	CN-52	PL-07	AS-16-	EO-07	AP-06	AP-02	AP-01	
PL-22		PL-11	AS-16-SM	EO-59	AP-08	AP-03	AP-02	
		PL-12	AP-07	EO-62	PL-01	AP-05	AS-05	
		PL-18	PL-02	CN-53	PL-09	AS-02	EO-09	
			PL-08	CN-59	PL-13	EO-09	EO-12	
			PL-10	CN-60		EO-12	EO-56	
			PL-14			EO-56	EO-57	
			PL-15			OP-06	EO-58	
			PL-16			PL-03	OP-06	
			PL-19				CN-51-	
			PL-20				CN-54-	
			LU-01				CN-55	
			LU-02				CN-56	
			LU-03				CN-58	
			LU-04					



a. DC Power, Inflight, Average



b. DC Power, Inflight, Peak



c. Total Electrical Energy

ORIGINAL PAGE IS
OF POOR QUALITY

Figure 4-2. Cumulative Summary of Payload/Tug Interface Requirements

Table 4-3. Tug Electrical Power Requirements for Payload Accommodation

Services Required	Single Payload	Multiple Payloads	Tug Impacts
Power (Avg kW)			Average power has no impact.
Post-deployment/ pre-separation	0.7	1.15	Peak power may require payload batteries as peaking supply
Transfer	0.2	0.2	
Power (Peak kW)	2.4	3.4	
Energy (kW-hr)	11.8	12.7	Additional reactants are required for payload needs

4.3 POWER QUALITY EFFECTS

Quality of electrical power systems is normally measured in terms of the sensitivity to anticipated power levels, peak loads, and surges. Largest response demands placed on power systems frequently produce voltage transients. In this respect, fuel cell power plants are high quality since they have no appreciable undershoot of voltage upon application of load, or overshoot upon removal of load. This results from the rapid internal response of a fuel cell source, and the large equivalent capacitance that reduces the sensitivity to load induced ripple. (See Figure 4-3.)

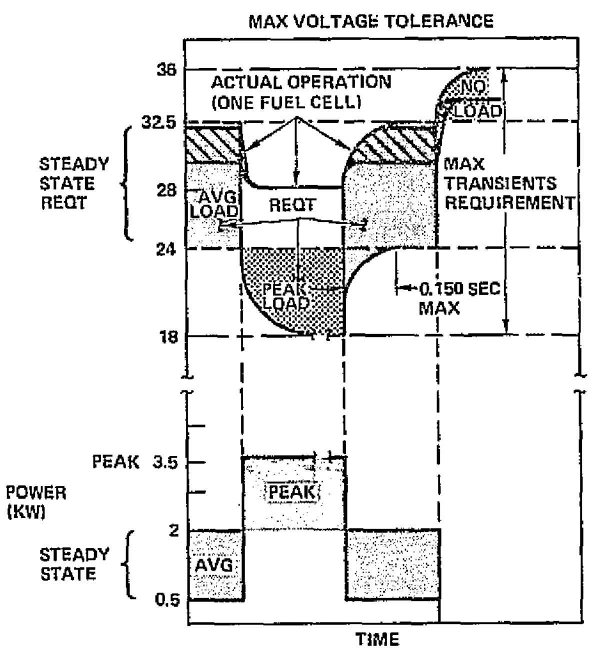


Figure 4-3. Space Tug Power Quality Requirements

Fuel cell power rating within a specified quality band is achieved for the particular type of cell structure by the matching of the number of cells/area within a stack. Allowable current density is a function of design and construction and is most directly influenced by the ability to remove heat from the cell and the venting of the exhaust products. Rating is therefore measured in terms of voltage drop against load. Figure 4-4 shows the effects of external parameters on the power plant voltage and quality.

Battery ratings are similar in derivation, and are dependent on the internal cell construction. Heat removal is still the dominant factor, where the amount of heat generated is a function of internal resistance and the load currents. The well known silver oxide/zinc battery type is manufactured

Power
Utility

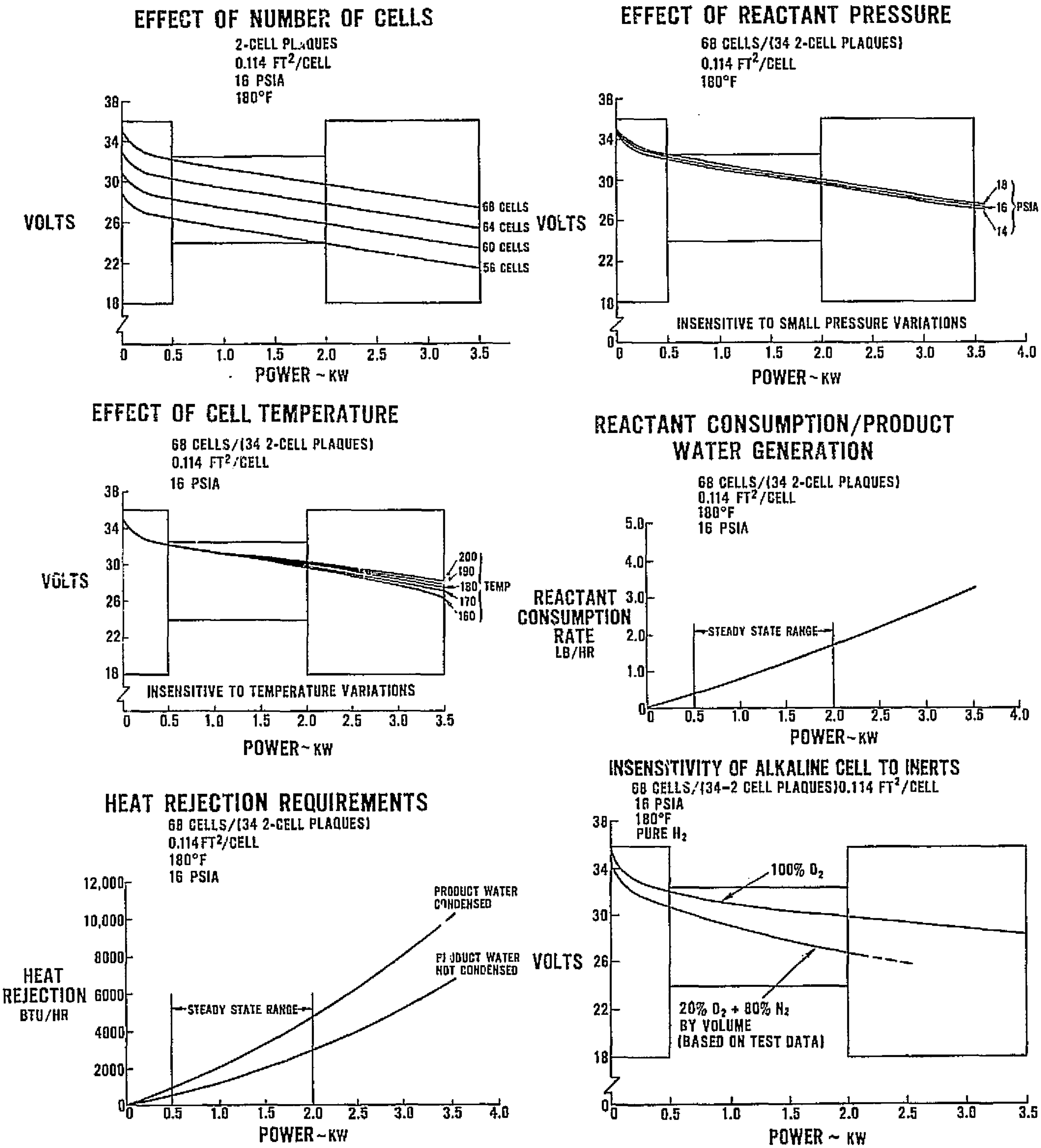


Figure 4-4. Effects of External Parameters on Power Plant Voltage and Quality

in several types with different capacities per pound. The heavier duty type used on the Centaur upper stage is typically 1.8 ampere hours/pound (3.96 ampere hours/kilogram), whereas lighter weight designs for extended duration orbit spacecraft realize 3 ampere hours/pound (6.6 ampere hours/kilogram). Compared to fuel cells, batteries do not generally conduct internally generated heat out of the battery housing as easily as is possible with the outflow of exhaust products in a fuel cell.

4.4 POWER PLANT OPTIONS SYNTHESIS

The initial baseline power plant was described in the NASA Tug document MSFC 68M00039-2 as a configuration utilizing a modified Orbiter fuel cell with dedicated supercritical storage of the hydrogen and oxygen reactants. Additional options were synthesized for the trade studies and consist of alternative methods of integration into the vehicle, optional peripherals, and newer "lightweight" cell structures. An early standard of comparison was the simple possibility of using a number of batteries onboard, which for the mission durations under consideration (mission 1, 164 hours) proved to be prohibitive in weight performance for the amount of energy needed for Tug and payload. The power plant trade then rapidly settled down to the detailed evaluation of the integrated fuel cell options and the selection of the design best suited to Tug requirements.

4.4.1 BATTERIES VERSUS FUEL CELLS. Previous Tug studies at Convair, the Space Tug Systems Study (STSS), NAS 8-29676, and the Reusable Centaur Study, NAS 8-30290, examined the available battery types suitable for upper stage use for vehicles very similar to the present Tug concept. Conclusions of those studies were that (for roughly the same power requirements and mission durations) while the electrical power could be made available from batteries, the payload performance penalties were on the order of several hundreds of pounds. Generally, when the mission flight times outside the Orbiter start to exceed 24 hours, a more efficient power plant such as fuel cells is needed in lieu of batteries.

Using the baseline power level requirements by mission phase as listed in Table 4-1 and the detailed Convair mission timeline for the Geosynchronous Placement and Retrieval mission (NASA mission 1), the kilowatt hour energy requirements for each major phase were computed (see Table 4-4). This shows a total requirement of 215 kW-hr for the nominal 164 hour mission; at 28 volts, this is a requirement of 7679 ampere hours. Based on the use of lightweight silver oxide/zinc spacecraft design batteries (400 A-hr, 134 pounds (61.1 kg) each battery, this would require approximately 19 batteries, which represents 2546 pounds (1154 kg) dry weight of batteries. An actual Tug design tailored for battery operation would necessitate revising the phasing requirements of Tables 4-1 and 4-4; however, even 50% of the projected power requirement would need 1273 pounds (577 kg) of batteries.

Fuel cells in the recommended lightweight thermally integrated configuration for the updated baseline Tug, including the reactants, 10% reserve, emergency battery weights,

Table 4-4. Energy Requirements by Mission Phase-Geosynchronous Placement and Retrieval Mission (164 hours)

Mission Phase	Duration (hours)	Power Level (watts)	Energy (kW-hr)	Current (amperes)
Ascent	10.8	1145	12.4	40.9
Predeploy C/O	1.0	1938	1.9	69.2
Deploy Tug	1.0	1822	1.8	65.1
On orbit PL C/O	1.0	1743	1.7	62.3
Coast	133.7	1313	175.6	46.9
Eng. Burn	1.7	1690	2.9	60.4
Guid. Update	7.0	1374	9.6	49.1
Rend. & Dock	5.5	1329	7.3	47.5
Norm. Retrieval	0.6	1149	0.7	41.0
Descent	1.4	738	1.0	26.4
		Total Energy	214.9 kW-hr	

and the added peripherals, would weigh 412 pounds (185 kg), and this represents a savings of 2134 pounds (968 kg) over the brute force battery solution.

4.4.2 FUEL CELL OPTIONS. At the start of the study, the initial modified Orbiter fuel cell with single string, supercritical reactant storage appeared as the least expensive power source to develop. A secondary alternate was the lightweight fuel cell under technology development by NASA Lewis Research Center and the Air Force. In all, three subconfigurations of each type were considered (Figure 4-5).

The two extremes were quickly eliminated as not being cost effective. A minimum modification Orbiter unit weighed 16 pounds (7.2 kilograms) more with greater technical risk. The lightweight technology design incorporating passive fluidics controls reduced the weight by only 5 pounds (2.3 kilograms) and had high development risk cost in the 3 to 5 million dollar category.

With the two remaining lightweight options, the cost difference between adapting existing fluid controls and developing specific units for the Tug was indeterminate. The largest gain was from waste heat/passive water removal concept and with thermal integration of the lightweight fuel cell. The next task was to define the two baselines in depth before expanding to other trade studies (Figure 4-6).

An alternative configuration that evolved was the modifying of the Orbiter fuel cell for low pressure operation similar to the lightweight unit. This would eliminate the need for supercritical storage to supply 60 psi (414 kN/m^2) to the fuel cell stack, while permitting retention of the Orbiter configuration. The result is a heavier, but a low risk/low cost compromise. Significant benefits for safety and interface simplicity occur

FUEL CELL OPTIONS	REACTANT STORAGE: SUPERCRITICAL	REACTANT STORAGE: MAIN TANKS & ZERO-G ACQUISITION SCREEN	REACTANT STORAGE: MAIN TANKS WITH He CONTAMINATION	WASTE HEAT REJECTION: SPACE RADIATORS	WASTE HEAT REJECTION: THERMALLY INTEGRATED	WASTE HEAT REJECTION: H_2 COOLED	PRODUCT WATER: TOTAL STORAGE	PRODUCT WATER: ZERO-G STORAGE & EJECT DURING FIRING	PRODUCT WATER: CONTINUOUS EJECTION	
ORBITER UNIT (MINIMUM MOD)										DROPPED
ORBITER STACK & RESIZED FLUID CONTROLS	X			X		X		X	DOD PROHIBITS	60 PSI ORBITER & SUPERCRITICAL
ORBITER STACK & LOW PRESSURE OPERATION *			X	X		X		X	DOD PROHIBITS	LOW PRESS ORBITER
LIGHTWEIGHT TECHNOLOGY (EXISTING FLUID CONTROLS)								X	DOD PROHIBITS	DROPPED
LIGHTWEIGHT TECHNOLOGY (RESIZED FLUID CONTROLS)			X	X	X	X		X		THERMALLY INTEGRATED LIGHTWEIGHT
LIGHTWEIGHT TECHNOLOGY (PASSIVE FLUIDS CONTROLS)								X		DROPPED

PERIPHERAL EQ. OPTIONS

NOTES:
1. EACH OPTION MUST BE RESIZED TO MATCH FUEL CELL OPTION SELECTION.
2. TOTAL SYSTEM WEIGHTS AND Δ COST AGAINST BASELINE SYSTEMS WILL BE USED. (EXAMPLE – WHEN OTHER SYSTEMS USE THE FUEL CELL WASTE HEAT, THE WEIGHT AND DEV. $ OF ELECTRICAL CONTROLS WILL BE SUBTRACTED FOR THAT SYSTEM.

Figure 4-5. Potential Power System Options

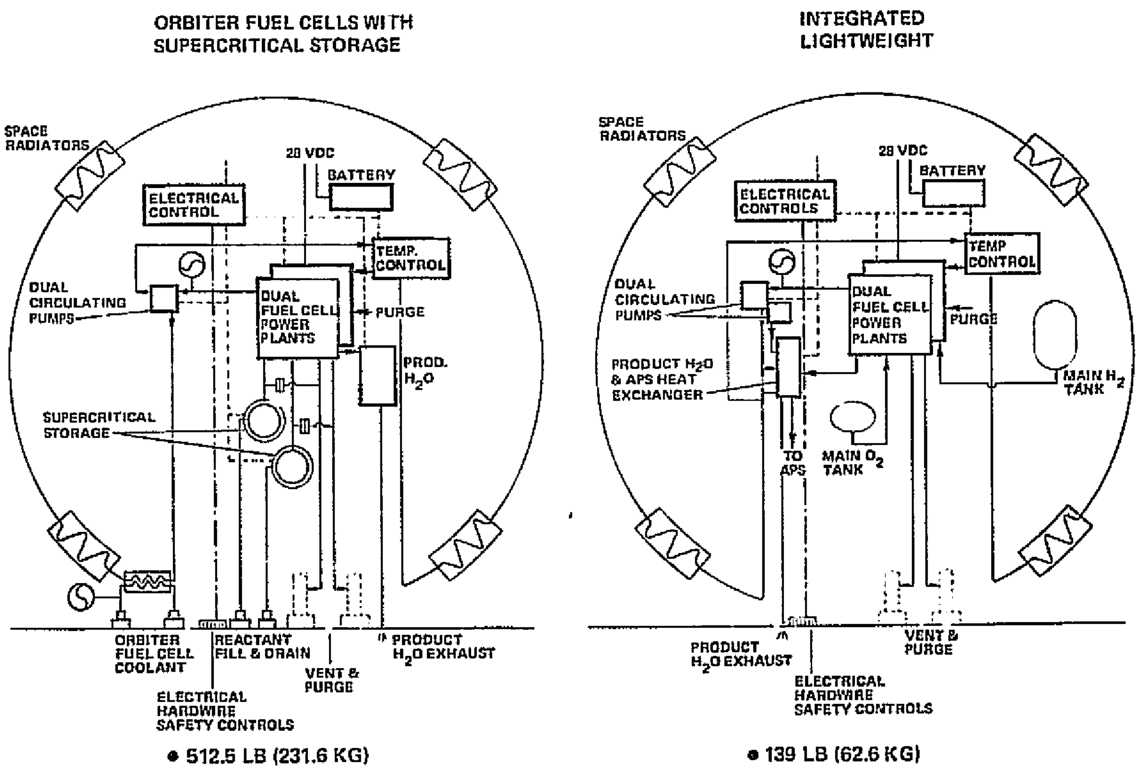


Figure 4-6. Power Source Options Selected for Analysis

when the fuel cell is operated low pressure at about 16 psi (110.4 kN/m^2), with direct extraction of reactants from the main propellant tanks. All reactant management — fill, drain, pressure control, venting, and safing — is done by the Tug main propellant system. In effect, the fuel cell gets the reactant management system free.

A further simplification is possible with lightweight system thermal integration for the Tug. The lightweight unit normally exhausts steam compared to the Orbiter type, which condenses the steam into usable water for the Orbiter crew. By using the APS heating requirements to condense the steam, thermostatic electrical APS heaters are also eliminated. Smaller space radiators are needed and electrical energy that would be required by the APS heating system is eliminated.

Fuel cell options narrow down to three basic alternatives:

a. A minimum modification of the 60 psi (414 kN/m^2) Orbiter unit with supporting dedicated supercritical storage.

b. A low pressure Orbiter unit modification that will extract reactants directly from the main propellant tanks.

c. A thermally integrated lightweight fuel cell specifically designed to meet Tug objectives.

4.4.2.1 Minimum Modification Orbiter Type. Supercritical storage is required for the minimum modified, 60 psi (414 kN/m^2), fuel cell power plant (Figure 4-7). The

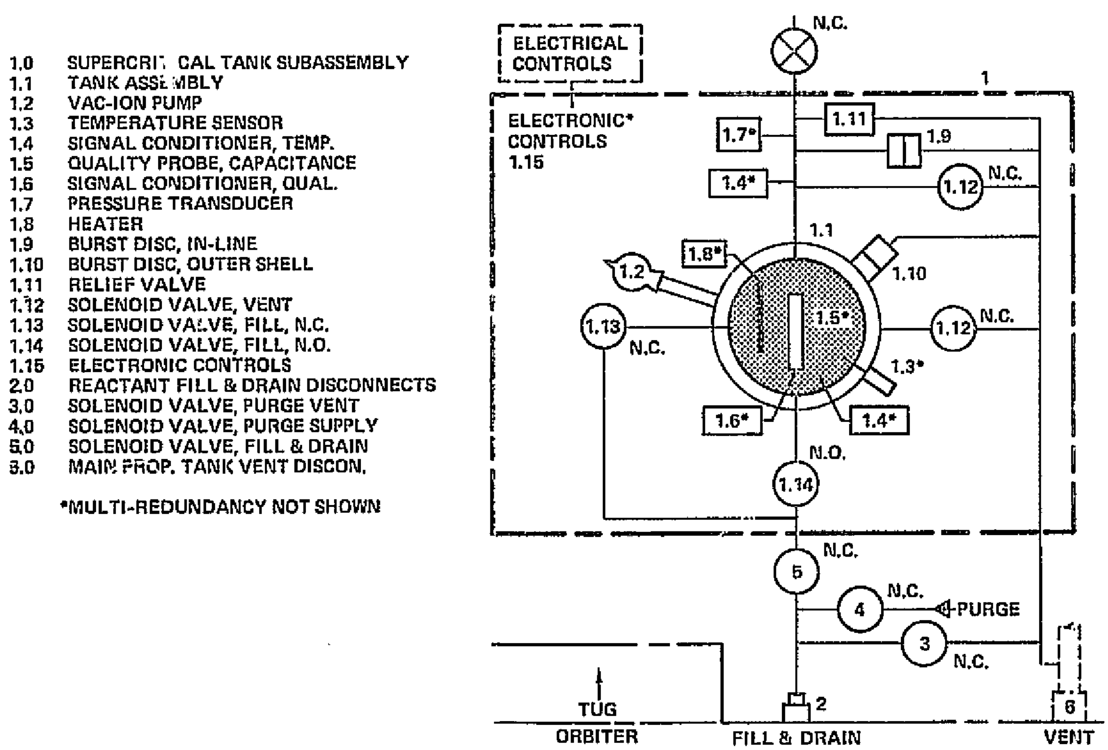


Figure 4-7. Baseline Supercritical Storage (Typical Each Reactant)

reactants are loaded in liquid form through separate fill and drain disconnects from the ground through the Orbiter into the Tug. Operating at 60 psi (414 kN/m^2) requires that the system be loaded to this pressure level and superinsulated for minimum boil-off. Where the Orbiter has redundant reactant storage, the Tug would be single string and must be compensated by ultra-high reliable control, sensing, and DMS redundancy management. Supercritical storage presents a dedicated kW-hr limit for long missions. Heavy tanks sized for the maximum must also be carried for shorter missions (Figure 4-8).

All sensors and controls must be triple redundant and DMS controlled. The added weight of safety/redundancy management to meet the Tug reliability requirement of 0.9991 or better has been added to each system weight comparison.

4.4.2.2 Low Pressure Modification — Orbiter Type. This alternative was not in the original baseline system options. Conceptually, this is a composite system with less capability at lower technology risk and has therefore been recommended as a backup option. Existing Orbiter stack technology is used and the number of cells in the stack are reduced for the 2 kW power rating at 16 psi (110.4 kN/m^2). The thermal control

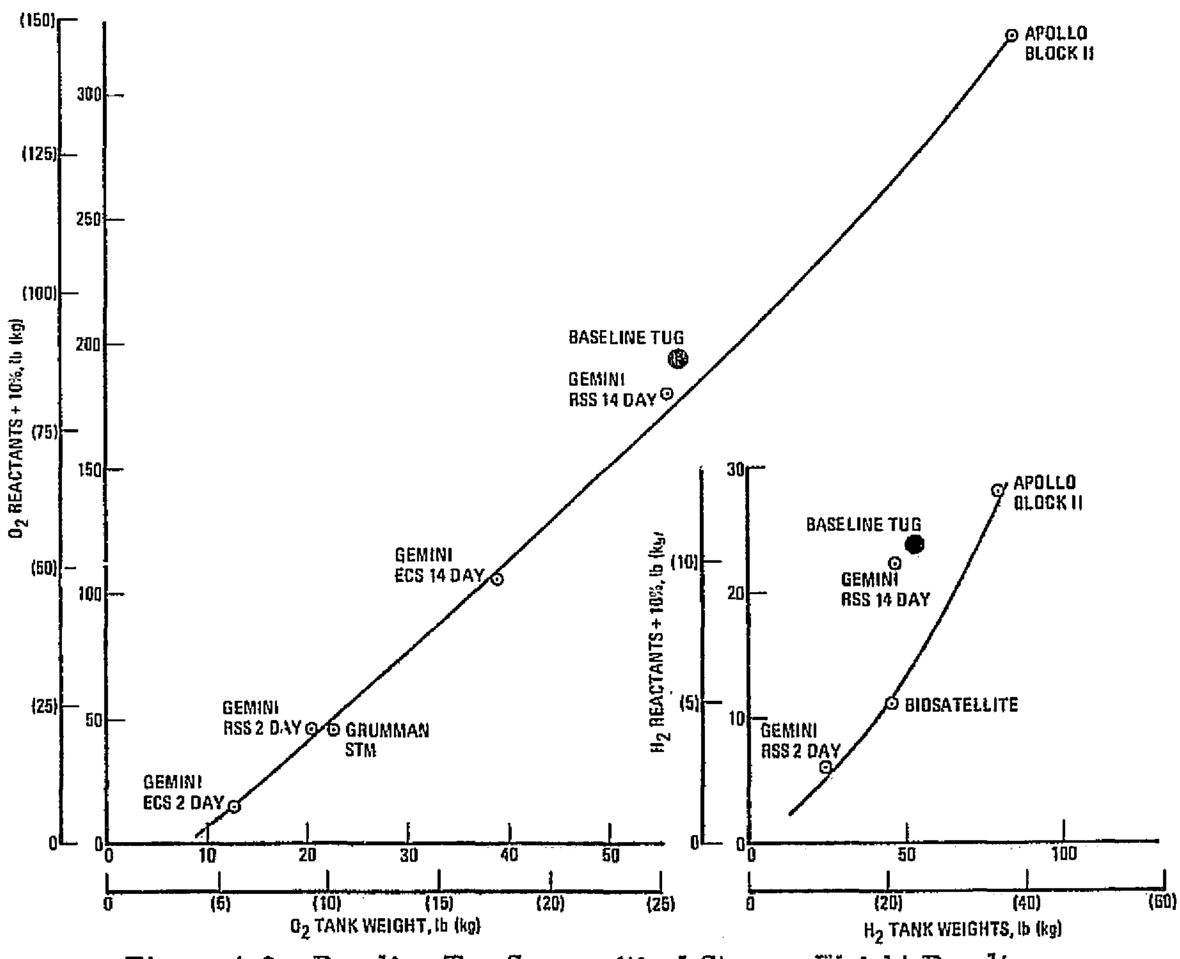


Figure 4-8. Baseline Tug Supercritical Storage Weight Penalty

system is identical to the high pressure unit (Figure 4-9). New qualification/reliability data at the lower tank feed pressures is needed and represents an additional power plant development task.

4.4.2.3 Thermally Integrated Lightweight Type. The lightweight concept (Figure 4-10) adapts well to Tug requirements, with low pressure 16 psi (110.4 kN/m^2) operation for direct reactant feed from the main propellant tanks and product water exhausted as 4 psi (27.6 kN/m^2) steam. A dedicated reactant management is eliminated through the use of the main propellant system.

Waste heat from the steam can now be utilized to heat the auxiliary propulsion system hydrazine fluid to the best operational condition, 93 ±13°F (307 ±7°K). Thermal integrating serves several purposes:

a. Provides circulating APS fluid for uniform temperature control.

b. Eliminates APS line and bottle heaters and multiple thermal controls.

c. Reduces criticality of APS insulation effectiveness.

d. Reduces electrical load, providing more power reserve for growth.

e. Reduces space radiator size and weight.

f. Increases thermal limited peak power times.

g. Permits waste heat absorption during ascent or abort.

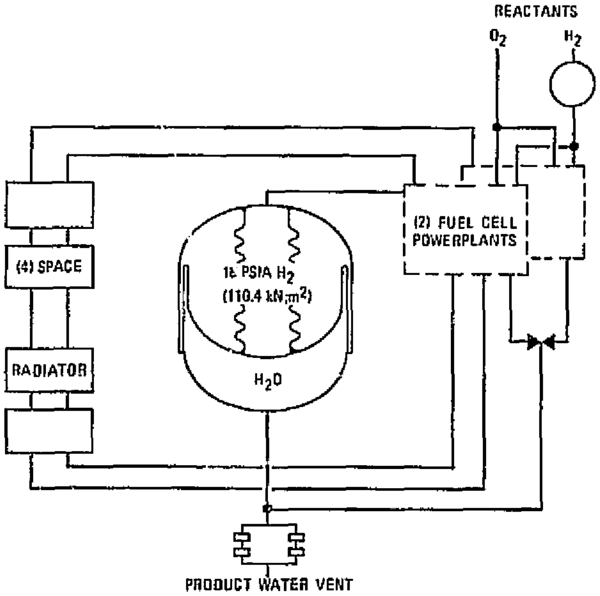


Figure 4-9. Low Pressure Modification – Orbiter Type – Peripheral Equipment (with Redundancy to Meet 185-hour Reliability (> .9991)

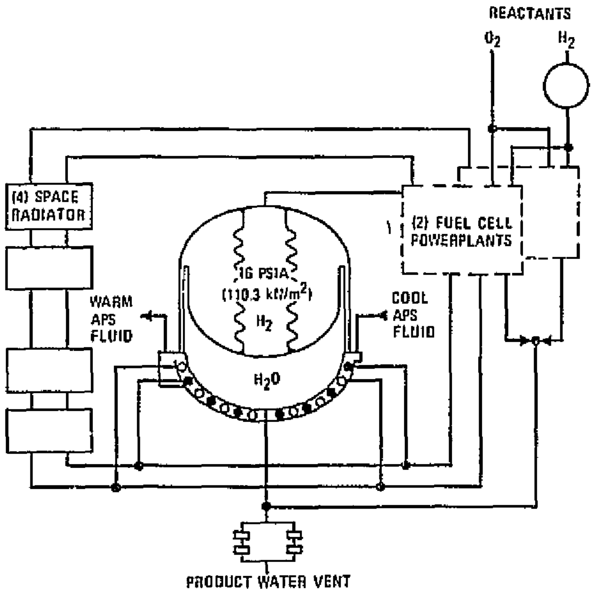


Figure 4-10. Thermally Integrated Lightweight Type Peripheral Equipment (with Redundancy to Meet 185-hour Reliability (> .9991)

4.4.3 PERIPHERAL EQUIPMENT COMPARISON. Peripheral equipments are unique for each power plant option. Major differences in the power plants dictate different peripheral equipment solutions, which are driven by three primary conditions:

a. Single string supercritical storage and its propellant management versus direct reactant acquisition.

b. Redundancy solutions to achieve system fail operational – fail safe/high reliability over a possible 185 hour mission time.

These differences are depicted in the respective reliability block diagrams, Figures 4-11 and 4-12. The lightweight and low pressure power plant system are functionally similar, but differ in where those functions are accomplished (Figure 4-11). Added safety and redundant hardware to meet the 0.9991 system reliability for 185 hours is shown by heavy outlined boxes and summarized later in Table 4-6 on page 4-20. Extensive system cross strapping is utilized to enable meeting the requirements.

4.4.3.1 Reactant Storage. Supercritical storage analysis highlighted these Tug vehicle concerns:

a. Single string storage/man rating, dedicated multiple vent, purge, and safing capability.

b. Incorporating safety redundancy lowered system reliability — often dictating quad valving to achieve both safety and reliability.

c. Sensing and electrical controls were deemed triple redundant with voting selection/control commands.

d. Dedicated ground fill and drain system was not included in study cost trades.

e. System reconfigurating control complexity with different solutions during different operating modes resulted in main computer control.

Direct acquisition from the main propellant tanks highlighted these vehicle concerns:

a. Main tanks propellant management system solved most safety oriented reactant problems.

b. Tug retrieval electrical power, after the main tanks are vented, created a need for small dedicated reactant supplies.

c. Main tank propellants may contain a variable amount of He inerts — depending on which engine/tank pressurization means is finally selected. Fuel cell power plant may require voltage droop controlled purging.

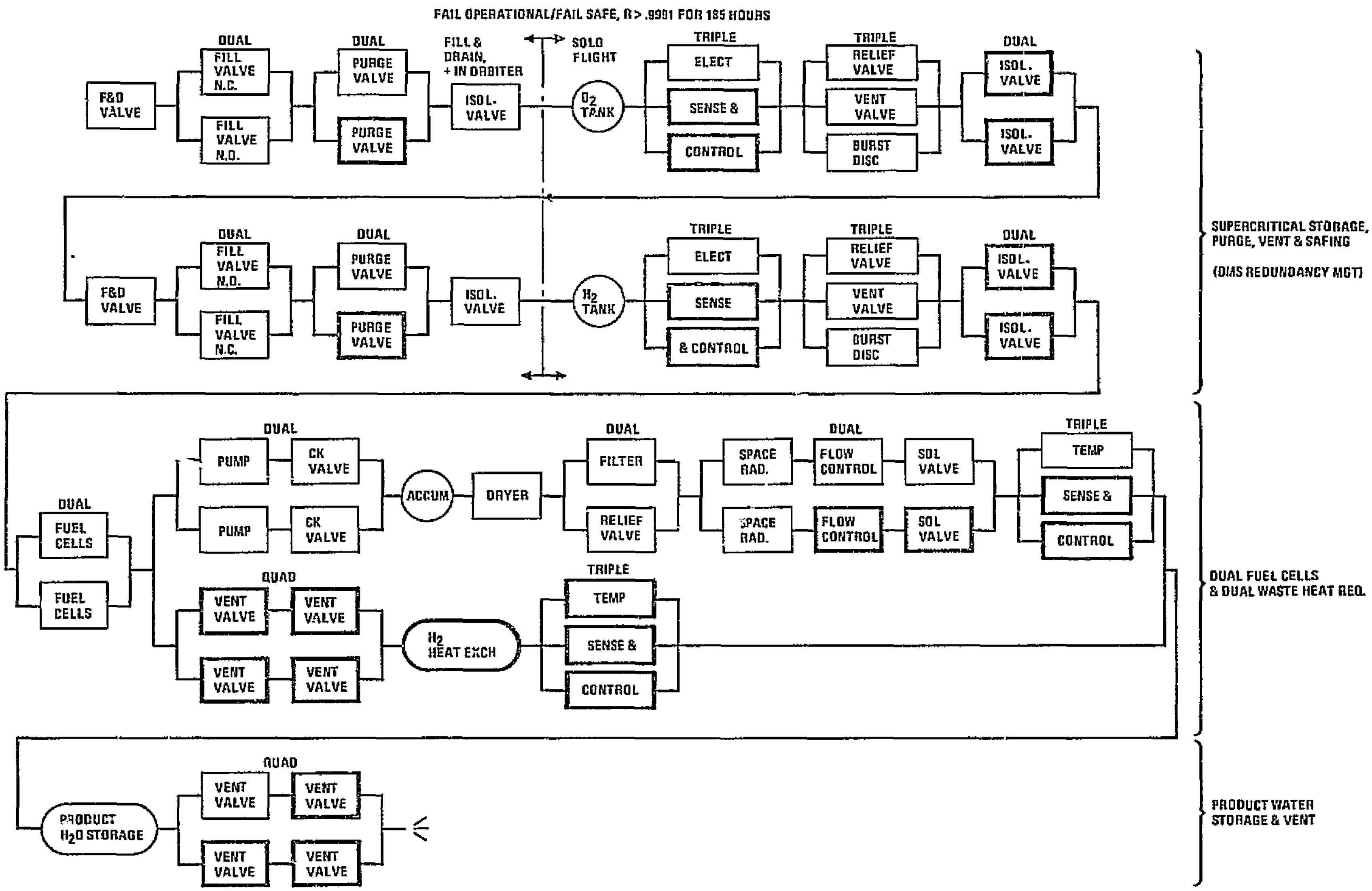


Figure 4-11. Modified Orbiter and Supercritical Storage – Redundancy Management

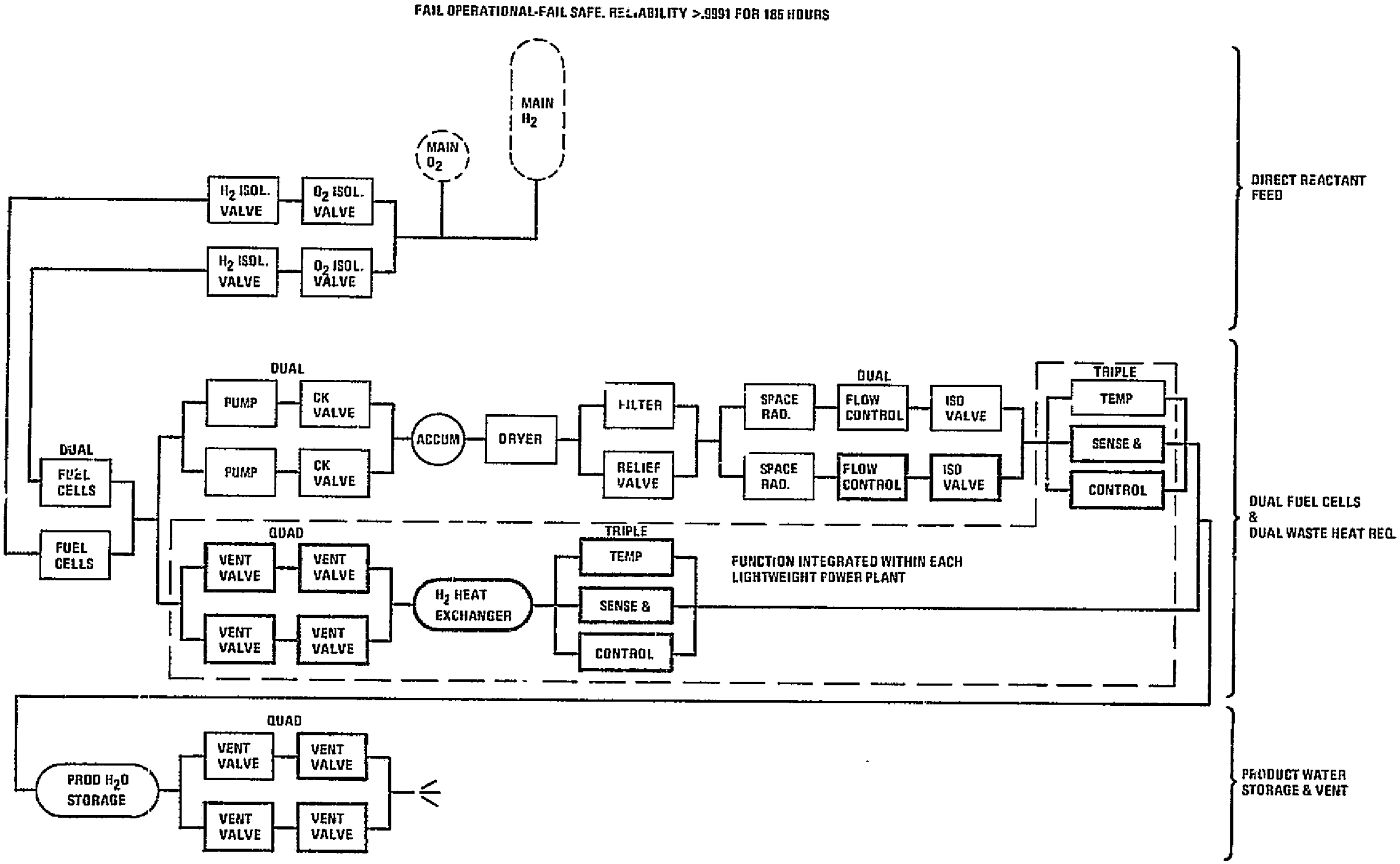


Figure 4-12. Low Pressure Fuel Cell Redundancy Management (Modified Orbiter or Lightweight)

4.4.3.2 Redundancy Management. The redundancy management requirements highlighted the impacts of crew safety — fail operational and high reliability over the long mission.

a. Fail operational/Orbiter crew safety while the Tug is in or near the Orbiter, dictates at least dual functional hardware solutions.

b. High reliability, over long missions, required cross strapping between peripheral equipment and autonomous system reconfiguration.

c. Redundancy solutions drive peripheral equipment design, which ii turn impacts power plants design.

d. Storage of product water during coast is of questionable value.

e. Tug performance is heavily affected by added redundancy management hardware weights.

f. Highest system risk associated with these redundancy management/autonomous reconfiguration — fail operational solutions.

To keep modifications to the existing Orbiter power plants to a minimum, all redundancy management solutions are achieved as external "add ons"; whereas, the new lightweight power plant can integrate the Tug's requirements into the initial design.

4.4.3.3 Thermal Control and Integration. The thermal control systems applicable to the two principal options are illustrated in Figures 4-13 and 4-14, which are the high pressure modified Orbiter power plant and the low pressure thermally integrated power plant respectively. Redundancy is not shown, but the relative "single string" complexities of the two types of thermal control requirement are reflected.

Integration of the thermal system for the fuel cells into the vehicle allows significant improvements in efficiency. Perhaps the most important element is the heating of the APS hydrazine fluid, which must be maintained at 93 ±13°F (307 ±7°K), and where the hydrazine supply bottles and interconnecting lines are located between the two cryogenic main propellant tanks. Equalization of the sun/deep space heating is not possible through simple Tug rotation (rotisserie) where the Tug is payload attitude constrained. A normal solution would utilize thermistatically controlled line heaters and electric blankets. Another alternative is the circulation of fluid through the lines with temperature control at a central location, similar to the hydraulic thrust vector control system. Both of these solutions require electrical heaters and power during the entire mission frame. The additional electrical power needed, in turn, produces more waste heat to be rejected.

By integrating and using the fuel cell waste heat to maintain the circulating hydrazine fluid within the desired operating temperature limits, the controlled electric heaters can be eliminated and, alternatively, the necessity to rotate the Tug for thermal equalization avoided. Interconnecting the power plant waste heat rejection with the heating of the attitude control system also provides a heat sink when the Tug is in the Orbiter during ascent and/or abort.

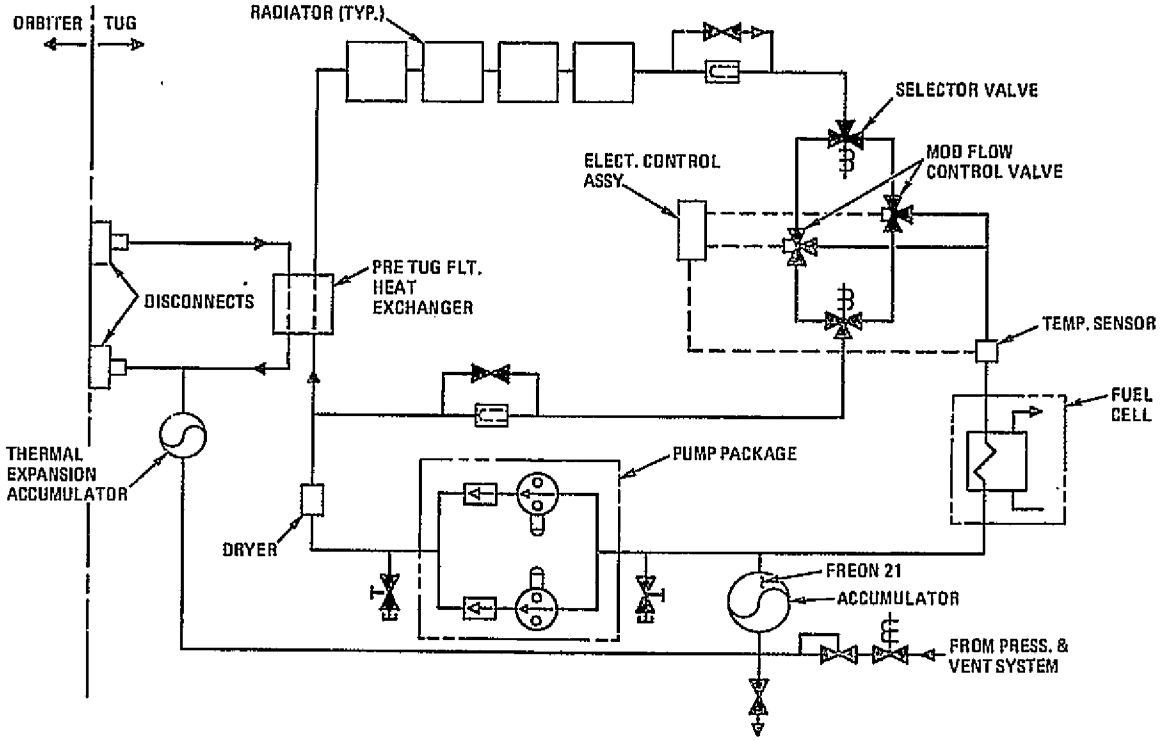


Figure 4-13. Orbiter Technology Fuel Cell Baseline Thermal Control

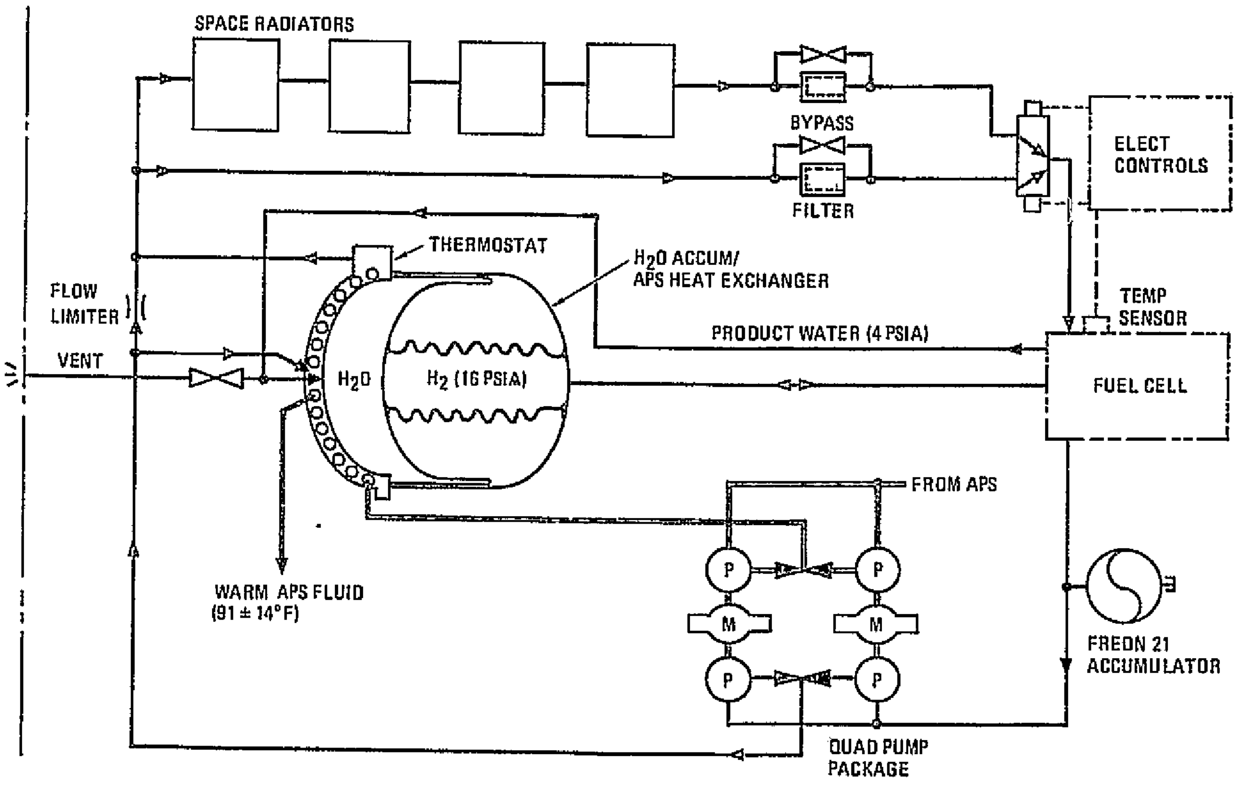


Figure 4-14. 1978-1980 Lightweight Fuel Cell Baseline Thermal Control

4.5 POWER SYSTEM OPTION TRADES

Specific comparisons of the options were performed in several supporting trade analyses. These looked at weight performance, reliability, DDT&E costs, interface sensitivities, growth potential, and the recommended selection for the electrical power system baseline.

4.5.1 WEIGHT COMPARISON. In the summary descriptions of the fuel cell power plants, the weight penalty associated with supercritical reactant storage was examined. For a nominal 2 kW power plant design, the three principal options were further analyzed considering peripherals, the fuel cells, and the redundancy additions. The detailed weight comparison is shown in Table 4-5. Reactants have not been included as they are essentially the same for all power plants at the 2 kW output level. Significant weight performance is achieved with the integrated lightweight design; a minimum of 200 pounds (90.7 kg) saving over the total system required for the low pressure Orbiter option which is the leading contender.

4.5.2 RELIABILITY/REDUNDANCY. All three options were subjected to safety analysis before the reliability requirements were finalized. The Convair safety and hazards/failure analyses are reported in the final report for the parallel Tug study, "Space Tug/Shuttle Interface Compatibility," Contract NAS 8-31012. In this overview, details of only two options are discussed, as the difference between the two low pressure cases (modified Orbiter or lightweight) are minor. For the power system single WBS element, the MSFC 68M00039-2 reliability goal of 0.9998 for the power system was revised to reflect feed, fill and drain, APS, and fuel cell thermal conditioning. The revised reliability apportionment is 0.9991 for the 185 hour mission (see Fig-4-15). This reliability value is equivalent to 23-1/2 years of power generation between power failures. Achievement of this level of fail-operational reliability requires instantaneous failure switching and continuous power production. Table 4-6 shows the reliability requirements for meeting the 0.991 goal.

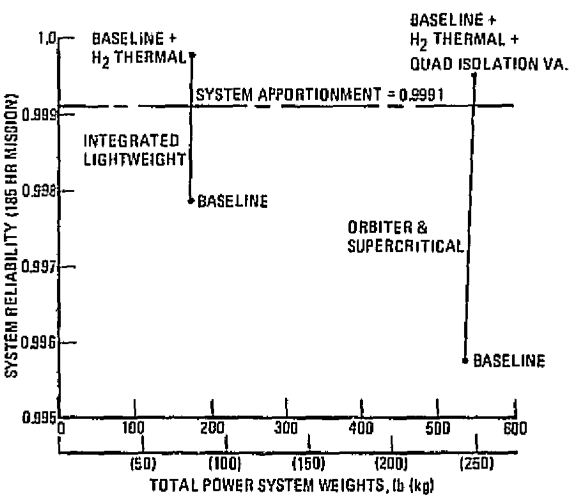


Figure 4-15. Reliability Apportionment

Additional redundancy management considerations took into account integration of the autonomous, DMS, and shared sensor and control options. Technical and cost elements examined were:

Table 4-5. Baseline Tug Power System Options (pounds)

Item	System Flight Weights (2 kW)		
	Orbiter and Supercritical	Low Pressure Orbiter	Integrated Lightweight
1. Fuel Cell Power Plants with Optimized Components	230	230	72.6
2. Reactant Supply	134	11	8
2.1 O_2 Supercritical Tanks	56	-	-
2.2 H_2 Supercritical Tanks	52	-	-
2.3 Controls and Feed Lines	26	11	8
3. Waste Heat Rejection	77.6	77.6	30
3.1 Space Radiators	35.4	35.4	27
3.2 Thermal Control Dist.	12	12	-
3.3 Pump/Dryer/Accum.	7	7	7
3.4 Preflight Heat Exch.	8	8	-
3.5 APS Circu. Pumps	-	-	4
3.6 APS Heaters Removed	-	-	-22
3.7 Freon 21/FC-40	15.2	15.2	14
4. Product Water/APS Heat Exch.	10.5	10.5	18.5
4.1 Accum/APS Heat Exch.	9	9	17
4.2 Dump Valves	1.5	1.5	1.5
5. Purge, Vent and Safing	4.8	3.0	3.0
6. Elec. Controls/Instru.	12.0	8.0	4.0
Totals	468.9	340.1	136.1
+Redundancy to Meet 0.9991 Reliability	+12.0	+8.0	+3.0
System Totals (minus emergency battery)	480.9	348.1	139.0

Table 4-6. Redundancy Required to Meet Reliability Goal ($R \geq 0.9991$ for 185-hour Mission)

POWER SYSTEM COMPONENTS	MODIFIED-SHUTTLE FUEL CELL	LIGHTWEIGHT FUEL CELL
REACTANT SUPPLY AND FEED	SINGLE SUPERCRITICAL TANKS DUAL ISOLATION VALVES TRIPLE SENSORS DMS REDUNDANCY MANAGEMENT	MAIN PROPELLANT TANKS DUAL FEED LINES
PURGE & VENT	DUAL PURGE VALVES FOR SUPERCRITICAL TANKS	NOT APPLICABLE (REACTANTS FROM MAIN PROPELLANT TANKS)
FUEL CELLS	DUAL FUEL CELL STACKS	DUAL FUEL CELL STACKS
WASTE HEAT	DUAL COOLANT LOOPS IN RADIATORS TRIPLE SENSORS FOR CONTROL H_2 HEAT EXCHANGER (BACKUP)	DUAL COOLANT LOOPS IN RADIATORS TRIPLE SENSORS FOR CONTROL H_2 HEAT EXCHANGER (BACKUP)
PRODUCT WATER	SINGLE STORAGE BOTTLE QUAD VENT VALVES TRIPLE SENSORS	SINGLE STORAGE BOTTLE QUAD VENT VALVES TRIPLE SENSORS
	• TEMP, PRESS SENSORS FOR CONTROL • 14 VALVES ADDED FOR SAFETY/ RELIABILITY • DMS MANAGEMENT	• TEMP SENSORS ONLY FOR CONTROL • 4 VALVES ADDED FOR SAFETY/ RELIABILITY • SELF CHECKING

Technical Elements Examined	Cost Elements Examined
Reaction Time	DMS Memory
DMS Emergency Interrupt	DMS Software
Multi Emergency Priority	Hardware
External/Internal Caused Problem	C/O Time/Crew/AGE
Weight/Reliability	Simulation/Demonstration
Maintenance Record	Qualification
Transition/Recovery	Vehicle System Test
Emergency Operations	Orbiter Integration
Orbiter Crew Override	
Orbiter/Ground Decision Knowledge	

4.5.3 DUAL FUEL CELLS. Implementation for all three options in a dual configuration can be accomplished in an optimal manner by simultaneous and parallel operation into the same load. Fuel cells are self compensating between each other, so that simultaneous operation can eliminate the high inrush current of switching a backup power plant into a demand circuit after the primary unit has been cut out. In the case of a shorted power plant, that fuel cell can be cut out before the currents reach the kiloampere range, thereby eliminating the power dead band or droop that would normally occur with switching in the backup. The inherent ability of the power plants to share the load avoids the more complex problem of keeping a backup and peripheral equipment warmed up for instant switch-in. To keep the peripheral equipment in a standby mode generated a second and unique idle mode. Peculiar sensing hardware would be needed just for the standby mode. The various problems of how to instantaneously activate redundant peripheral equipment were not addressed during this trade study.

Redundancy management for the modified Orbiter type power plant involves man-rated safety and venting redundancy for the supercritical storage. The complexity requires the use of DMS/data bus management as the lightest and most cost effective means to monitor and select the failure compensating cross-strapping route.

Both of the low pressure options receive their reactants directly from the main tanks. The main propellant tank management system performs the safing and venting of reactants. An exception occurs during Tug retrieval and abort when the main propellants are dumped, vented, and safed. Under these conditions, tank isolation valves in each feedline are closed, entrapping reactants within insulated feedline storage volumes. This permits an additional five hours of operation of the fuel cells. Internal regulation and vent controls are used during this situation for reactant control.

Dual waste heat rejection is required, even with the baseline dual circulating pumps and dual coolant loops. Structurally severe micrometeorite damage to any of the four space radiators would shut down both fuel cell power plants. A backup H_2 heat exchanger system, fed from each fuel cell reactant supply or H_2 zero-g exhaust, would cool the fuel cells in the event that a space radiator system were damaged. It needs 2 to 5 pounds per hour (0.9 to 2.3 kg per hour) of H_2 to function. Product water venting was made quadruple redundant to preclude inadvertent ventings and to assure venting during main engine burn. Lack of venting would shut down both fuel cell power plants.

4.5.4 INTERFACE SENSITIVITY. Figure 4-16 shows that the major difference in power plants is not in cell design but in the reactant storage/propellant management. Supercritical storage reactants are loaded and dumped separately from the main propellant tanks. Direct feeding of reactants from the main tanks eliminates separate GSE, Orbiter, and Tug fill and drain hardware, and additional management for supercritical storage. A significant simplification is elimination of supercritical safing and purge controls.

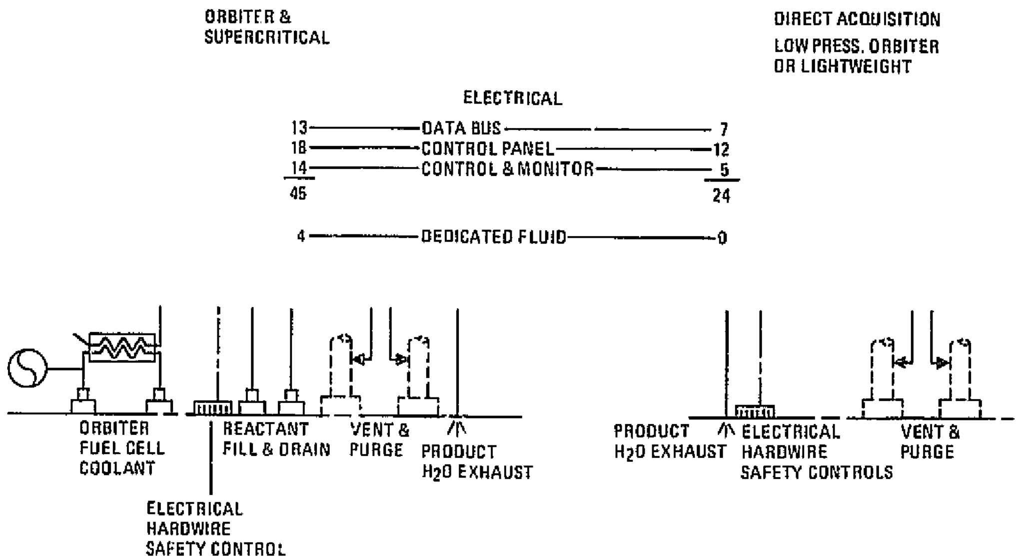


Figure 4-16. Interface Sensitivities

4.5.5 PERFORMANCE VERSUS DDT&E COST. Evaluation of the cost elements indicates that the least expensive development centers around using a minor modification to the Orbiter fuel cell power plant. The unique and troublesome feature of using the Orbiter design is that the lack of cryogenics on board the Orbiter means that reactants must be held in supercritical storage for the duration of the 30-day mission. Adapting the Orbiter fuel cells with the new peripherals needed for Tug is a costly process.

An alternative to the direct translation of the Orbiter high pressure system is to change the feed method of the fuel stack to a low pressure system taking reactants from the main propellant tanks of the Tug. Requalification of this power plant for low pressure operation increases cost, but the overall system development cost and the risk would be reduced by the avoidance of the supercritical storage. This low risk configuration does not meet Tug performance objectives because of heavy fuel cell stack weight and somewhat greater peripherals weight for the nominal 2 kW design (see Table 4-5).

Technical program risk for the integrated lightweight cell option can be reduced with an early simulation/demonstration program and with continued SRT activity that will progressively remove risk as the technology gradually evolves. Development cost elements for the three options analyzed as considered for various payload capability differences are shown in Figure 4-17. These values are for all the elements associated with the fuel cell power plants so as to realize a proper comparison. The normal WBS for cost estimating accounts from some of the thermal elements and peripherals outside of the Avionics System.

Total program costs are compared in Table 4-7. In looking at the total program cost effects, the option with Orbiter type cells and the supercritical storage of reactants shows to be the greatest cost. The other two options with the low pressure direct feed of propellants are lower in cost with different program cost virtues. The lightweight option has a higher DDT&E dollar requirement and a lower expected production cost; whereas, the modified Orbiter cell type costs less during development, but would involve greater production costs. From an overall cost standpoint, either of the low pressure versions are probably acceptable.

Performance and the growth flexibility of the lightweight cell design are superior. Also the costs seem to be slightly lower, so that both technical and cost factors favor the selection of the lightweight fuel cell power plant.

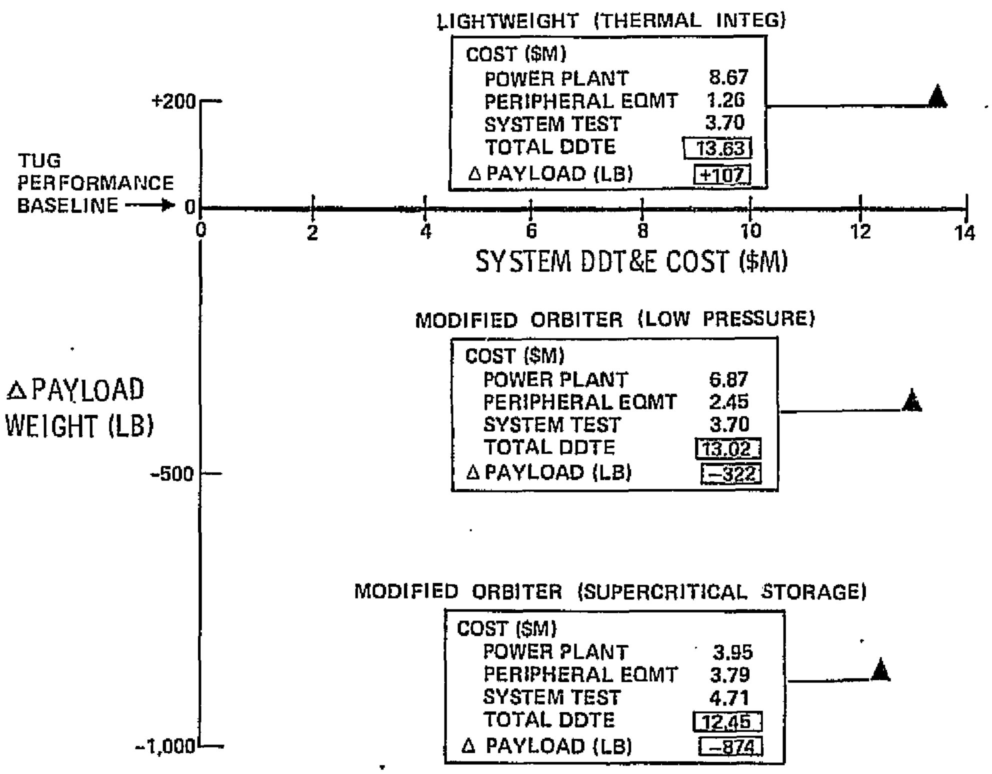


Figure 4-17. Power System Options Trade

Table 4-7. Summary Costs, Fuel Cell System Options Trade

	SYSTEM COSTS ($1,000)		
	LIGHTWEIGHT INTEGRATED	MOD ORBITER LOW PRESSURE	ORBITER SUPER-CRITICAL STORAGE
DDT&E	(13,631)	(13,022)	(12,445)
FUEL CELL	8,670	6,868	3,947
THERMAL/WATER	856	1,949	3,104
ELECT CONTROLS	405	505	684
SYSTEM TEST	3,700	3,700	4,710
PRODUCTION	(6,920)	(11,167)	(13,544)
TUGS (15)	5,531	9,206	11,391
INITIAL SPARES	1,389	1,961	2,153
OPERATIONS	(4,478)	(4,879)	(5,222)
FUEL CELL	4,082	3,961	3,888
OTHER	396	918	1,334
TOTAL PROGRAM	25,029	29,068	31,211

4.5.6 RECOMMENDED DEVELOPMENT PLAN. Development plans have been generated for the two viable options: the low pressure Orbiter type of power plant as a backup, and the thermally integrated lightweight cell design. Figure 4-18 shows a summary of these suggested plans.

Redundancy management design and development necessary for system reliability and operational safety are major cost drivers. They require a long time to evolve the operational system and prototype hardware must be available early in mid-1978 to permit meaningful system development. Some differences exist between the two alternative power plants that directly feed from the main propellant tanks. Modifying the Orbiter units requires some of the redundancy management provisions to be applied external to the power plant proper. With the lightweight type cell, much of the redundancy control can be incorporated within the fuel cell package that will be designed for Tug usage.

For purposes of the plan, the low pressure Orbiter unit was assumed to be a redirection of the current MSFC Tug fuel cell development program. Enough development plan detail was laid out to scope critical events and cost elements.

4.5.7 GROWTH POTENTIAL. Three factors predominate in the estimation of Tug power plant growth:

a. Future maximum payload requirements — including multiple spacecraft.

b. Primary electrical loads — flexibility to accommodate design changes.

c. Practical performance limitations of implementation.

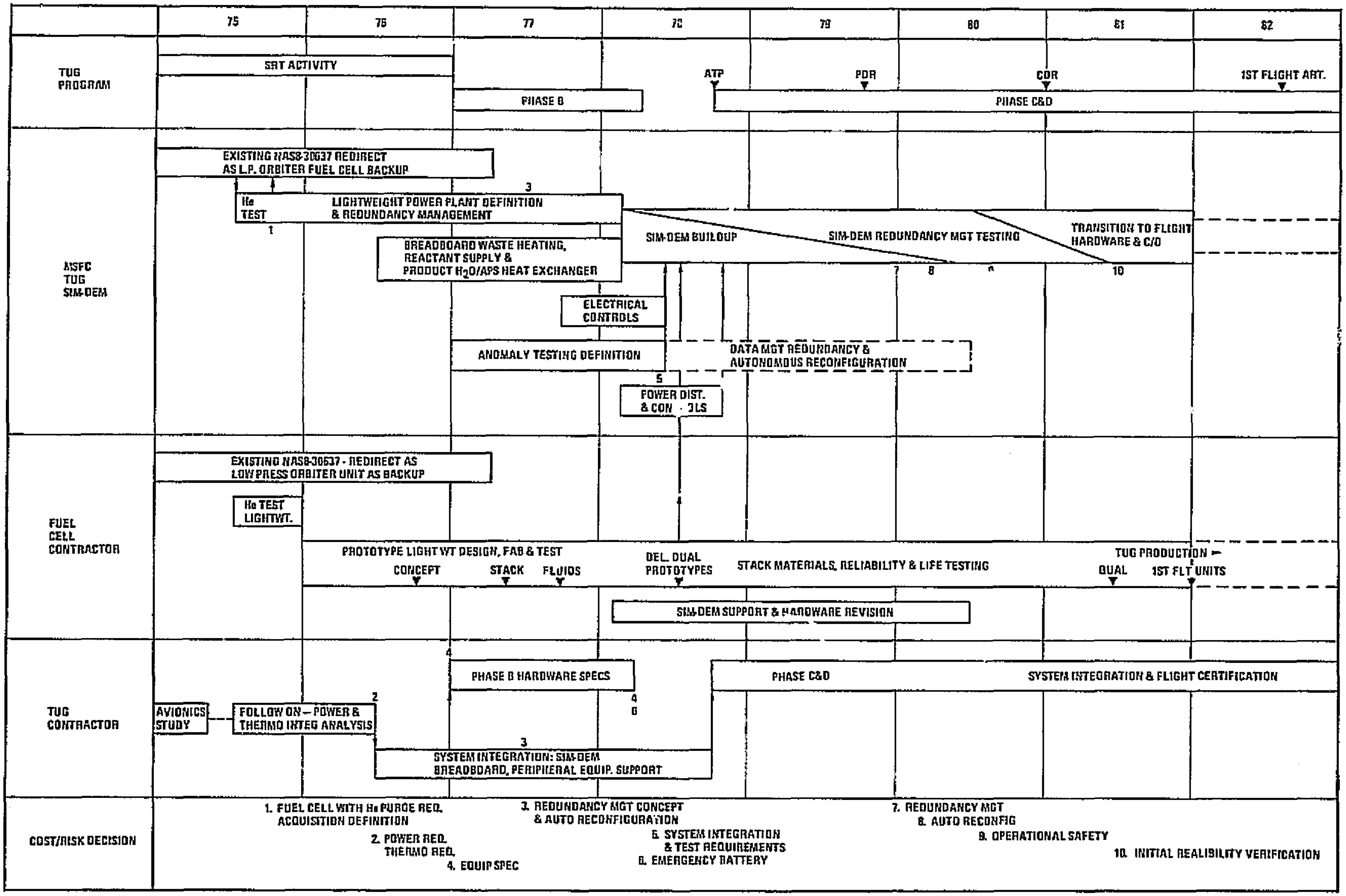


Figure 4-18. Thermally Integrated, Lightweight, Electrical Power System Development Plan

For the latest updated Tug baseline, power requirements are 1.2 kW average (nominal) with occasional accommodation of 1.5 kW peaks. These Tug vehicle requirements are increased by the payload requirements as specified in the MDAC Tug Study, "IUS/Tug Payload Requirements Compatibility Study," Contract NAS 8-31013, which indicate for multiples and additional 1.15 kW average and 3.4 kW peaks. High short term payload peaking loads are presently planned to be accommodated by dedicated batteries. If this power were furnished by the Tug fuel cells, the required capacity would be 2.35 kW average and 4.6 kW peak. A conservative margin based on an estimated 50% growth of these requirements would be 3.5 kW continuous for eight hours during the first payload delivery. Possible growth implementation solutions that have been examined are:

a. Triple 2 kW power plants with 4 kW continuous output.

b. Dual fuel cells simultaneously operating into the same load — each capable of 3.5 kW output for eight hours.

The recommendation is to design the dual power plants, cells and associated peripherals, for operation at a 3.5 kW rating for eight hours. This will provide sufficient capacity for the longest multiple (dual) payload delivery time, and represents a 55% reserve above the present requirements. The power level limitation is primarily a thermal system design boundary, and the fuel cells could provide additional output with proper heat removal and venting of waste products. For the lightweight cell power plants, the design penalty over the 2 kW nominal model that was used for the options comparison is 98 pounds (40.5 kg) for the improved peak load capability. This solution is within the original baseline values projected for the Tug electrical power plant, MSFC 68M00039-2, at 2 kW.

If this recommended solution is adopted for the Orbiter low pressure option, which is not fully thermally integrated, a larger waste heat rejection requirement exists. This requires increased space radiator size and the weight performance is degraded by an additional 126 pounds (56.7 kg) over the original penalty of 550 pounds (247.5 kg), which is a total of 676 pounds (304.2 kg). Therefore the flexibility with the lightweight cell to extend the power capability for future demand increases is considerably better.

4.5.8 BASELINE ELECTRICAL POWER SYSTEM. After consideration of the criteria of performance, weight, cost, development risk, reliability/redundancy implementation, and growth potential as they have been compared in the preceding paragraphs, the thermally integrated lightweight dual fuel cell configuration is recommended for the Tug Electrical Power System. The baseline Avionics configuration has been updated to this selection and is more fully detailed in Volume III, the configuration volume of this final report.

The thermally integrated power plant (Figure 4-19) accepts propellant grade reactants directly from the main propellant tanks. These reactants may be supplied in liquid,

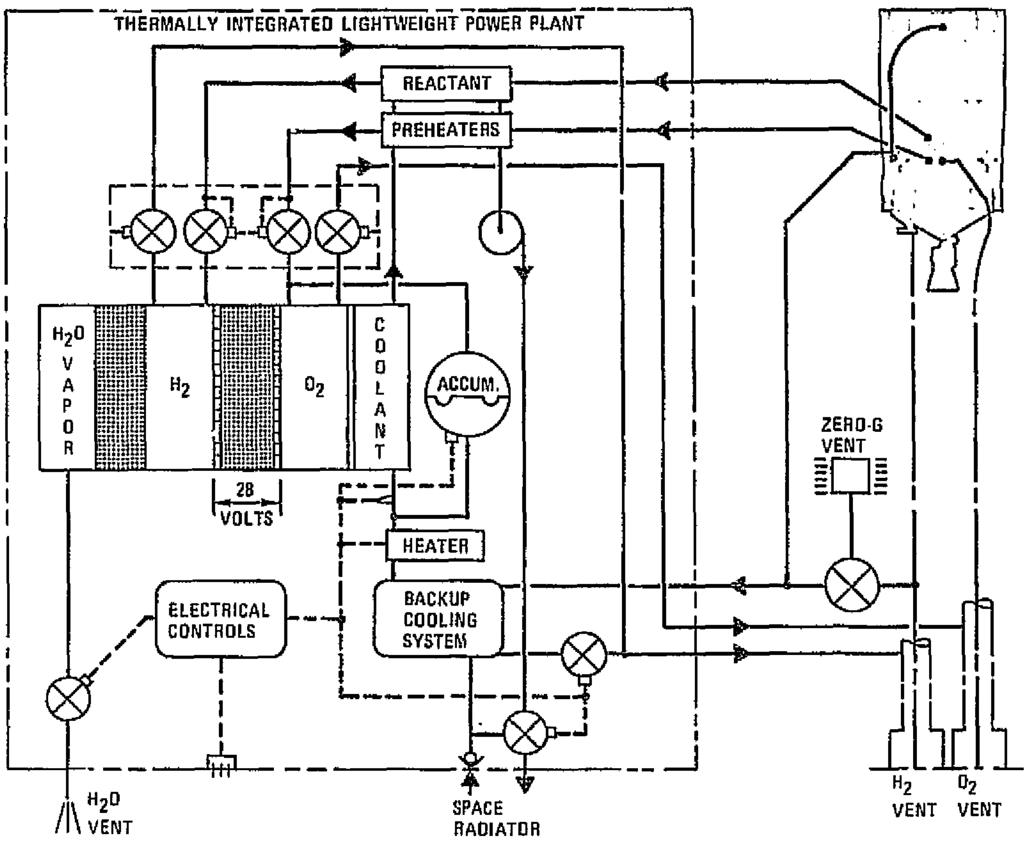


Figure 4-19. Thermally Integrated Power System

gas, or mixed phase condition at 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 can cause flight voltage drop, the power plant controls will automatically initiate venting. This venting will flush and dilute the inerts in the reactant chambers until the voltage drop has been reduced to an acceptable limit.

Product water is exhausted from the power plant at 4 psia (2716 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 venting is autonomously controlled by the power plant controls. A product water condenser and storage accumulator provide the capability to retain water during the longest payload delivery coast phase. The circulating APS fluid is used to condense the product steam into water at the APS fluid temperature of approximately 90° F (306° K).

The unit contains a fuel cell coolant loop. Waste heat from the cell raises the circulating APS hydrazine fluid within operating temperature limits when the APS heat losses except 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.

Each power plant has a dual dedicated heat rejection system: a primary space radiator assembly and a backup H_2 heat exchanger within the power plant. The space radiator consists of four separate radiators in series which 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 package. Leakage of coolant is sensed by the power plant coolant accumulator low volume position switch, which when actuated shuts down that power plant space radiator coolant system and switches over to the backup H_2 heat exchanger. The defective power plant is taken out of the power sharing mode and placed on standby.

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 line is sized to hold a liquid volume of reactants to sustain power plant operation during Tug retrieval or abort. The volume is sufficient to operate the power plant during the time frame from the last main engine burn to retrieval by the Orbiter. 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.

The majority of electrical controls and all the instrumentation are an integral part of the dual power plant. A separate redundancy management microprocessor controls the voting and autonomous reconfiguration. Commands are 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 meters) of the Orbiter and uninterrupted power during the mission phase.

The dual redundant fuel cells and peripherals provide adequate failure protection to satisfy safety and fail safe goals. As an extra precaution against primary power multiple failures, an emergency battery is provided for a short term backup Tug power supply during the last 3000 feet (914 meters) of Tug retrieval. The sizing and description of this battery appear in Section 4.7.

4.6 AVAILABLE ORBITER POWER VERSUS REQUIREMENTS

Power required by the Tug and the payload(s) during ascent and abort phases exceeds the allocated accommodations available from the Orbiter. NASA document JSC 07700, as modified by "Level II Program Requirements Control Board Directive, S00620-RI-Orbiter Electrical Power and Distribution Accommodations for Payload" dated 4 November 1974, indicates the availability of 1000 watts during the ascent phase (1500 watts peak for two minutes maximum) for all electrical loads in the cargo bay. On orbit power availability from the Orbiter is 7000 watts average. Payload electrical loads include the deployment adapter valves, actuators, motors, Tug avionics, other Tug subsystems, and the spacecraft power needs.

Two approaches were reviewed, reduction of the power requirements and the utilization of alternate power sources. The analysis showed 425 watts needed for valve control, propellant mixers, and circulating pumps, which was then reduced to about 200 watts by the adoption of latching type valves. The remaining load then represents a powered down condition. Total power requirements during ascent still exceed the 1000 watts allocated from the Orbiter:

a. Ascent – 1478 watts for 15 minutes.

b. RTLS Abort – 1783 watts average for 1 hour
2360 watts peak for 5 minutes

Figure 4-20 presents the power requirements for the mission phases that are pertinent to the Tug being in the cargo bay.

Solutions that were adopted were to select the latching type valves and to operate the Tug fuel cells in the cargo bay in preference to obtaining power from an additional alternate source. Recommendation is that the Tug fuel cells be turned on at liftoff for the supplying of all Tug and spacecraft power in the Orbiter bay. The fuel cells waste heat can be processed by the product water heat exchanger which slightly heats up the APS propellant, 9°F/hr (5°K/hr) for 1500 watts output. Advantages of using the Tug fuel cells during ascent are:

a. Reduction of the impact on safety, in the event the Orbiter power goes to zero during abort.

b. Future changes in the power requirements for spacecraft support do not impact Orbiter power accommodations.

POWER REQUIREMENTS

	ASCENT PHASE	PREDEPLOY CHECKOUT	DEPLOY	ABORT SAFING
DMS	99	114	114	114
GN&C		382	382	
R&D		50		
COMM	10	10	72	10
INSTR	66	66	66	66
EPS	115	140	130	115
HEATERS	30	230	37	30
VALVE CONTROL MIXERS & PUMPS	225	296	321	768
TOTAL TUG	545	1,288	1,122	1,103
SINGLE SPACECRAFT	600	650	700	
D/ADAPTER	333	333	781	680
TOTAL TUG & P/L REQUIREMENTS	1,478	2,263	2,603	1,783
ORBITER POWER AVAIL	1,000	7,000	7,000	1,000

POWER SOURCE

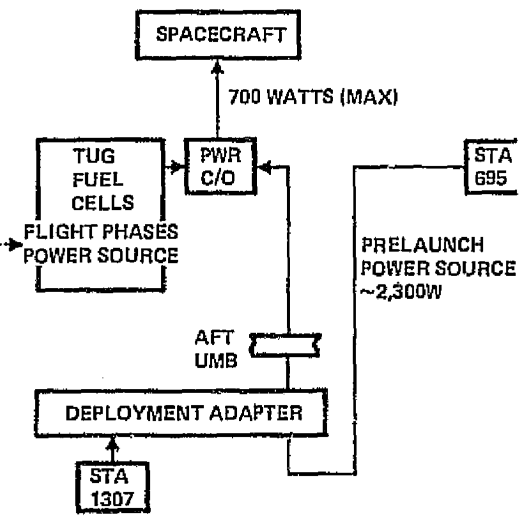


Figure 4-20. Cargo Bay Power Sources/Uses

c. Tug power on and electrical checkout before launch.

d. Simplified switching.

Concerns of having the cells on in the Orbiter bay are:

a. APS fluid as a heat sink would increase 2.25°F (1.25°K) during ascent.

b. APS fluid as heat sink would increase 8.23°F (4.57°K) during ascent/RTLS abort.

c. Dedicated H_2 and O_2 accumulators to hold and supply reactants after main tank dump on RTLS abort: 6.2 pounds (2.8 kg).

d. 4 psia (27.6 kN/m^2) vacuum pump on deployment adapter.

4.7 EMERGENCY BATTERY REQUIREMENTS

While the basic safety protection is provided by the dual redundant fuel cell power plant system, a further backup has been added to the baseline system in the form of an emergency battery. This is not required, but is an extra precaution in the event of multiple failures. If the entire primary power system should fail in the Orbiter payload bay or during the terminal retrieval process, the Tug emergency battery can be switched in. The battery is in addition to the dual redundancy and four to five hours reactants reserve and is sized to ensure that the Tug will have power for stability and safety status communication in excess of the normal mission sequence time when the Tug is within 3000 feet (930 meters) of the Orbiter.

Longest nominal mission time for emergency retrieval by the Orbiter at the end of the Tug free flight is 0.28 hour (16.8 minutes) for a nominal sequence. Therefore calculations for a nominal battery size are:

847 watts (emergency retrieval) × 0.28 hour = 237 watt hours

At 28 volts, 237 watt hours = 8.5 ampere hours

Current rate: 847/28 = 30.3 amperes

Table 4-8 lists characteristics of the batteries. Selection of the 36 pound (16.3 kg), 23 ampere hour battery results in approximately 170 percent additional capacity over the "nominal" requirement.

Table 4-8. Battery Selection (Silver Oxide/Zinc Batteries)

Capacity (A-hr)	23	100	150
Normal Current (amperes)	40	50	50
Peak Current (amperes)	115	80	80
Weight (pounds)	36	70	86
Dimensions (inches)	11 × 8 × 7	13 × 12 × 8	13 × 12 × 8
Allowable Emergency Duration (minutes)	45	198 (3.3 hours)	298 (5 hours)

4.8 POWER DISTRIBUTION AND CONTROL

The preferred subsystem configuration for electrical power distribution and control is very similar to the original MSFC baseline (see Figure 4-21). An additional arm-safe switch has been added for further Orbiter safety, inhibiting the main propulsion system and the APS from any inadvertent operation near the Orbiter.

In the event of a fault, an automatic backup switchover uses the primary distribution and control system. This is intended to accommodate most emergencies. Additional protection is provided by manual actuation of the separate hardware controls of the Orbiter Mission Specialist Station (MSS) panel with bypassing the Tug primary control system. A third mode for backup permits the dropping out of payload and Tug non-essential power needs using the emergency bus; with power, manual abort and safing control is obtained from the Orbiter.

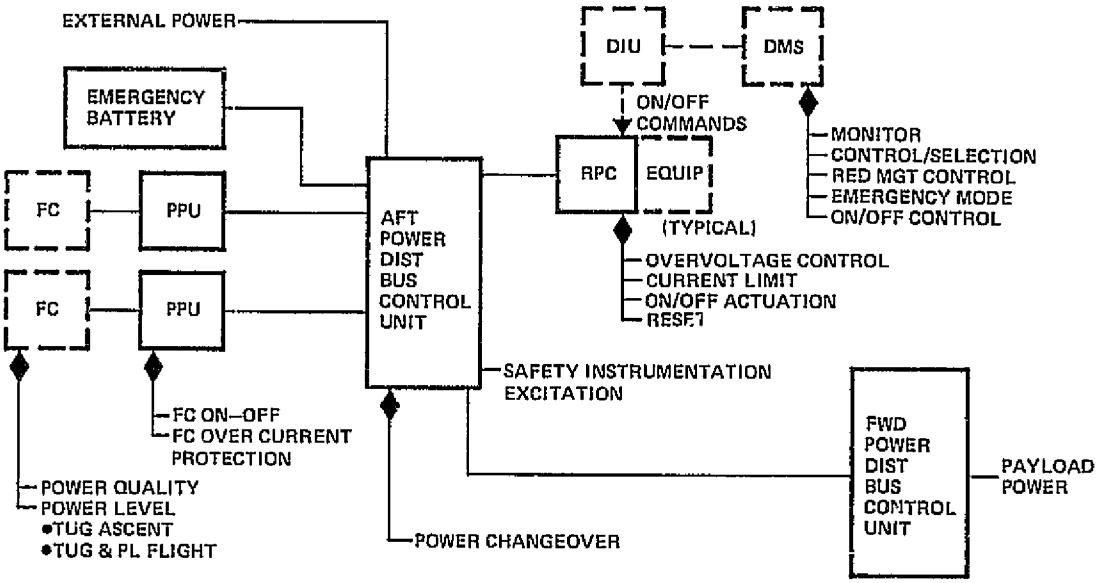


Figure 4-21. Electrical Power Distribution and Control Functional Division and Control Hierarchy

NASA technology developments now in progress and needed for Tug and requiring completion within the Tug program development times include:

a. Solid state, high current, lightweight circuit breaker/on-off switches for the fuel cells. In a downstream short condition, the cells will respond and provide over 1000 amperes short circuit current. Each Power Processing Unit (PPU) near the fuel cell should incorporate solid-state circuit breakers capable of handling these current levels.

b. Continued evolution of the Remote Power Controllers (RPC's), particularly their Digital Interface Unit (DIU) control from the data bus, reset, and emergency power bus override.

SECTION 5

RENDEZVOUS AND DOCKING

Investigation and comparison of the potential methods for rendezvous and docking of the Tug with free-flying spacecraft have involved the orbital approach to the spacecraft; candidate sensor evaluation; guidance, navigation and control subsystem capability assessment; and the examination of terminal approach and docking for both remote-manned control and fully autonomous techniques. This group of related trade studies has resulted in the recommended selection of ladar and TV as the best sensors to support closure and docking of the Tug with the target spacecraft. Of the possible navigation schemes, direct ascent insertion for rendezvous is favored as the most efficient and appears to be feasible. The excellent navigational performance of the GN&C subsystem, with the high accuracy ILT position and velocity update system, allows the Tug to perform the insertion burn at apogee in the near vicinity of the spacecraft.

5.1 FUNCTIONAL ELEMENTS

Figure 5-1 shows the functional elements that are associated with the rendezvous and docking of the Space Tug with a spacecraft. Acquisition entails either the searching of the dispersion volume produced by uncertainty in the Tug position, or an accurate knowledge of the pointing vector from Tug to spacecraft. When the pointing vector is

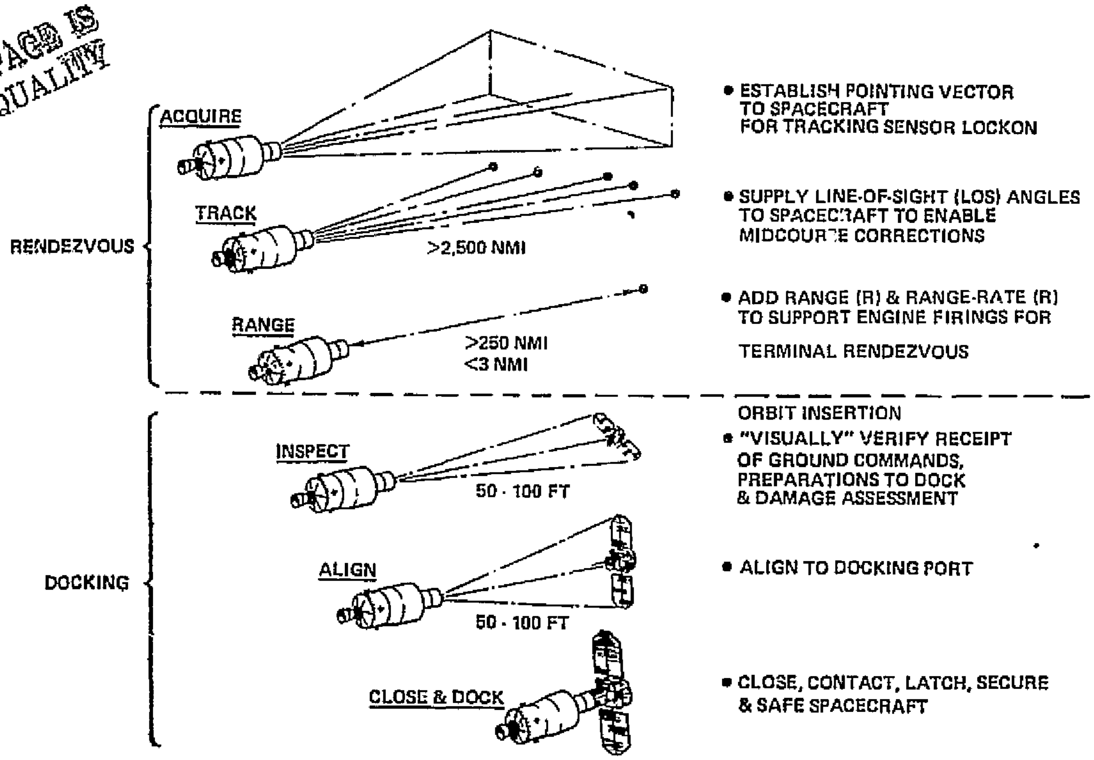


Figure 5-1. Functional Elements of Rendezvous and Docking

established, the subsystem locks the tracking sensor into the tracking mode. Tracking involves the supplying of line-of-sight (LOS) angles to the spacecraft. The required precision is inversely proportional to slant range and can be somewhat crude at the acquisition point. Midcourse corrections become necessary to ensure that the desired target point for the initiation of the insertion burn is achieved. The addition of LOS information can reduce the relative ephemeris errors between Tug and spacecraft. The degree to which this can be achieved is a function of the maximum range of the sensor as well as the basic navigation accuracy of Tug.

Ranging is a requirement when the rendezvous guidance begins. This can be before or after the insertion burn depending on the strategy employed. The more desirable approach is to insert in close proximity to the spacecraft, reacquire after the burn to correct the insertion velocity dispersions, and to close to within inspection distances. If a direct ascent maneuver is employed, ranging must be accomplished after insertion; or, alternatively, the ranging sensor must have a maximum range on the order of several hundred miles. Required range accuracy is inversely proportional to range. Inspection is accomplished at a standoff of typically 50 to 100 feet (15.2 to 30.5 meters) and requires only gross range information. If the spacecraft is active, it can be interrogated for health/status and commanded to latency. Unless the spacecraft can respond to reorientation commands, the Tug must orbit the spacecraft to achieve a gross alignment on the docking port, hence obtain docking sensor lockon. Docking is then accomplished by closing at a controlled range rate. Spacecraft relative pitch, yaw, and roll information is required to dock in addition to range and LOS angles.

5.2 SENSOR TRADES

5.2.1 SENSOR CANDIDATE SYNTHESIS. The initial screening of candidate sensors for rendezvous and docking was based on a number of factors. Systems requiring actively cooperative targets were eliminated a priori; such a system would be inconsistent with servicing or retrieval of a failed spacecraft. Sensors operating at wavelengths greater than in the microwave region were rejected because of the impractically large apertures necessary to meet spatial resolution requirements. The range of wavelength of passive sensors was determined by consideration of the spectrum of solar reflectance and target thermal emission. Active sensors were selected on the basis of the availability of reliable high power sources.

Four generic types of sensors passed the initial screening: radars (radio detection and ranging systems) in the 2-30 cm wavelength region, ladars (laser detection and ranging system) in the 0.8 to 11 μm region, passive LWIR (long wavelength infrared) sensors utilizing target thermal emission from 6 to 16 μm, and passive sensors utilizing reflected solar radiation in the visible region (0.4 to 0.8 μm).

The approach to synthesizing candidate sensor systems was to select a set of standard conditions — field of view, aperture size, frame rate, etc. — that would permit an objective comparison of candidates. The selection of these parameters (see Table 5-1)

Table 5-1. Standard Conditions for Sensor Performance Calculations

Target		
Projected Area:	10 m^2	
Infrared Emissivity:	0.8	
Infrared Bidirectional Reflectivity:	3×10^{-3} sr^{-1}	
Visible Bidirectional Reflectivity:	1.6×10^{-2} sr^{-1}	
Effective Surface Temperature:	275°K	
Retroreflector Cross Section:	$\frac{1.56 \times 10^3}{\lambda^2}$ cm^2 sr^{-1}	
Solar Irradiance:	540 W/m^2	0.4 to 0.7 μm
	365 W/m^2	0.7 to 1.1 μm
	300 W/m^2	1.0 to 3.0 μm
Sensor		
Acquisition Field of View:	0.274 sr	
Tracking Field of View:	3×10^{-4} sr	
Docking Field of View:	3×10^{-2} sr	
Frame Time:	140 sec (acquisition), 14 sec (tracking)	
Optical Aperture:	10 cm	

was somewhat arbitrary in that system-level trade information was not available in time to structure the subsystem trades. As a result, the candidate sensors were not fully optimized for total system performance.

Several candidates in each class of sensor were evaluated. Radars included C, S, L, X, and Ku-bands. LWIR concepts considered ranged from uncooled detectors to cryogenically cooled detectors with cooled, baffled optics; scanning techniques included radar scans, linear-scanned arrays, and staring mosaics. Ladar candidates included scanning and non-scanning devices utilizing solid state and gas lasers. Television employing SEC and SIT vidicons was examined. (It is recommended that future studies include CCD and CID arrays.)

The next four sections present performance calculations for each sensor candidate applicable to the acquisition, tracking, port search, and docking phases of the mission. Results of these calculations are summarized in Tables 5-2 and 5-3, providing a comparison of candidate sensor characteristics. (It is noted again that these numbers do not represent utlimate performance to be expected, but provide a comparison on a normalized basis.)

Table 5-2. Sensor Performance and Cost

Sensor System	Applicable Mission Phase: Acquisition	Tracking	Ranging	Docking	Estimate Subsystem Costs ($M): Total	DDT&E	Prod	Oper‡	Volume (ft^3)	Ranging Range: Max (n.mi.)	Min (ft)	Range Rate (ft/sec): Max	Min	Maximum Angular Rate (deg/sec): Far	Near	Range Accuracy (ft): Far	Near	Angular Accuracy (deg): LOS	Attitude
Scanning Ladars																			
† GaAs Active	X	X	X	X	20.7	10.1	9.9	0.7	1.0	85	1.0	36,000	0.05	1.1	1.6	3.28	0.33	0.06	0.06
Passive*	X	X								NA	NA	NA	NA	0.4	NA	NA	NA	0.06	NA
CO_2 Active	X	X	X	X					3.0	270	10	15,000	0.23	1.0	—	13.8	3.28	0.04	—
Passive**	X	X								NA	NA	NA	NA	0.07	NA	NA	NA	1.0	NA
HF Active	X	X	X	X					3.5	480	10	4,900	0.23	0.8	—	13.8	3.28	0.06	—
Passive*	X	X								NA	NA	NA	NA	0.07	NA	NA	NA	1.0	NA
Scanning Radars																			
Ku Band	X	X	X	X												25	5	0.06	NA
X Band	X	X	X	X	43.[illegible]	34	8.8	0.6	14.1	215						25	5	0.06	NA
S Band	X	X	X																
L Band	X	X	X																
Longwave Infrared	X	X			7.1	1.8	4.9	0.3	1.0	NA	NA	NA	NA	0.07	NA	NA	NA	1.0	NA
† Low Light Level TV																			
SEC*	X	X		X					0.2					0.1		NA	f(R)	0.06	
SIT*	X	X		X	4.35	1.4	2.8	0.2	0.2					0.1		NA	f(R)	0.06	
Ladars (10 deg FOV)																			
† GaAs			X	X	7.9	5.0	2.7	0.2	0.5	4.0	1.0	36,000	0.05	NA	NA	NA	0.33	NA	NA
Tricolor			X	X	9.0	5.2	3.5	0.2	0.6	4.0	1.0	36,000	0.05	NA	NA	NA	0.33	NA	0.5

† Retained for further evaluation
* Presumes sun illumination
‡ Single spare only; does not include ground or flight operations costs
** Used as an LWIR sensor (cooled detector & optics)

Table 5-3. Sensor System Characteristics

SENSOR SYSTEM	APPLICABLE MISSION PHASE: ACQUISITION	TRACKING	RANGING	DOCKING	SYS MTBF (HR)	ON-ORBIT WEIGHT (LB): TUG	SC	POWER (W)	ACQ MAX RANGE (NMI): RETRO	SKIN	TRACKING MAX RANGE (NMI): RETRO	SKIN	HARDWARE TECHNOLOGY DEVELOPMENT
SCANNING LADARS													
✓ • GaAs ACTIVE	X	X	X	X	7,000	39	6	40	44	0.4	85	1.0	DEVELOPMENT
PASSIVE*	X	X			30,000			30	NA	1,190	NA	3,690	
• CO_2 ACTIVE	X	X	X	X	900	65	6	180	150	1.4	270	8.3	PREDEVELOPMENT
PASSIVE**	X	X			5,000			50	NA	268	NA	827	
• HF ACTIVE	X	X	X	X	200	80	6	340	190	2.0	480	11.5	RESEARCH
PASSIVE*	X	X			5,000			50	NA	44	NA	57.7	
SCANNING RADARS													
• Ku BAND	X	X	X	X	1,100	150	NA	150	NA	240	NA	340	OPERATIONAL
• X BAND	X	X	X	X	1,100	165	NA	150	NA	215	NA	300	OPERATIONAL
• S BAND	X	X	X		1,250	190	NA	150	NA	185	NA	260	OPERATIONAL
• L BAND	X	X	X		1,500	215	NA	150	NA	160	NA	225	OPERATIONAL
LONGWAVE INFRARED	X	X			5,000	30	NA	30	NA	332	NA	1,030	SIMILAR TO OPERATIONAL
LOW LIGHT LEVEL TV SEC*	X	X		X	15,000	8	NA	8	NA	1,400	NA	1,400	OPERATIONAL
✓ SIT*	X	X		X	15,000	8	NA	8	NA	4,110	NA	4,110	OPERATIONAL
LADARS (10 DEG FOV)													
✓ • GaAs			X	X	15,000	19	6	40	NA	NA	4.0	NA	DEVELOPMENT
• TRICOLOR			X	X	15,000	20	6	40	NA	NA	4.0	NA	DEVELOPMENT

✓ RETAINED FOR FURTHER EVALUATION
*PRESUMES SUN ILLUMINATION
**USED AS AN LWIR SENSOR (COOLED DETECTOR & OPTICS)

Based on performance, reliability, weight, stage of development, and system-level redundancy, a combination of the GaAs scanning ladar and television is recommended for rendezvous and docking. An important aspect of this selection is that it permits an orderly progression from manned to autonomous operation. The modes of operation and the complementary nature of these two sensors are discussed in Section 5.5.

5.2.2 RADAR (2-30 cm). The search-mode radar range equation for a microwave radar may be expressed:

$$\frac{PA}{L} = \frac{4\pi R^4 \Omega KT (s/n)}{\sigma T_F}$$

where

P	=	transmitted power (watts)
A	=	antenna area or aperture (meter^2)
L	=	total system losses
R	=	target range (meters)
Ω	=	angular search volume (steradians)
K	=	Boltzman's constant (1.38×10^{-23} joule/deg)
T	=	system temperature (deg Kelvin)
(s/n)	=	signal-to-noise ratio required for an acceptable detection probability
σ	=	target radar cross-section (meter^2)
T_F	=	frame time (or time to search the angular search volume)

The basic radar design trade is that of power aperture (PA) and detection range (R). These factors are therefore treated as the variables in the following discussion.

Losses (L) — Since the tug radar operates in a vacuum, the only rf loss is that associated with system microwave components. This is conservatively assumed to be 2 db. Since a simple modulation waveform can be used in this application, a 1.5 db matched filter loss is assumed; if a non-optimum modulation waveform is selected, an additional 3 db loss could result. In addition, it is customary to assume a maintenance or field service degradation loss to account for equipment performance variations, caused by use and normal maintenance. A 3 db loss is considered adequate for this factor. Therefore, the total system loss is 2 + 1.5 + 3 = 6.5 db.

System Temperature (T) — Since the radar operates in space, the major noise contributing element is the receiver. Assuming a non-exotic receiver, a noise temperature of 500°K is to be expected by the use of either a tunnel diode amplifier or a conventional uncooled parametric amplifier.

If the connecting microwave elements are assumed to be at a temperature of 200°K, the assumed 1 db receiver microwave loss results in a noise temperature of 41°K. The antenna temperature will vary from 290°K when the antenna is looking towards the earth to essentially 0°K when the radar is looking towards deep space; the resulting effective antenna temperature will vary from 0 to 233°K. Therefore, the system temperature will vary from 500+41+0 = 541°K to 500+41+233 = 774°K.

Signal-to-Noise Ratio (S/N) — If a basic frame detection probability of 0.5 is assumed, the following cumulative detection probability results:

Frame	Average Detection Time (sec)	Cumulative Detection Probability
1	70	0.5
2	210	0.75
3	350	0.88
4	490	0.94
5	640	0.97

Since this application permits a relatively high false alarm probability of 10^{-6}, a signal-to-noise ratio of 13 db is required assuming the target radar cross section is decorrelated from scan to scan.

Target Radar Cross Section (σ) — The target behavior may be assumed to be bounded by the limits of an isotropic and a flat plate scatterer. Therefore, the radar cross section can vary from 10 meter^2 to $4\pi A^2/\lambda^2 = 1.4 \times 10^6$ meter^2 at $\lambda = 3$ cm. A median and therefore probable cross section is $10\pi = 31.4$ meter^2.

Expected Detection Range (R) – Accumulating these system baseline parameters (expressing all values in db) results in the following summation:

	PA/R^4 (minimum)		PA/R^4 (maximum)	
	+	–	+	–
Losses (L)	3.5		6.5	
4π	11		11	
Ω(0.274 sr)	4.35	10	4.35	10
K	1.4	230	1.4	230
T (541/774°K)	27.34		28.89	
s/n	13		13	
σ (10/31.4)		15		10
T_F (140)		21.47		21.47
Totals	-215.88		-206.33	

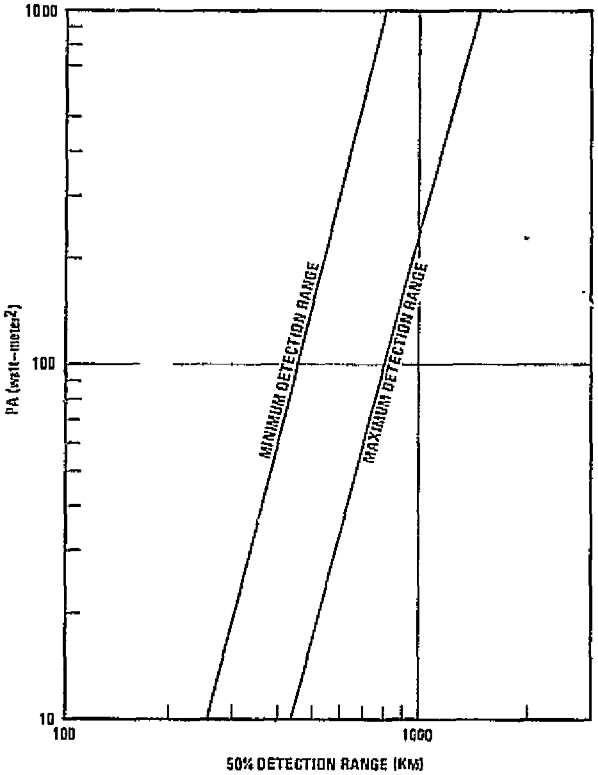


Figure 5-2. Expected Detection Ranges of Candidate Radars

Therefore PA/R^4 is in the range of $PA/R^4 = 2.62 \times 10^{-10}$ to 2.35×10^{-9} where P is expressed in watts, A in meter^2 and R in kilometers. This range is plotted in Figure 5-2. It is seen that a radar power aperture product of 100 watt-meter^2 results in a detection range of 455 to 790 km (246 to 427 n. mi.).

Selected Candidates — A baseline radar requirement of 100 watt-meter^2 will be examined. If it is assumed that the aperture is 1.0 meter^2, the transmitted power required is 100 watts. Since overall efficiency will be in the range of 10 to 20%, this candidate radar requires an input power of 500 to 1000 watts.

The 500 watt average input power is a significant detraction from the performance of this candidate. It is noted from Figure 5-2 that if the detection range could be reduced to 280 to 420 km (150 to 230 n. mi.), a power aperture of 10 watt-meter^2 would be adequate. This option would reduce the prime power requirement to 50 to 100 watts. (The 100 watt value is more probable since the transmitter efficiency is a monotonically increasing function of transmitter power.)

Both the long range and short range candidates will be considered in what follows.

Weight and Volume Estimates — The candidate radar is similar to most aircraft fire control system radars.[1] (The major difference is that typical aircraft radars have apertures of approximately 0.5 meter^2.) Therefore it is informative to examine similar aircraft systems.

Figure 5-3 summarizes the weight-to-transmitter power relationship of several operational X-band ($\lambda = 3$ cm) radars. In addition, data for three currently proposed radars are included. This data suggests that a 1974 state-of-the-art X-band radar would weigh 200+ pounds if built to aircraft specifications. Assuming the Shuttle-Tug environment

[1] Phased array antenna systems — such as the one selected for the Communications Subsystem — cannot achieve the power density required. For example, using the Communications Subsystem S-band module, a 100 W-m^2 power aperture would require 40,000 modules!

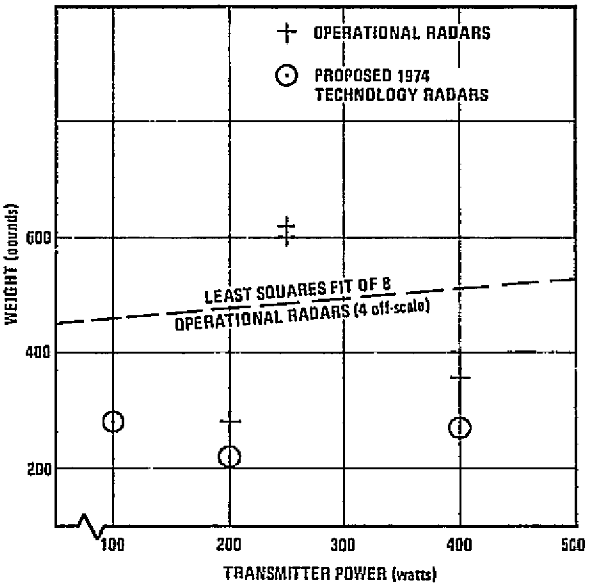


Figure 5-3. Transmitter Power to Radar System Weight Relationship of Aircraft X-band Radars

is less severe and spacecraft lightweight technology were used, a weight approaching 175 pounds (79.4 kg) could result. Since the currently proposed radars use a maximum of solid state technology, it is not anticipated that 1978 technology will result in significant further weight savings. The principal benefit of 1978 technology should be to improve reliability.

The relative insensitivity of the weight-to-power function suggests flexibility in the power aperture trade. However, radar weight is a significant function of aperture. In addition, input power is essentially a linear function of transmitter power. It is believed the 1.0 meter^2 aperture, 100 watt (transmitted power) candidate is a near optimum design point to balance the weight/power penalties.

The following weight and volume breakdown is an estimate of the distribution of the 175 pound (79.4 kg) candidate radar:

	Weight, lb (kg)	Volume, ft^3 (m^3)
Antenna	35 (15.9)	10 (0.283)
Transmitter	50 (22.7)	1.7 (0.048)
Receiver	25 (11.3)	0.8 (0.022)
Synchronizer	25 (11.3)	0.6 (0.017)
Power Supply	40 (18.1)	1.0 (0.0283)
Total	175 (79.4)	14.1 (0.399)

It is noted that LSI techniques apply only to parts of the receiver and to the synchronizer; therefore, dramatic further weight reductions are not anticipated.

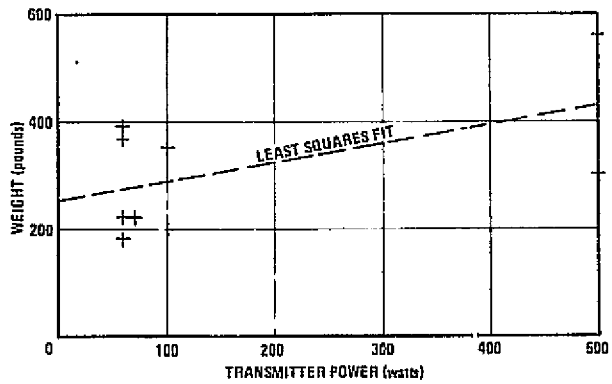


Figure 5-4. Transmitter Power to Radar System Weight Relationship of Operational Aircraft Ku-band Radars

Similarly, Figure 5-4 summarizes the weight-to-transmitter power relationship of a number of operational Ku-band ($\lambda = 2$ cm) aircraft radars. Several points are to be noted in comparing Figures 5-3 and 5-4. The lighter weight of the existing Ku-band radars is attributed to 1) transmitter and receiver weights vary directly with wavelength, 2) the particular Ku-band radars contained in the figure tend to be simpler configurations than the X-band radars, and 3) these Ku-band radars tend to be newer designs with significant solid state technology already employed. The greater slope of the weight-to-transmitter function is attributed to the decreased efficiency of power generation at higher frequencies.

It is concluded that a candidate radar at Ku-band would not have a significantly different weight from the X-band candidate, since weight savings due to the smaller size of microwave components would be partially compensated by power supply increases due to decreased power generation efficiency.

Similar data is not available for the lower frequencies because of limited use of C, S, and L-band radars in aircraft applications. However, it is noted that weight tends to increase significantly as the wavelength increases. For example, a proposed 2 kW L-band, solid state aircraft radar weighs 1100 pounds (499 kg) without its antenna. (The data of Figures 5-3 and 5-4 predict corresponding weights of 807 pounds (366 kg) at X-band and 966 pounds (438 kg) at Ku-band).

Since atmospheric attenuation is not a factor, either X- or Ku-band is recommended for the Tug application. Higher frequencies are not recommended because the receiver noise temperature increases dramatically beyond the Ku-band.

Reliability — Past examinations of field reliability data have demonstrated the utility of equipment weight as an estimator of equipment complexity and therefore of equipment reliability. The hyperbolic function (weight) (MTBF) = constant is a logical parametric estimating relationship.

The weight and reliability of many operational aircraft radars are plotted in Figure 5-5. Field data is noisy and MTBF is therefore difficult to define precisely. Nevertheless,

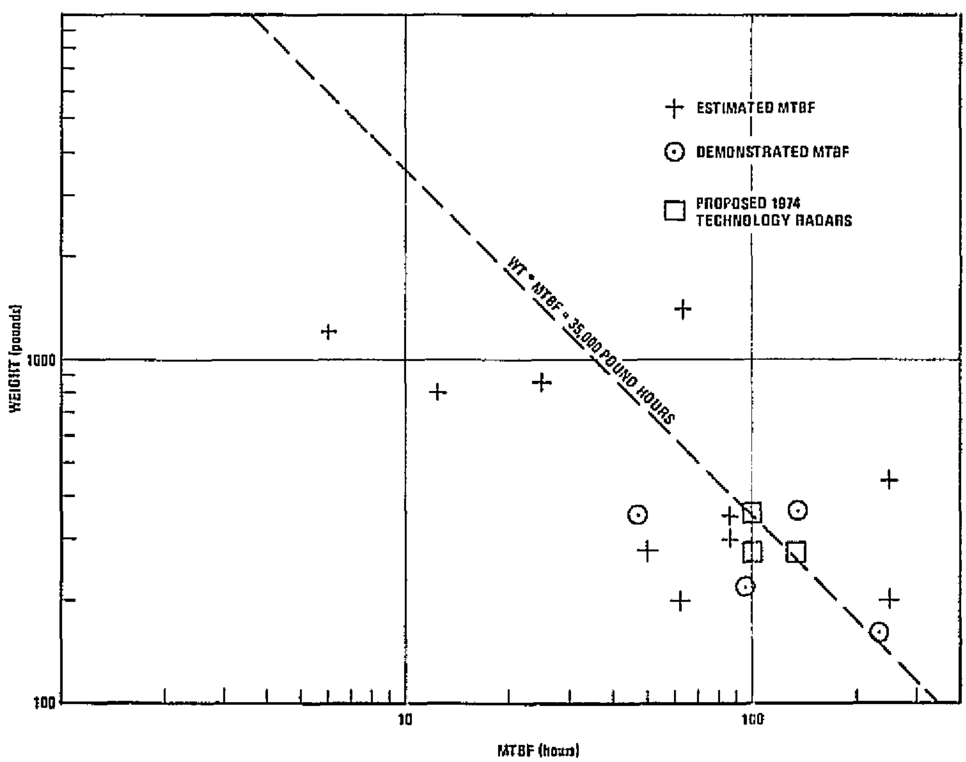


Figure 5-5. Operation Aircraft Radar Reliability Experience

Figure 5-5 demonstrates the utility of the estimator (weight) (MTBF) = 35,000 pound-hours (15,876 kg-hours). (This value of the constant was derived by averaging the data of Figure 5-5.) In particular, the close fit of the estimator to the predicted reliability of three proposed 1974 radars is to be noted.

The radars of Figure 5-5 are in service in manned aircraft. Since the space environment is generally more benign and since the Tug radar will not suffer the typical maintenance abuse of aircraft field maintenance, a higher reliability is expected. Available data suggests an approximate six-fold improvement in space. Therefore the estimating relationship (weight) (MTBF) = 200,000 is recommended.

The use of this estimator results in the prediction of a MTBF of 1143 hours for the baseline X-band radar.

Summary — The proposed Tug radar high-power candidate is summarized below:

Type	Simple, non-coherent, pulsed
Wavelength	3 cm (X-band)
PRF	50 Hz & 500 Hz
Peak Power	1 MW
Pulse Width	2 μsec and 0.2 μsec
Antenna	1.0 meter diameter paraboloid
Angular Coverage	$\pm$15 degrees ($\pm$0.26 radian) (azimuth & elevation)
Beamwidth	1.4 degrees (0.024 radian)
Augular Tracking Accuracy	1 mrad
Range Resolution	150 & 15 meters
Range Tracking Accuracy	15 & 1.5 meters
Typical Detection Range on a 31 m^2 (radar cross section) target	665 km (360 n. mi.)
Weight	175 pounds (79.4 kg)
Input Power	500 watts
MTBF	1100 hours
Volume	
Antenna	10 ft^3 (0.283 m^3)
Electronics	4 ft^3 (0.113 m^3)

The optional low-power candidate would result in the following changed characteristics:

Peak Power	250 kW
Pulse Width	1 μs & 0.2 μs
Range Resolution	75 & 15 meters
Range Tracking Accuracy	7.5 & 1.5 meters
Typical Detection Range on a 31 m^2 (radar cross section) target	395 km (215 n. mi.)
Weight	150 pounds (68 kg)
Input Power	150 watts

5.2.3 LONG WAVELENGTH INFRARED (LWIR) SENSORS. The peak thermal emission from typical spacecraft occurs in the vicinity of 10 μm wavelength. Detectors are available covering the 6 to 16 μm spectral band. This band includes approximately 50% of the thermal emission from a target spacecraft at temperatures ranging from 250 to 300°K.

Design options for an LWIR system include raster scanning with a single detector, line scanning with a linear array of detectors, and detector mosaics in a staring mode. Detector temperatures can range from cryogenic to ambient. To realize maximum sensitivity from detectors at 4°K against a space background, it is necessary to cool and baffle the optical system. Detectors at 77°K are less sensitive, but do not require cooled optics. Ambient temperature detectors are considerably less sensitive.

Preliminary calculations indicated that an LWIR system utilizing a linear array of detectors at 77°K would compete favorably with the baseline GaAs scanning ladar, in terms of acquisition range. A system with these characteristics represents a reasonable compromise between cost and performance, and was thus chosen to represent this class of sensor.

In concept, the LWIR sensor employs a 30 element HgCdTe detector array cooled to 77°K by means of H_2 boil-off from the Tug propellant system. A single axis, scanning mirror provides spatial coverage transverse to the linear detector array. Aperture, field-of-view, etc., are specified in Table 5-1.

The range equation for a passive scanning device can be expressed as

$$R^2 = \frac{J A_r}{s/n \, (NEP) \, L}$$

where

J = target intensity in the spectral band (W/sr)
A_r = receiver area (cm^2)
s/n = signal-to-noise ratio
NEP = noise-equivalent power (W)
L = optical and electronic loss factor

The NEP is a function of detector figure of merit D* (cm $He^{1/2}$ W^{-1}), detector area A_d (cm), and electrical bandwidth (Hz), given by

$$NEP = \frac{\sqrt{A_d \Delta_f}}{D^*}$$

Assuming a 30 degree (0.523 radian) field of view, the detector area corresponding to a 10 cm f/1 optical system is 3×10^{-2} cm^2. A bandwidth of 36 Hz is required in the acquisition mode. A detector D* of 1.6×10^{11} cm $Hz^{1/2}$ W^{-1} is calculated, assuming background-limited performance, with thermal emission of the optical system predominating. The NEP in acquisition mode is thus 6.7×10^{-12} W.

Against a 10 m^2 target with an emissivity of 0.8, at a temperature of 275°K, the maximum range in acquisition is 332 n.mi. (612 km), increasing to 1030 n.mi. (1900 km) in track mode.

At long range, a crude measurement of range can be derived from a knowledge of target intensity and sensor calibration. To first order, the range uncertainty ΔR can be computed from

$$\left(\frac{\Delta R}{R}\right)^2 = \frac{\Delta J}{J \cdot K \cdot S}$$

where

J = uncertainty in target intensity J (W/m^2)
K = calibration constant (V/W)
S = signal (V)

Assuming an uncertainty of ±f factor of 2 in target intensity due to variations in temperature, emissivity, and projected area as a function of aspect angle, the range can be estimated to an accuracy of about ±40%. An improvement could be achieved by controlling the area-emissivity product of the target as a function of aspect angle, but this would place unreasonable constraints on the spacecraft design.

At short range, where the target is resolved by the LWIR system, the range can be determined with better accuracy by measuring the angular subtense of the target.

Measurement of target size becomes impractical at very short range due to image blurring, for a fixed-focus system. If the sensor is focussed at infinity, the image will be blurred by 10% at a range of approximately 10 times the focal length. This criterion was used to establish minimum range. For a 10 cm, f/2 system, the minimum range is on the order of two meters.

Because of the large uncertainty in the range measurement, the LWIR system has virtually no capability for range rate determination.

An angular resolution of 1 degree (0.017 radian) is predicated arbitrarily on an array of 30 essentially square detectors covering a 30 degree (0.523 radian) field of view. This could be improved with a larger number of smaller square detectors, or with rectangular detectors, resolution in the along-scan being determined by detector width, and in the cross-scan direction by detector length. Cross-scan resolution could be improved with Kalman filtering. This detailed trade was not conducted prior to elimination of LWIR in favor of ladars and/or TV.

The major considerations in eliminating the LWIR candidate were as follows (not necessarily in order of importance):

Judged as a complete subsystem, LWIR provides only a very crude range measurement and essentially no range rate or target attitude information. Judged as a line-of-sight sensor to be used in conjunction with range and attitude sensors, its primary advantage is that it does not constrain the rendezvous tactics to approach the target from its sunlit side. This advantage has only a small impact on overall mission performance, as discussed in Section 5.2.6, and is outweighed by a number of other factors. Essentially no capability increment results from adding an LWIR system to the existing television, which is required for visual inspection. Substitution of an LWIR scanner in place of television to satisfy both line-of-sight tracking and visual inspection requirements, even if shown to be feasible, would result in higher system cost, weight, and volume with reduced performance and reliability (see Tables 5-2 and 5-3).

5.2.4 LADAR SENSORS. Two obvious candidates in this class of sensors are the GaAs scanning ladar (prototypes that have been developed for MSFC by ITT) and the CO_2 scanning ladar currently under study at Norden. Because of recent developments in high-frequency lasers, and the advantages of higher lasing efficiency and shorter wavelength in comparison with CO_2, a high-frequency scanning ladar was postulated, assuming a design similar to the Norden CO_2 design, but with performance scaled from 10.6 μm to 2.8 μm.

In addition to the scanning ladars, two non-scanning ladars were synthesized to complement passive sensors limited to a line-of-sight capability. One provides range only information; the other provides range and spacecraft-relative attitude.

The performance of these sensors in each applicable mission phase is discussed below, based on nominal system parameters appearing in Table 5-1.

The range equation used to evaluate the ladar sensors is given by:

$$R^4 = \frac{P\sigma A_R \eta\lambda}{w(s/n)hc\,\Delta f\,L}$$

where

P = peak transmitter power (W)

A_r = receiver area (cm^2)

η = detector quantum efficiency

λ = wavelength (cm)

w = transmitter solid angle (sr)

s/n = signal-to-noise ratio

h = Planck's constant (6.7×10^{-34} j sec)

c = velocity of light (3×10^{10} cm/sec)

Δf = electrical bandwidth (Hz)

L = optical and electronic loss factor

Minimum range is device dependent, and is treated separately in the discussions of individual candidates. The maximum range rate is highly variable, depending on range, frame time, scanning geometry, and the extent ot which Kalman filtering is employed in ephemeris computations. To provide a rule-of-thumb comparison of candidates, maximum range rate was calculated singly on the basis of maximum range divided by frame time (admittedly an oversimplification). Similarly, minimum range was calculated as range resolution divided by frame time.

Range resolution is given by

$$\Delta R = \frac{c\tau}{2}$$

where τ is the resolution of the measurement of pulse transit time.

Angular resolution was equated with instantaneous field of view (again, an oversimplification, but adequate for comparison of candidates). Maximum acquisition angular rate was calculated by means of the expression.

$$\dot{\theta}_{max} = \frac{\theta_i - \theta_s}{2\eta_a t_d + \eta_t t_d}$$

where

θ_i = instantaneous field of view

θ_s = acquisition step angle

η_a = number of acquisition steps per line

η_t = number of track steps per cross line

t_d = acquisition step dwell time

Minimum angular rate was determined as angular resolution divided by frame time. Target attitude accuracy is device dependent, and is discussed individually. To avoid unnecessary repetition in the discussions of sensor performance that follow, parameters common to all candidates are listed in Table 5-1.

5.2.4.1 GaAs Scanning Ladar. The ITT GaAs scanning ladar, which has been under development for several years, employs a GaAs pulsed laser and a piezo-electrically driven scanning mirror system. The detector is an ITT image-dissecting photomultiplier, which is scanned electronically in synchronization with the laser scan. A schematic of the system appears in Figure 5-6.

Assuming a peak transmitted power of six watts, (GaAs lasers are currently available with three watts peak power; a development program with a goal of 60 watts is in progress) and a 1 kHz pulse repetition rate, the maximum range against the standard cooperative target is 44 n.mi. (81.5 km) in acquisition and 85 n.mi. (157.4 km) in tracking.

Used as a passive sensor, which requires removal of the narrow band spectral filter used for rejection of solar radiation, the maximum range can be extended to 1190 n.mi. (2000 km) in acquisition and 3690 n.mi. (6900 km) in tracking. Passive performance is limited by the star background; a discussion of the star-background-limited performance appears in Section 5.2.5.

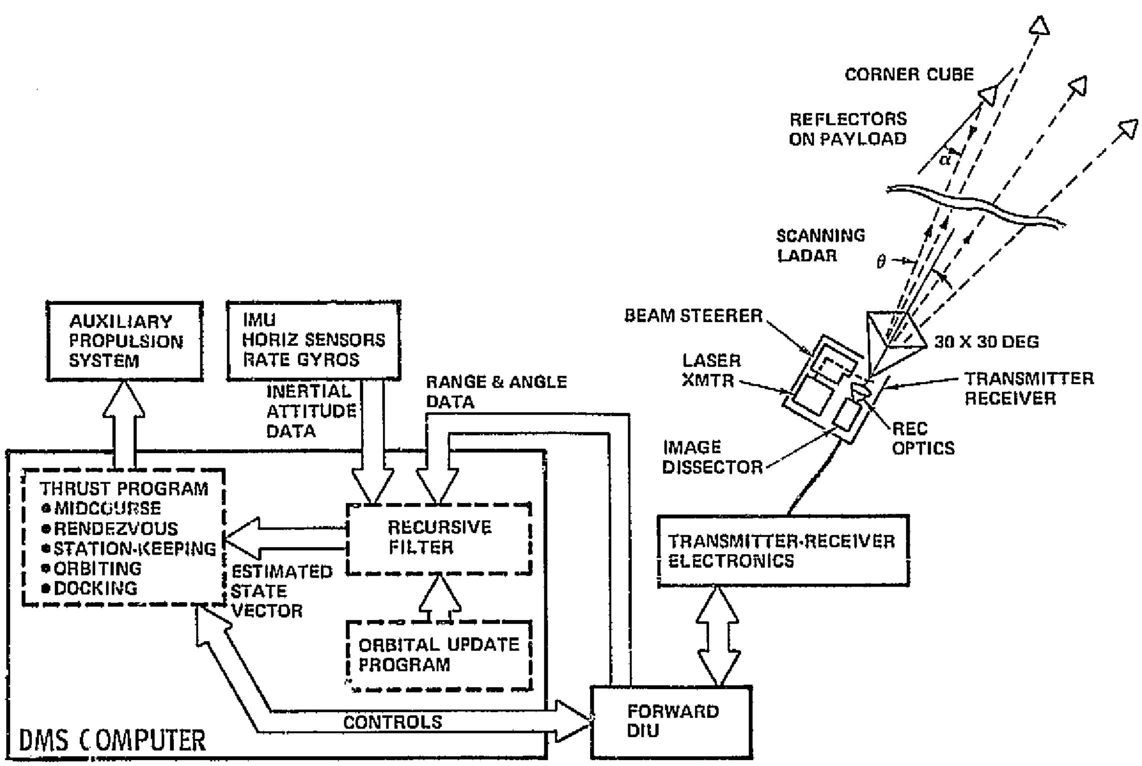


Figure 5-6. Rendezvous and Docking Subsystem
Autonomous GaAs SLR Candidate

Tests of the GaAs ladar under development have demonstrated a temporal resolution of 0.67 μsec in pulse transmit time, corresponding to a range resolution of ±10 cm. The minimum range capability is of the same order for a range only measurement. As discussed below, a further limitation on minimum range is imposed for target attitude measurements.

Assuming an instantaneous field of view of 0.1 degree (0.0017 radian), an acquisition step angle of 0.8 degree (0.014 radian), 376 acquisition steps per line, 64 track steps per line, and a step dwell time of 1 μsec, the maximum angular rate for this sensor is approximately 0.025 deg/sec (0.0004 rad/sec) in the acquisition mode. In track mode, with a 64 × 64 element field of view, the maximum angular rate increases to 1.1 deg/sec (0.019 rad/sec).

The minimum angular rate (on a frame-to-frame basis) is 0.0004 deg/sec (0.6 E-05 rad/sec) for the 140 sec acquisition frame time, and 0.004 deg/sec (0.6 E-04 rad/sec) for the 14 second tracking frame time.

Measurement of target attitude is accomplished by measuring the range to each of three retroreflectors in a T-shaped pattern of known dimensions. A fourth retroreflector is used only in the initial measurement to resolve roll ambiquity. A limitation, in terms of minimum range, is that the entire three-retroreflector pattern must be contained within the field of view to permit measurement of target attitude. For near-normal incidence, the minimum range is given by

$$R_{min} = \frac{2L}{\sin \frac{\theta}{2}}$$

where L is the retro separation and θ is the total field of view angle. The minimum range assuming a retro separation of 1.8m and a 10 degree (0.174 radian) field of view is 9.4m. This could be reduced by increasing the field of view, or by reducing the retro separation. Increasing the field of view beyond about 30 degrees (0.523 radian) is impractical. Reducing the retro separation is at the expense of attitude accuracy, which, on a single measurement basis at near-normal incidence is given by

$$\Delta \theta \approx \frac{\Delta R}{L}$$

In the ITT system, the attitude accuracy is improved by an algorithm that compute rms values of pitch, yaw, and roll angles on a multiple-measurement basis. An accuracy of ±1 degree (0.0174 radian)in attitude has been demonstrated against a stable target. Additional analysis is required to determine the performance of this type of system against a target that possesses angular momentum. In this analysis, the Tug guidance and propulsion system characteristics, in addition to sensor characteristics, must be taken into account, since Tug must orbit a rotating target to keep the retro pattern in the field of view.

5.2.4.2 CO_2 Scanning Ladar. The CO_2 scanning ladar currently under study by Norden employs a pulsed CO_2 gas laser. Two-axis scanning is torque-motor actuated. Heterodyne detection is employed to achieve photon-noise limited performance. A schematic of the system appears in Figure 5-7.

The primary advantages of the CO_2 system over GaAs include the availability of high power CO_2 lasers and considerably better quantum efficiency (0.5 at 10.6 μm as opposed to 9×10^{-3} at 0.9 μm).

Assuming 185 watts peak power (six watts average for 1 μsec pulses at a pulse repetition rate of approximately 30 kHz), the maximum range for the standard conditions listed in Table 5-1 is 150 n.mi. (278 km) in acquisition and 270 n.mi. (500 km) in tracking, comparing favorably with the GaAs system. Considerably greater maximum range capability is possible within current technology; the practical limitation on laser power is essentially one of system weight and electrical power requirements.

Operation in a passive mode would require the use of a cryogenically cooled detector (not required for heterodyne detection in the active mode). Assuming a detector D* of 1.6×10^{11} cm $Hz^{1/2}$ W^{-1}, the maximum range in acquisition is 268 n.mi. (496 km), and 827 n.mi. (1532 km) in the tracking mode, using the range equation for LWIR passive detection presented in Section 5.2.3. Thus either the GaAs or the CO_2 scanning ladars could be configured to provide line-of-sight tracking at very respectable ranges in the event of laser failure, or simply to conserve power during those portions of the mission where line-of-sight information is adequate.

Range resolution for the CO_2 ladar is estimated by Norden to be 4.2m on a single pulse basis and 0.42m averaged (filtered) over 100 pulses. This is more than adequate at long range, but is marginal at close range. Minimum range in the docking mode (see later discussion), based on a criterion of 10% defocussing and assuming a 10 cm, f/3 system, is three meters, although further analysis is required to refine this estimate.

The maximum angular rate, based on a 20% overlap of successive scan lines, as in the case of the GaAs ladar, and an acquisition step time of 3×10^{-5} sec, is approximately 0.1 deg/sec (0.0175 rad/sec). Frame-to-frame minimum angular rate is 0.0003 deg/sec (0.052 mr/sec).in the acquisition mode.

The marginal range resolution capability of the CO_2 ladar makes it impractical to employ an attitude-sensing scheme such as that used in the GaAs system. Norden has proposed a technique that utilizes a circularly scanned low power laser (the local oscillator in the scanning ladar system) in combination with a special circular target on the target spacecraft.

The target consists of two matched but staggered patterns on each side of a transparent substrate. The top pattern is opaque; the bottom pattern is diffusely reflective. The relationship of the pattern grid size and the substrate thickness is such that the

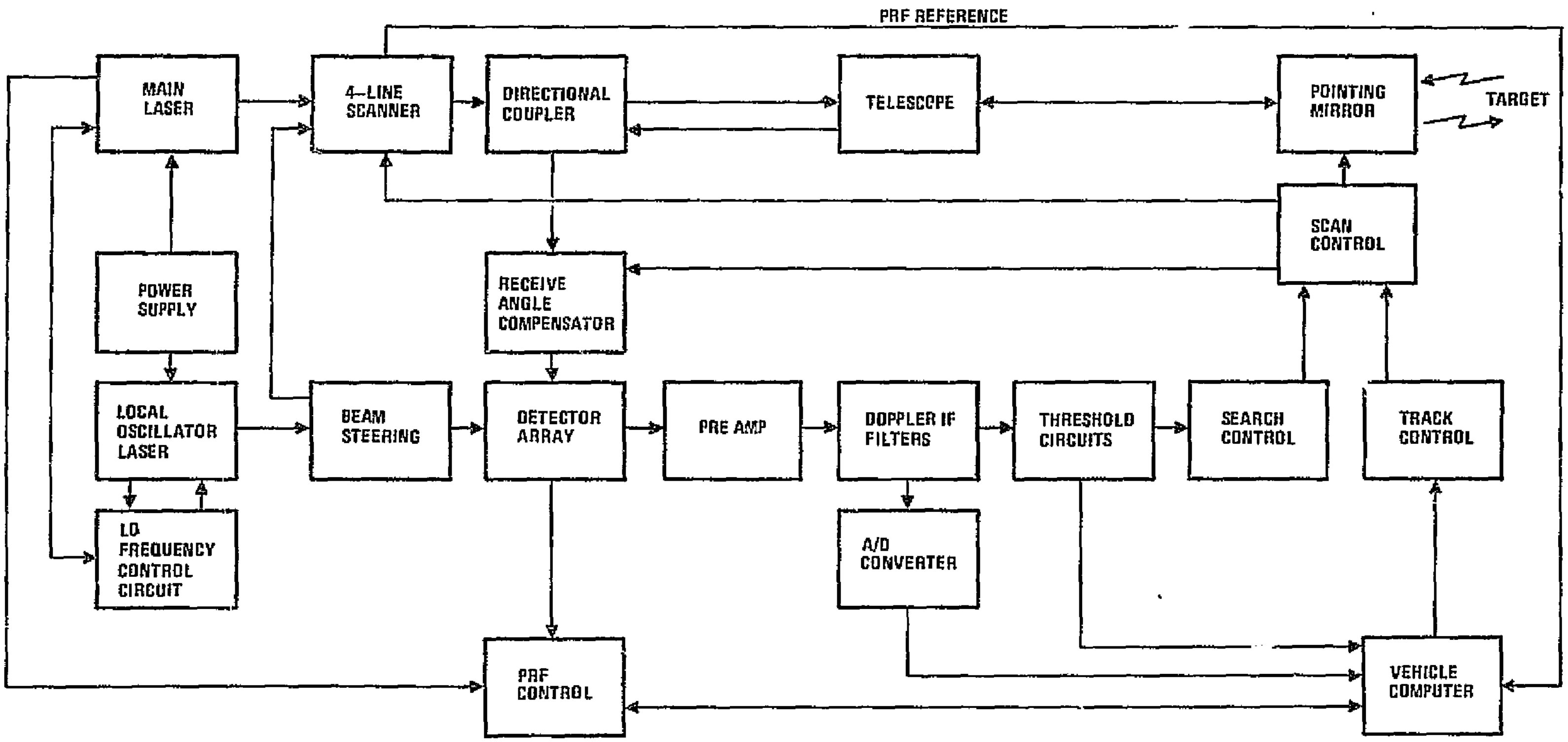


Figure 5-7. CO_2 Ladar

reflectivity is a maximum at normal incidence, falling off with increasing angle of incidence. A second pattern, concentric with the first, consists of a series of annular rings of alternately high and low diffuse reflectivity whose width is varied so that the reflectivity is a linear function of distance from the center of the pattern.

The laser is scanned in a circular pattern over a small section of the target. The signal is approximately sinusoidal; from the relative phase of the signal with respect to the scan angle, the angular position of the misalignment is obtained. Using this error signal, Tug is maneuvered to maximize the reflected return from the central target corresponding to zero relative attitude.

To obtain a range measurement in this mode, the scan cone angle is adjusted until the ratio of the maximum and minimum signals over a scan period is equal to a predetermined value. For a pattern of radius r and a scan circle of radius r_1, the ratio of maximum and minimum signals is:

$$\frac{k\,e^{-a(r-r_1)}}{k\,e^{-a(r+r_1)}} = e^{2ar_1}$$

Adjusting the scan cone angle β to obtain a ratio equal to a predetermined value e^{2ar_0}, the range is then r_0/β where β is the cone angle of the scan.

In principle, this concept appears to be feasible. Further analysis is required, however, to determine the accuracy in range and attitude achievable with this approach. Of particular concern is the effect on its performance with a spinning or tumbling target.

5.2.4.3 High-Frequency Scanning Ladar. High frequency lasers, of considerable interest in the field of high-energy lasers, have undergone considerable development in the past few years. For the Tug application, high frequency offers two advantages over CO_2: higher lasing efficiency and shorter wavelength (2.8 μm as opposed to 10.6 for CO_2). For this reason, a brief analysis was conducted to determine the performance of a system similar to the Norden CO_2 ladar, but operating at 2.8 μm.

To first order, the s/n ratio is inversely proportional to the wavelength for an active system employing retroreflectors; the retroreflector cross section is proportional to the square of the wavelength, and the photo-noise-limited NEP is proportional to the wavelength. Referring to the range equation at the beginning of this section, the range advantage of high frequency over CO_2 (all other factors being equal) is the fourth root of the ratio of wavelengths, or a factor of approximately 1.4. Except for minor design details, the performance number in Tables 5-2 and 5-3 reflect this advantage.

Calculations of performance in a passive mode were based on reflected solar radiation in the 1-3 μm wavelength region.

Detailed analysis of the high frequency candidate was deferred pending comparison of the CO_2 ladar with other candidate sensors. In view of the selection of a combined GaAs/TV system, no further effort was expended to obtain a detailed comparison of HF vs CO_2 ladars.

5.2.4.4 Ranging Ladar. In considering passive sensor candidates with limited capability for measuring range and range rate, a complementary sensor was postulated to perform these functions.

The concept is simply to use a pulsed GaAs emitting diode with a relatively wide, fixed field of view boresighted with the line-of-sight sensor. A large area photodiode, covering the field of view of the emitter, is used for detection.

Using the standard set of conditions for comparison with other sensors, the maximum range in tracking mode is 4 n.mi. (7.4 km), assuming six watts peak transmitted power and a bandwidth of 150 MHz. In practice, the field of view for this sensor could be considerably reduced to allow for line-of-sight errors in passive tracking, boresight error between the ladar and the passive sensor, and the Tug pointing jitter. Based on a 1 × 1 degree (0.174 × 0.174 radian) field of view, the maximum range increases to 22 n.mi. (40.7 km).

Using the processing technique employed for the GaAs ladar system, a range resolution of ±10 cm is achievable. Minimum range is also approximately 10 cm. Since the field of view is fixed, no angular information is provided by this sensor.

5.2.4.5 Tricolor Ladar. The tricolor laser diode docking sensor (Figure 5-8) is typical of the configuring of a candidate to meet specific limitations of other subsystems. It is a simple (hence reliable), light, and low-cost sensor system recently investigated under company funding. It shows promise as a terminal rendezvous and docking sensor supporting LLLTV in a hybrid subsystem. It combines the best features of ladar yet avoids any requirement to scan either the laser beam or the detector. The addition of a self-contained roll detector can extend it to an autonomous candidate.

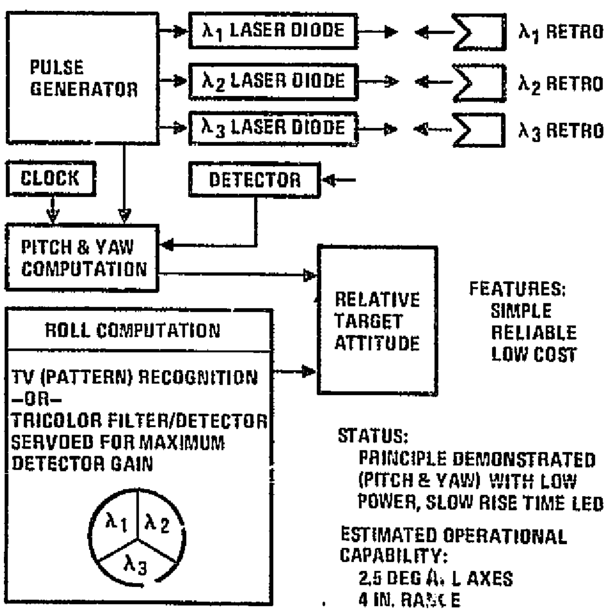


Figure 5-8. Tricolor Laser Diodes Docking Sensor

Based on a peak transmitted power of 1 watt and a bandwidth of 150 MHz, the maximum range for this sensor is estimated to be 1.77 n.mi. (3.28 km), adequate for attitude sensing, but marginal for ranging, if used in combination with a passive line-of-sight sensor. The range resolution, assuming the use of the data processing technique developed for the GaAs ladar, is ±10 cm. Ranging can be accomplished down to a minimum range of about 10 cm.

The minimum range for attitude sensing is limited to approximately 9.4 meters, below which the retroreflector pattern is larger than the field of view. This is a serious drawback for any system using a retroreflector pattern for attitude sensing. One possible solution would be to employ an interferometric technique for the range measurement. The high precision possible with interferometry would permit the use of a much smaller target pattern. However, the complexity of an interferometric system presents a reliability problem, and would involve a costly development program.

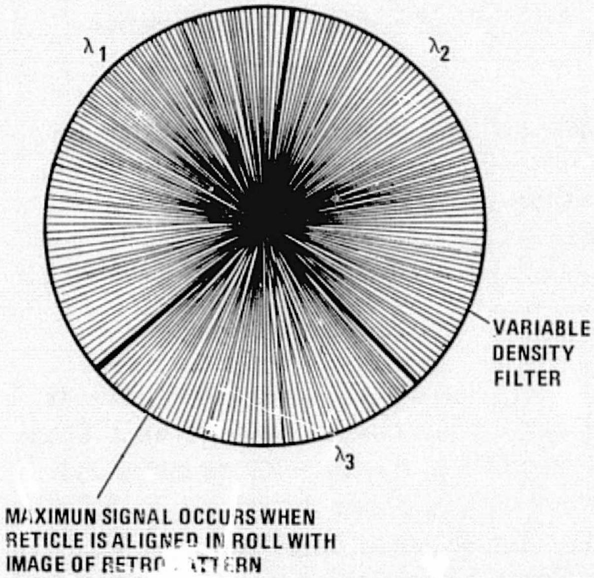


Figure 5-9. Reticle for Determination of Target Roll Angle

The accuracy of the tricolor ladar in measuring target attitude, based on the ratio of range resolution to retro separation distance, is approximately 5 on a single-pulse basis. This can be improved by filtering in a manner similar to that employed by ITT for the GaAs scanning ladar.

The tricolor ladar measurement does not provide roll information. The angle that is measured is the resultant of pitch and yaw. Thus, an independent measurement of the roll angle is required. This can be accomplished with TV (see TV discussion). Alternatively, the retro pattern can be imaged on a three-segment tricolor reticle with a servo loop to rotate the reticle for maximum signal strength. One possible reticle design is shown in Figure 5-9.

5.2.5 TELEVISION. From a system standpoint, the use of television as a rendezvous sensor is quite logical; since it is required for satellite inspection, it can also be used to perform most of the rendezvous functions. A few additional requirements are imposed on the TV system in making it serve a dual purpose, but the basic capability required for satellite inspection is for the most part adequate for rendezvous sensing.

A description of the proposed system is given in Section 5.5. The range equation is given by

$$R^2 = \frac{I \sigma A_R}{s/n\,(NEP)\,L}$$

where I is the solar irradiance on the target within the spectral bandpass of the detector, and the remaining terms are as previously defined. The exoatmospheric solar irradiance in the 0.4 to 0.7 μm wavelength interval is approximately 540 W/m^2. Assuming a 10 m^2 target, 5% of which is diffusely reflecting, the effective target cross section is 0.16 m^2/sr.

Although the frame time is 140 seconds in acquisition and 14 seconds in tracking, the exposure time for the TV system must be restricted to approximately 0.2 second to avoid smearing of the target due to target angular rate and Tug pointing instability. For an integration time of 0.2 seconds, the NEP for a Secondary Electron Conduction (SEC) vidicon is approximately 2×10^{-15} watts. Silicon-Intensifier Target (SIT) performance is better by an order of magnitude.

For the standard conditions listed in Table 5-1, the acquisition range is 1400 n. mi. (2593 km) for a SEC vidicon, and 4110 n. mi. (7612 km) for a SIT vidicon, assuming detector-limited operation. Against an earth background, the LLLTV system would be severely background limited; the use of TV as an acquisition sensor thus would place a constraint on the target approach trajectory to ensure operation against a space background.

Discrimination against stars by observing target angular velocity relative to the stars is impractical because of the long observation times required. However, the number of stars of visual magnitude greater than that of the target contained within the solid angle corresponding to the GN&C system pointing uncertainty is relatively small, and discrimination can be accomplished by comparison of the data field with a star catalog (see discussion in Section 5.3.1).

At long range, where the target is effectively a point source, no range information is available from LLLTV data. The target intensity may fluctuate by as much as a factor of 10^4 due to specular glints, and thus it is not possible to measure its intensity and calculate range from a $1/R^2$ dependence. At short range, where the target can be resolved, the range can be determined from the angular subtense of the target. With man in the loop, using interactive graphics to display a silhouette superimposed on the target image, the accuracy of the range determination will closely approach the ratio of angular resolution to target angular subtense. For example, with an angular resolution of 1 μrad, the range can be determined to approximately 10% for a three meter target at 300 meters. Range rate can be derived by differentiation of the range measurement as a function of time.

For a frame time of 140 seconds and an angular resolution of 0.06 degree (0.0105 radian) the average angular rate can be determined to an accuracy of 0.0004 deg/sec (0.7 mr/sec). Angular rate accuracy corresponding to a 14 second frame time is 0.004 deg/sec (7 mr/sec).

LLLTV can also be used to determine target attitude, employing interactive graphics, with operator control of attitude and size of a simulated target superimposed on the target image. Further study is required to determine the accuracy achievable with this technique as a function of target size and shape. For targets with rotational symmetry, it may be necessary to provide a pattern on the target vehicle to aid in attitude determination.

5.2.6 THE PRACTICALITY OF SOLAR ILLUMINATION OF TARGET SPACECRAFT.

The preceeding investigation illustrated the maximum acquisition and tracking range improvements that can be achieved utilizing solar illumination of the target spacecraft. The requirement for solar illumination constrains both the rendezvous trajectory as well as the surface properties of the target spacecraft (size, shape, and reflectance properties). An analysis was undertaken to evaluate these two constraints for rendezvous missions.

5.2.6.1 Ideal Diffuse Targets. Two ideal cases were initially considered: a cube with sides of area A and a cylinder of radius R and length L. For a flat plate,

$$J = I_s A \rho' \cos \theta_i \cos \theta_r ,$$

where

ρ' is the bidirectional reflectivity (= ρ/π for a diffuse surface)

θ_i is the angle of incidence of solar radiation from the surface normal to the sunline

θ_r is the observation angle, from the surface normal to the line of sight

For two flat plates at right angles (two sides of the cube)

$$J = I_s A \rho' (\cos \theta_i \cos \theta_r + \sin \theta_i \sin \theta_r)$$
$$= I_s A \rho' \cos (\theta_i - \theta_r)$$

The normalized signature of a cube as a function of the angle between the sun line and the Tug-target LOS, $(\theta_i - \theta_r)$, is presented in Figure 5-10. For a cylinder with geometry as depicted in Figure 5-11, the minimum signature occurs when the sun line is normal to either the body axis or the end surface, and depends on the ratio of the length to the diameter. The effect of $(\theta_i - \theta_r)$ is also presented in Figure 5-10 for both cases. Similar plots appear in Figure 5-12 for three different length-to-diameter ratios.

Target intensity is most critical for high-altitude missions (such as geosynchronous) because of the potentially larger dispersions in Tug position. For rendezvous at geosynchronous altitudes, the sun will be approximately in the plane of the trajectory. Referring to Figures 5-10 and 5-12, the acquisition window for a target in a 24-hour orbit is approximately $(\theta_i - \theta_r/360) \times 24$ hours. The minimum intensity within a given window is shown in Figure 5-13 for a cube and for a cylinder with an L/D ratio of 2, based on a solar irradiation of 500 W/m^2 in the 0.4 to 0.7 μm spectral band, and a diffuse white target.

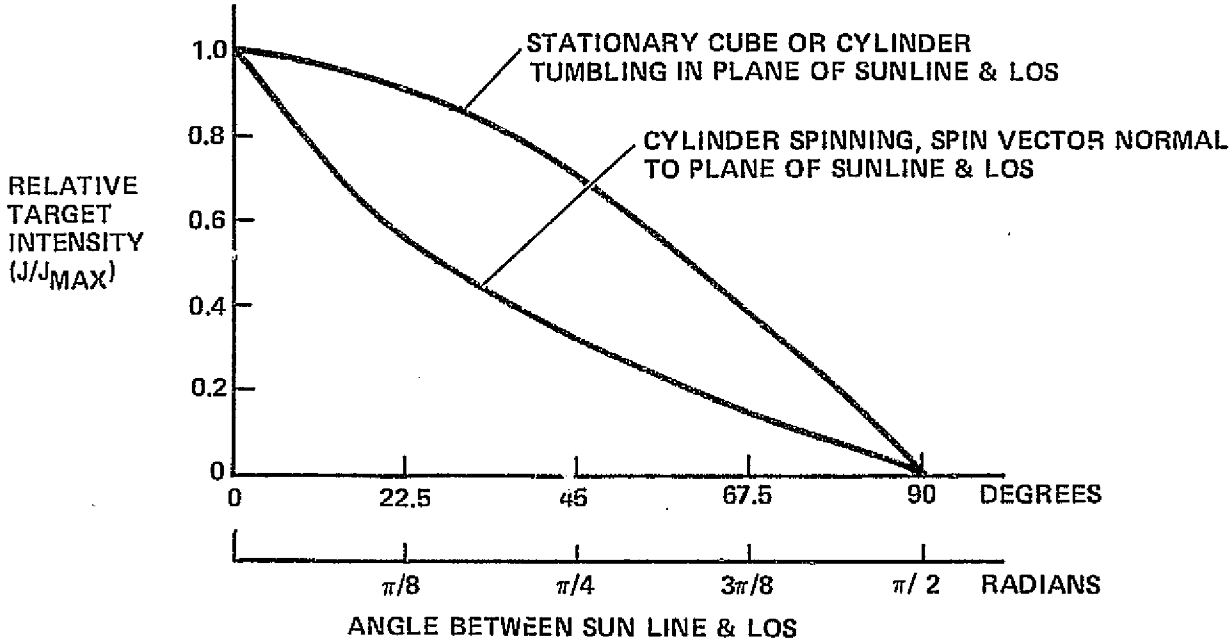


Figure 5-10. Effect of Sun/LOS Angle on Target Intensity

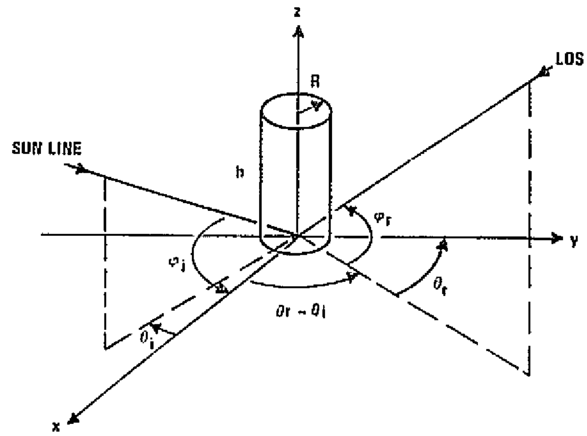


Figure 5-11. Cylindrical Target Geometry

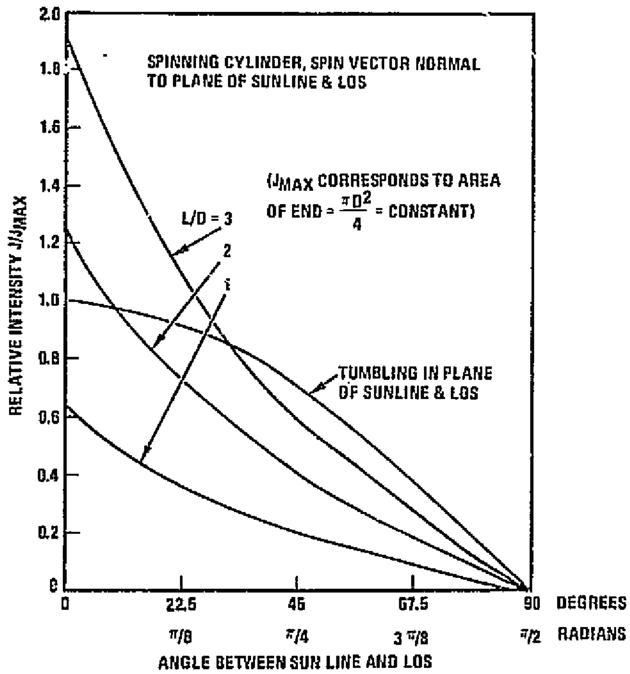


Figure 5-12. Effect of Sun/LOS Angle on Minimum Intensity of Cylinder as a Function of Ratio of Length to Diameter

The nominal target used in calculations of rendezvous sensor performance in preceeding subsections was 10 m^2, with 5% of the surface white and diffusely reflecting, resulting in an intensity of 86 W/sr. From Figure 5-13 it can be seen that a diffuse white cube, 1 m^2 on a side, is equivalent to the nominal target within a window of ±4 hours, and a diffuse white cylinder 1 m^2 on the ends and a length-to-diameter ratio of 2 is equivalent to the nominal target within a window of ±2 hours. These times are quite sufficient for rendezvous trajectories that we have investigated (principally geosynchronous).

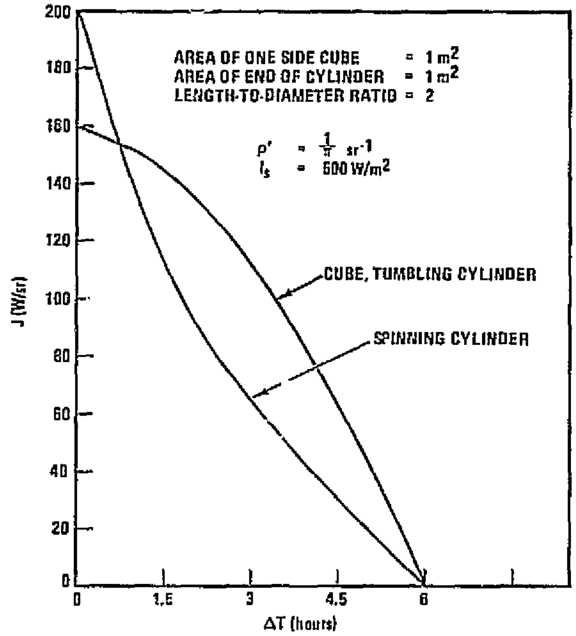


Figure 5-13. Minimum Target Intensity as a Function of $\Delta t = |$Optimum Acquisition Time — Actual Acquisition Time$|$ with Target at Synchronous Altitude

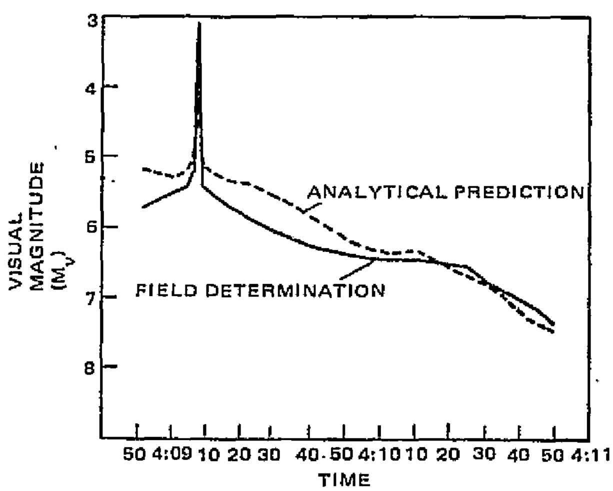


Figure 5-14. Comparison of Field Determination and Analytical Prediction Demonstrates Accuracy of Visual Magnitude Prediction

5.2.6.2 Representative Spacecraft. As an example of the extent to which existing satellites (or future satellites not specifically designed to Tug retrieval requirements) might depart from an ideal diffuse target, the visible signature of the Air Force P72-2 satellite was calculated, first using measured bidirectional reflectivities of the materials actually used in its construction, and then assuming that all surfaces were diffuse (but with the same total reflectance as in the first case).

Convair has developed a sophisticated computer program specifically for calculating satellite signatures. The first major application of this program was to predict the signature of the Apollo Lunar Module (LM) as seen from the Command and Service Module (CSM). Based on our signature calculations for LM (a very complicated shape), NASA personnel calculated the anticipated time of acquisition from the CSM. Actual acquisition occurred within one second of the predicted time. Figure 5-14 is a comparison of the predicted signature and photometric observations of the RADCAT satellite, providing another verification of the accuracy of this prediction technique.

The P72-2 satellite is illustrated in Figure 5-15. Note that an appreciable percentage of the surface is comprised of specular materials.

Figure 5-16 presents plots of the radiant intensity of the target as a function of time for three different sun-target-observer geometries. The target was in a 400 n.mi. (740 km) polar orbit; each pass constituted 1/6 of an orbit.

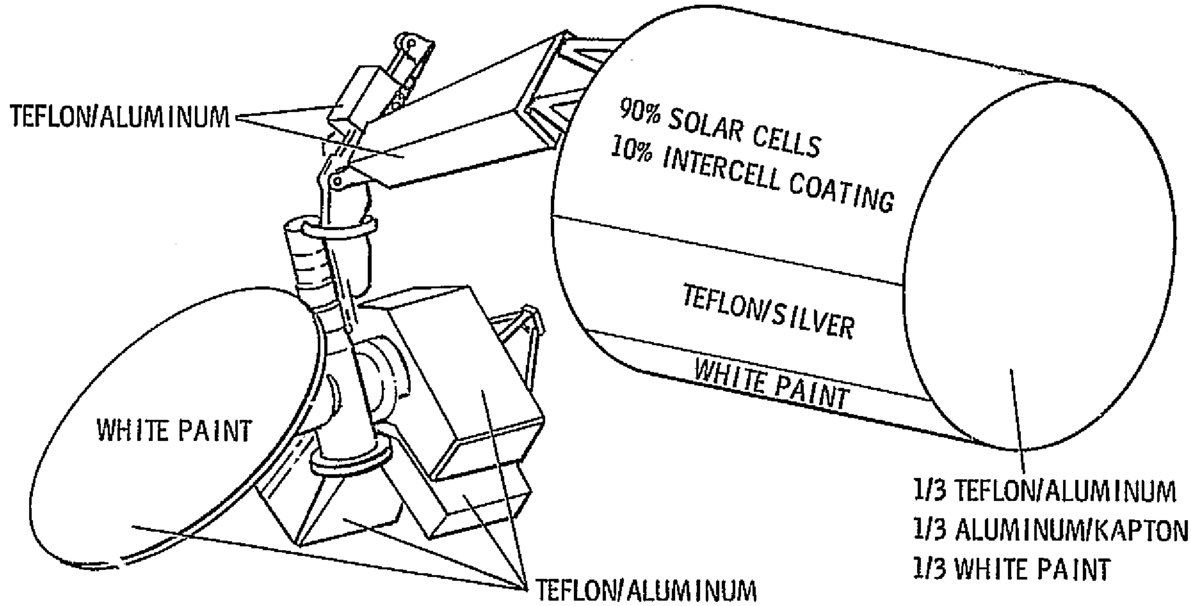


Figure 5-15. P72-2 Spacecraft

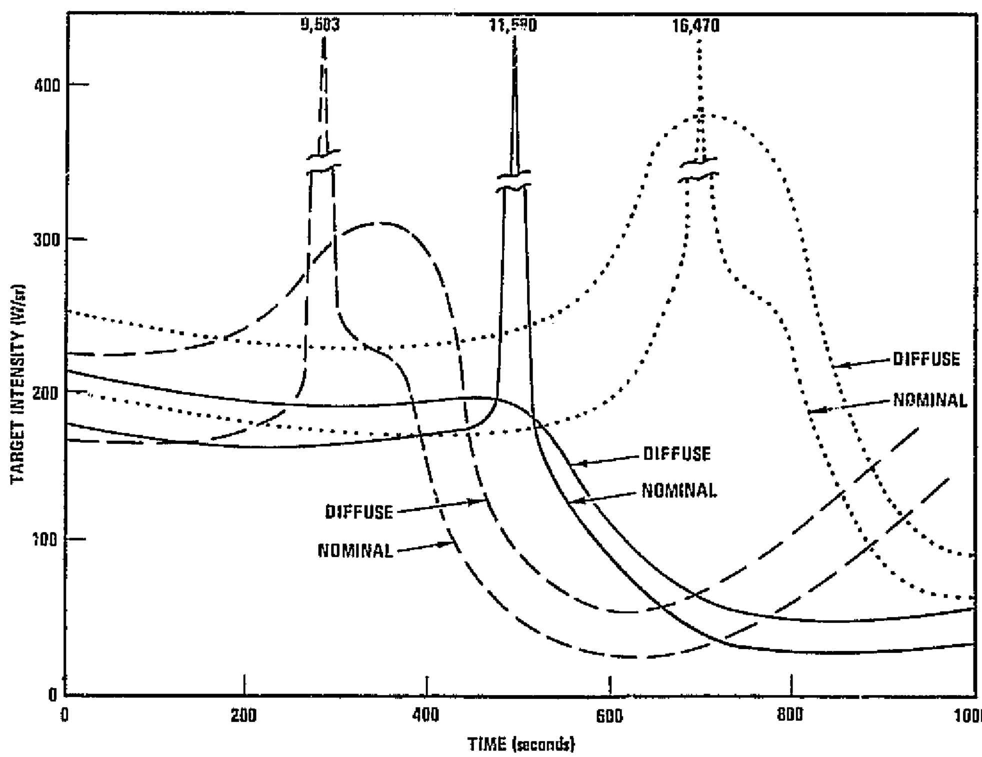


Figure 5-16. Air Force P72-2 Satellite Signature (0.4-0.7μ) as Viewed from Three Different Aspect Angles During 60-degree Pass

It is significant that nowhere is the nominal signature less than 50% of the ideal diffuse signature, despite the high percentage of specular surfaces on the nominal target. The minimum intensity of the satellite as constructed is approximately 25 W/sr. Acquisition range for such a target with either the GaAs SLR or LLLTV sensors would be approximately 1600 n.mi. (3000 km).

For rendezvous at low altitude, the acquisition range requirement will be reduced by at least a factor of 10 because of the much lower Tug dispersion. Using the baseline acquisition sensor, a target intensity of less than 1 W/sr is more than adequate for 300 n.mi. (560 km) acquisition. This corresponds to a diffuse white surface of 0.005 m^2. It would be extremely difficult to design a satellite with such a low signature; there should be virtually no impact on the design of low altitude satellites to satisfy Tug acquisition requirements.

5.2.6.3 Spacecraft Surface Conditioning. For spacecraft of the size typical of those being considered for Tug retrieval, it is quite feasible to design the spacecraft to provide a minimum radiant intensity of 86 W/sr from any aspect within the constraints of thermal control requirements. Thermal control is normally accomplished by the use of diffuse white paints (such as Z-93), with second-surface mirrors, or with aluminized or silvered flexible materials (such as Teflon and Kapton). Frequently, a large percentage of the total surface of a satellite is covered with solar cells.

Using white paint, the area required for a $\rho' A$ product of 10 (0.05) $1/\pi = 0.16\ m^2/sr$, assuming a total reflectance of 0.9 in the 0.4 to 0.7 μm regions, is approximately 0.55 m^2. This is not a stringent requirement; the majority of satellites utilizing white paints for thermal control will meet this requirement automatically.

In the case of spacecraft using second-surface mirrors, the situation is different. Second-surface mirrors currently in use are specularly reflecting, and can be characterized by an effective $\rho' = 10^4\ sr^{-1}$ (corresponding to the reciprocal of the solid angle subtended by the sun), but only at the angle of specular reflectance. Thus second-surface mirrors will produce glints of high intensity — 500 W/sr per cm^2 of surface — at the specular angle, but essentially zero elsewhere.

In the special case where the satellite is roughly spherical, and the radius of curvature is such that adjacent mirrors are canted at an angle of less than 0.5 degree, the signature will be essentially the same as for a diffuse target of the same size and shape. For cylindrical or rectangular geometry, however, the signature is highly dependent on target attitude and cannot be counted on to provide a continuous source. Some modification of the optical properties of second surface mirrors will therefore be required. The situation is similar for aluminized flexible materials.

Convair has recently completed a program under SAMSO funding to develop diffusely reflecting second-surface mirros and aluminized flexible materials, while at the same time maintaining thermal performance (less than 1% degradation in solar reflectance).

The development effort was successful, and the process is quite inexpensive. In quantity, diffusely reflecting second-surface mirrors can be produced at a lower cost than for the conventional type. Diffusely reflecting mirrors of the Teflon/aluminum type can be produced at a slight premium over conventional costs. Approximately 0.5 m^2 of diffuse second-surface mirrors are required for the nominal signature.

The Air Force is currently investigating techniques for achieving diffuse reflectance from solar cells. Convair is participating in these investigations. Based on the success with second-surface mirrors, and the similarities in construction of solar cells and second-surface mirrors, we believe that diffusely reflecting solar cells will be well within the state of the art by 1978. It is also quite likely that such cells will be slightly more efficient (by 1 or 2%) than specularly reflecting cells. The total reflectance of solar cells in the 0.4 to 0.7μm region is approximately 0.2. Assuming diffuse reflectance, a solar array area of 2.5 m^2 will be equivalent to the nominal target.

5.2.6.4 Summary. Thus, for cooperative targets, the thermal control designer will have a number of options. An effectively diffuse target with a radiant intensity of 86 W/sr can be achieved (assuming a specular surface and roughly spherical geometry) by providing 0.55 m^2 of diffuse white paint, 0.5 m^2 of diffuse second-surface mirrors, 2.5 m^2 of diffuse solar cells, or any appropriate combination thereof.

In those very few cases where mission constraints preclude a thermal control system with these characteristics, a less efficient (from the standpoint of performance) rendezvous maneuver may be required, based on a reduced acquisition range.

It should be noted in either event that the vicinity of the spacecraft docking port must be surface conditioned for the application of scanning ladars or LLLTV to avoid the problems of spurious specular reflections from adjacent surfaces.

Other than the simplified approach taken in Figure 5-13, no analysis has been conducted to evaluate the impact of solar illumination on orbital operations.

5.3 GN&C SUBSYSTEM CAPABILITY ASSESSMENT AND CANDIDATE SUBSYSTEMS SELECTION

The objectives of this system-level trade were to evaluate the contribution that the GN&C subsystem can make to the Rendezvous & Docking subsystem, and to further narrow the selection to a single autonomous subsystem candidate and a single remote-manned subsystem candidate.

The candidate options are those that were defined within the sensor options trade. The selection criteria in this section are principally performance adequacy, followed by reliability, weight, cost, and power (in ranked order) as derived in preceding section. This section principally deals with establishing performance requirements and assessing the GN&C subsystem.

C-2

5.3.1 RENDEZVOUS AND DOCKING, DIRECT ASCENT APPROACH. The initial rendezvous of a retrieval or servicing mission may be accomplished either by direct ascent (insertion in close proximity to the target spacecraft) or by the phasing approach (insertion into a — usually lower altitude — phasing orbit and drift into proximity of the spacecraft). A direct ascent rendezvous is near optimum in impulse and time, and is the more efficient choice, if it can be demonstrated. The maneuvers (Figure 5-17) consist of an injection into an elliptical phasing orbit with perigee at the Shuttle orbit and apogee adjusted to provide the required phasing with the orbiting spacecraft and accomplish this phasing in the minimum time (viz., at the lowest altitude). The phasing orbit will also generally include a plane change. Not shown on the figure are the small midcourse corrections required to trim the phasing orbit.

Ideally, injection into the primary ascent ellipse occurs at perigee (the node) of the phasing orbit and generally includes a plane change. Midcourse corrections tend to cluster toward the end of the transfer to minimize the position (phase) uncertainty at the insertion point.

Insertion into the target spacecraft's orbit ideally occurs at apogee of the perturbed transfer ellipse and consist of the velocity required to match the spacecraft velocity at this point. Insertion should occur in close proximity to the spacecraft to minimize the time/impulse required to close to within inspection distances. This, in turn, requires precise navigation to ascertain Tug's current state and precise guidance to

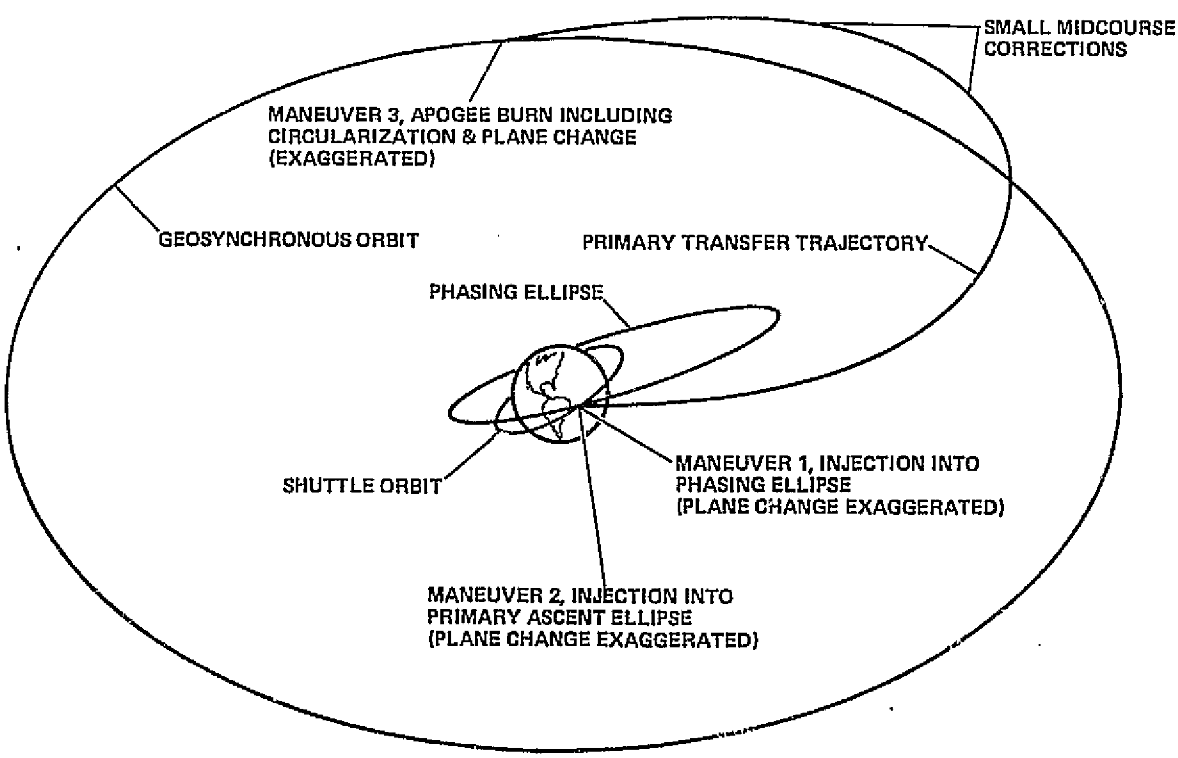


Figure 5-17. Gross Rendezvous Mission Profile (Direct Ascent to Geosynchronous Orbit)

match the state of the target spacecraft at the conclusion of the insertion burn. If this could be accomplished exactly — that is, if rendezvous with the assumed target position could be accomplished without error — the remaining (position) uncertainty would be that uncertainty of the target spacecraft orbit, viz., 1.0 n.mi. (1.95 km) spherical radius at geosynchronous altitude. Such dispersions are well within the capabilities of docking sensors investigated in the previous section. The extent that this can be accomplished will determine whether a short, medium, or long range rendezvous sensor would be required and whether an insertion into an intermediate phasing orbit would be necessary.

Figure 5-18 illustrates selected target-related trajectory parameters, on a direct ascent rendezvous to geosynchronous altitude including expected dispersions (Section 2). Superimposed on the figure is the tracking performance of the candidate sensors. Recall that, in track, the sensors provide only line of sight (LOS) data. Range data — to be of use — must occur prior to the reorientation maneuver for the insertion burn, which occurs approximately 250 n.mi. (460 km) from the target. Further, as is apparent from the dispersion regions noted in the figure, just prior to the insertion burn is actually too late to effect a correction of the insertion target point, so that range data would be required, at even larger ranges. Only the gas laser ladars are capable of this extended performance. What is required is early measurement of the LOS, as afforded by the GaAs SLR or LLLTV sensors with a solar illuminated spacecraft.

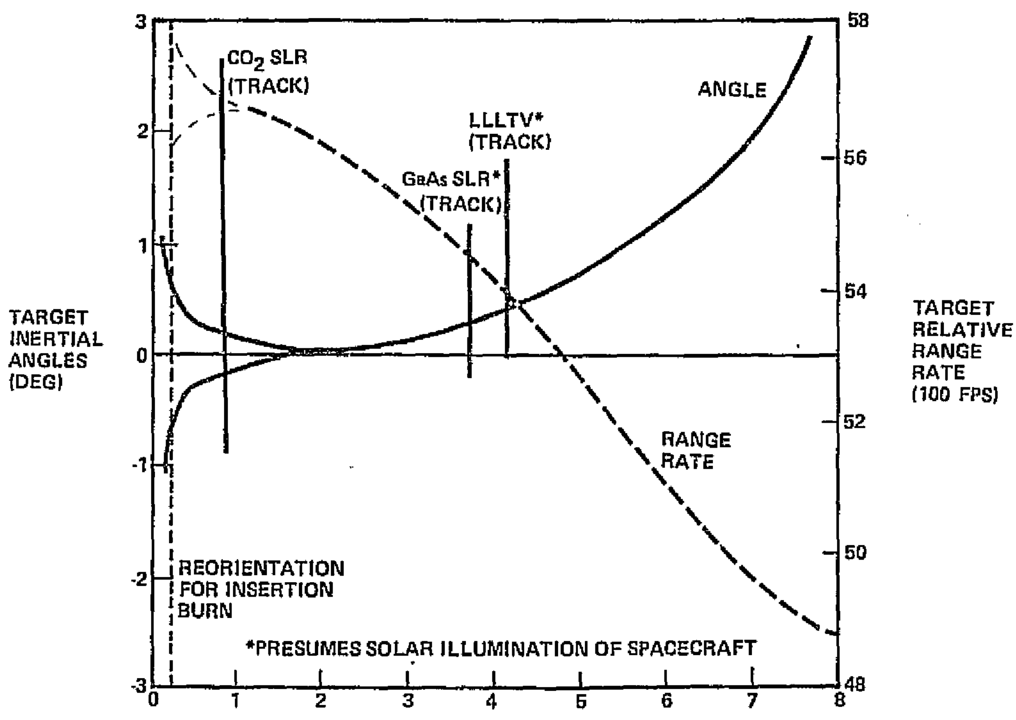


Figure 5-18. Selected Target Relative Trajectory Parameters, Direct Ascent to Orbit Insertion (Geosynchronous Mission)

Figure 5-19 illustrates LOS accuracy available versus relative range to the target spacecraft. The knowledge of the target spacecraft's position is fixed at 1.0 n.mi. (1.85 km) spherical radius. At large distances this constitutes an excellent knowledge of the LOS from Tug to spacecraft, even with Tug's navigation uncertainty in its own position included. (See Table 2-3 of Section 2 for justification of the 1.5 n.mi. (2.8 km) 3σ navigation performance, utilizing the baselined Interferometric Landmark Tracker update sensor, which is incorporated in Figure 5-19.) Since this onboard knowledge of LOS degrades inversely with relative range, the measurement accuracy available from the candidate long range tracking sensors could improve onboard knwoledge at ranges less than 2500 n.mi. (4630 km). Either of the long range tracking sensors (employing solar illumination of the spacecraft) could lock on before the tracking uncertainty had degraded beyond 0.06 degree (0.001 radian). Thus, 0.06 degree (0.001 radian) is an upper limit of tracking uncertainty providing that the tracking sensor can discriminate the target from its background.

Discriminating the target from its stellar background requires knowledge of the location of the stars within the tracking sensor's field of view (FOV). Restricting the observed FOV can result in a star catalog of moderate size (Figure 5-20). The limited pointing accuracy is that of the Tug attitude control system in "fine" mode and is 0.1 degree (0.0017 radian); that is, a 0.2 degree (0.0034 radian) FOV would suffice at the maximum tracking range of the candidate sensors (preceding figure). A star catalog to +9 MV for a 3 x 3 degree (0.052 x 0.052 radian) (dispersed) FOV would result in a storage requirement of less than 70 words. Even smaller catalogs would suffice if uploaded via ground stations.

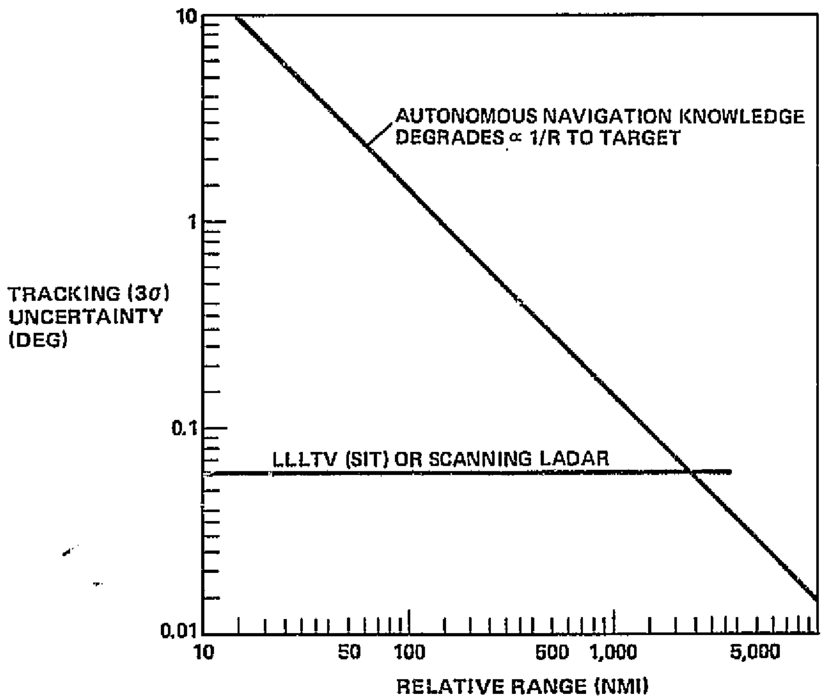


Figure 5-19. Tracking Uncertainty

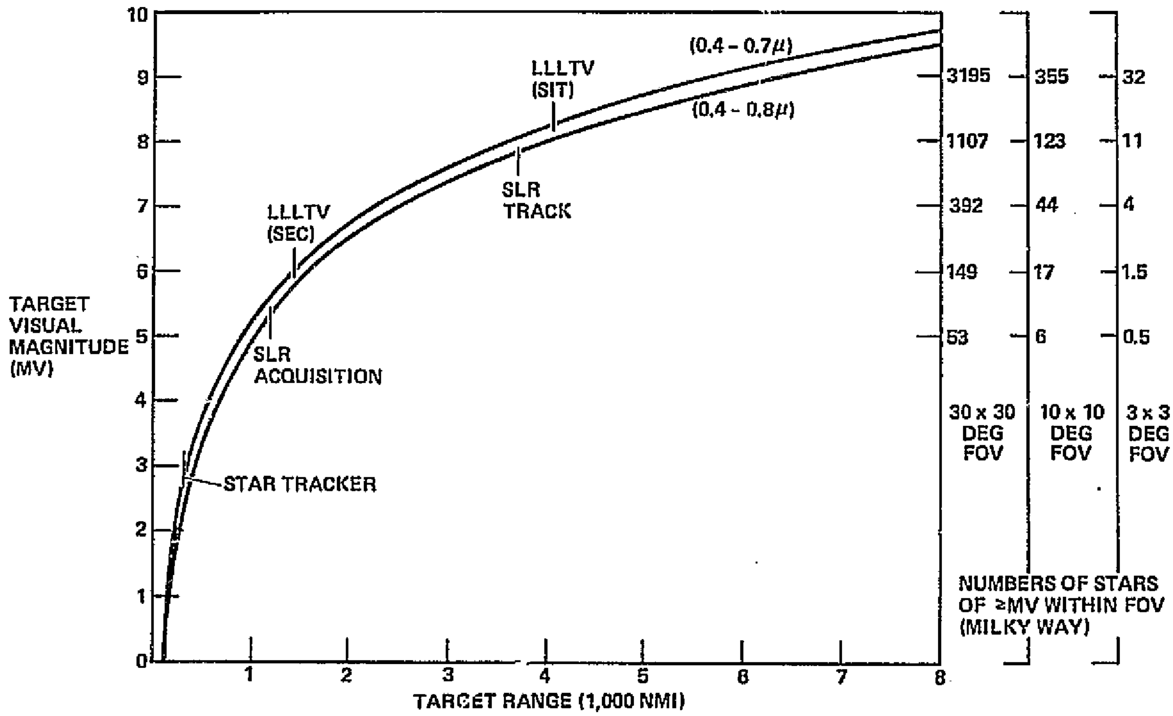


Figure 5-20. Long-Range Tracking Approach

A star tracker is included in the baseline GN&C subsystem and its performance was investigated as noted on the figure. Since this device is only meant to track bright stars (brighter than +3 MV). it is of little use in long-range tracking.

It was concluded that the requirement for spacecraft acquisition can be circumvented by pointing the tracking sensor (Tug) at the spacecraft and discriminating between the spacecraft and its background via a star catalog. The navigation subsystem has the required accuracy for this pointing (Figure 5-19).

Figure 5-21 illustrates how knowledge of LOS, when added to the navigation Kalman filter, can improve overall knowledge of the target relative orbit including reduction in range uncertainty. A conceptual explanation follows.

The ±0.06 degree (0.001 radian) sighting uncertainty represents a sighting error cone with a 0.06 degree (0.001 radian) half angle. Since range information is lacking, the full error volume represents the target uncertainty volume for ranges exceeding the maximum tracking range of the sensor (i.e., the quoted ranges are conservative). However, knowledge of orbital mechanics and the approximate apex of the error cone allows conic propagation of the cone to the next sighting point. Thus, it is the intersection of the two error cones that represents the uncertainty at t_1, greatly reducing the range uncertainty.

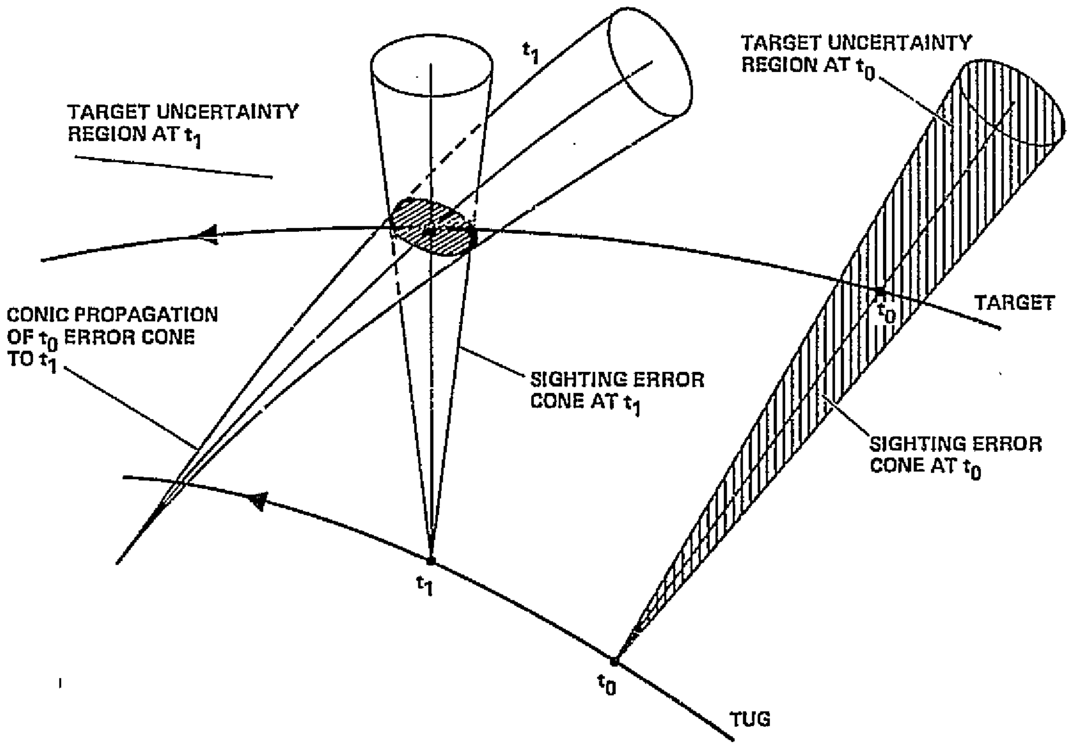


Figure 5-21. A Conceptual Explanation of Range Determination Via Kalman Filtering

In the actual application, it is more than LOS information that is being fed into the navigation Kalman filter, ensuring rapid convergence of the knowledge of the relative motion. This is summarized in Figure 5-22, which represents an extension to the analysis.

Continuous guidance and numerous — but small — midcourse corrections (APS propulsion) can provide insertion to within 2.8 n.mi. (5.2 km) of the aiming point. Since the target is known to within 1.0 n.mi. (1.95 km), this insertion accuracy is equivalent to within 3.0 n.mi. (5.6 km) of the target. It is estimated (simulation results were incomplete) that the addition of LOS angles into the navigation Kalman filter will result in an insertion on the order of 1.0 n.mi. (1.85 km).

Accurate knowledge of insertion position can minimize intentional biases (to avoid searching 4π steradians) and the time/impulse required to close to within inspection distances. Once insertion to within close proximity is accomplished, insertion velocity dispersions are quickly and accurately removed by nulling out LOS angular rates and establishing a suitable closure velocity with the spacecraft.

Ranging is not easily accomplished with the candidate sensors (Figure 5-23). If a direct ascent approach is employed, reorientation for the orbital injection burn must be accomplished around 250 n.mi. (463 km). Only the high powered gas ladars (CO_2 and high frequency) are ensured this capability, although MSFC is investigating extending the GaAs SLR to this range.

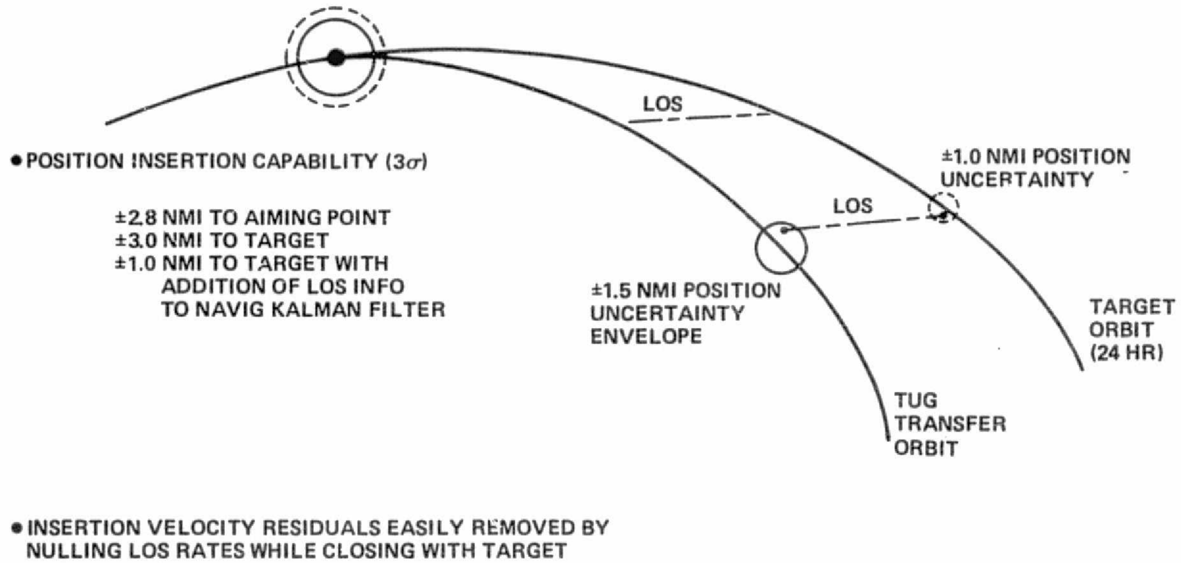


Figure 5-22. Autonomous Navigation Rendezvous Capability (Direct Ascent to Geosynchronous Altitude)

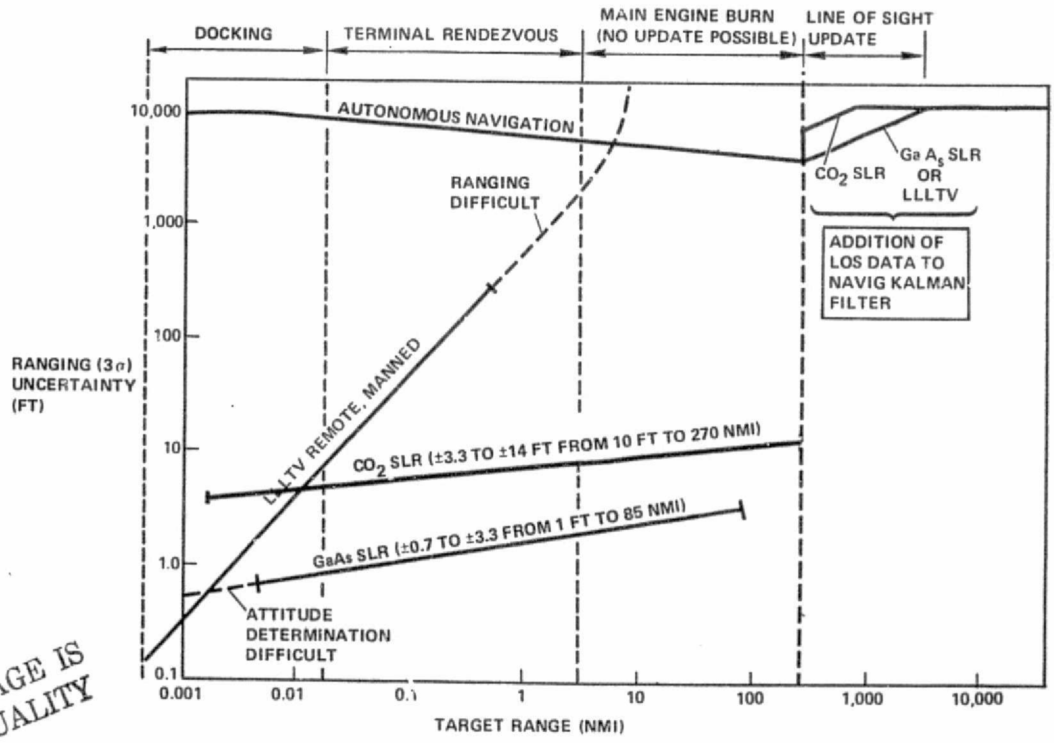


Figure 5-23. Sensor Capabilities Versus Target Range

After insertion, all of the candidate sensor can range — but to varying accuracy. In particular, LLLTV is only marginally adequate in this application.

Note that the autonomous navigation capability in itself is not adequate for terminal rendezvous. The navigation improvement shown results from adding LOS measurements into the navigation Kalman filter; if range information were also available, insertion accuracy could safely support placement of Tug within 3000 feet (914 meters) of the target.

The SLRs provide excellent terminal rendezvous sensors but are somewhat lacking when it comes to docking. CO_2 can approach within 10 feet (3.05 meters) but suffers in accuracy unless digital filtering is employed within the unit itself (i.e., at the pulsing rate). GaAs has sufficient accuracy but is presently limited to 31 feet (9.14 meters) due to an inability to retain all three retroreflectors with its restricted FOV, 20 degrees (9.35 radians); ranging can continue to final docking if attitude determination is foregone.

Conversely, LLLTV is marginally adequate for terminal rendezvous but potentially excellent when it comes to docking. The performance illustrated is based upon a variable 2 to 20 degrees (0.035 to 0.35 radian) FOV (zoom or turret lens) and cannot be much improved. Simulation studies are required to evaluate LLLTVs applicability to terminal rendezvous and docking (see Section 5.4).

It was concluded that the GN&C Subsystem can insert within 3.0 n.mi. (5.6 km) of a target-relative aiming point unassisted, and within 1.0 n.mi. (1.85 km) of a target-relative aiming point utilizing LOS measurement from 2500 n.mi. (4630 km) to the initiation of the insertion (main engine) burn. (These LOS measurements presume solar illumination of the target.) Either approach obviates the requirement to obtain range prior to insertion on direct ascent; the candidate sensors can all supply post-insertion range but to considerably varying accuracies.

5.3.2 RENDEZVOUS AND DOCKING, PHASING ORBIT APPROACH. Although the preceding analysis has shown that direct ascent rendezvous is preferred (to an injection into a phasing orbit) on an initial rendezvous, a subsequent rendezvous — such as would be required on a placement/retrieval mission or a servicing sortie — generally requires phasing to another spacecraft in the same (or nearly the same) orbit. This presents a considerably different approach profile.

Figure 5-24 illustrates a simplified view of the terminal rendezvous profile utilizing a phasing orbit approach. Both circular and elliptical phasing are shown, the difference being only whether a circularization burn is executed at perigee of elliptical phasing orbit (lower point 2 of the figure). The principal advantage of circular phasing is that the approach is at a constant relative altitude difference, eliminating any requirement to adjust the orbit period so that the reinjection point (point 4 of the figure) occurs near apogee of the phasing ellipse. The principal advantage of elliptical phasing is that it provides minimum impulse phasing in minimum time when phasing through

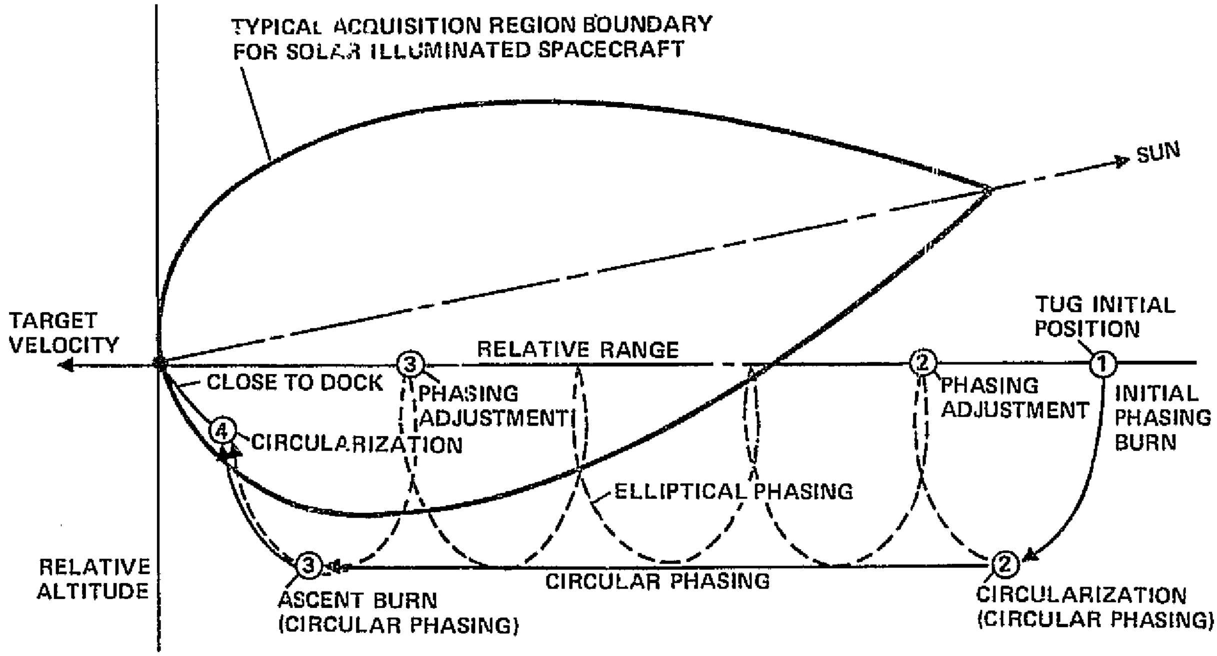


Figure 5-24. Terminal Rendezvous Profile, Phasing Orbit Approach

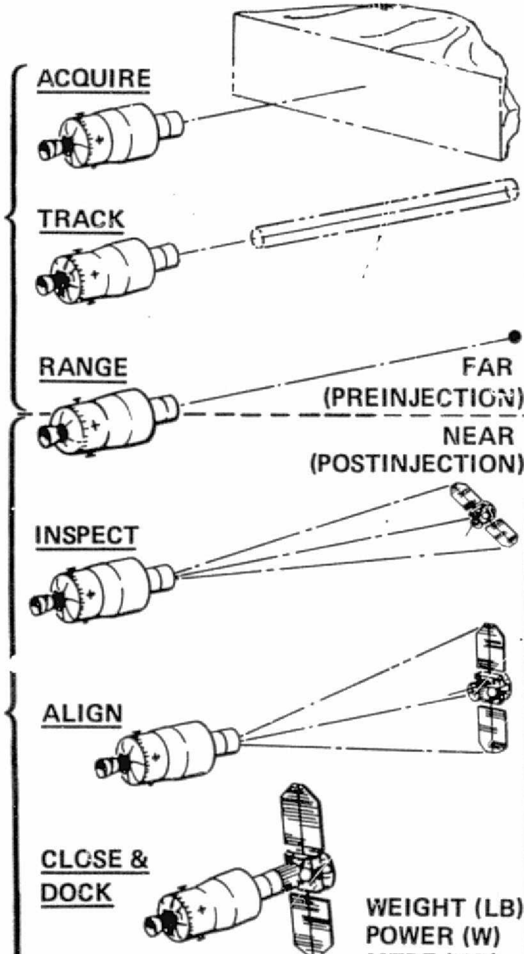
large angles, viz., through several complete orbits (five orbits — that is, five days at geosynchronous altitude — are shown on the figure).

The selection of circular or elliptical phasing in a specific circumstance depends on the initial phase angle and whether time or ΔV is more critical. If the time constraint can tolerate several phasing orbits, elliptical phasing will generally result in a smaller ΔV.

In either event, the initiation of the insertion burn (point 4 on the figure) can be accomplished with navigation information only, with navigation together with LOS measurements added to the navigation Kalman filter (as before), or with navigation information together with LOS and range obtained from a scanning ladar. This latter case is possible because the approach velocities are much reduced over the direct ascent case (particularly with elliptical phasing, which theoretically rendezvous at apogee) and the ΔV burn required correspondingly much shorter.

Due to the long coast and small burn required in either case, the GN&C insertion capability will be correspondingly better than that for direct ascent insertion and is sufficient in itself. Thus the direct ascent mission profile provides the driving requirements for subsystem operation.

5.3.3 SUMMARY AND CANDIDATE SUBSYSTEM SELECTION. Figure 5-25 is an updated summary of the candidate sensors (including their generic derivatives) that were evaluated for each of the six functional phases.



APPLICABILITY TO EACH PHASE

		STAR TRACKER (8 x 8 DEG)	SCANNING** LADAR (30 x 30 DEG)	LADAR** (10 x 10 DEG)	SLOW SCAN LLLTV (10 x 10 DEG)	TRICOLOR LADAR** (10 x 10 DEG)	AUTONOMOUS NAVIGATION CAPABILITY
RENDEZVOUS	ACQUIRE	POOR* (330 NMI)	FAIR* (1,200 NMI)	NA	GOOD* (4,100 NMI)	NA	EXCELLENT (UNLIMITED) ✔
	TRACK	POOR*	GOOD* (3,700 NMI) ✓	NA	GOOD* (4,100 NMI) ✓	NA	FAIR (DEGRADES ≈ i/R)
	RANGE — FAR (PREINJECTION)	NA	POOR (TO 85 NMI)	POOR (TO 10 NMI)	NA	NA	EXCELLENT (±3.0 NMI) ✔
DOCKING	NEAR (POSTINJECTION)	NA	EXCELLENT (±3 FT UP TO 85 NMI) ✓	EXCELLENT (±3 FT UP TO 10 NMI) ✓	FAIR (ACCURACY f(R) TO 3 NMI)	EXCELLENT (±3 FT UP TO 4 NMI) ✓	POOR (±3.0 NMI)
	INSPECT	NA	NA	NA	EXCELLENT ✔	NA	NA
	ALIGN	NA	POOR (PATTERN RECOG-NITION & TRANSFER)	NA	EXCELLENT ✔	NA; GOOD(±4 IN. TO 1 FT & ±2.5 DEG TO 31 FT)	NA
	CLOSE & DOCK	NA	GOOD (±4 IN. TO 1 FT & ±1 DEG TO 31 FT) ✓	NA; EXCELLENT (±4 IN. TO 1 FT) ✓	GOOD (SLOW) ✓; EXCELLENT (±4 IN. TO 1 FT) ✓		NA
	WEIGHT (LB)	NA	39	20	9	15	NA
	POWER (W)	NA	40	40	8	40	NA
	MTBF (HR)	NA	7,000	10,000	15,000	15,000	NA
	COST	NA	HIGH	MEDIUM	VERY LOW	MEDIUM	NA

*PRESUMES SOLAR ILLUMINATION OF SPACECRAFT
**REQUIRES RETROREFLECTORS ON SPACECRAFT

✔ SELECTED CANDIDATE
✓ POSSIBLE CANDIDATE

Figure 5-25. Sensor Options Study Summary Update

The navigation subsystem's star tracker was added as a candidate since it is on board and essentially a no-cost system in terms of dollars, weight, power, etc., and can also provide spacecraft position (which appears as a star on a star background). Unfortunately, its performance is too poor as a competitive candidate.

LLLTV must be a part of the Rendezvous and Docking Subsystem since it is the only candidate for inspection, and the only reasonable candidate for docking port search and alignment.

Autonomous navigation capability is essential for the acquisition phase and will be utilized throughout the rendezvous phase. In this respect, the tracking sensor provides additional input to the navigation (Kalman) filter enabling insertion in closer proximity to the spacecraft. Both the Scanning Ladar and LLLTV can provide this capability.

None of the candidate sensors remaining from the initial screening can provide preinjection range for the direct ascent rendezvous due to the required range (greater than 300 n.mi. or 560 km). However, autonomous navigation can provide insertion to within 3 n.mi. (5.6 km) alone, and to within 1 n.mi. (1.9 km) employing the tracking sensor (Figure 5-22). Hence range is only required postinjection. Any of the three Ladars can provide this capability.

The Scanning Ladar provides the inherent capability for autonomous docking, although insufficient analysis and testing have been accomplished to confirm its capabilities. LLLTV alone is believed to enable docking through a remotely situated supervisor, although this remains to be demonstrated (see Section 5.4). Its principal flaw is being — of necessity — very slow; coupling with either the non-scanning Ladar or the Tricolor Ladar ranging sensor greatly improves its performance.

It is concluded that the GaAs SLR and LLLTV, the Ladar and LLLTV, or the Tricolor Ladar ranging sensor and LLLTV can accomplish the mission. If insertion accuracies to less than one mile (1.9 km) can be achieved employing LOS, LLLTV alone might suffice.

Note however that of remaining candidates, only the GaAs Scanning Ladar and its generic equivalent, the Tricolor Ladar docking sensor, provide the possibility for fully autonomous operation. Since the GaAs SLR is currently in development, it is the obvious choice. What remains then is to evaluate a fully autonomous subsystem (utilizing GaAs SLR) and a remote-manned subsystem (utilizing the LLLTV sensor) — principally in the docking phase of the mission (see Figure 5-25) — to ascertain their adequacy. This is a prerequisite to final selection of the baseline subsystem.

5.4 REMOTE-MANNED RENDEZVOUS AND DOCKING FEASIBILITY DEMONSTRATIONS

The development of a simulation for remote-manned rendezvous and docking utilizing slow-scan, low light level TV (LLLTV) was conceived and initiated as a company-funded indepndent research and development (IRAD) effort in mid-1974 in response to separate technical discussions held with both NASA (MSFC) and Air Force (SAMSO) personnel. The IRAD's (10 month) effort was limited to modification, checkout, and validation of simulator hardware specific to this task in Convair's Visual Simulator Laboratory and to the development of simulation (digital computer) software.

The docking feasibility demonstration (Section 5.4.7) for the Space Tug Avionics Definition Study (NAS8-31010) was accomplished using this facility.

5.4.1 OVERVIEW. Figure 5-26 illustrates the simulation study area. The docking supervisor's console appears in the background, the test conductor's console in the foreground, and a teletype is located between these two consoles for communication with the remotely located simulation (digital) computer.

The test conductor's console has three television displays. The one in the center is identical to that on the docking supervisor's console. On the left is a continuous space view as would be seen by Tug via a continuous (rather than frame-by-frame) TV camera. On the right is a wing view camera designed to prevent collision between Tug's camera (on a carriage riding on rails) and the sting-mounted spacecraft.

The study area is located in a quiet area with an observation area behind floor-to-ceiling glass walls. Throughout simulation runs, the console operator has his back to the displays on the test conductor's console, hence avoiding false cues.

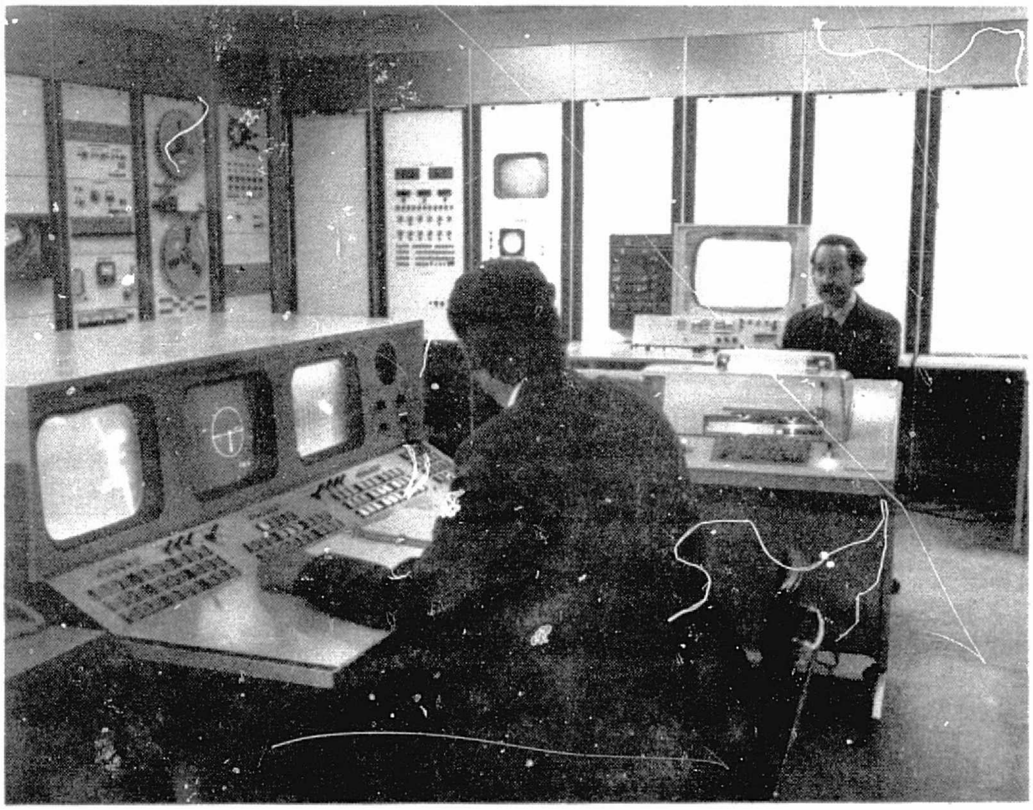


Figure 5-26 Convair's Rendezvous and Docking Simulation Study Area

The novel LLLTV sensor system envisioned for manned, remote rendezvous and docking is more like a still camera than a broadcast TV (Figure 5-27). A single "still" frame is "exposed" by an electronic shutter (grid gate) and slow-scanned (destructively read) as it is being sent to the ground. This substantially reduces transmission bandwidth and is well within the requirements established by manned, remote operations 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.

There are no technical concerns associated with the LLLTV sensor system. Analysis has shown that LLLTV, such as would be required for docking inspection, could provide a suitable acquisition sensor for either an autonomous or remote manned rendezvous/docking subsystem. A manned, remote simulation has demonstrated its feasibility during the rendezvous (Section 5.4.6) and docking (Section 5.4.7) phases. Should it prove fully adequate in subsequent detailed simulation studies and tests, an order-of magnitude savings can be obtained for early rendezvous and docking operations.

5.4.2 <u>OPERATIONAL SCENARIO</u>. An actual scenario of the terminal phase of remote, manned rendezvous and docking might be as follows: at an assigned time, the supervisor situates himself at a TV monitor or large video projected display, such as

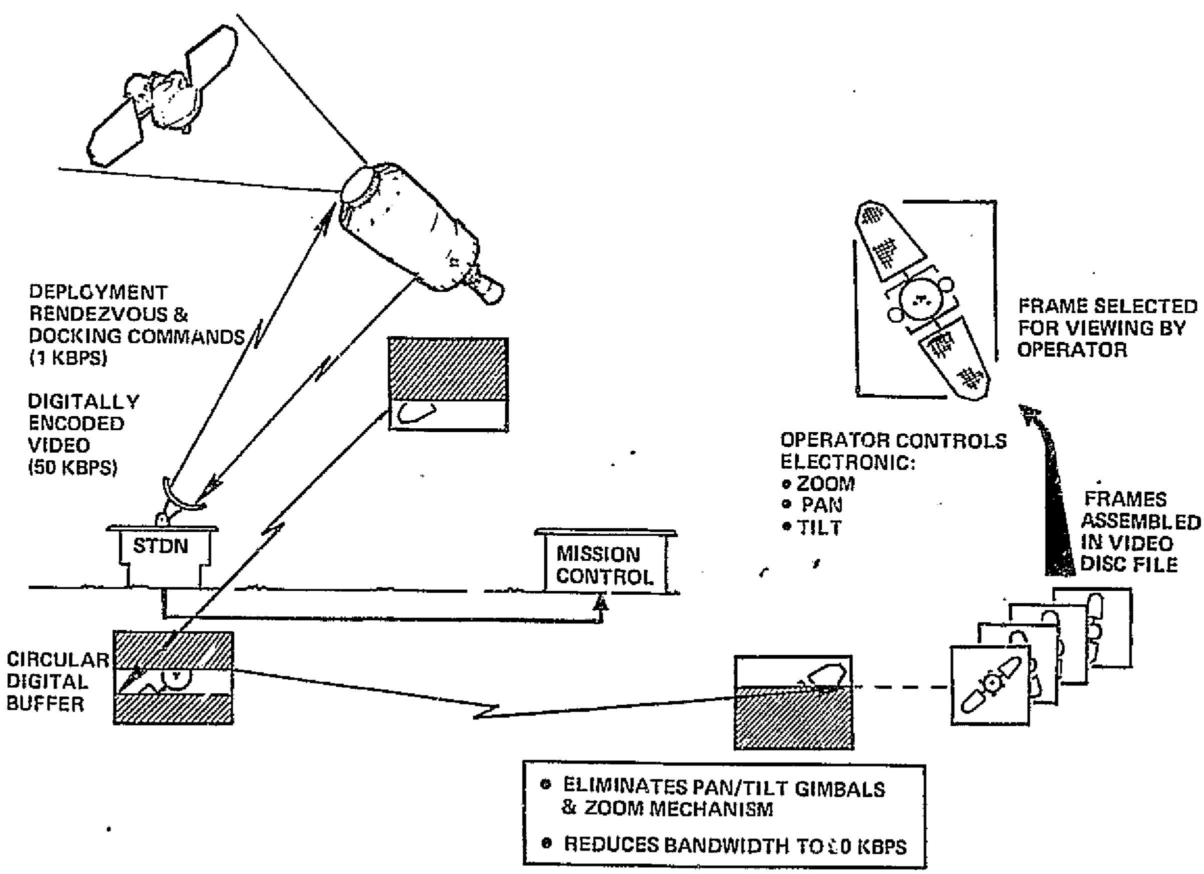


Figure 5-27. Frame-by-Frame Communication System Concept

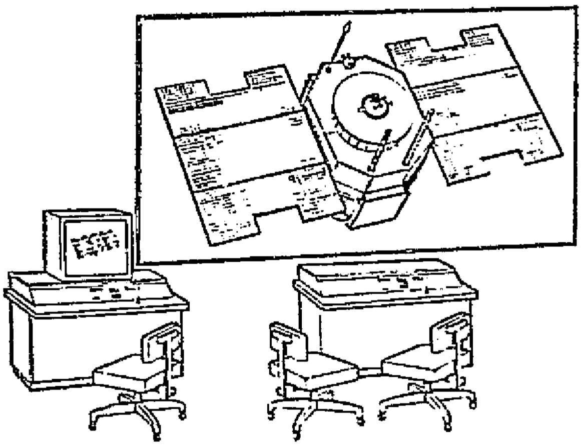


Figure 5-28. Supervisor's Control Station

shown in Figure 5-28, to perform the initial checks. Following a variable delay (typical of transmission delay between the ground and space networks), a picture appears. The supervisor periodically (or automatically) requests a new image and scans the last score or so of frames by means of a video disc recorder, attempting visual sighting by means of standard time-compression techniques or with the aid of known star sightings. Following visual sighting, the supervisor initiates the tracking mode, wherein he locates a range reticle on the target. This provides line-of-sight (LOS) angles for each display and LOS angular rates (which can be derived by difference techniques). These sightings support the injection burn to place Tug in close proximity to the spacecraft.

Following the insertion burn, visual sighting is once again established and the LOS information is provided for guidance corrections. Eventually, the target's known cross section permits the supervisor to make a crude range (hence range rate) measurement by adjusting a range reticle ring to the cross section. Once target details can be discerned, a standard orientation can also be commanded by rotating the range reticle index to line up with a desired target feature. These estimates get progressively better and allow closure to the near proximity of the spacecraft (e.g., to 100 feet, 30.5 meters) for visual inspection.

Inspection entails a slow orbiting maneuver about the spacecraft, with the upper stage longitudinal axis essentially aligned with the upper stage-to-spacecraft vector. During inspection, the spacecraft's docking adapter is located, the spacecraft commanded to latency, and the orbiting rate adjusted to align with the docking adapter. Controlled closure then achieves docking.

The fundamental 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.

The supervisor operates on each frame (frames are received approximately on 16-second centers) as illustrated in Figure 5-29. The reticle is envisioned as being ground-computer driven based on information known to the upper stage at exposure time. If the supervisor detects a discrepancy, he takes control of the reticle by

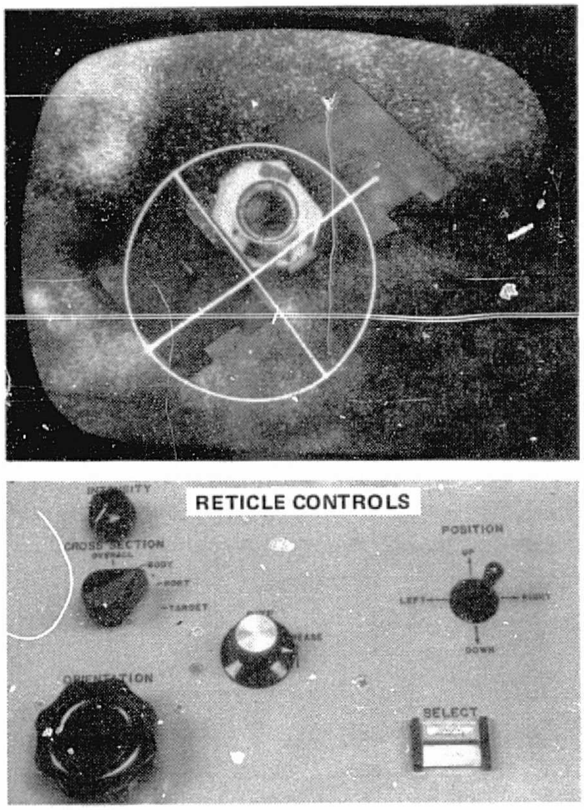


Figure 5-29. Remote, Manned Docking Procedure

pressing the mode SELECT switch, putting it in LOCAL control. He then positions (joystick), sizes (rotary potentiometer), and orients (large rotary control) the reticle to the spacecraft docking port (presuming this had shown a 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 upper-stage pitch and yaw (from reticle location), relative roll (from orientation), and range (from size).

If the discrepancy noted is with the target T (shown within the port), the crosshair need only be positioned on the T (joystick) to enable a computation of spacecraft — relative pitch (about the horizontal) and yaw (about the vertical). To indicate TARGET measurements, the design concept requires the supervisor to hold the spring loaded CROSS SECTION swtich in this position while returning to REMOTE control.

5.4.3 <u>SIMULATOR OVERVIEW</u>. The geosynchronous mission was the obvious choice for the initial manned, remote rendezvous and docking studies. It is estimated that well over half of the retrieval and on-orbit service missions will be at this altitude. The satellite is rarely eclipsed and so should be in view longer. Because of its distance from the satellite, the earth need not be simulated separately but could be added to the star field (if desired). Path curvature is moderate at the near rendezvous distances subordinating the Coriolis effects. The geosynchronous retrieval mission also brings with it a wealth of data from previous studies documented in the open literature.

Display generation for the rendezvous and docking studies were accomplished using Convair's Visual Display Simulator illustrated in Figure 5-30. The display is generated using standard commercial video techniques, employing a fine-line (945 lines vertical) screen in contrast to commercial TV (545 lines vertical). The simulated ground station display is a composite of separate displays representing: 1) the far background (stars and moon), 2) optional near background (earth with clouds), 3) foreground (target satellite), and 4) near foreground (the range reticle). The selected target satellite model employed was an existing fifth-scale Global Positioning System (GPS) three-axis-stabilized model, which has spiral antennas and gimbaled solar arrays

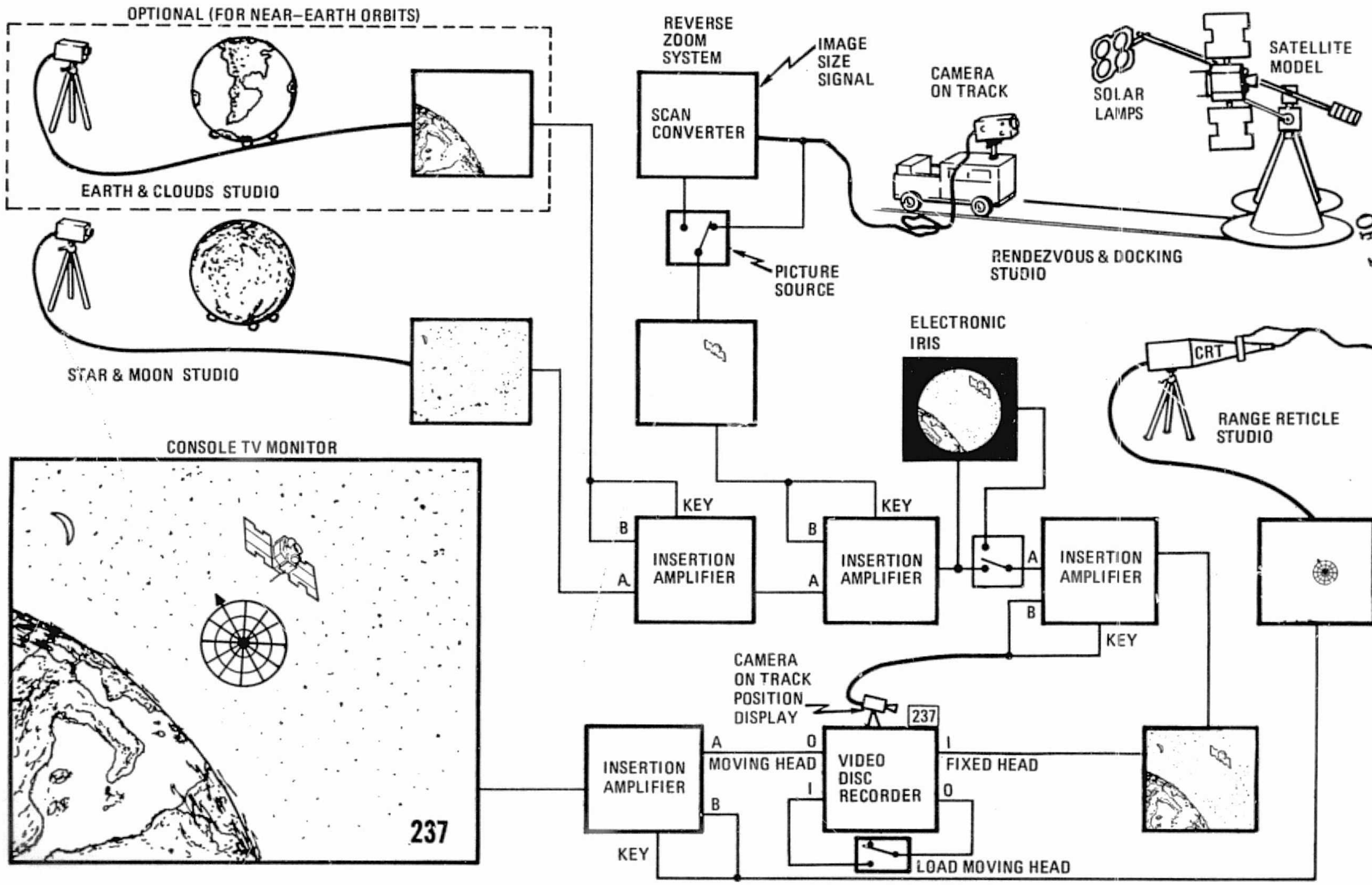


Figure 5-30. A Realistic Composite Can Be Built from Separate Images (Convair's Existing Visual Display Simulator)

that present typical obstacles to be avoided during docking. A range reticle is superimposed on the video output of the TV sensor by the simulated ground station computer. The range reticle is independently maneuverable by the remote supervisor.

At geosynchronous altitudes, the earth does not dominate the scene and can be appropriately affixed by means of a decal to the star background as is done with the moon. To compromise between a model sufficiently detailed for docking studies (about 1/5-scale) and fixed-length tracks, a scan converter is employed to "reverse-zoom" the model image to nearly a single dot. Operationally, the vidicon must be protected from direct rays from the sun and its corona; this is accomplished by an iris positioned at an intermediate image plane within the simulated TV sensor (to protect the camera-on-track vidicon) and an electronic iris downstream of the composite image (to occlude the background).

A video disc recorder is employed in the same manner as envisioned during actual operation from the ground station, whereby at any time the remote supervisor can recall preceding images at random for detailed re-examination, or a block of images serially for time compression. The video track position digital display is keyed onto the composite from a light-emitting diode (LED) display on the video disc controller before being stored on the disc. The availability of the last picture is delayed (fixed and random) before being placed on the moving track, simulating transmission delay.

The range reticle and/or target symbol is generated by a laboratory cathode ray tube (CRT) scope. Due to a size limitation of the digital computer (which has since been replaced with a larger, faster computer), the capability to provide measurements on the target "T" could not be provided. Instead, the supervisor utilized his ORBIT control to achieve a visually satisfactory alignment with the docking port before closing to contact (dock).

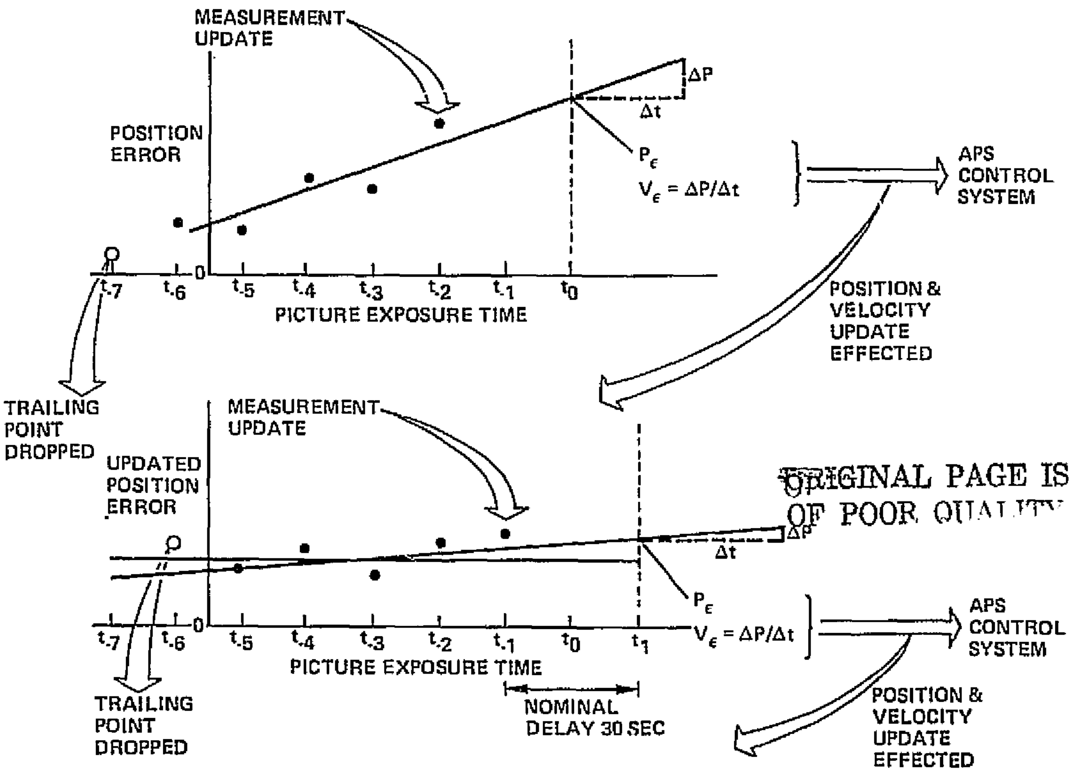
5.4.4 SIMULATOR SOFTWARE. The rendezvous and docking kinematics and control simulation is implemented by a digital program composed of rigid-body dynamics, simulated navigation (via a reference ephemeris supplied as an initial condition), target acquisition scanning logic, line-of-sight terminal rendezvous algorithms, and an optional automatic docking subsystem algorithm supporting an autonomous docking study. Propellant sloshing dynamics had been planned for incorporation but the sloshing models (being developed under contract to NASA MSFC) were not available in time to support incorporation.

For remote, manned rendezvous and docking, the digital simulation has been augmented to handle the simulators' display drive and control signals. Interfaces with the remote controls and displays has been similarly treated. The objective was to develop a single simulation that will handle either autonomous or remote, manned terminal rendezvous and docking. A detailed computer simulation of terminal rendezvous from several nautical miles to contact provides the capability to accomplish terminal rendezvous and/or docking either manually, automatically, or automatically with manual override (hybrid subsystem).

The simulator software effort began as an interface integration task and progressed toward a complete digital simulation of terminal rendezvous and docking. The software subsets consist of the target spacecraft (three degrees of freedom), Tug (six degrees of freedom), the supervisor's console command/display interface, the guidance and control algorithms, the orbital kinematics, the simulator (servo) drives, the video disc recorder digital command/control interface, and the test conductor's console command/display interface. Software development has concentrated on the docking control algorithm.

The docking control law employed was a simple least-squares fit of the measurement data when transformed to Tug coordinates (Figure 5-31). The current simulation uses up to the last 10 measurements (selectable by the test conductor) to compute the current position errors (intercept P_E) and velocity errors (slope V_E) in all three axes, together with establishing the LOS and rotations about the LOS.

If too many stages (measurement vectors) are employed in the digital filter, the system becomes highly susceptible to noise. Repeated success has been obtained with four to eight filter stages except at very large range, where too little target detail is available (discernible) from the simulator on which to range. (The present simulator employs a 30 degree (0.52 radian) FOV fixed lens rather than a zoom or turret lens to simulate the envisioned operational system. Hence, it lacks the ranging performance necessary to simulate terminal rendezvous.) Modification of this least-squares technique to a recursive filter is currently underway.



ORIGINAL PAGE IS OF POOR QUALITY

Figure 5-31. Docking Control Law (Five-Stage Least-Squares Fit Illustrated)

The velocity and attitude control systems initially employed were simple "bang-bang" techniques with the intent to replace them with a more sophisticated technique employing dual thresholds. Replacement was precluded by a lack of space within the simulator computer thus rendering the simulation data-on-impulse-expended essentially useless. This is currently being rectified by the incorporation of a Centaur-type detailed control system model (Figure 5-32) on the new, larger simulation computer.

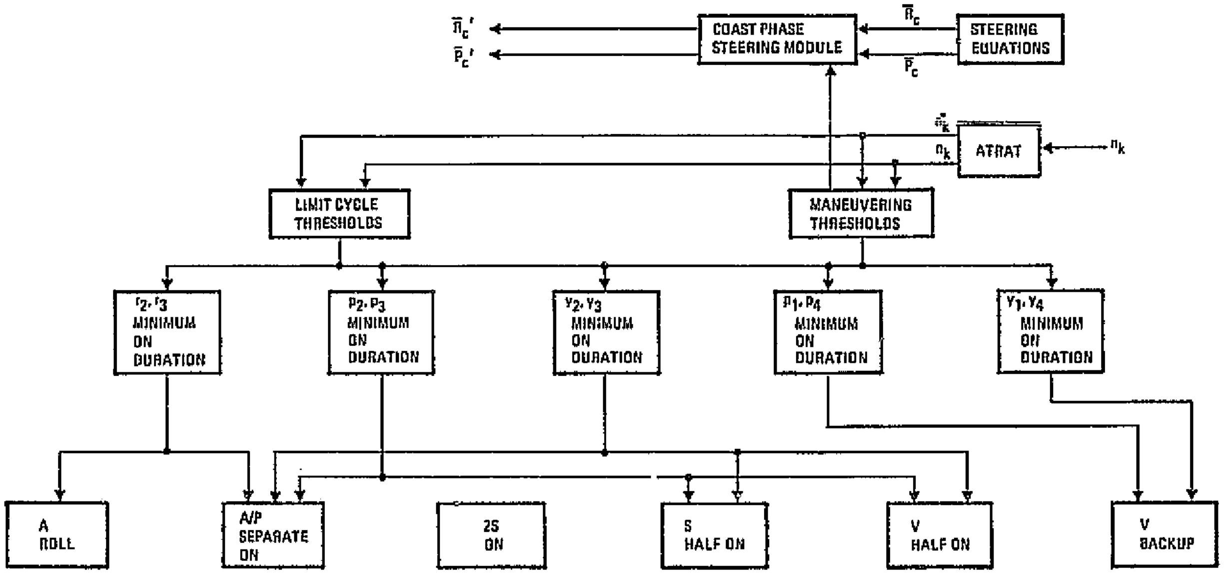


Figure 5-32. Centaur-Type Reaction Control System Schematic

The Auxiliary Propulsion System (APS) performance was modeled from vehicular data for an initial rendezvous on a servicing sortie. Thrust accelerations were determined from the geometry in Figure 5-33; each engine pair has a rated (altitude) thrust of 50 pounds (22.5 kg). The APS propellant (monopropellant hydrazine) is consumed at an I_{sp} of 160 seconds if pulsing or an I_{sp} of 230 seconds if continuous. A budget of 96.5 pounds (43.4 kg) was used for terminal rendezvous and docking.

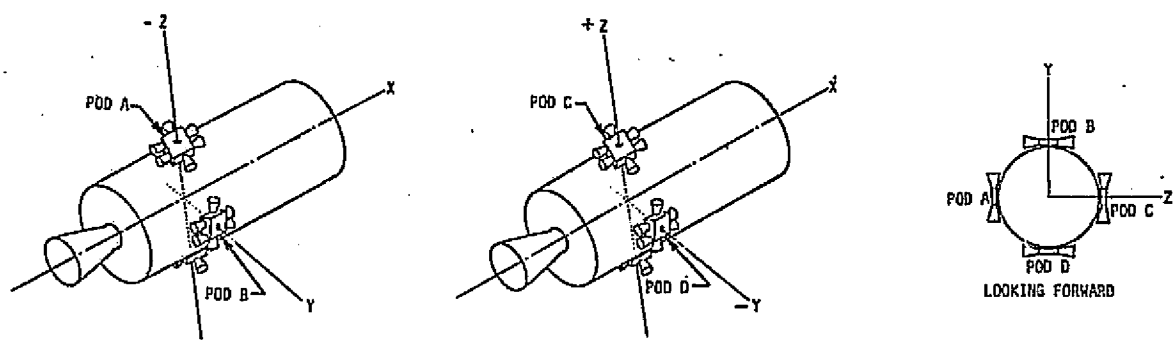


Figure 5-33. APS Installation Geometry

5.4.5 SIMULATOR HARDWARE. Figure 5-34 illustrates the principal simulator hardware. An existing Global Positioning System (GPS) three-axis stabilized spacecraft model (1/5-scale) was modified and mounted onto an available three-axis sting. It was necessary to add a ring fluorescent within the docking port to ensure that its dark interior keyed black onto the star (or earth) background so that the background did not show through. Addition of fill lights prevented breakdown of keying for all orientations of the model. To provide for simulating spin-stabilized spacecraft and for unrestricted spacecraft roll motion (inner gimbal), a slipring assembly was developed through which to pass the port light and gimbaled solar array motor power leads.

A solar lamp assembly consisting of four photoflood lamps was constructed and mounted on an existing pedestal-mounted servo motor. By gimballing normal to the camera-target axis, sun cone angles from near zero (behind the lens) to 180 degrees (3.14 radians) (into the lens) can be simulated. The proper sun clock angle (about the line of sight) is obtained by rolling the model spacecraft about the line of sight and compensating for this motion by counter-rolling the camera image within the camera-on-track optics (Perchant prism).

The carriage camera pallet was detached from its Y-axis servo frame and suspended on three jackscrews to simulate Z-axis (vertical) motion during docking (Figure 5-35). Four roundways were incorporated into the design to achieve the required structural

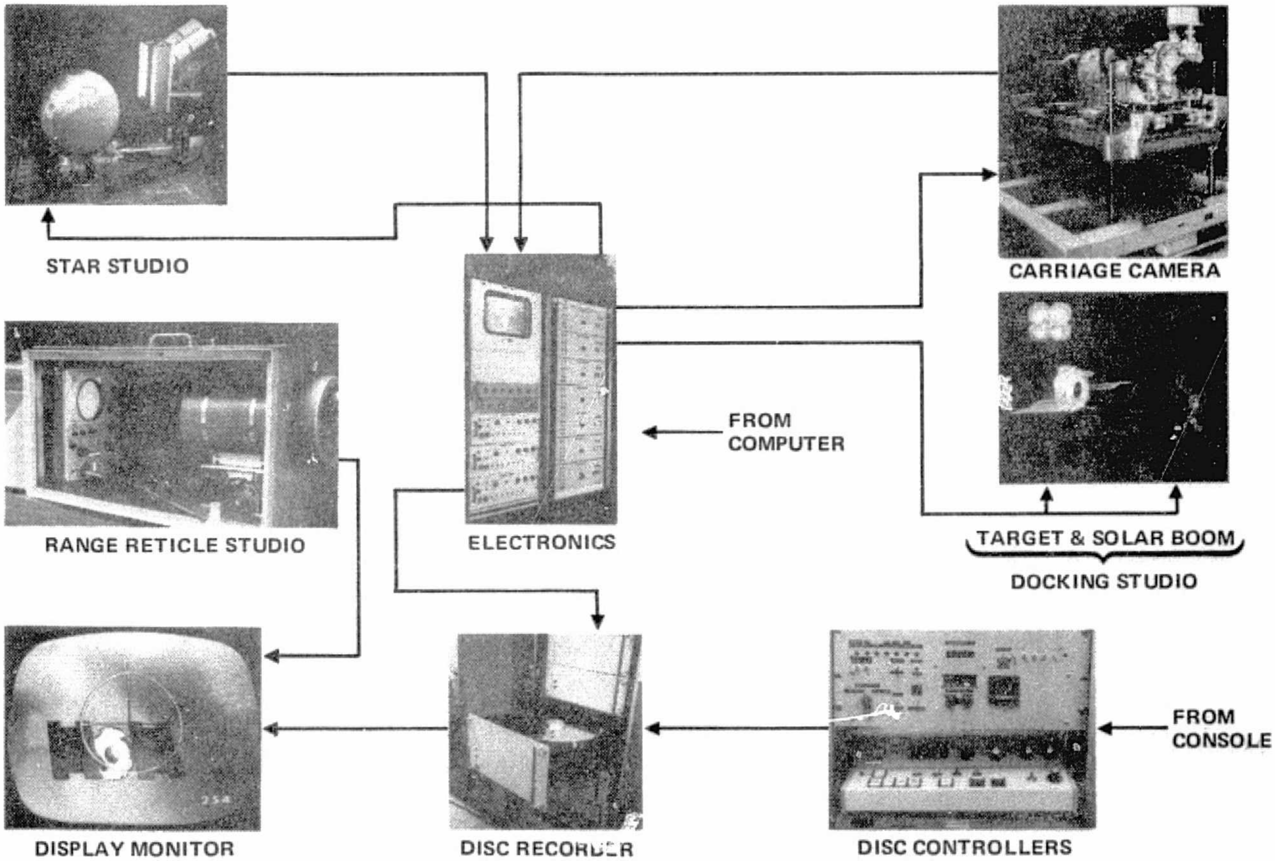


Figure 5-34. Rendezvous and Docking Simulator Hardware

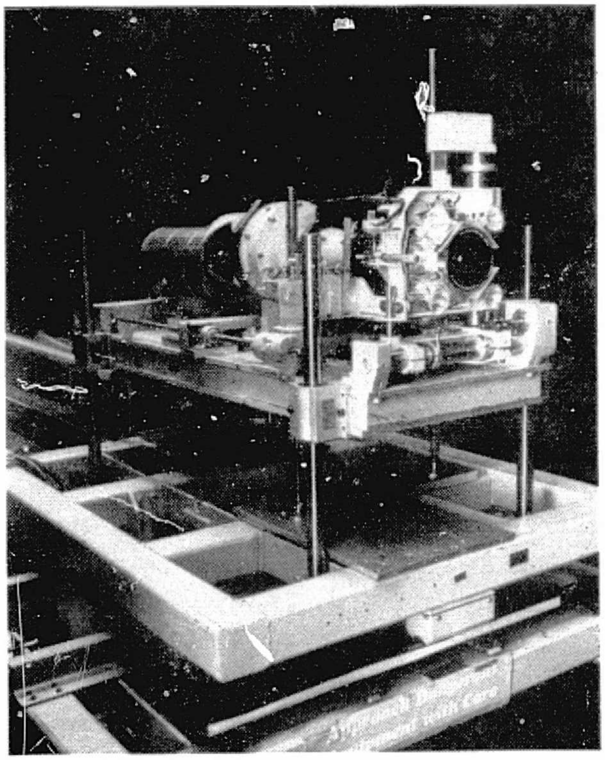


Figure 5-35. Carriage Camera Vertical Drive

rigidity to prevent X and Y motion from inducing an oscillation due to the high pallet mass (camera, optics, docking lamps and servo motors) and high offset center of mass (at the top of its stroke). The pictured installation proved entirely satisfactory after addition of the third jack-screw (only two were provided in the initial design).

An additional rack of servo-amplifier electronics (Figure 5-34) was developed to drive the additional degrees of freedom necessitated by the docking application.

The existing Video Disc Recorder (VDS) local and remote controllers were modified for 945-line remote, computer-controlled operation (Figure 5-34). A light-emitting diode (LED) display was enlarged and keyed onto the resulting composite image before it was written onto the VDS. The range reticle (generated on a laboratory CRT via a dedicated special purpose computer built for this purpose) was keyed onto the composite image received from the disc recorder when displayed at the remote supervisor's console.

Figure 5-36 illustrates the remote supervisor's console developed for this investigation. The display elements are the TV monitor, an inclined panel containing caution, warning, and status lights, and a digital readout display of selected flight parameters. The controller elements are separated into four horizontal panels (from right to left): SCANNING CONTROLS for manual, remote control of the Video Disc Recorder; FLIGHT MODE controls for commanding the rendezvous/docking vehicle; RETICLE CONTROLS for manually controlling the range reticle; and SPACECRAFT CONTROLS for commanding the spacecraft (via a ground RF link) if it is designated "active-cooperative."

5.4.6 SIMULATION MINISTUDIES. Two ministudies were conducted on company funds using the supervisor's console in a partially completed condition. In the Spacecraft Acquisition ministudy (Figure 5-37) operator performance was investigated while searching for the spacecraft, which appeared as a faint star among hundreds of stars of the Milky Way. The video disc recorder contained a sequence of frames on which the spacecraft became progressively grighter until it finally dominated the scene. There were typically six such scenarios loaded onto the video disc.

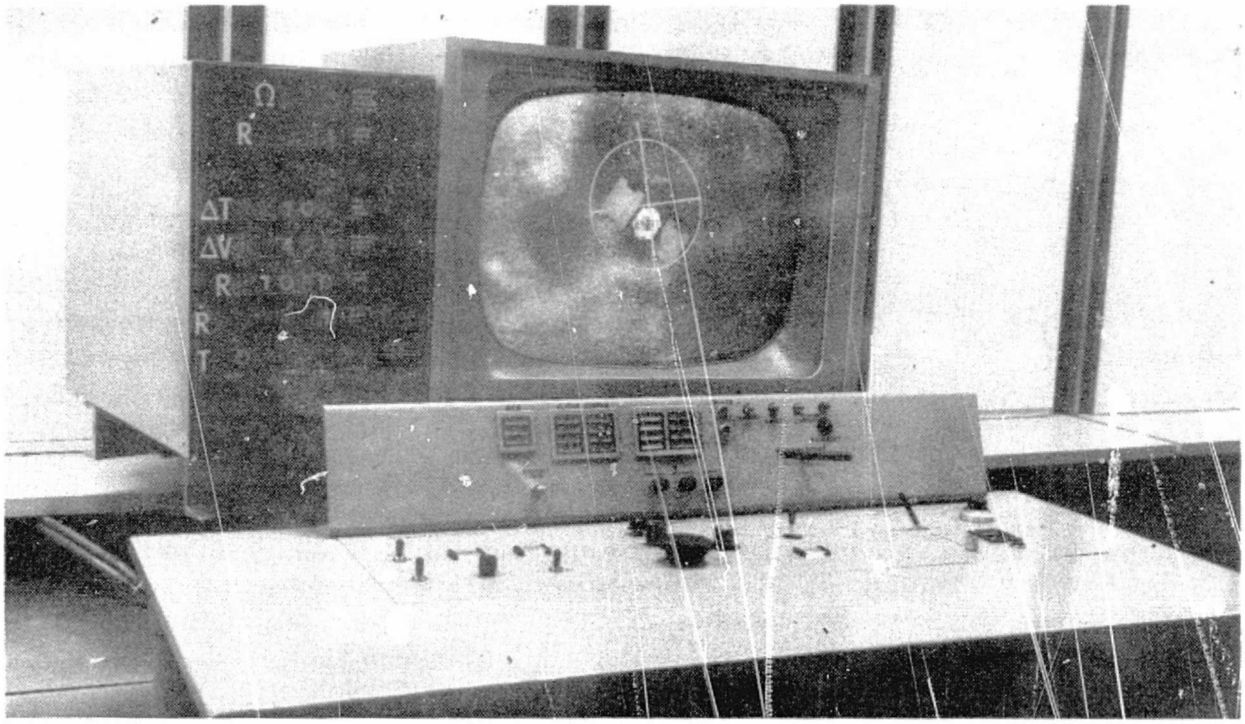


Figure 5-36. Remote Supervisor's Console, Convair's Simulation Study

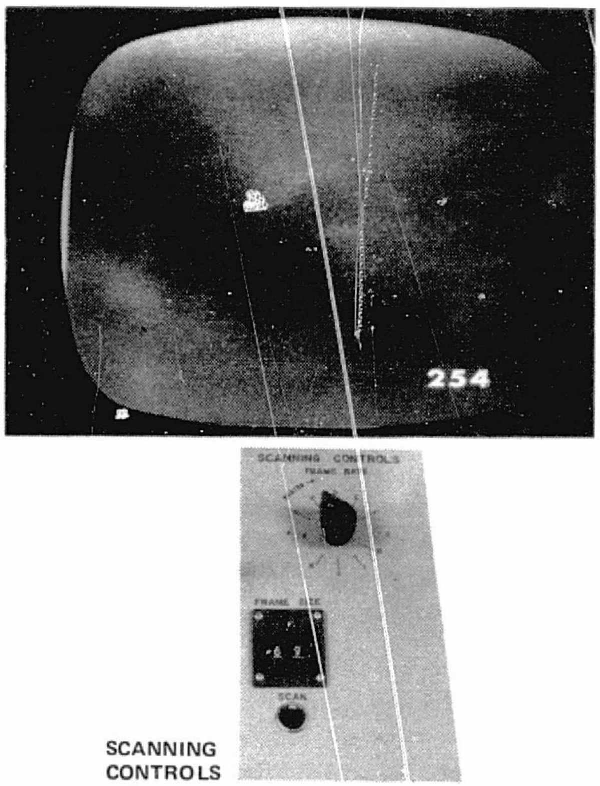


Figure 5-37. Spacecraft Acquisition Ministudy

By means of the console SCANNING CONTROLS, the operator searched for the spacecraft by scanning the frames looking for a streaking target (top to bottom of the screen). Although noise bursts produced false indications, these were easily discounted with additional frames. The spacecraft was typically acquired and confirmed within six to eight frames (about two degrees of visual angle) at an operator-selected (optimum) scan rate of three frames per second. The need for adjustment of brightness and contrast resulted in adding these controls to the console inclined panel.

The second ministudy, Spacecraft Tracking and Ranging (Figure 5-38), sought to identify mean operator performance and performance dispersions, principally in task time. The scenario loaded onto the disc consisted of the spacecraft at various orientations and at progressively closer

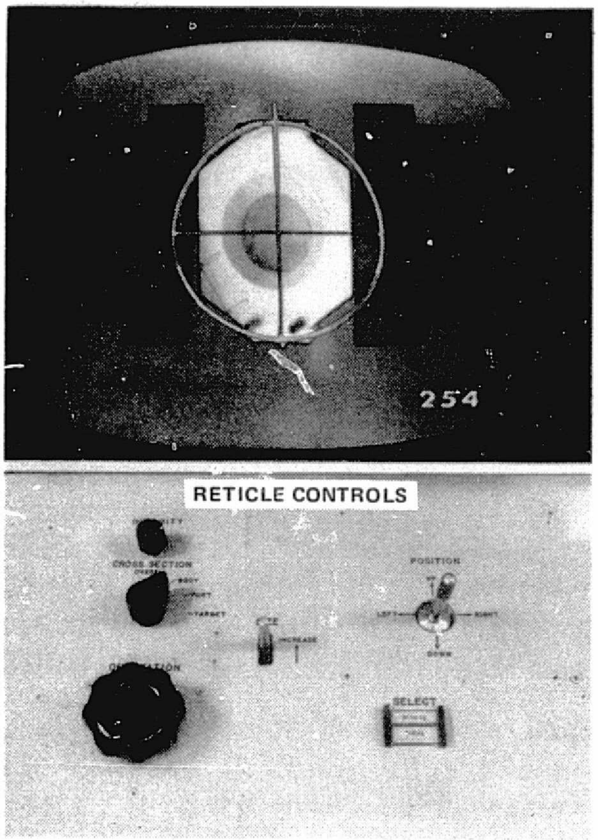


Figure 5-38. Spacecraft Tracking and Ranging Ministudy

ranges up to the docking position. The test conductor supplied each frame of the sequence and timed the operator, who positioned, sized, and oriented the reticle on the central body and, subsequently, the docking port. A joystick supplied the horizontal and vertical positioning, a throttle controlled the sizing, and a knob the heads-up orientation. Prior to each frame, these controls were randomly disturbed and the reticle moved off the screen to the lower left by the test conductor.

Results indicated that the reticle task took an average of 11 seconds, although a substantial improvement was observed with additional practice. The shorter times correlated with decreased accuracy, and the longer times were observed to be attributed to a sticking joystick. Although orientation was quickly and accurately accomplished, placement accuracy was invalidated by the sticking joystick, and sizing accuracy was invalidated by varying operator interpretations of task directions.

This study was subsequently repeated with reworked controls and reticle, and with additional spacecraft docking target markings. Results confirmed that performance substantially improved with the reworked reticle and controls. Subject placement performance averaged 0.2 degree (0.087 radian) visual angle (measured at his eye). With sufficient practice, the basic placement, and orientation task could repeatedly be accomplished in less than 5 seconds.

5.4.7 DOCKING FEASIBILITY DEMONSTRATION. Repeated simulations of remote, manned docking were conducted on the simulator during simulation development. Difficulties encountered were subjected to analysis and often resulted in alternation of the simulator hardware, software, or operating procedures. As the simulator neared checkout, successful docking operations became commonplace. Figure 5-39 illustrates typical computer output.

Only representative results have been obtained to date but these clearly establish the feasibility of remote, manned docking utilizing a low light level television (LLLTV) camera (Table 5-4). Initial displacements simulated ranged from 190 to 1500 feet (58 to 460 meters); some approach orientations required orbiting the spacecraft

```
RETICLE IN LOCAL AT     770.00 SECONDS

RETICLE IN REMOTE AT    778.50 SECONDS
DU=    -.2676   DV=    -.1803   DW=    -.1431   TIME=  752.00

DATA ENTERED AT   781.50 SECONDS

OLD COMPUTED STATE-
       •          •          •
       U          V          W          U          V          W
    -.0999     .0007      .0053      3.43       -.03       -.18

TRUE  STATE-
       U          V          W          U          V          W
       •          •          •
    -.1193     .0056      .0069      1.62        .07       -.13

NEW COMPUTED STATE-
       U          V          W          U          V          W
       •          •          •
    -.1045    -.0034      .0023      3.11       -.29       -.38

************************************************************************

CONTACT TIME=     795.00 SECONDS

POSITION ERRORS (FEET)

       Y          Z          R
      .34       -.18        .39

VELOCITY ERRORS (FEET PER SECOND)
       •          •          •          •
       X          Y          Z          R
     .0153     -.0034      .0035      .0048

ANGLE ERRORS (DEGREES AND DEGREES PER SECOND)

    ROLL   PITCH    YAW    TOTAL      PR       PR       YR      TOTAL
    2.68    1.07   -1.43    3.22     .001     .000    -.002     .002

FUEL REMAINING FROM BUDGET    13.67 POUNDS
```


Figure 5-39. Typical Computer Output from Docking Simulation

Table 5-4. Representative Simulation Results

PARAMETER	UNITS	RQMT	RUN NUMBER 1	2	3	4	5
INITIAL DISPLACEMENT	FT		190	190	1000	190	1500
TIME TO CONTACT	MIN		11.2	11.5	32.0	16.0	34.3
POSITION ERROR (RADIAL)	FT	1.0*	0.43	0.08	0.16	0.46	0.47
VELOCITY ERROR (RADIAL)	FPS	0.3*	0.154	0.002	0.002	0.001	0.001
ANGULAR ERROR (RADIAL)	DEG	5.0*	2.03	2.45	1.35	3.96	7.75
ANGULAR RATE ERROR	DEG/SEC	0.5*	0.863	0.001	0.006	0.003	0.004
ROLL INDEX ERROR	DEG		INCLUDED IN ANGULAR ERROR				
NUMBER OF FILTER STATES			8	5	8	5	6
REQUIRED ORBITAL ARC	DEG		0	0	30	0	20
APS PROPELLANT CONSUMPTION	FPS	10**	OVERSIMPLIFIED (BANG-BANG) CONTROL SYSTEM INVALIDATES DATA				

*OBTAINED FROM ARTICLE 3.2.1.1.1.4.2 OF MSFC 68M00039-1, BASELINE SPACE TUG SYSTEM REQUIREMENTS & GUIDELINES, 7/15/74

**OBTAINED FROM FIGURES 2.1-7 and -10 OF MSFC 68 00039-2, BASELINE SPACE TUG CONFIGURATION DEFINITION, 7/15/74 although angles up to 30 degrees (0.52 radian) before alignment with the docking port was achieved. The time to contact reflected the initial displacement, the requirement to orbit, and whether any stationkeeping was employed.

Two problems have been uncovered with this simulation. The first concerns the angular error at docking. The TARGET measurement capability, although envisioned for an operational system, had not yet been added to the simulation due to a limited availability of computer space and since it was found that this capability could be emulated via the console's ORBIT control. However, this expedient caused additional delays, higher APS propellant consumption and occasional failures to "dock." This capability is currently being added now that a larger simulation computer has become available. The TARGET measurement capability will enable direct measurements on the docking "T" as soon as gross alignment on the docking port has been established.

The second problem relates to the maximum practical range for LLLTV in a ranging mode. Since the simulator employs a camera with a fixed, 30-degree (0.52 radians) FOV, it was found that 1500 feet (500 meters) was a practical limit on ranging owing both to FOV restrictions and simulator error sources. This result is in agreement with the ranging performance derived for LLLTV in Section 5.2.4.

5.4.8 CONCLUSION. The use of LLLTV in a remote, manned configuration for rendezvous and docking with a solar illuminated spacecraft has been demonstrated (via simulation) as being feasible. Due to the limitation imposed by a fixed, 30-degree (0.52 radian) FOV in Convair's Docking Simulator, the initial portion of the extended docking phase, viz., that portion of terminal rendezvous immediately following the insertion burn — could not be simulated.

Two approaches to long range tracking of a solar illuminated spacecraft were investigated. The use of a video disc recorder to acquire a streaking target (such as would be the case during a phasing orbit approach) was simulated in an environment close to the anticipated operational environment. The target was quickly and accurately acquired and confirmed despite the presence of television image noise.

The use of a star catalog to position registration marks at known star and assumed target locations was investigated but not simulated. The remote, manned application of this technique would utilize the console supervisor to shift the star registration pattern to achieve precise registration with the brighter stars, and then to reposition the assumed target location to agree with the observed target location (if indeed it is observable on that particular frame). Based on simulation of the scanning technique, the registration technique would pose no problems.

Remote, manned docking was conducted in an environment identical to that envisioned for the operational system except for the application of stress on the console supervisor. Even though the observation measurement on the target "T" (located within the docking port) was not available to the console supervisor during simulation studies, approach, station keeping, orbiting (of the target), inspection, docking port location, controlled closure, and docking were all routinely accomplished from several hundred feet (circa 100 meters) from the spacecraft. Although simulator restrictions (fixed FOV) precluded simulating postinsertion velocity capture and range lock, docking from ranges up to 1500 feet (500 meters) was successfully accomplished.

5.5 AUTONOMOUS SUBSYSTEM ANALYSIS AND SUBSYSTEM SELECTION TRADE

The performance of the autonomous subsystem (employing the GaAs SLR sensor) was investigated through similarity to the remote manned (LLLTV) subsystem operation. Both subsystems must provide equivalent data to Tug. The principal differences lie in the speed and precision available from the GaAs SLR, which will result in much improved rendezvous and early docking performance, and in the degree of confidence to be assessed to their respective docking performance.

The approach to selecting an operational Rendezvous and Docking subsystem was to develop a low weight, low cost, low risk alternative to a fully autonomous scanning ladar subsystem. This trade gave rise to the hybrid subsystem illustrated in Figure 5-40 and whose functions are allocated in Table 5-5. A fully autonomous subsystem, based on the SLR sensor, could then evolve either during DDT&E (if sufficient time/budget is available) or subsequently during initial operation. The principal drivers of this evolutionary approach are time, cost, and risk. During the transition period, functions currently allocated to slow-scan LLLTV (those designated primary in Table 5-5 except for inspection) would gradually revert to the SLR subsystem.

Table 5-5. Primary/Backup Allocation

Item	Scanning Ladar	Slow-Scan LLLTV
Acquisition Confirmation	Primary	Backup
Tracking	Primary	Backup
Ranging		
Preinjection	Primary	--*
Post Injection	Primary	Backup
Inspection	—	Primary/Backup
Alignment to Axes	—	Primary/Backup
Closure & Docking		
Initial Operational Capability (ICC)	Backup	Primary
Fully Operational	Primary	Backup

*Autonomous Navigation Subsystem is backup

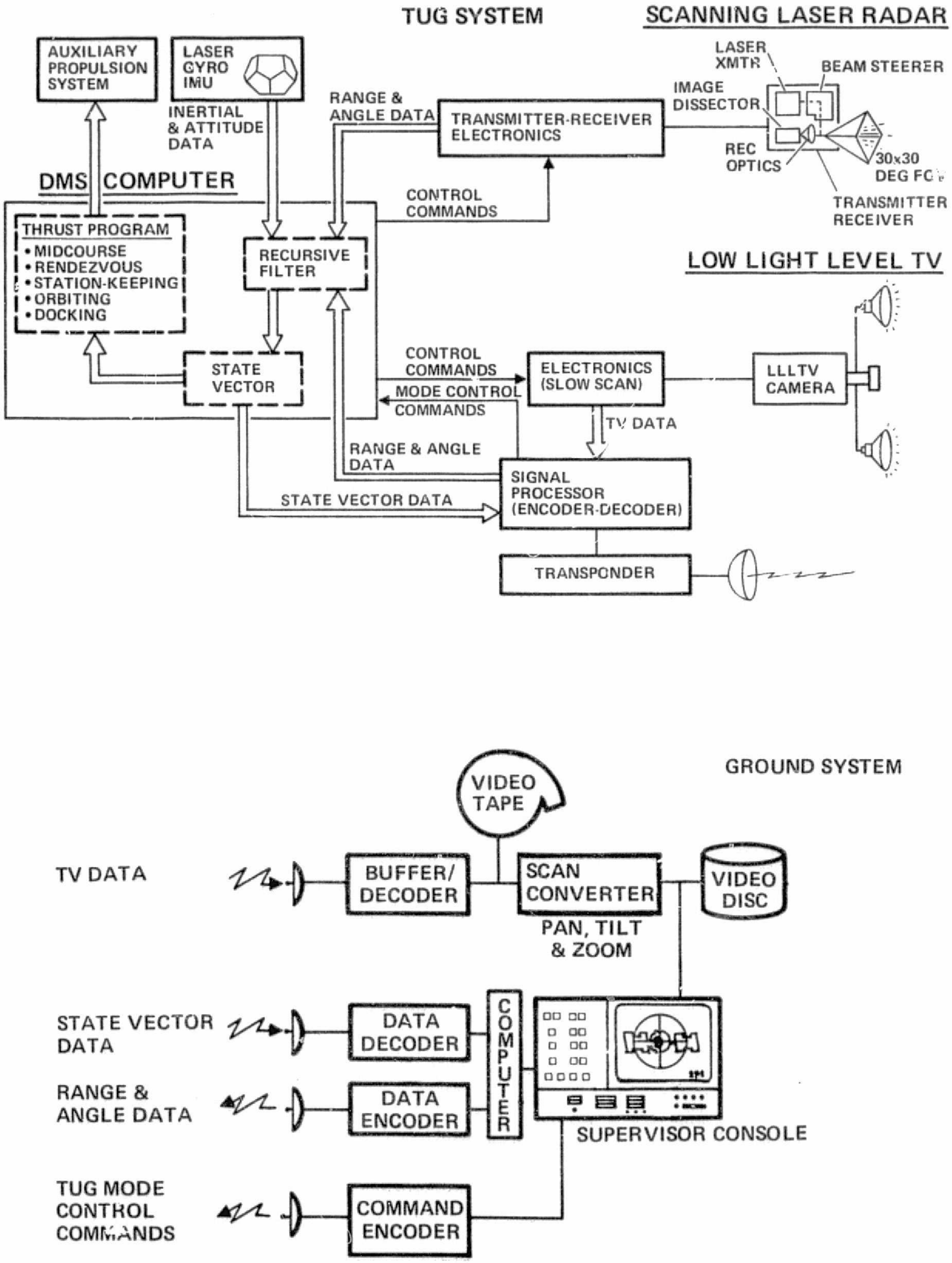


Figure 5-40. Baseline Rendezvous and Docking Subsystem

Operationally, LLLTV is envisioned to be fully utilized in a monitoring role (except during visual inspection and gross alignment with the docking axes) with functional backup capability. Gross subsystem failures will be detected by the DMS computer, which will suspend current operations and call for the remote-manned backup subsystem. Performance assessment will be accomplished by LLLTV in a monitoring role; performance degradation or subtle failures will be resolved by selected human interaction (e.g., request to Tug's DMS computer to re-initiate retroreflector acquisition) or a human decision to resort to the manned backup subsystem. This approach minimizes operational risks.

The remaining technical concerns with the SLR baseline subsystem hardware are its performance during the port search and docking phases, the development status of the improved GaAs wafer laser (for reliability and maintainability improvement), and the qualification of the piezoelectric beam steerer. Principally, insufficient testing has been performed to establish that the SLR will be able to discern skin reflections from retroreflectors, particularly while employing AGC. A related issue is its ability to track a spin-stabilized spacecraft (to 100 rpm) unless it employs a counterrotating retroreflector array. During terminal docking, the peripheral retroreflectors will begin to fall outside the active FOV necessitating open-loop control (for some axes) over the last 30 feet (9.14 meters).

There are no technical concerns with LLLTV hardware; however, manned-remote operation — although demonstrated — remains to be fully assessed.

The remainder of this section summarizes the operational rationale for a hybrid subsystem.

5.5.1 SPACECRAFT ACQUISITION CONFIRMATION. Figure 5-19 illustrated 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 spacecraft (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 since its operation is fully autonomous and easily verified. (Using the GaAs SLR in the passive acquisition mode will necessitate the removal of the $0.9\mu m$ bandpass filter from the receiver optics of the Gen-3 (generation 3) prototype.) The information required is simply line-of-sight (LOS) angles to the target and can optionally be supplied by the LLLTV camera in the event of a failure. LOS will be obtained by scanning the complete detection pattern into the computer (via "stars" of visual magnitude up to the capability of the sensor). After "viewing" a number of frames, the computer will make an actual target determination. The GaAs SLR 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.

5.5.2 SPACECRAFT TRACKING. Below 2500 n. mi. (4650 km), again see Figure 5-19, either of the subsystem sensors can provide LOS measurements to an accuracy better than that of onboard knowledge. These 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 altitude). 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 circa 20 degrees (0.35 radian) 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 utilizing 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 postinjection uncertainty cone. 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 insures the 0.06 degree (0.001 radian) tracking accuracy for these sensors (Figure 5-19). The required star catalog is quite manageable, i.e., about 70 words (representing a 3-degree (0.052 radian) FOV).

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

5.5.3 SPACECRAFT RANGING, PREINJECTION. The range at which preinjection ranging information must become available (if it is to be utilized) 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 altitude, 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 re-insertion 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 5.5.4), as was illustrated in Figure 5-22. This means that long-range tracking and (particularly) preinjection ranging should be justified on the improvements gained beyond that available from the navigation subsystem.

Note that LLLTV has no preinjection ranging capability (see also below). Thus, in event of a failure of the primary sensor (possible extended-range SLR) in this phase, the GN&C subsystem will effect rendezvous from onboard knowledge.

5.5.4 SPACECRAFT RANGING, POSTINJECTION. The range at which post-injection 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 GN&C 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 (Figure 5-23).

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 judgemental 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.)

Due to its superior performance and autonomous operation, GaAs SLR will serve as the primary sensor during the postinjection ranging phase, supplying LOS and range. Using the angular coordinates generated during the track phase, a narrow field of view active track (laser operating) will be used 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.

Active tracking 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 passively track 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 range- or track-lock on the false target. Frame time during this phase is to be 14 seconds.

5.5.5 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 & 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 utilizing 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 utilize a fixed, 30-degree (0.52 radian) 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 eight pounds (3.6 kg).

It should be noted that 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.

5.5.6 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 predetermined 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 the inspection functions.

The port search/alignment phase is to consist of Tug slowly orbiting the target vehicle at a radius of approximately 75 feet (25 meters). Location of the docking port utilizing 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 feet (25 meters) 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 retroreflector axis performance), the SLR will revert to the acquisition mode supplying range, LOS angles, and target intensity for each target in the field of view at the reduced intensity. This will require additional gain in the discrimination logic. The computer will discriminate another retroreflector in the field of view and command 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).

5.5.7 CLOSURE AND DOCKING. The primary subsystem selected 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 (±0.087 radian) (coarse) alignment to the docking port and range equal to approximately 75 feet (25 meters).

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 (±0.087 radian) 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.

5.6 CONCLUSIONS

Final recommendation of one Rendezvous and Docking Subsystem over another was determined to be the proper subject of additional, more in-depth studies. Basic problems were identified and the initial feasibility established for several techniques, with a significant laboratory simulation effort being accomplished in exploring the remote manned subsystem operation. Limited time and manhours for this task within the present Avionics Study precluded full resolution of a final "best" system.

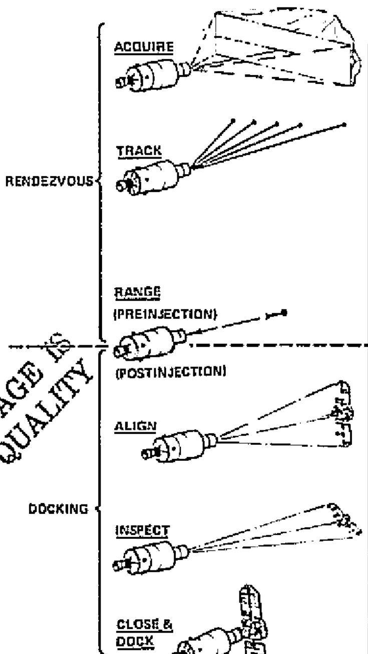
In addition to the simulation/demonstration of the remote manned rendezvous and docking, these trades have examined the GN&C autonomous capability and the application of candidate sensors to the basic functional requirements. Figure 5-41 shows the evaluation results for the various sensors as applied to the functional requirements. The large check marks show where the conditional requirements are met by the candidates.

From the evaluation of the sensors, the following were determined:

a. Autonomous navigation capability is essential for the acquisition phase, and will be utilized throughout the rendezvous phase. Additional input to the navigation filter (Kalman) by the tracking sensor allows insertion in closer proximity to the spacecraft. Both a scanning Ladar and LLLTV can provide this capability.

b. None of the candidate sensors can reliably furnish preinjection range for the direct ascent rendezvous due to the required range, about 300 n. mi. (555 km). However, insertion to within 3 n. mi. (5.6 km) can be accomplished with autonomous navigation alone, and to within 1 n. mi. (1.9 km) employing the tracking sensor. Range would only then be required post injection. Either of the Ladars or the tricolor docking Ladar can provide this capability.

c. For the docking process, the scanning Ladars can furnish the inherent capability for autonomous docking. LLLTV must be a part of the Rendezvous and Docking subsystem since it is the only candidate for inspection, and the only reasonable candidate for port search and alignment. With LLLTV alone, docking can be done through a remotely situated supervisor, as has been demonstrated. The TV alone is, of necessity, rather slow, and the performance can be greatly improved by coupling with either the non-scanning Ladar or the tricolor docking Ladar.



CONDITIONAL REQUIREMENTS	GaAs SLADAR** (30 DEG FOV)	CO_2 SLADAR** (30 DEG FOV)	SLOW SCAN LLLTV (2 - 30 DEG FOV)	TRICOLOR DOCKING LADAR** (10 DEG FOV)	AUTONOMOUS NAVIGATION CAPABILITY
ACQUISITION RANGE LOCK BY 2,500 NMI	1,200 NMI*	NA	4,100 NMI* ✓	NA	INDIRECTLY "ACQUIRED" AT ALL TIMES
TRACK FOR NAV UPDATE LOS INFO ±0.1° BY 2,500 NMI	3,700 NMI* ✓	NA	4,100 NMI* ✓	NA	±0.1 DEG (DEGRADES BELOW 1,500 NMI)
RANGING BEFORE 250 NMI	NA	±14 FT TO 270 NMI ✓	NA	NA	INDIRECT: ±1.8 NMI AT ALL TIMES
RANGING 2-4 NMI	±2 FT ✓	±14 FT ✓	±600 FT* AT 2 NMI	±2 FT ✓	NA
VISUAL INSPECTION	NA	NA	✓	NA	NA
ALIGNMENT <10 DEG (50 TO 100 FT)	± 1 DEG (>31 FT) ✓	± 1 DEG (>10 FT) ✓	<2 DEG ✓	±2.5 DEG (>31 FT) ✓	NA
CLOSURE & DOCKING AXIAL: 0.1-1.0 FPS RADIAL: 1.0 FT 0.3 FPS ANGULAR: 5.0 DEG 0.5 D/S	NOT STUDIED	NOT STUDIED	MANNED, REMOTE SIMULATION STUDY ✓	NOT STUDIED	NA

*REQUIRES SOLAR ILLUMINATION OF SPACECRAFT
**REQUIRED RETROREFLECTORS ON SPACECRAFT
✓ MEETS CONDITIONAL REQUIREMENTS

Figure 5-41. Rendezvous and Docking Summary Study

SECTION 6

DATA MANAGEMENT SUBSYSTEM

6.1 COMPUTER REQUIREMENTS ANALYSIS

Computer requirements are established through the analysis of vehicle functions and mission timelines. These requirements are converted to computer characteristics that define the configuration, size, and processing capability. The general characteristics of immediate concern are things such as:

System concept or organization	Input-output configuration and speed
Word length	Safety, reliability
Memory size and speed	Weight, size
Processing speed	Instruction repertoire
Logic technology	Utility software

Table 6-1 summarizes the identified requirements that have a direct influence on computer characteristics.

6.1.1 SOFTWARE. The 32 bit data word requirement is established by the precision needed in guidance and navigation computations. A 16 bit data word would make double precision calculations necessary with associated increases in execution times.

Software estimates for the mission functions represent the minimum memory size the computer should be expected to have. Experience has shown that as systems develop, new functions are identified and operational improvements are requested and the size of the software grows. If there is no growth capability, the cost of the software is drastically affected as more and more effort is expended in packing data and reducing computer program size. A minimum of 40% growth allowance for initial estimates is considered adequate. This criterion indicates that a 48K memory is desirable for Tug.

A processing speed greater than 400, 000 operations per second can result when the processing associated with a dodecahedron IMU is included with normal system functions. This would require an extremely fast computer, which is generally not available with margin for processing speed growth.

Table 6-1. DMS Computer Requirements

Item	Requirement	Driver
Data Word	32 bits	GN&C calculations
Memory Addressing	Up to 48K words	Software estimate 30,469 words
Processing Speed	> 400 Kops	Vehicle & IMU processing
Instruction Set	IBM 360 compatible	Computer lab simulations
Desired Features:	Floating point hardware	Reduction in coding effort and scaling errors
	Microprogram control	Special functions, high speed
	Direct memory access	Data bus, IOP and orbiter Data interface
	High order language	Reduced coding effort and easier revision

Compatibility of the computer instruction set with that of a powerful ground based computer is needed for system simulations before flight hardware is available.

Microprogram control and floating point hardware provide high speed execution of special functions and reduce the effort for coding of application programs. Higher order languages use these functions to improve the accuracy of the programmer's work and reduce the verification time for functions otherwise created in software.

Direct memory access reduces the burden on the CPU for control of storage for system data and data transfers to the data bus. This data is needed in the central computer memory to accomplish the vehicle functions, but much of it is being generated, or used continuously, in the other subsystems without relation to the functions being performed by the central computer.

A high order language will reduce the coding effort required. The alternative is the use of an assembly language. A high order language can furnish greater visibility into program structure and the intended logic. Coding changes and validation corrections are quicker and easier. Coding in a compiler language has proved to be possible at nearly twice the rate of coding in assembly language and the coding errors are reduced 10 to 70% for different sized programs.

6.1.2 PROCESSING SPEED. The requirement for processing speed in excess of 400 thousand equivalent operations per second during engine burns forces consideration of dedicating a separate processing unit for the IMU calculations which represents a constant computing load of approximately 200 Kops. Two configurations for implementing this dedicated processor have been considered in this study. The effects on weight and power are shown in Table 6-2.

Table 6-2. Configurations to Raise Processing Speed Capability to Level of Requirement

Requirement		Capability	Alternate Configurations			
Processing Speed			Hardware Impact	Weight lb (kg)	Power, Watts	Dev Cost
Vehicle Management	216 Kops	Simplex Computer 300 → 400 Kops	Add separate redundant IMU processor (8K x 32)	Δ 30 (13.5)	Δ 50	Low
IMU Processing	200 Kops					
	416 Kops	Modular Computer 250 to 350 Kops per Module	Add CPU to IOP module	Δ 5 (2.3)	Δ 4	Low
Conclusion • A dual CPU modular computer will meet DMS processing speed requirements with small increase in weight or power						

The throughput capability of available simplex computers is from 300 to 400 Kops. To meet the processing speed requirement and provide growth capability, separate dual redundant computers dedicated to IMU processing can be added. An 8K memory is sufficient for this task. Dual redundant IMU processors would add 30 pounds (13.5 kilograms) and 50 watts to the system requirements.

6.1.3 ARCHITECTURE. Modular computers are made with the same basic logic technologies as simplex computers so that processing speed is not improved enough to overcome the processing speed limitation. However, the architecture permits independent modules, such as a CPU, to be added where necessary. The input/output processor (IOP) has direct access to the main memory so that a CPU added to the IOP could feed the processed navigation data into the main memory for use by the vehicle management software. This extra CPU module would add 5 pounds (2.3 kilograms) and 4 watts to the system requirements. The preferred configuration is the dual modular computer with a change in the IOP to add a dual processor dedicated to the task of processing IMU data.

6.1.4 SAFETY/RELIABILITY REDUNDANCY. Safety, reliability, and mission functions are additional factors that have a major impact on the computer configuration. Mission timeliness and the functional analysis of the mission establish the functions to be performed. The safety analysis establishes the safety limitations that impact hardware characteristics. The reliability goal is derived from an apportionment of failure rates to components in a manner that will achieve the 97% probability of mission success for the total vehicle.

The analyses for reliability and safety indicated a redundant data management subsystem is required. This means at least one backup unit is required for each element in the data management subsystem. Derivation of the minimum redundancy requirements is summarized in Figure 6-1. The safety analysis indicated a backup is required for the IMU and the main elements of the data management subsystem to guarantee stability during operations in the vicinity of the Orbiter. Flawless performance in this vicinity is necessary to ensure the safety of personnel. The reliability analysis provided the apportionment of unreliability of the total vehicle to each of its subsystems including the avionics subsystem. The reliability goal assigned to avionics in this manner was 0.992. The unreliability was further apportioned and the reliability goal assigned to the data management subsystem was 0.9953.

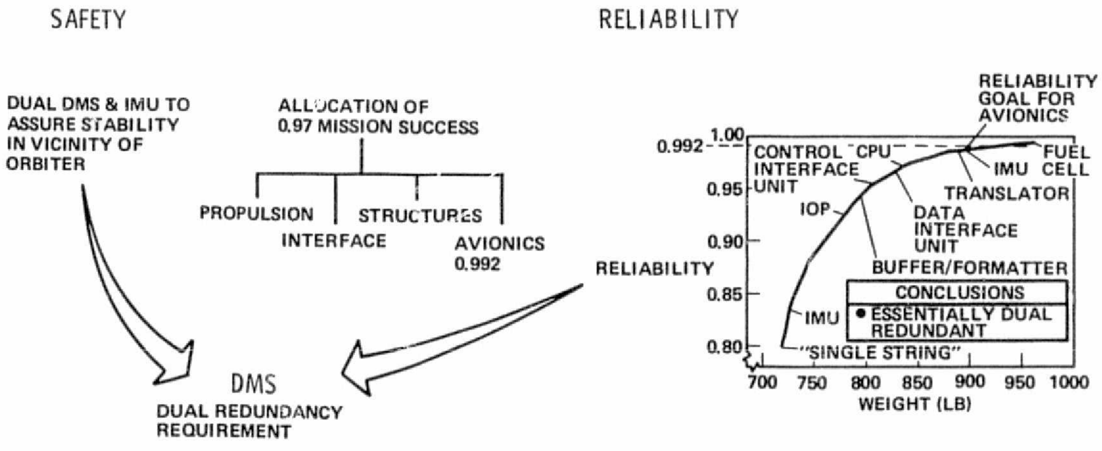


Figure 6-1. DMS Minimum Redundancy Requirements

A redundancy analysis of the avionics subsystem was made with a model that begins with a single string set of elements. For each configuration, the element is added that creates the greatest gain in reliability for the least addition to weight. This process was repeated in this analysis until the avionics subsystem goal of 0.992 was reached. This analysis showed that the avionics system goal of 0.992 could be achieved with a dual redundant CPU, IOP, CIU, DIU, and buffer formatter in a system with a modular computer using a memory with error detection and correction. A similar analysis using a simplex computer with the memory, CPU, and I/O packages to operate as a unit also showed a dual redundant computer would be needed to achieve the reliability goal of the avionics system.

Even though it is known that a backup computer is required, this information does not provide a guide to the choice of the computer configuration to meet the requirement. The reliability of the various configurations available was analyzed to determine the impact of redundancy configuration on the level of reliability. It was found that every configuration with a spare computer can exceed the reliability goal assigned to the data management subsystem, so that reliability does not limit the choice of a particular configuration. Results are shown in Figure 6-2. The data management subsystem goal for reliability is set at 0.9953. Dual redundant simplex computers with a coverage of 0.9 have a reliability of 0.9964. Other redundant systems have even higher reliabilities. There is a significant improvement in reliability with the dual modular

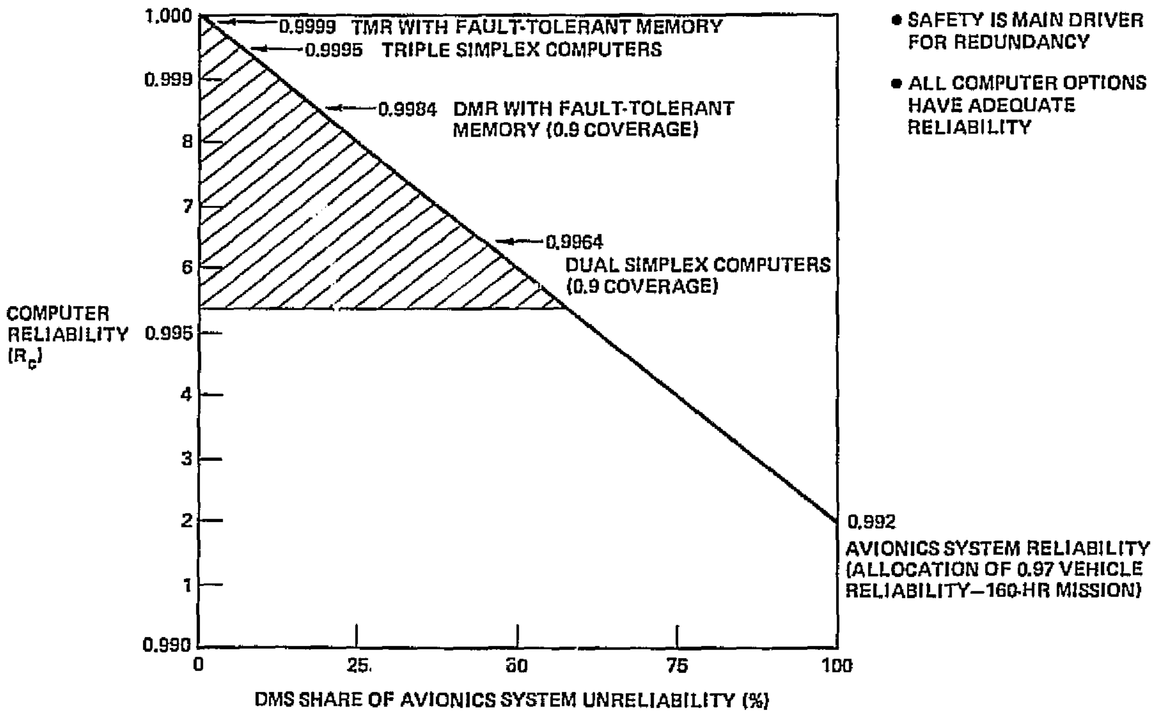


Figure 6-2. Evaluation of Redundant Computer Options

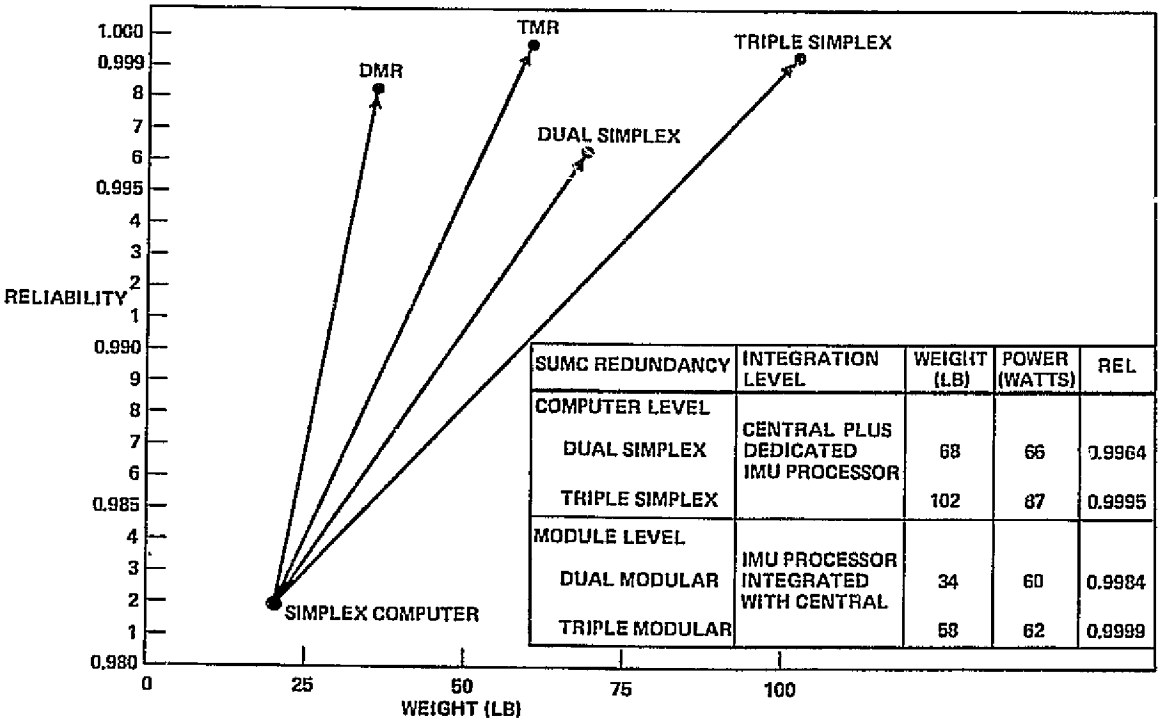
redundant (DMR) configuration as compared to dual simplex computers and triple modular redundant (TMR) computers have a higher reliability than triple simplex computers.

Even though calculations showed that every redundant configuration could meet the computer reliability goal, there was a significant difference in the weight and power. Figure 6-3 summarizes the results of a comparison of these characteristics. The DMR computer is the lowest weight configuration. Its weight is 34 pounds (15.3 kilograms) as compared to 58 pounds (26.1 kilograms) for a TMR computer. Both modular configurations have a tolerant memory that provides error detection and correction.

The results of this reliability and weight study of various redundant computer configurations lead to a recommendation of a DMR computer architecture with a fault tolerant memory as the preferred computer for the Tug Data Management Subsystem.

6.2 COMPUTER OPTIONS

The processing speed, memory capacity, word resolution, circuit technology, reliability, weight, and size goals are primary criteria for the choice of computer. Other characteristics are desirable features that can reinforce the selection rationale of a particular machine or concept.



SUMC REDUNDANCY	INTEGRATION LEVEL	WEIGHT (LB)	POWER (WATTS)	REL
COMPUTER LEVEL				
DUAL SIMPLEX	CENTRAL PLUS DEDICATED IMU PROCESSOR	68	66	0.9964
TRIPLE SIMPLEX		102	67	0.9995
MODULE LEVEL				
DUAL MODULAR	IMU PROCESSOR INTEGRATED WITH CENTRAL	34	60	0.9984
TRIPLE MODULAR		58	62	0.9999

Figure 6-3. Redundant Computer Weight and Power Comparison

Transistor transistor logic (TTL) circuit technology dominates the field of present generation airborne computers. Medium and large scale integrated circuits have been available for a number of years, establishing a proven performance and reliability history. TTL circuits offer high throughput speeds, good immur to natural or man made radiation, and interface commonality that has become standard throughout the computer and digital equipment industry. However, drawbacks do exist with TTL devices. TTL has high power requirements with system considerations for thermal design and reliability degradation due to thermal stress. Noise margins are relatively low, requiring careful circuit design to eliminate power supply transients and capacitive coupling of fast rise time noise into adjacent circuits.

Newer computer developments have been based on metal oxide semiconductor (MOS) integrated circuits. The significant advantage of these circuits is the very high density that can be attained on a single chip and the low power requirements of the MOS devices. Disadvantages with the early MOS structures are relatively slow propagation speeds and very high failure rates when exposed to long term radiation.

Future generation computer development will most probably utilize the emerging complementary MOS (CMOS) technology. CMOS technology was introduced six years ago and is now available from several semiconductor manufacturers. CMOS establishes an ideal set of operating characteristics for a spaceborne computer application. Advantages are zero quiescent power, high noise immunity, wide power supply range (regulation not critical), and high input impedance.

Silicon-on-sapphire (SOS) devices have a lower parasitic capacitance that permits three to five times speed improvement over corresponding bulk silicon devices. This speed improvement makes them speed competitive with older bipolar technologies without the large power penalty. In the SOS process, the substrate is an insulator that eliminates parasitic capacitances that decrease speed and increase power consumption, as commonly related to reverse-biased junctions and drain-source junctions in other MOS devices.

Figure 6-4 displays the spectrum of digital circuit technologies and maps the various computer candidates with their respective technology. CMOS/SOS large scale integrated circuits are recommended for Tug computer and other digital components since significant reliability, weight, and power advantages are gained. Current trends support a technology growth that should result in fully developed devices with proven performance history within the time frame of Tug development.

The MSFC baseline SUMC computer has redundant modules for its processor and input/output elements. The main memory provides error detection and correction on data and instruction transfers so that the whole computer is fault tolerant. The provision of error correction and backup hardware units enables the computer to meet the apportioned reliability goal associated with 97% probability of mission success.

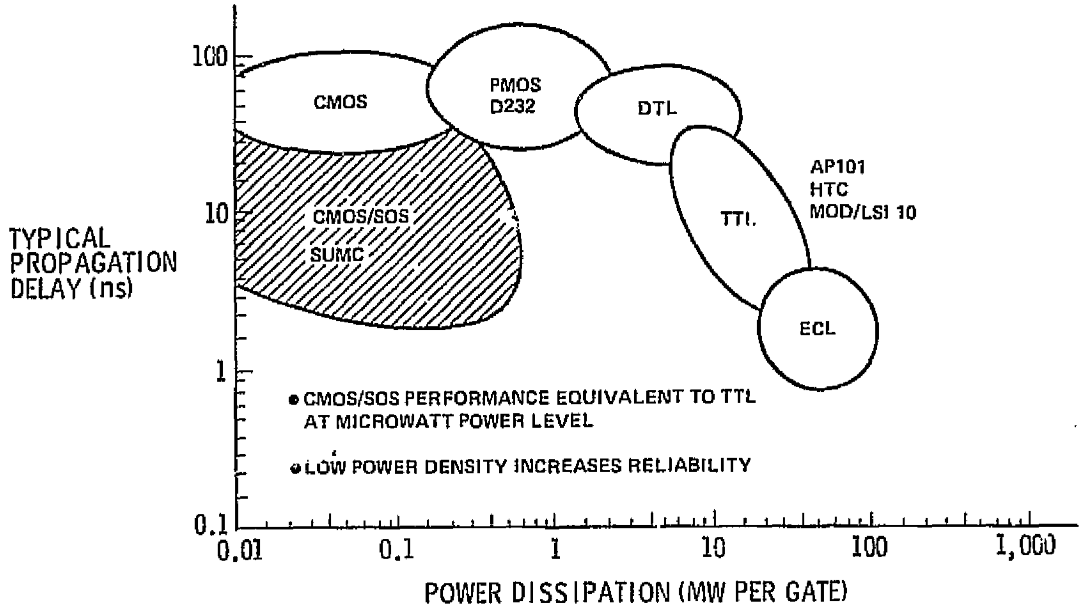


Figure 6-4. Computer Technology Spectrum

Other options exist in computer configurations with performance capability similar to the baseline computer. Computer suppliers were consulted to identify existing or developmental computers that can perform the Tug mission and are implemented with circuit technology expected of the 1980 time frame. The final choice of a computer may be strongly influenced by the cost of development, as well as production, but cost is not a factor in identifying machines and developments that will be available. Information was obtained on several machines, and the most promising candidates were examined in greater detail. Those listed in Table 6-3 should be considered as typical candidates for use on the Tug.

The Autonetics D232 is an advanced computer now being evaluated for other applications with the expectation that it would be fully qualified by 1978. Its use would represent a configuration with low hardware development cost.

The IBM AP 101 is a computer used on the Orbiter, with a unique Orbiter input/output, and it was examined for possible use as common hardware. The present packaging is air cooled, which represents an impact on cost, since redesign for cold-plate cooling and requalification are required.

The SUMC baseline computer prototypes were examined and compared to the other candidates. The modular architecture is particularly advantageous in overcoming processing speed limitations and implementing modular redundancy for the most weight effective approach to increased reliability. SUMC circuits are advanced

ORIGINAL PAGE IS
OF POOR QUALITY

Table 6-3. Characteristics of Candidate Computers

COMPUTER	CIRCUIT TECH	WORD LENGTH DATA BITS	WORD LENGTH INSTR BITS	NO. OF INSTR	INSTR COMPAT	HARDWR FLTING POINT	DMA CHANS	EXECUTION TIME ADD (µS)	EXECUTION TIME MULT (µS)	EXECUTION TIME DIV (µS)	MEMORY WORDS (MAX)	MEMORY TECH	FAULT DETECT	MICRO PROG CONT	8K × 32 BIT COMPUTER SIZE (CU IN.)	8K × 32 BIT COMPUTER WT (LB)	8K × 32 BIT COMPUTER PWR (WATTS)	MTBF (HR)	ASSESSMENT
AUTONETICS D232	PMOS/LSI	16/32	16/32	125	–	YES	1	2	11	21	64K × 32	PLATED WIRE	–	YES	935 (64K × 32 BIT MEMORY)	30	140	5,000	SIMPLEX ONLY; INSTR SET NOT 360 COMPATIBLE
IBM AP101	TTL/LSI	16/32	16/32	148	–	YES	1	1.65	5.45	8.65	32K × 32	CORE	PARITY & STORE PROTECT	YES	1,500	40	290	–	SHUTTLE COMPUTER–SIMPLEX VERSION ONLY. THERMAL DESIGN & REPACKAGING REQ'D FOR TUG
HTC	TTL/LSI	8/16/32	16/32/48	88	S/360	NO	1	2.4	27		16K × 32	NMOS	PARITY & STORE PROTECT	YES	350	18	94	10,000	MODULAR SUMC ARCHITECTURE TTL TECHNOLOGY DEGRADES RELIABILITY & THERMAL PERFORMANCE. PROTOTYPE 16-BIT VERSION AVAIL
SUMC	CMOS/SOS LSI	8/16/32/64	16/32/48	160	S/360	YES	5	7.5	10	35	256K × 32	CMOS	PARITY	YES	600	15	20	23,400	MODULAR SUMC ARCHITECTURE ADVANCED CIRCUIT TECHNOLOGY PROTOTYPE AVAIL MID-76
HONEYWELL MOD/LSI 10	TTL/LSI	32	16/32/48	85		YES	7	1.7	5.8	11.7	256K × 32	MOS	–	YES	400 (20K × 32 BIT MEMORY)	17	105	–	SMALL INSTRUCTION SET TTL TECHNOLOGY

technology CMOS/SOS large scale integrated circuits representing a low power, highly reliable implementation with little sacrifice in speed over present generation TTL computers.

Other modular computer options considered in this study were the IBM HTC and the Honeywell MOD/LSI 10, both representing advanced modular architecture. The HTC computer is well along in the development cycle with prototype hardware delivered to MSFC for a strapdown laser gyro experiment. A major disqualifier of the HTC for consideration as the Tug central computer is its limited main memory addressing capability. The Honeywell machine appears to have adequate performance specifications for the Tug mission; however, at this time, data on reliability and cost are not available for evaluation against the other options.

Consideration of modular architecture, large S/360 type instruction set, ample memory addressing capability, and high reliability through advanced CMOS/SOS circuitry leads to a recommendation of the SUMC computer for Tug DMS. Some reservations must be noted with this recommendation. The SUMC, at present, is being developed under MSFC contract to RCA for prototype delivery of a simplex modular computer equipped with a parity checked semiconductor memory by approximately mid-1976. To meet Tug requirements for a redundant modular configuration and fault tolerant memory, additional on-going development must occur to modify the CPU and IOP modules, data paths, and memory units for redundant operation with automatic failure sense and switch to backup elements. The time frame for the requirement of a Tug computer with full redundancy implemented is satisfactory for an orderly development, provided funds are available for additional module and circuit development work.

Since funding for future development can not be adequately assessed, alternative redundant configurations have been studied with low development cost as a key parameter in the selection criteria. Results of this study are shown in Table 6-4.

This data was generated by examining dual computer level redundant configurations of candidate machines listed in Table 6-3. Cost data shown represents total program costs for 15 Tug vehicles including spares and test units. Included in this analysis is the Autonetics DF224, a recently developed modular redundant computer with automatic reconfiguration control and 24 bit data and instruction paths. This machine was studied in two configurations, modular redundant and computer level redundant.

Results of the computer level redundant study lead to selection of a dual central computer with a dual dedicated processor for IMU management as shown in Figure 6-5. Dedicated IMU processing is recommended since the additional throughput approaches execution speed capability of the fastest machines during high dynamic mission phases such as main engine operation and rendezvous and docking.

Computer level redundancy implementation is a bulky solution to attain satisfactory system reliability and safety. A weight of 158 pounds (71.1 kilograms) for dual central and dual IMU processors represents a substantial penalty over the recommended SUMC modular redundant configuration of 34 pounds (15.3 kilograms).

Development cost for the dual simplex SUMC configuration is high since four computers are involved in system testing and the external reconfiguration control unit must be developed.

Table 6-4. Analysis of Redundant Low Development Computer Configurations

COMPUTER	CONFIGURATION	CHARACTERISTICS						PROGRAM COST ($1,000)			
			CIRCUIT TECH		DATA WORD (BITS)	WEIGHT (LB/KG)	POWER (IN.)				
		REF	CPU	MEM				DDT&E	PROD	OPS	TOTAL
SUMC	DUAL MODULAR FAULT TOL MEM	RCA	CMOS/ SOS	CMOS/ SOS	8/16/32/64	34/15	60	6,962	4,531	266	11,759
SUMC	DUAL 48K DUAL 8K	RCA	CMOS/ SOS	CMOS/ SOS	8/16/32/64	83/38	76	7,529	5,695	326	13,550
DF224	DUAL 48K DUAL 8K	RI	PMOS	PLATED WIRE	24	220/100	520	4,267	6,951	408	11,626
D232	DUAL 48K DUAL 8K	RI	PMOS	MNOS	16/32	79/36	576	4,297	5,898	346	10,539
AP101	DUAL 48K	IBM	TTL	CORE	16/32	125/56	525	4,109	6,937	309	11,355

SUMC – BEST CHOICE FOR TUG REQUIREMENT

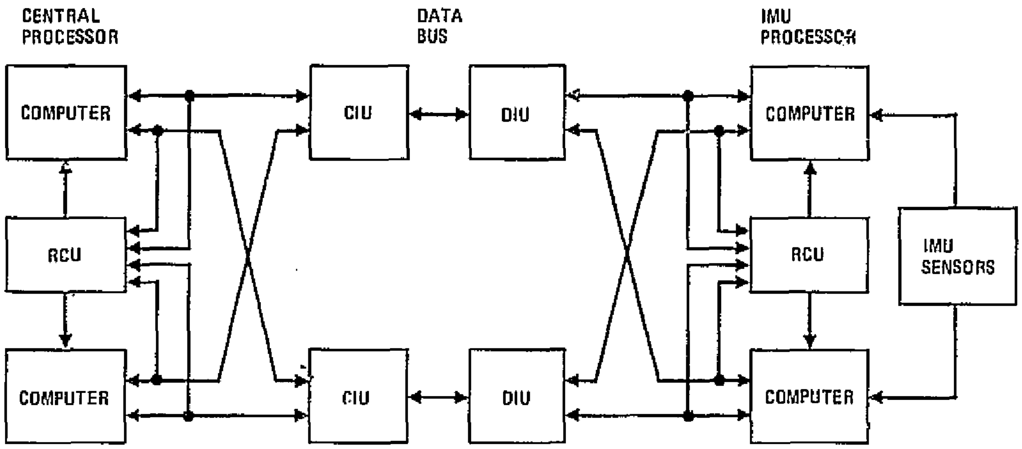


Figure 6-5. Computer Level Redundancy Configuration

Switching to the backup computer is controlled by an external reconfiguration controller monitoring memory parity, I/O wraparound tests, program timing, and software executed self tests. Fault coverage of up to 90% is estimated for a reconfiguration controller employing these techniques. Any detected malfunction causes automatic selection of the backup unit. The backup unit is synchronized and operating in parallel with the primary during critical mission phases so that switchover is nearly instantaneous and is transparent to the overall avionics system.

Finally, a survey of industry was conducted to determine availability of a low development modular redundant computer configuration. Autonetics DF224 fulfills most of the Tug computer requirements. It has a large instruction set, high reliability plated wire memory, fast execution speed, and input/output suitable for data bus operation. However, its 24 bit word length does not permit S/360 compatibility, and memory addressing is limited to 64K words. These limitations are not so significant to preclude consideration of the DF224 in a redundant modular configuration.

Two units house the dual redundant computer system as shown in Figure 6-6. The central computer contains 48K of memory and the IMU processor contains 8K of memory. A reconfiguration control unit, dedicated to each computer unit, monitors computer health and activates an external interrupt discrete upon fault detection of its associated computer. This module, in conjunction with the on-line backup computer, provides a fault tolerant system with very fast reconfiguration.

Each computer chassis, containing one 8K string, one 48K string, and two reconfiguration control units, has external dimensions of 18.0 in. (45.7 cm) × 18.7 in. (47.5 cm) × 12.8 in. (32.5 cm) and weighs 110 pounds (49.5 kilograms). This results in a total system weight of approximately 220 pounds (99 kilograms) and recurring cost of about $450,000 per shipset. Recurring cost and weight can be substantially reduced by substituting semiconductor memory for the present standard configuration

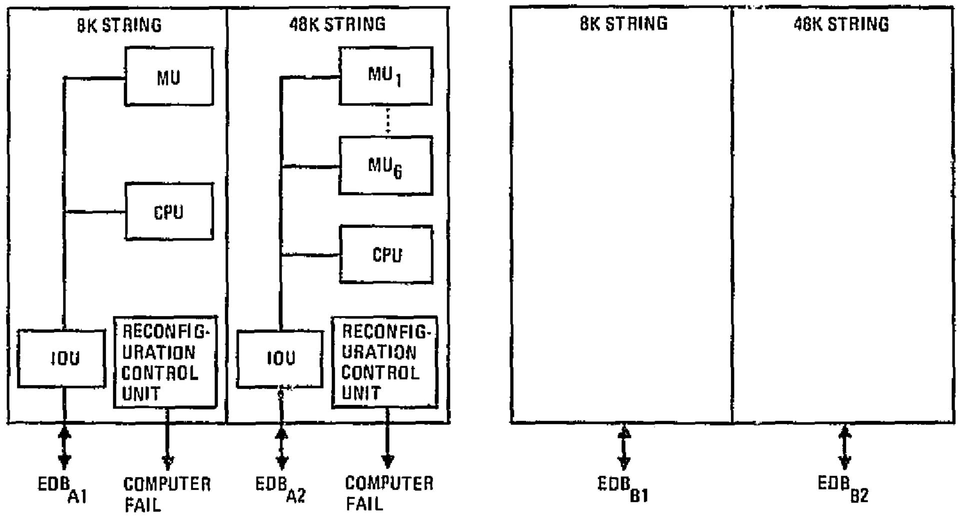


Figure 6-6. Dual Redundant Configurations Based on the DF224 Modular Computer

implemented with plated wire memory. However, this departs from the qualified baseline and would raise development costs.

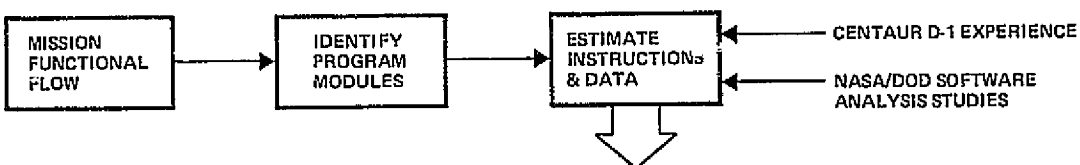
No low development computer level redundancy, or existing modular level redundancy implementation was found that satisfactorily met Tug requirements, without imposing large weight and power penalties and substantial recurring costs. If development of a redundant modular computer, such as SUMC, can not be funded, then dual simplex computers with 48K × 32 bit memory capacity, semiconductor memory technology, S/360 instruction compatibility, and an external reconfiguration control unit would be the recommended alternate configuration. Assuming selection of an existing computer, development costs for minor changes and documentation should be less than $5 million, and total program recurring costs should lie in the range of $10 million per shipset.

6.3 SOFTWARE REQUIREMENTS ANALYSIS

Software requirements for the Tug DMS are highly sensitive to the level of autonomy imposed on the DMS. In this analysis, Level II autonomy is assumed with the DMS controlling guidance, navigation, flight control, sequencing, and redundancy management of the various subsystem elements. Ground override or update capability can be exercised through the Communications subsystem for any of these computerized functions.

In developing an estimate of the total software package, the mission functional flow diagrams were analyzed to determine a set of software modules satisfying all phases of the Tug mission from deployment, transfer to injection orbit, injection and/or rendezvous and docking, and return to Orbiter for retrieval. Complexity of the module set identified was then evaluated against existing executive and applications programs recently generated for the Titan Centaur launch vehicle flight program. Where additional complexity was apparent, due to new or advanced mission capability, a software delta was estimated for the Tug module. Figure 6-7 summarizes the results of this effort to arrive at an estimate of the total amount of software (instructions and data) required to implement the DMS.

Software for navigation and guidance sustained substantial increases due to strapdown guidance calculations, Kalman filtering, dodecahedron IMU management, and storage of the S-band radar coordinate catalog for the Interferometric Landmark Tracker (ILT). Four telemetry formats have been identified to meet Tug data requirements. These formats are a low level status format representing minimal system status information, high level status format containing more extensive status information and intermediate results of calculations, maintenance data format for vehicle instrumentation, and a rendezvous and docking format containing encoded TV with associated position data. Compared to the Centaur upper stage, these represent about twice the present telemetry format software. The sequencing software module was also much larger due to the requirements for multiburn missions, rendezvous and docking, and multiple spacecraft sequencing. Rendezvous and docking software is an entirely new software package. The 3500 words of instruction and data represents the necessary filtering and control algorithms to accomplish a semiautonomous rendezvous and docking mission phase. Finally, checkout, redundancy management, and the attendant utility programs comprise a substantial software increment. These programs



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

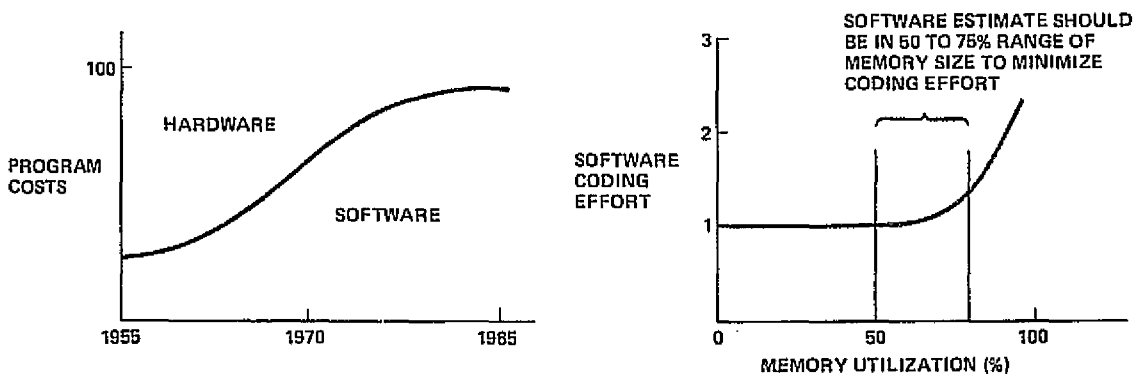
Figure 6-7. Development of Software Estimate

must evaluate system health, reconfigure if a faulty condition occurs, and maintain a log of redundant equipment status. All other software modules were similar to existing functions requiring very small increases in instruction code and data storage.

The baseline computer for DMS is a 32 bit/word machine operating with a S/360 instruction set. An important advantage of this instruction set is the capability to operate with both long (32 bit) and short (16 bit) instructions. Previous experience indicates that a typical mix of instructions is roughly 75% short instruction and 25% long instruction. Applying this instruction mix ratio to the previous instruction estimate leads to a memory requirement of 25, 391 words.

High order language (HOL) program generation has been shown to be effective for a large share of the Tug software. The software compiler, which converts HOL programming to machine language, performs less efficiently than optimum in assigning memory to machine code. Various multipliers are used in the software industry to estimate the additional memory burden, usually ranging from 10 to 25% increase. A conservative 20% overhead was assumed for the HOL inefficiency to arrive at a total estimated memory storage requirement of 30, 469 words.

Selection of memory size from initial software estimates depends upon several factors; confidence in the estimate, cost of memory, and expected growth in system capability. The study approach of using existing software from a similar space vehicle control computer as the basis for establishing estimate rationale lends a high degree of confidence in the estimate of data and instruction required for the total Tug



REF:
RAND REPORT
RM-6213-PR
(JAN 70)

SOFTWARE ESTIMATE	30,469 WORDS
SELECTED MEMORY SIZE	49,152 WORDS
MEMORY UTILIZATION	62%

Figure 6-8. Memory Size Selection

mission. A recent report by the Rand Corporation shows the rising cost of software and declining hardware costs projected for the next 10 years (Figure 6-8) leads to selection of a large capacity memory to reduce software development effort. Generally the memory utilization factor, based on initial software estimates, should be in the range of 50 to 75% of selected memory size depending on the confidence level of the software estimate and expected system capability growth.

The recommended memory size of 49,152 words results in a memory utilization factor of 62% and should provide sufficient margin to minimize software coding effort.

An important factor in reducing software coding effort is a high order language (HOL) for convenient interaction of programmer and machine. A HOL allows management visibility into the software structure and its capability without requiring an intimate knowledge of machine architecture and instruction set. Several languages exist with varying indices of performance and program coding effectiveness, based on the ratio of machine code generated by a set of HOL statements. Assembly or mnemonic language with a coding effectiveness of one is the least efficient method of generating program code. However, several advantages of assembly level programming make it a vital part of a programmer language repertoire. Code generation at the assembly language level allows experienced programmers flexibility in optimizing program code either for execution time or minimum memory utilization. Usually these two optimization criteria are in conflict, and this conflict represents the foremost criticism of HOL programming, especially when one considers the throughput and loop servicing constraints for a central computer of the DMS.

This study attempted to evaluate several programming languages and relate the programming effectiveness of each HOL against assembly language. A summary of results is shown in Figure 6-9. Present generation languages, such as Fortran, Jovial, Goal, and SPL/J6 showed an improvement factor of two to four machine instructions generated for each programmer statement. Higher order Algorithmic Language (HAL) showed the best performance index, averaging six instructions per program statement over a wide variety of benchmark programs, such as matrix manipulation, sorting, typical navigation and guidance subroutines. HAL is under development for application to the Shuttle programming problem, and several compilers are in existence at this time to generate machine code for host or target flight computers. Estimates of memory size increase due to inefficiency in memory assignment of the HAL compiler range from 15 to 20%, which is typical of other languages evaluated. Real time throughput and processing constraints are handled by software time tagging of execution priorities and insertion of assembly language subroutines for highly critical processing.

The total software development cycle is typically made up of three major phases of effort, analysis and design — 35%, coding and documentation — 20%, validation and test — 45%. A high order language, such as HAL, could substantially reduce the time and effort required in the analysis, design, coding, and documentation phases

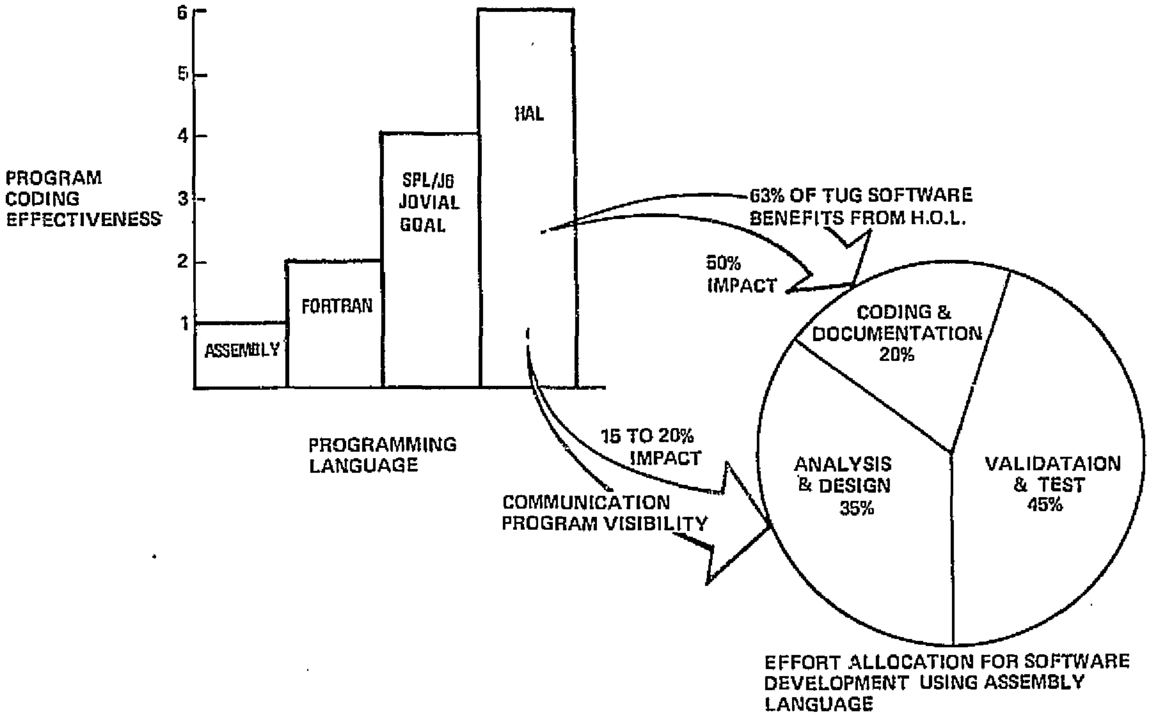


Figure 6-9. Language Recommendation

with very little impact on the validation and test phases. Evaluation of the programs and subroutines required for Tug software indicates roughly 63% of the total software is amenable to HOL programming, mathematical calculations, simple sorting, comparing, and control sequencing. Reduction in the total software development time and effort with HAL programming would most likely range from 20 to 40% and result in a highly visible program documentation that allows program management techniques to be applied similar to those imposed on a hardware development task, such as design reviews and establishment of critical milestones for an orderly development cycle.

6.4 SYSTEM REDUNDANCY MANAGEMENT TECHNIQUES

The data management subsystem is essentially dual redundant to meet both the reliability goal and the fail operational requirement for system safety. 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.

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 were shown in Figure 6-1. 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.

Table 6-5 summarizes the types of redundancy utilized throughout the Tug avionics system.

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 rapid 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.

Table 6-5. 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
GN&C			DIU CROSSTRAPPED TO LRUS
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

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 making of failures. Figure 6-10 depicts the techniques used in management of the primary and backup units in the Tug computer.

A basic premise for 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.

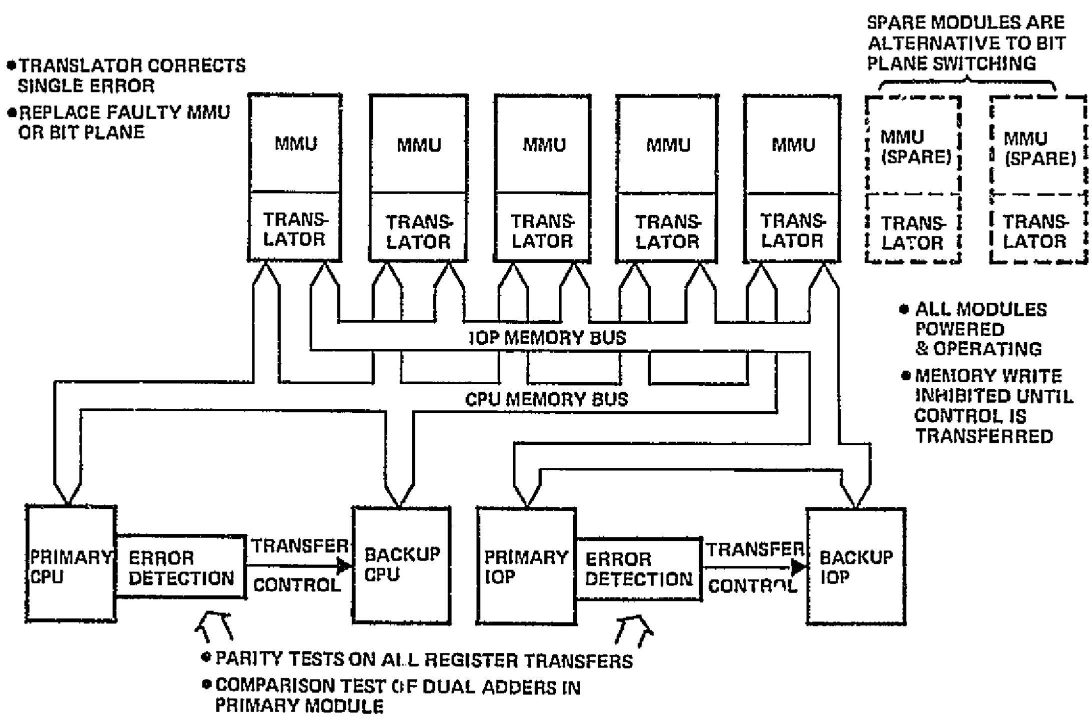


Figure 6-10. Fault-Tolerant Computer Redundancy Management

SECTION 7

TUG CHECKOUT

Three major trade studies served as the basic drivers in establishing the Tug checkout system configuration. These studies involve the issues of checkout philosophy, allocation, and implementation.

A summary of the checkout system resulting from the final trade iterations is provided as follows:

a. The general philosophy is Condition Monitored Maintenance with preflight testing (CMM_{PF})

b. The Data Management subsystem is utilized for executing test instructions, and storing and processing data.

c. Tug onboard checkout software subtotal is 8.9K words.

   1. Safety - 1.5K words
   2. Status - 6.2K
   3. Initialization - 0.6K
   4. Partial functional - 0.6K

d. 1200 words of memory set aside for payload checkout.

e. 320 Mbits write/read maintenance data storage capability on NASA standard tape recorder.

Volume III contains a complete description of the Tug checkout system.

7.1 CHECKOUT PHILOSOPHY TRADE

The intent of any checkout effort is to establish confidence that the item being checked will perform to expectations. Six checkout philosophies and their characteristics were defined to cover the philosophy spectrum from no testing to extensive testing in that order:

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_{PF}) — full trend analysis

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

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

"Hands Off" Test and Checkout Philosophy — This test and checkout philosophy is essentially one of no test and checkout. Once a component, subsystem or system has been installed and checked out, there are no further checks conducted during the operational life. Simply fly it or use it until it fails.

This philosophy does not lend itself to detecting random failures or to providing data for use in trend analysis to support preplanned maintenance and refurbishment actions. On the other hand, there is very little impact on Tug design since the concept requires no special features to accommodate tests, checkouts or planned refurbishment. Further, system degradation caused by disturbing system integrity to make test connections, can be avoided. It was experience with this latter problem that led the USAF to adopt a "Hands Off" philosophy on the Atlas ICBM. Another general application — practiced to a greater or lesser degree — is the family car; it is usually driven until it breaks down. While the consequences of such a philosophy are not too serious with the family car or an ICBM in a standby alert mode, Tug inflight failures can result in lost Tugs, payloads and missions and add considerably to overall program costs.

"Hard Time — Remove and Replace" Test and Checkout Philosophy — This philosophy like hands off, is another form of a no test and checkout philosophy. In this case, components or LRUs are removed and replaced every X hours, cycles, or events without regard to their actual status or condition. The removal time period is determined by engineering analysis supported by development test results. This philosophy is based largely on the assumption that there is a straightline relationship between wearout or performance degradation and age. Further it assumes that component/ LRU replacement at stated intervals can prevent failures.

This philosophy does not lend itself to detecting random failures or predicting data for trend analysis. It is however, readily applicable to certain electromechanical and mechanical components such as relays and fluid valves. Application of this philosophy for the whole Tug can result in the requirement for extensive spares inventory. It can also result in removal of items which still have a useful life, resulting in unnecessary costs.

This philosophy has been used extensively on USAF and commercial aircraft, but, was not considered applicable for Tug systems.

"Hard Time — Test" Test and Checkout Philosophy — This philosophy calls for test or checkout at prescribed time/cycle intervals to ascertain that performance falls within parameter limits. The assumption here is that inspection and test at fixed intervals can detect, and thus prevent, failures. The decision to remove and replace is based on test data rather than engineering analysis and predicted useful life. The concept is similar to test and retest, but is time-related rather than event-related where event-related refers to events such as discrete events or functions in the turnaround cycle.

Preliminary analysis indicates that turnaround costs can be expected to be about the same as hard time — remove and replace. Under this test philosophy, more ground checkout equipment and tests are balanced against more spares and unnecessary replacements associated with hard time — remove and replace philosophy. This concept was employed on the USAF Atlas ICBM program and is still selectively used by commercial airlines where it is referred to as "on-condition" maintenance.

"Condition Monitored Maintenance With No Preflight ($CMM_{\overline{PF}}$)" Test and Checkout Philosophy — The $CMM_{\overline{PF}}$ philosophy is the concept now practiced by most major commercial airlines (supplemented by on-condition for selected items). This philosophy relies solely on technical analysis of flight data and crew reports as a basis for committing to a subsequent mission. There is no planned testing between flights.

Application of this philosophy to the Tug would require a high degree of confidence in vehicle performance and analysis of performance data. It might be useful as an operations goal after several years of Tug flight experience.

"Condition Monitored Maintenance With Preflight Testing (CMM_{PF})" — Test and Checkout Philosophy — The CMM_{PF} philosophy is based on the assumption that performing a mission is the best system test. In this concept, system performance is monitored during the mission to detect data trends toward parameter limits. These data, combined with an integrated systems preflight checkout, provide the basis for committing to the next mission.

Remove and replace action is based on an engineering analysis of the flight data. This permits preplanned maintenance.

The CMM_{PF} philosophy is a direct outgrowth of the CMM concept developed and used by commercial airlines. It does have a significant impact on the Tug in that it requires considerable on-board performance monitoring capability. This capability results in reliable, fast and low cost turnaround.

"Test and Retest" Test and Checkout Philosophy — This philosophy calls for extensive, detailed testing during turnaround and prelaunch operations at all levels (i.e., component, subsystem, system, and integrated vehicle level). In general, the system and integrated vehicle tests will simulate, to the maximum extent possible, the actual operational conditions. This philosophy leads to such time consuming checkouts as cryogenic tankings and full scale countdown demonstrations.

Implementation of this philosophy provides the best opportunity to detect random failures and provides data useful for trend analysis and permits the establishment of useful life times based on specific test data. It results in a high confidence that all systems are operating within prescribed parameter limits before committing to launch.

This philosophy will result in the longest turnaround time. This in turn generates high turnaround costs associated with the larger number of test personnel utilized for longer times as well as more extensive ground checkout equipment required to accomplish the tests. It will also have some impact on Tug design in that more test points must be provided.

This philosophy is presently used on the expendable launch vehicles and spacecraft.

Summary — The CMM_{PF} philosophy is a direct outgrowth of the CMM concept developed and used by commercial airlines. It has a significant impact on the Tug in that is requires considerable on-board performance monitoring capability. This capability results in reliable, fast and low cost turnaround. It is recognized that not all Tug systems or components will lend themselves to the CMM concept. In areas such as structures, performance monitoring during flight is not practical nor within the baseline on board instrumentation capabilities. For these systems, the Hard Time-Test or Hard Time-Remove and Replace concept would be used to supplement CMM in determining required maintenance actions and ascertaining flight readiness status.

7.2 TEST REQUIREMENTS, SUPPORT, AND ALLOCATION TRADE

7.2.1 TEST REQUIREMENTS. The logical steps or sequences used to implement a philosophy are known as tests. Basic elements of a test have been identified and six categories of test defined. These categories, as shown in Figure 7-1, typify the various test characteristics required to accommodate the implementation of the checkout philosophy within system operational constraints.

Measurements obtained from the unit under test have been categorized as operational, functional, and instrumentation-derived, in anticipation of the need to differentiate between the methods required to access each category. Shaded portions of the six test categories indicate the support/measurement mix between them, which, in fact, establishes their unique characteristics.

When these six tests should be performed involves the judgement as to which components each of the test type should be applied, taking into consideration the particular flight or ground operational phase. This judgement for Tug checkout is shown in Tables 7-1 through 7-6. Each table is a matrix showing which elements are being tested by a particular test during the 10 different operational flight and ground phases. The distribution of test activity formed by these matrices provides a basis for allocating the responsibility for performing these tests as discussed in Section 7.3.

7.2.2 SUPPORT REQUIREMENTS. Along the the basic tests required to be performed during each mission phase, the individual component test parameter support requirements need identification. The test parameters provide the means of assessing the functional operability or state of the component. They are implemented by specifying the proper combination of input and output measurements required to

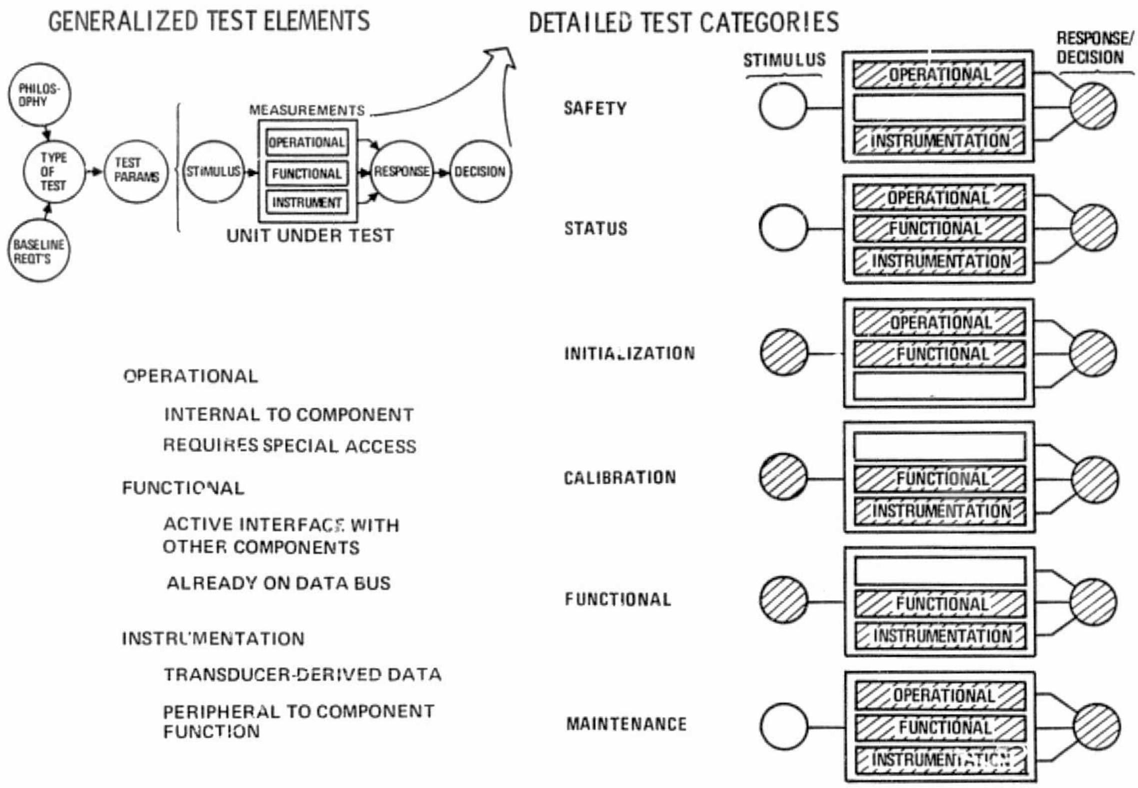


Figure 7-1. Test Implementation Philosophy

synthesize the parameter. Processing of the measurements, or the actual parameter synthesis, is another implementation/support requirement along with a criterion comparison and decision on the acceptability of the parameter.

A representative set of test parameters was generated for each component at each level of test. Parameters required to be available for each test category were then converted to measurement (or test data) and support requirements. These requirements are provided in terms of

a. Software required to execute the test, evaluate the results, and apply an acceptability criterion.

b. Measurements required as input data to the evaluation.

The total component support requirements are applied to each test category and summed to provide a quantitative basis for the checkout structure.

Total software support is shown categorically in Figure 7-2 and represents that software directly chargeable to each of the checkout tests. Note that only a small amount of coding is anticipated in support of Tug peculiar maintenance data processing since basic trend analysis subroutines and methods are being examined under KSC contract and are expected to be available for Tug. The total number of measurements to be

Table 7-1. Elements Undergoing Functional Tests

Elements Undergoing Test	Operational Phases									
	Pre Launch	Orbiter Ascent	On-Orbit	Tug Deploy	Tug Ascent	P/L Deploy/ Ret	Tug Descent	Orbiter Capture	Orbiter Descent	Grnd Ops
1. Computer	x									x
2. CIU	x									x
3. DIU										x
4. Tape Recorder	x									x
5. IMU & Electronics	x		x							x
6. Sun Sensor	x		x							x
7. Star Tracker	x		x							x
8. ILT			x							x
9. Control Electronics	x		x							x
10. Rate Gyros	x		x							x
11. Laser Radar/Electronics	x		x							x
12. TV Camera & Electronics	x		x							x
13. Strobe Lights/Electronics	x									x
14. Phased Array	x		x							x
15. Network/Switch Assemb/Omni	x									x
16. Transponder	x									x
17. Signal Processing	x									x
18. CMD Distribution	x									x
19. Sensors	x		x	x						x
20. Signal Conditioner	x									x
21. Fuel Cell		x								x
22. Emergency Battery										x
23. Power Processing		x								x
24. Power Distribution	x	x								x
25. Harnesses	x									x
26. Waste Heat Loop	x		x							x
27. Plumbing										
28. Thermal Control	x		x							x
29. Main Engine										x
30. APS										x
31. FF&D										x
32. VAPS										x
33. Tanks										x
34. Interface Orbiter	x									x
35. Interface Ground	x									x
36. Interface Spacecraft	x									x
Totals	25	3	12	1	0	0	0	0	0	35

Table 7-2. Elements Undergoing Status Verification Tests

Elements Undergoing Test	Operational Phases									
	Pre Launch	Orbiter Ascent	On-Orbit	Tug Deploy	Tug Ascent	P/L Deploy/Ret	Tug Descent	Orbiter Capture	Orbiter Descent	Grnd Ops
1. Computer	x	x	x	x	x	x	x	x	x	x
2. CIU	x	x	x	x	x	x	x	x	x	
3. DIU	x	x	x	x	x	x	x	x	x	
4. Tape Recorder			x	x	x	x	x			
5. IMU & Electronics		x	x	x	x	x	x	x		
6. Sun Sensor			x	x	x	x	x			
7. Star Tracker			x	x	x	x	x			
8. ILT			x	x	x	x	x	x		
9. Control Electronics		x	x	x	x	x	x	x		
10. Rate Gyros		x	x	x	x	x	x	x		
11. Laser Radar/Electronics			x			x	x			
12. TV Camera & Electronics			x			x	x			
13. Strobe Lights/Electronics			x			x	x			
14. Phased Array			x	x	x	x	x	x		
15. Network/Switch Assemb/Omni		x	x	x	x	x	x	x		
16. Transponder		x	x	x	x	x	x	x	x	
17. Signal Processing		x	x	x	x	x	x	x	x	
18. CMD Distribution		x	x	x	x	x	x	x	x	
19. Sensors	x	x						x	x	x
20. Signal Conditioner	x	x	x	x	x	x	x	x	x	x
21. Fuel Cell		x	x	x	x	x	x	x		
22. Emergency Battery		x	x	x			x	x		
23. Power Processing		x	x	x	x	x	x	x		
24. Power Distribution		x	x	x	x	x	x	x		
25. Harnesses										
26. Waste Heat Loop		x	x	x		x	x	x		
27. Plumbing										
28. Thermal Control		x	x	x				x		
29. Main Engine			x	x				x		
30. APS			x	x				x		
31. FF&D	x	x	x	x				x	x	
32. VAPS	x	x	x	x	x	x	x	x	x	x
33. Tanks										
34. Interface Orbiter	x	x		x				x	x	
35. Interface Ground										
36. Interface Spacecraft		x	x	x				x	x	
Totals	8	22	30	28	20	24	24	26	12	4

Table 7-3. Elements Undergoing Safety Monitoring

Elements Undergoing Test	Operational Phases									
	Pre Launch	Orbiter Ascent	On-Orbit	Tug Deploy	Tug Ascent	P/L Deploy/Ret	Tug Descent	Orbiter Capture	Orbiter Descent	Grnd Ops
1. Computer				x						
2. CIU										
3. DIU										
4. Tape Recorder										
5. IMU & Electronics										
6. Sun Sensor										
7. Star Tracker										
8. ILT										
9. Control Electronics				x			x	x		
10. Rate Gyros										
11. Laser Radar/Electronics										
12. TV Camera & Electronics										
13. Strobe Lights/Electronics										
14. Phased Array										
15. Network/Switch Assemb/Omni										
16. Transponder										
17. Signal Processing										
18. CMD Distribution										
19. Sensors	x	x	x	x			x	x	x	x
20. Signal Conditioner										
21. Fuel Cell		x	x	x			x	x		
22. Emergency Battery		x	x	x			x	x		
23. Power Processing										
24. Power Distribution										
25. Harnesses										
26. Waste Heat Loop		x	x							
27. Plumbing										
28. Thermal Control										
29. Main Engine		x	x	x			x	x		
30. APS		x	x	x			x	x		
31. FF&D		x	x					x		
32. VAPS	x	x	x	x			x	x	x	
33. Tanks										
34. Interface Orbiter			x	x			x	x		
35. Interface Ground										
36. Interface Spacecraft										
Totals	2	8	9	9	0	0	8	9	2	1

Table 7-4. Elements Undergoing Calibration Tests

Elements Undergoing Test	Operational Phases: Pre Launch	Orbiter Ascent	On-Orbit	Tug Deploy	Tug Ascent	P/L Deploy/Ret	Tug Descent	Orbiter Capture	Orbiter Descent	Grnd Ops
1. Computer										
2. CIU										
3. DIU										
4. Tape Recorder	x									x
5. IMU & Electronics	x	x	x							x
6. Sun Sensor	x								x	
7. Star Tracker	x									x
8. ILT	x									x
9. Control Electronics										
10. Rate Gyros										
11. Laser Radar/Electronics	x									x
12. TV Camera & Electronics	x									x
13. Strobe Lights/Electronics										
14. Phased Array										
15. Network/Switch Assemb/Omni										
16. Transponder										
17. Signal Processing										
18. CMD Distribution										
19. Sensors	x									x
20. Signal Conditioner	x									x
21. Fuel Cell										
22. Emergency Battery										
23. Power Processing										
24. Power Distribution										
25. Harnesses										
26. Waste Heat Loop										
27. Plumbing										
28. Thermal Control										
29. Main Engine										
30. APS										
31. FF&D										
32. VAPS	x									x
33. Tanks										
34. Interface Orbiter										
35. Interface Ground										
36. Interface Spacecraft										
Totals	10	1	1							10

Table 7-5. Elements Undergoing Maintenance Data Acquisition Monitoring

Elements Undergoing Test	Operational Phases									
	Pre Launch	Orbiter Ascent	On-Orbit	Tug Deploy	Tug Ascent	P/L Deploy/Ret	Tug Descent	Orbiter Capture	Orbiter Descent	Grnd Ops
1. Computer										
2. CIU										
3. DIU										x
4. Tape Recorder										
5. IMU & Electronics										
6. Sun Sensor					x		x			
7. Star Tracker					x		x			
8. ILT					x		x			
9. Control Electronics										
10. Rate Gyros										
11. Laser Radar/Electronics						x	x			
12. TV Camera & Electronics										
13. Strobe Lights/Electronics										
14. Phased Array										x
15. Network/Switch Assemb/Omni										x
16. Transponder										x
17. Signal Processing										x
18. CMD Distribution										x
19. Sensors										
20. Signal Conditioner										x
21. Fuel Cell		x	x		x	x				x
22. Emergency Battery										
23. Power Processing										
24. Power Distribution										
25. Harnesses										
26. Waste Heat Loop		x	x	x	x					
27. Plumbing				x						
28. Thermal Control				x		x				
29. Main Engine				x	x	x	x			
30. APS				x		x				
31. FF&D				x		x	x			
32. VAPS				x		x	x			
33. Tanks				x						
34. Interface Orbiter										
35. Interface Ground										
36. Interface Spacecraft										
Totals	0	2	2	8	6	7	7	0	0	11

Table 7-6. Elements Undergoing Initialization Tests

Elements Undergoing Test	Operational Phases									
	Pre Launch	Orbiter Ascent	On-Orbit	Tug Deploy	Tug Ascent	P/L Deploy/Ret	Tug Descent	Orbiter Capture	Orbiter Descent	Grnd Ops
1. Computer	x	x	x							x
2. CIU		x								x
3. DIU		x								x
4. Tape Recorder			x							x
5. IMU & Electronics			x							x
6. Sun Sensor										
7. Star Tracker				x						
8. ILT				x						
9. Control Electronics		x								
10. Rate Gyros			x							
11. Laser Radar/Electronics			x			x				
12. TV Camera & Electronics						x				
13. Strobe Lights/Electronics						x				
14. Phased Array			x							x
15. Network/Switch Assemb/Omni		x								x
16. Transponder		x								x
17. Signal Processing		x	x							x
18. CMD Distribution		x	x							x
19. Sensors		x								
20. Signal Conditioner		x								x
21. Fuel Cell	x									x
22. Emergency Battery										x
23. Power Processing		x	x							x
24. Power Distribution	x		x							x
25. Harnesses										x
26. Waste Heat Loop	x		x							x
27. Plumbing										
28. Thermal Control		x								
29. Main Engine										
30. APS			x							
31. FF&D	x	x			x					x
32. VAPS		x								x
33. Tanks										
34. Interface Orbiter										
35. Interface Ground										
36. Interface Spacecraft										
Totals	5	14	12	2	0	3	1	0	0	20

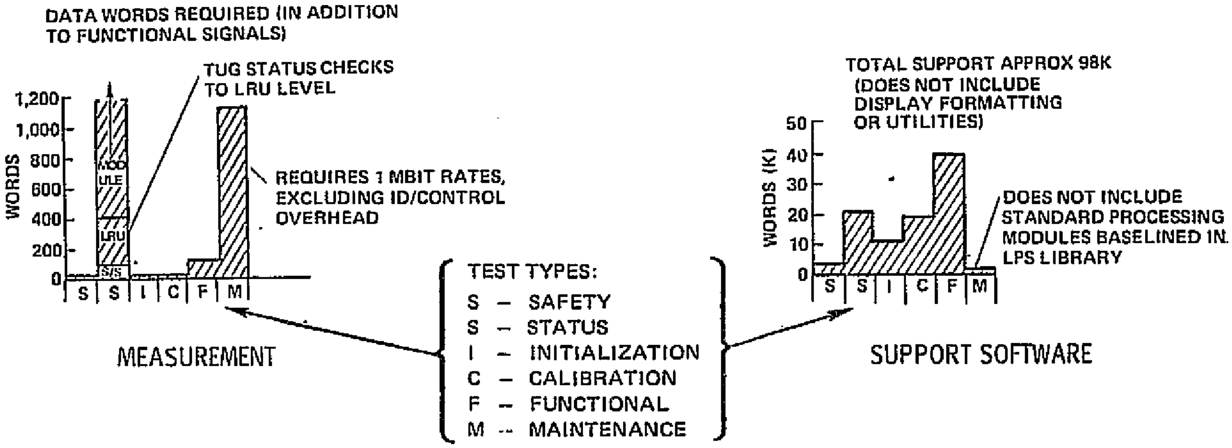


Figure 7-2. Software Support

acquired and processed to support Tug checkout is also shown. Again, these values reflect those directly chargeable to the checkout function. Functional measurements (those appearing normally in the system just to make it work) are not considered checkout overhead. Status checks are assumed to be to the component or line replaceable unit (LRU) level to ensure compliance with requirements and to limit the geometric expansion of the measurement array when moving down to module level isolation. Total Tug checkout software is 98K words (32 bits each), and does not include software required to process maintenance data. Measurements required to support the CMM_{PF} Tug checkout philosophy account for an additional 1825 words of data.

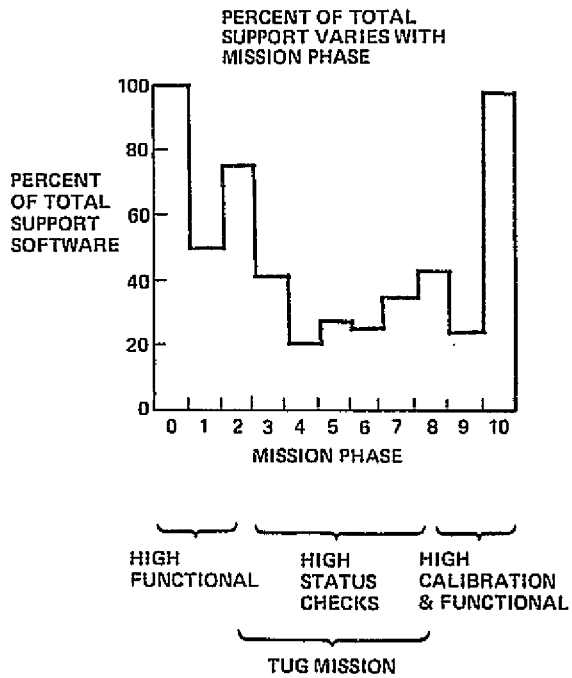


Figure 7-3. Total Software Support Versus Mission Phase

Tug total support, categorized by the tests it implements and depicted in Figure 7-3, is spread throughout the Tug operations cycle/mission profile to provide the basis for establishing the checkout allocation. Both test and support requirements were primarily a requirements coordinating effort necessary to provide a basis for the allocation, which is the second major area of trade.

7.2.3 ALLOCATION. The distribution of components being tested from the matrices on the previous tables (summarized in Table 7-7) leads to an allocation of where the tests should be run, based on the following criteria: recurring test demands (status tests), phase-peculiar testing (safety), and the requirement for high software memory storage with little usage (functional tests). The allocation is shown in Figure 7-4 including the amount of software memory associated with the Tug Shuttle and ground.

Table 7-7. Elements Undergoing Tests

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

	GROUND PHASES	ORBITER PHASES	TUG PHASES
	GROUND ALLOCATION	ORBITER ALLOCATION	TUG ALLOCATION
	FUNCTIONAL TEST CALIBRATION MAINTENANCE PROCESSING	SAFETY (MONITOR)	SAFETY (REACTION SEQUENCE) STATUS INITIALIZATION PARTIAL FUNCTIONAL MAINTENANCE ACQUISITION
TOTAL:	88 K WORDS	1.5 K WORDS	8.9 K WORDS

Figure 7-4. Checkout Software Allocation

7.3 ONBOARD CHECKOUT IMPLEMENTATION OPTIONS

7.3.1 STATUS AS A DRIVER. Since all support requirements up to this point have been predicated on an unintegrated system, the status requirements driving the implementation reflect negligible built-in-test-equipment (BITE) in the system. All trades involving BITE tend to drive the system software and measurement requirements down. Therefore, the trade was basically one of determining the desirability of reducing DMS responsibility for executing the tests. The approach taken was to assess the impact of the relatively large amount of data and support and then define a DMS capability threshold wherein BITE becomes necessary. As shown in Figure 7-5, the DMS displayed a high tolerance and capability with a low BITE implementation and did not in itself impose BITE development on the Tug.

The analysis required to support the trade was based on a modular software architecture. The dominant feature of this architecture with respect to the Tug implementation is that a single subsystem component interface with the DMS requires only a small software link so that, in terms of storage, the DMS is relatively insensitive to the number of things plugged into it. DMS duty cycle overhead is driven by critical subsystems requiring very high speed status data necessary to support redundancy management. These few critical subsystems that have been flagged as requiring BITE are:

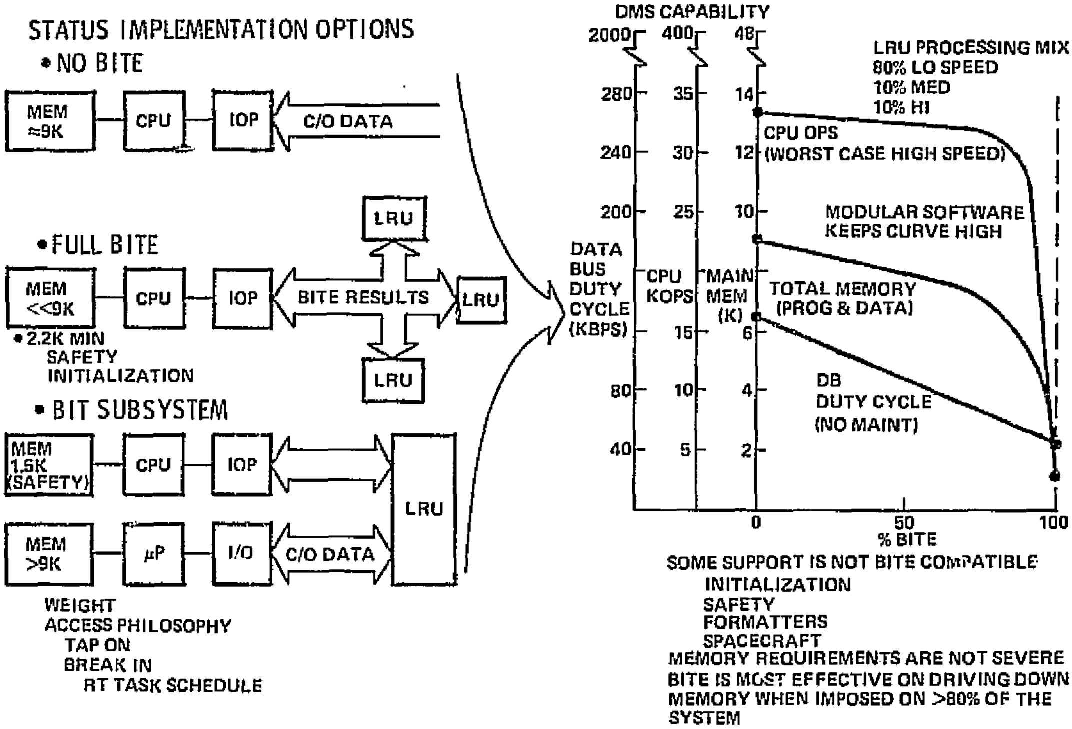


Figure 7-5. Onboard Checkout Implementation Options

a. Continuous BITE components:

1. Fuel cells (fast detection/recovery)
2. DMS (full self check — software/hardware implementation)

b. Commanded BITE components:

1. Scanning LADAR (full-DMS assist)
2. TV camera/electronics (full-DMS assist)
3. Signal conditioners (limited - A/D (PGA) converter check)
4. Engine control electronics (full-end-to-end checks)
5. IMU (partial - quick look technique)
6. Star tracker (limited - functional)
7. Sun sensor (limited - functional)
8. ILT (limited - functional/gain)

A second possible onboard checkout system architecture involves the use of a built-in-test (BIT) subsystem. The BIT subsystem is considered to be the best alternative to resolving a status and maintenance implementation conflict. The resolution would implement a second high speed data system dedicated to the checkout task and additionally provide the potential for onboard processing of maintenance data. Considering the absence of a firm need for peripheral (outside the DMS) status processing and onboard maintenance processing, the penalty in terms of cost and weight associated with a BIT subsystem eliminates it as a Tug candidate.

7.3.2 MAINTENANCE AS A DRIVER. The maintenance approach associated with the CMM_{PF} philosophy is that the best data on a subsystem/component is that generated during the last operations cycle under true environment. In general, electronics units can be easily and adequately stimulated to emulate environment and are therefore easily tested on the ground. Mechanical systems are best tested in flight during periods of maximum stress. A full maintenance program for the Tug requires data bus rates on the order of 1 Mbit for raw data only (i.e., no time tag or ID overhead) as shown in the support requirements development discussion.

Continuous or commanded status checks of electronic components will provide first order maintenance data to the line replaceable unit (LRU). In fact, compacted processed status data will in most instances provide isolation to a functional element within the device. For those subsystems/components best tested in flight under stress and requiring more detailed maintenance data, special maintenance action is accommodated. This is achieved in two ways: 1) with real-time responsive active measurements (these impact the Tug DMS/Communications subsystems), and 2) with limit detecting passive sensors for on-ground maintenance tasks.

The two maintenance accommodations and the driving requirements are shown in Figure 7-6.

As indicated, the baseline maintenance data is generated at a 160 Kbps rate during periods of full operability. The trade of what to do with this data involves onboard storage, telemetry rates, and telemetry coverage. The minimum weight/development implementation utilizes a standard NASA 3.2×10^8 bit tape recorder. Ground expense associated with this implementation, as shown in Figure 7-7, is on the order of 2.9 hours when restricted to a 16 Kbps downlink.

An iteration of the Communications subsystem trade for communications downlinks (refer to Section 3) subsequently identified the preference for a selectable 16, 64, 256 Kbps downlink. The 256 Kbps easily accommodates real-time transmission of maintenance data. During periods when telemetry coverage is unavailable, both the normal Tug telemetry and maintenance data will be recorded on the standard recorder as required.

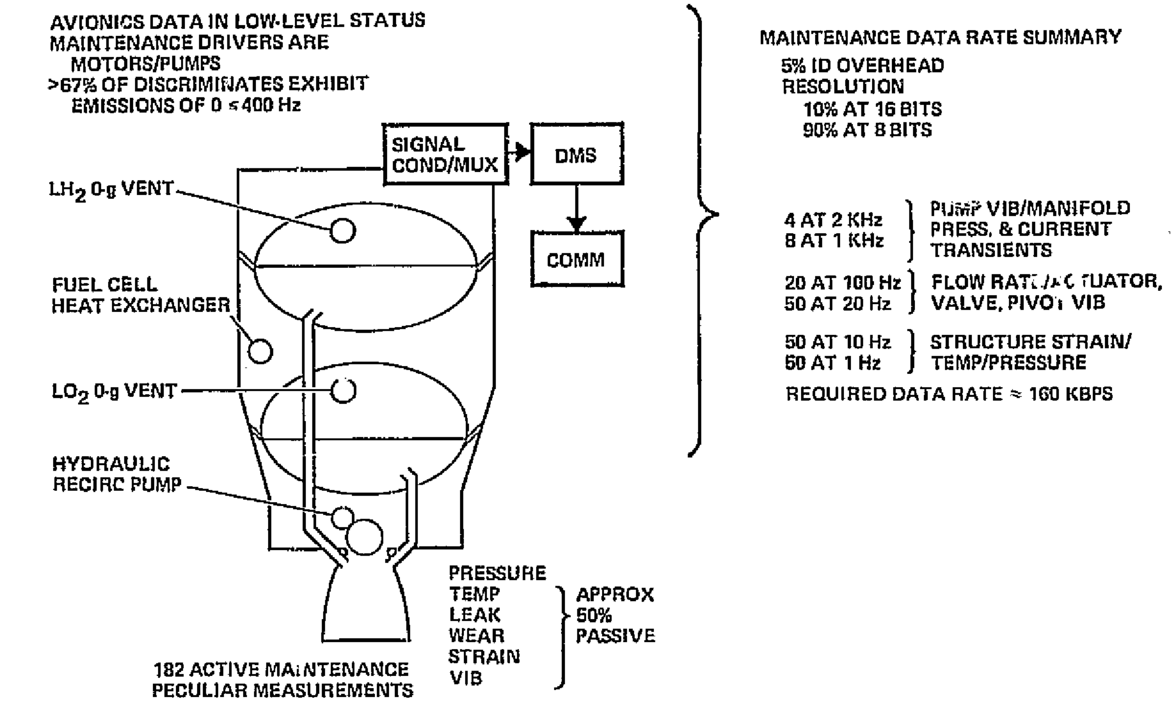


Figure 7-6. Maintenance Required Data

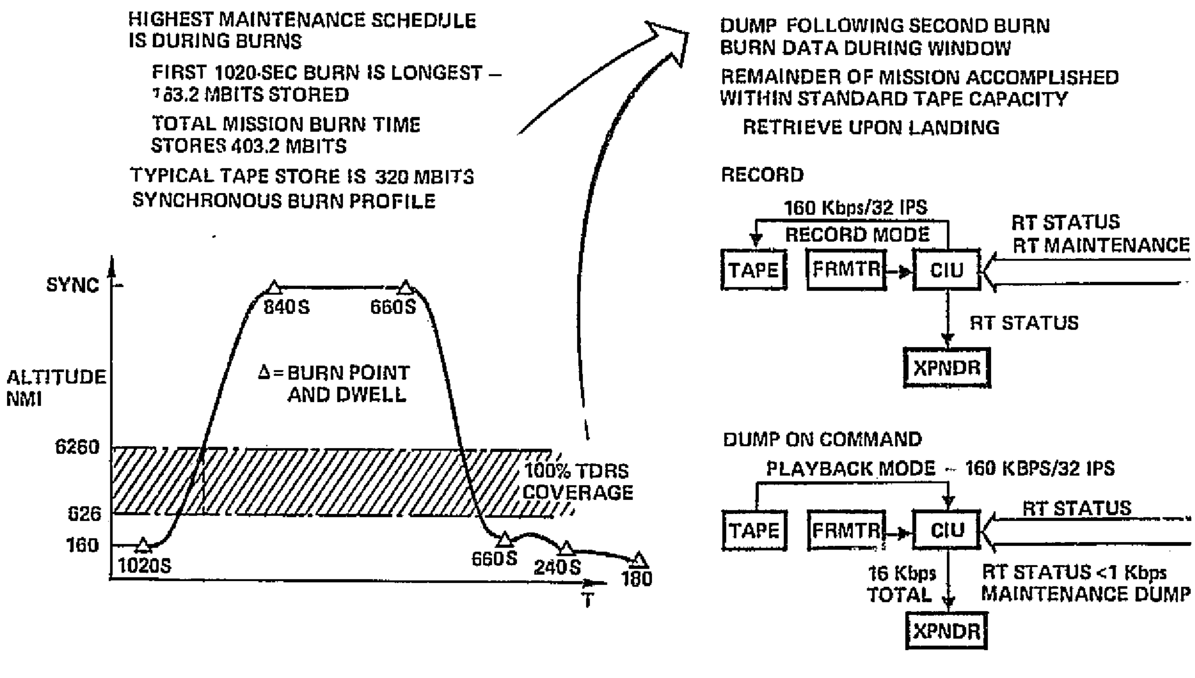


Figure 7-7. Maintenance Implementation

SECTION 8

AVIONICS INSTALLATION

8.1 AVIONICS EQUIPMENT LIST

As the major trade studies have been completed, the baseline Avionics System has been updated to reflect these recommendations. Table 8-1 presents the latest equipment list for the updated baseline system as evolved from the outputs of this Avionics Definition Study. There have been changes incorporated in each of the principal subsystems, with weight reductions in some and added weight estimated for others. In spite of this, the overall weight shown, 898 pounds (407.3 kilograms), is less than 10 pounds (4.5 kilograms) increased from the MSFC value of the 15 July 1974 configuration document MSFC 68 M 00039-2. Significant differences with respect to the equipment list are covered by subsystem below:

<u>Data Management Subsystem</u>

MSFC Configuration	158 lb (71.7 kg)
Study Baseline	100 lb (45.4 kg)
	-58 lb (26.3 kg)

Changes: (1) Computer size and weight reduced -31 lb (14.1 kg)

(2) Elimination of Auxiliary Memory -20 lb (9.1 kg)

(3) Buffer Formatter incorporated into CIU

<u>Guidance, Navigation, and Control Subsystem</u>

MSFC Configuration	153 lb (69.4 kg)
Study Baseline	190 lb (86.2 kg)
	+37 lb (16.8 kg)

Changes: (1) IMU and Electronics weight increased +13 lb (5.9 kg)

(2) Rate Gyros weight reduced -7 lb (3.2 kg)

(3) Star Tracker weight increased +7 lb (3.2 kg)

(4) ILT added, +24 lb (10.9 kg)

Table 8-1. Space Tug Avionics Equipment List - Baseline System

Equipment	No. Reqd	Dimensions Inches	Dimensions (cm)	Unit Op. Power, Watts	Unit Wt. lb	Unit Wt. (kg)	Subsys. Wt. lb	Subsys. Wt. (kg)
DATA MANAGEMENT								
Digital Computer	(1)	10 x 14 x 9.5	(25.4 x 35.6 x 24.1)	60	34	(15.4)	100	(45.4)
Computer I/F Unit	(2)	5 x 5 x 6.5	(12.7 x 12.7 x 16.5)	7	6.5	(2.9)		
Digital I/F Unit	(8)	5 x 5 x 6.5	(12.7 x 12.7 x 16.5)	5	5	(2.3)		
Tape Recorder	(1)	10 x 8 x 5	(25.4 x 20.3 x 12.7)	20	13	(5.9)		
GUID, NAV. AND CONTROL								
Inertial Measure. Unit	(1)	9 x 9 dia	(22.9 x 22.9 dia)	100	25	(11.3)	190	(86.2)
IMU Electronics	(1)	10 x 20 x 5	(25.4 x 50.8 x 12.7)	100	30	(13.6)		
Rate Gyro Package	(1)	10 x 10 x 6	(25.4 x 25.4 x 15.2)	100	20	(9.1)		
Star Tracker	(2)	6 x 8 x 12	(15.2 x 20.3 x 30.5)	12	16	(7.3)		
Sun Sensor	(2)	6.9 x 6.5 x 3	(17.5 x 16.5 x 7.5)	5	4.5	(2.0)		
Control Electronics	(1)	12 x 12 x 18	(30.5 x 30.5 x 45.7)	50	50	(22.7)		
Interferometric Landmark Tracker								
ILT Antenna	(4)	2 x 6 dia	(5.1 x 15.2 dia)	-	1	(0.5)		
ILT Receiver	(1)	12 x 10 x 9	(30.5 x 25.4 x 27.9)	15	20	(9.1)		
RENDEZVOUS & DOCKING							63	(28.6)
Scanning Laser Radar	(1)	6 x 8 x 20	(15.2 x 20.3 x 50.8)	10	28	(12.7)		
Ladar Electronics	(1)	9 x 9 x 11	(22.9 x 22.9 x 27.9)	30	11	(5.0)		
TV Camera & Electronics	(2)	6 x 6 x 15	(15.2 x 15.2 x 38.1)	10	8/13	(3.6/5.0)		
TV Strobe Lamps	(4)	3.5 x 3.5 x 3.5	(8.9 x 8.9 x 8.9)	Negl.	0.25	(0.5)		
Strobe Electronics	(2)	2 x 3.5 x 2.5	(5.1 x 8.9 x 6.4)	Negl.	1	(0.9)		

Table 8-1. Space Tug Avionics Equipment List - Baseline System, Contd

Equipment	No. Reqd	Dimensions: Inches	Dimensions: (cm)	Unit Op. Power, Watts	Unit Wt. lb	Unit Wt. (kg)	Subsys. Wt. lb	Subsys. Wt. (kg)
COMMUNICATIONS							149	(67.6)
Phased Array Antenna	(3)	3.5 x 15 dia	(8.9 x 38.1 dia)	93	16	(7.3)		
Hemispherical Antenna	(2)	4.6 x 6 x 2	(11.7 x 15.2 x 5.1)	-	1	(0.5)		
RF Network	(1)	3.3 x 3.8 x 1	(8.4 x 9.7 x 2.5)	-	2	(0.9)		
RF Switch	(1)	5 x 5 x 6.3	(12.7 x 12.7 x 16)	3	7.3	(3.3)		
Transponder	(2)	15 x 7 x 6	(38.1 x 17.8 x 15.2)	16	16.5	(7.5)		
Signal Processor	(2)	13.5 x 6 x 5.6	(34.3 x 15.2 x 14.2)	18	11	(5.0)		
Cmmd Dist. Unit	(1)	5 x 5 x 4	(12.7 x 12.7 x 10.2)	35	18	(8.2)		
Encrypter	(2)	5.8 x 4.3 x 5.3	(14.7 x 10.9 x 13.5)	7	4.3	(2.0)		
Decrypter	(2)	6 x 3.6 x 5.8	(15.2 x 9.1 x 14.7)	2.4	4.1	(1.9)		
INSTRUMENTATION							74	(33.6)
Transducers	(243)			(Total)	20	(9.1)		
Sig. Conditioners/MUX	(3)	12 x 10 x 6	(30.5 x 25.4 x 15.2)	22	18	(8.2)		
ELECTRICAL POWER							120	(54.4)
Fuel Cell Power Plant	(2)	12 x 6 x 15	(30.5 x 15.2 x 38.1)	20	42	(19.1)		
Emergency Battery	(1)	8 x 11 x 7	(20.3 x 27.9 x 17.8)	8	36	(16.3)		
PWR DISTRIBUTION & CONT.				82 (avg)			202	(91.6)
Fwd Control Unit	(1)	10 x 6 x 8	(25.4 x 15.2 x 20.3)		10	(4.5)		
Aft Control Unit	(1)	12 x 15 x 8	(30.5 x 38.1 x 20.3)		24	(10.9)		
Pwr. Processing Unit	(2)	9 x 9 x 8	(22.9 x 22.9 x 20.3)		8	(3.6)		
Remote Pwr. Controller	(59)	0.5 x 1 x 1	(1.3 x 2.5 x 2.5)		0.2	(0.1)		
Harnesses/Connectors				(Total)	130	(59.0)		
Arm/Safe Switches	(2)				5	(2.3)		
			SYSTEM TOTAL AVIONICS WEIGHT			898		(407.3)

Rendezvous and Docking Subsystem

MSFC Configuration	35 lb (15.9 kg)
Study Baseline	63 lb (28.6 kg)
	+28 lb (12.7 kg)

Changes: (1) Original subsystem was Laser Radar only

(2) TV Camera and Electronics added, +16 lb (7.3 kg)

(3) TV Strobe Lamps and Electronics added, +8 lb (3.6 kg)

Communications Subsystem

MSFC Configuration	72 lb (32.7 kg)
Study Baseline	149 lb (67.6 kg)
	+77 lb (34.9 kg)

Changes: (1) Revised AESPA, transmit only, 3 arrays -4 lb (1.8 kg)

(2) TV moved to Rend. & Docking subsystem -14 lb (6.4 kg)

(3) Hemispherical Antennas added, +2 lb (0.9 kg)

(4) RF Network & Switch added, +9 lb (4.1 kg)

(5) Separate Transponders, +33 lb (15.0 kg)

(6) Signal Processors added, +22 lb (10.0 kg)

(7) Decoder included in Sig. Processor, -3 lb (1.4 kg)

(8) Added Encrypters, Decrypters, +17 lb (7.7 kg)

(9) Command Distributer weight increased, +15 lb (6.8 kg)

Instrumentation Subsystem

MSFC Configuration	61 lb (27.7 kg)
Study Baseline	74 lb (33.6 kg)
	+13 lb (5.9 kg)

Changes: (1) Incorporation of MUX into Sig. Cond. +18 lb (8.2 kg)

(2) Reduction of Sensor/Transducer weight -5 lb (2.3 kg)

Electrical Power & Power Distribution Subsystems

MSFC Configuration	410 lb (186.0 kg)
Study Baseline	322 lb (146.1 kg)
	-88 lb (39.9 kg)

Changes: (1) Fuel cell assembly weight reduced, -38 lb (17.2 kg)

(2) Tankage eliminated, -15 lb (6.8 kg)

(3) Plumbing weight reduced, some outside avionics

(4) Thermal Control Distributor eliminated, -12 lb (5.4 kg)

(5) Battery weight increased, +16 lb (7.3 kg)

(6) Other distributer weights reduced (see Section 4 for power system weight trades)

8.2 INSTALLATION OPTIONS TRADE

The objective of the Tug avionics mounting options analysis was to achieve a minimum weight maintainable installation suitable for mission requirements. Three principal factors constitute the major criteria for installation: accessibility, proximity to related hardware, and thermal control.

A survey was made of some 50 components to acquire as much information as currently available to make an initial mounting arrangement and layout. A list of suggested mounting groups is given in Table 8-2. Basic mounting data that was gathered is presented in Table 8-3. The data provided identification of component size, weight, angular location, power dissipation, and view orientation. Included in the criteria that were applied in the development of the initial mounting scheme were: accessibility for maintenance/replacement, proximity constraints, thermal control, minimum electrical interference, physical mounting requirements, and view orientation for optics and antennas. Most heavily weighted in this first cut were proximity and view orientation, to a lesser extent accessibility, and with limited thermal control consideration.

Location of the avionic components has been made in accordance with the subsystem and system functional requirements. Mounting recommendations for the avionics installation are shown in the preliminary "rough" layout of Figure 8-1. Each mounting group has been located around the periphery of the square payload support structure on a special equipment support structure. Stable mounting platforms are provided for the Rendezvous and Docking components group with the laser radar and TV cameras, and for the GN&C group, which contains the IMU, star and sun trackers, and the rate gyro package. Three phased array transmitting antennas for communications

Table 8-2. Tug Avionics Mounting Groups

F - mount in space forward of LH_2 tank
A - mount in intertank space
S - mount on shroud, generally with external exposure

Mount each group generally on a common mounting plate unless otherwise indicated. Plate will ultimately be used for thermal control as a heat sink and heat distribution medium.

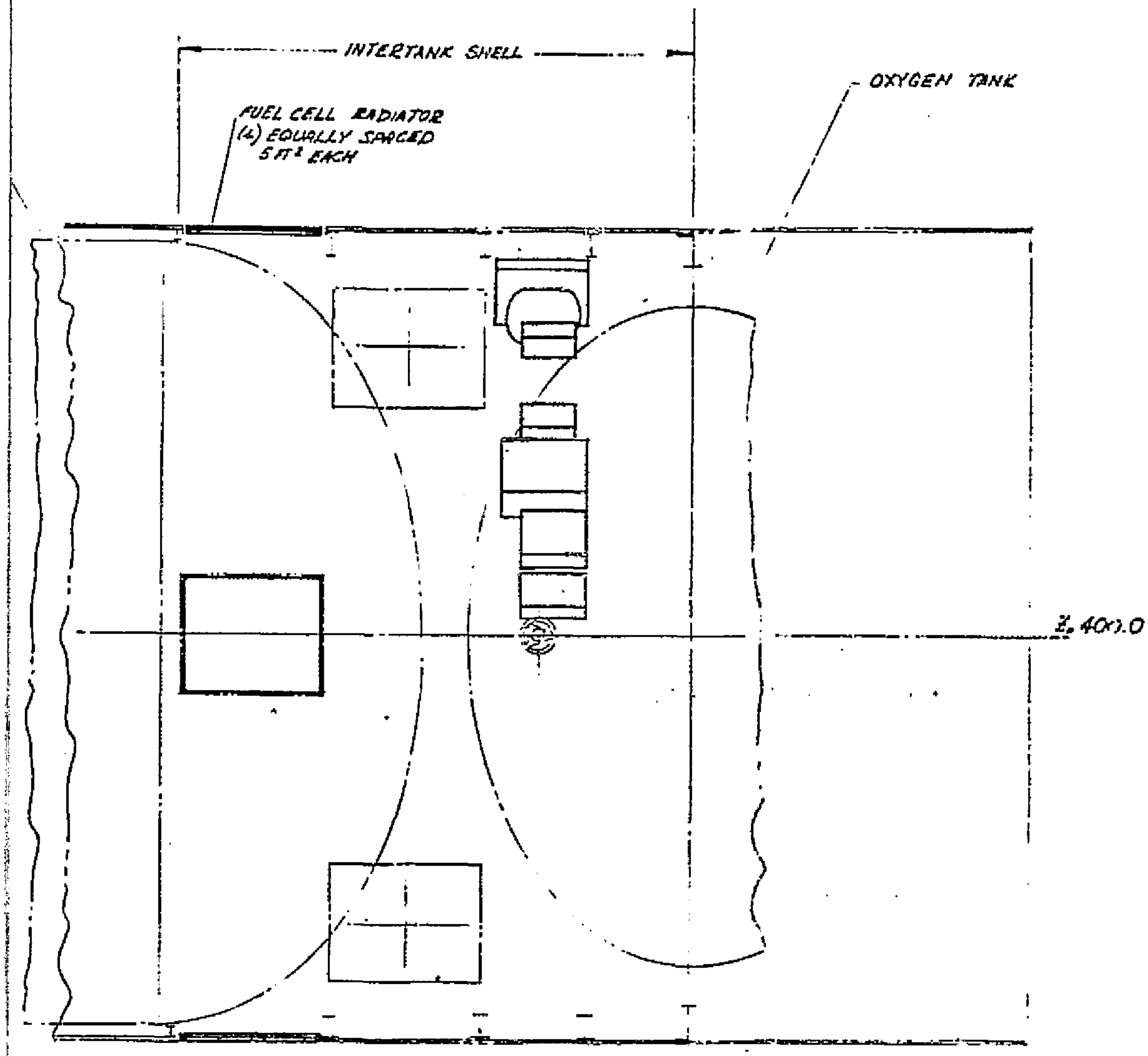
F1	IMU, Star Tracker No. 1, Star Tracker No. 2, Sun Sensor No. 1, Sun Sensor No. 2, DIU Pair No. 2.
F2	TV Camera and Electronics No. 1, TV Strobe Lamps Pair No. 1, TV Camera and Electronics No. 2, TV Strobe Lamps Pair No. 2, Scanning Laser Radar.
F3	TV Strobe Electronics No. 1, SLR Electronics, TV Strobe Electronics No. 2, DIU Pair No. 1.
F4	Arm Safe Switches No. 1 and 2, Forward Power Distrib. Control.
F5	Digital Computer, Tape Recorder, CIU No. 1, CIU No. 2.
F6	RF Network Assembly, RF Switch Assembly, Transponder No. 1, Transponder No. 2.
F7	Signal Processor No. 1, Signal Processor No. 2, Command Distribution, Encryption No. 1, Encryption No. 2, Decryption No. 1, Decryption No. 2.
F8	Signal Conditioner No. 1, DIU Pair No. 4.
F9	Phased Array Antennas 0 deg (0 rad), 120 deg (2.09 rad), 240 deg (4.19 rad); Hemi Antennas 90 deg (1.57 rad), 270 deg (4.71 rad). (Flush mounted angularly around shroud.)
A1	GN&C Control Electronics, DIU Pair No. 3.
A2	Signal Conditioner No. 2, Signal Conditioner No. 3.
A3	Fuel Cell No. 1, Fuel Cell No. 2.
A4	Emergency Battery, Aft Power Distr. Control.
A5	H_2O Heat Exchanger, ACS Pumps
S1	ILT-ANTS Antennas, ILT-ANTS Receiver Electronics. (Flush mount antennas in 40 in. (101.6 cm) square pattern anywhere on shroud surface with receiver electronics inside nearby.)
S2	Space Radiators. Flush mount four, 1 in. (2.54 cm) t, radiator panels angularly separated 90 deg (1.57 rad) around shroud. Freon plumbing connects them to fuel cells.

Table 8-3. Avionics Equipment Mounting Data – Supplement to Table 8-1
(for dimensions, power, and weights, see Table 8-1)

Equipment	No. Required	Group No.	Base Area inch	Base Area (cm)	Location Angle	View	Comments
Data Management							
Digital Computer	1	F5	10 x 14	(25.4 x 35.6)		None	
Computer I/F Unit	2	F5	10 x 6.5	(25.4 x 16.5)			
Digital I/F Unit, Pair No. 1	2	F3	10 x 6.5	(25.4 x 16.5)			
Pair No. 2	2	F1	10 x 6.5	(25.4 x 16.5)			
Pair No. 3	2	A1	10 x 6.5	(25.4 x 16.5)			
Pair No. 4	2	F8	10 x 6.5	(25.4 x 16.5)			
Tape Recorder	1	F5	10 x 8	(25.4 x 20.3)			
Guidance, Navigation and Control							
Inertial Measuring Unit	1	F1	9 x 9	(22.9 x 22.9)			
IMU Electronics	1	F1	10 x 20	(25.4 x 50.8)			Mount with DIU No. 2
Rate Gyro Package	1	F1	10 x 10	(25.4 x 25.4)		None	
Star Tracker	2	F1	24 x 6	(61.0 x 15.2)		45 deg (0.785 rad)	Unobstructed view
Sun Sensor, No. 1	1	F1	7 x 6.5	(17.8 x 16.5)		64 deg (0.1 rad)	Unobstructed view
No. 2	1	F1	7 x 6.5	(17.8 x 16.5)		64 deg (0.1 rad)	Unobstructed view
Control Electronics	1	A1	12 x 18	(30.5 x 45.7)		None	Mount with DIU No. 3
Interferometric							
Landmark Tracker Antenna	4	S1	6 x 12	(15.2 x 30.5)		Unobstructed	Mount as 40 in. (10.2 cm) array
Receiver	1	S1	10 x 12	(25.4 x 30.5)		None	Adjust to array
Rendezvous and Docking							
Scanning Laser Radar	1	F2	20 x 8	(50.8 x 20.3)		10 deg (0.174 rad) cone	
LADAR Electronics	1	F3	9 x 11	(22.8 x 27.9)		None	
TV Camera and Electronics	2	F2	6 x 15	(15.2 x 38.1)		10 deg (0.174 rad) cone	
TV Strobe Lamps	4	F2	3.5 x 3.5	(8.9 x 8.9)		10 deg (0.174 rad) cone	
Strobe Electronics	2	F3	2 x 3.5	(5.1 x 8.9)			

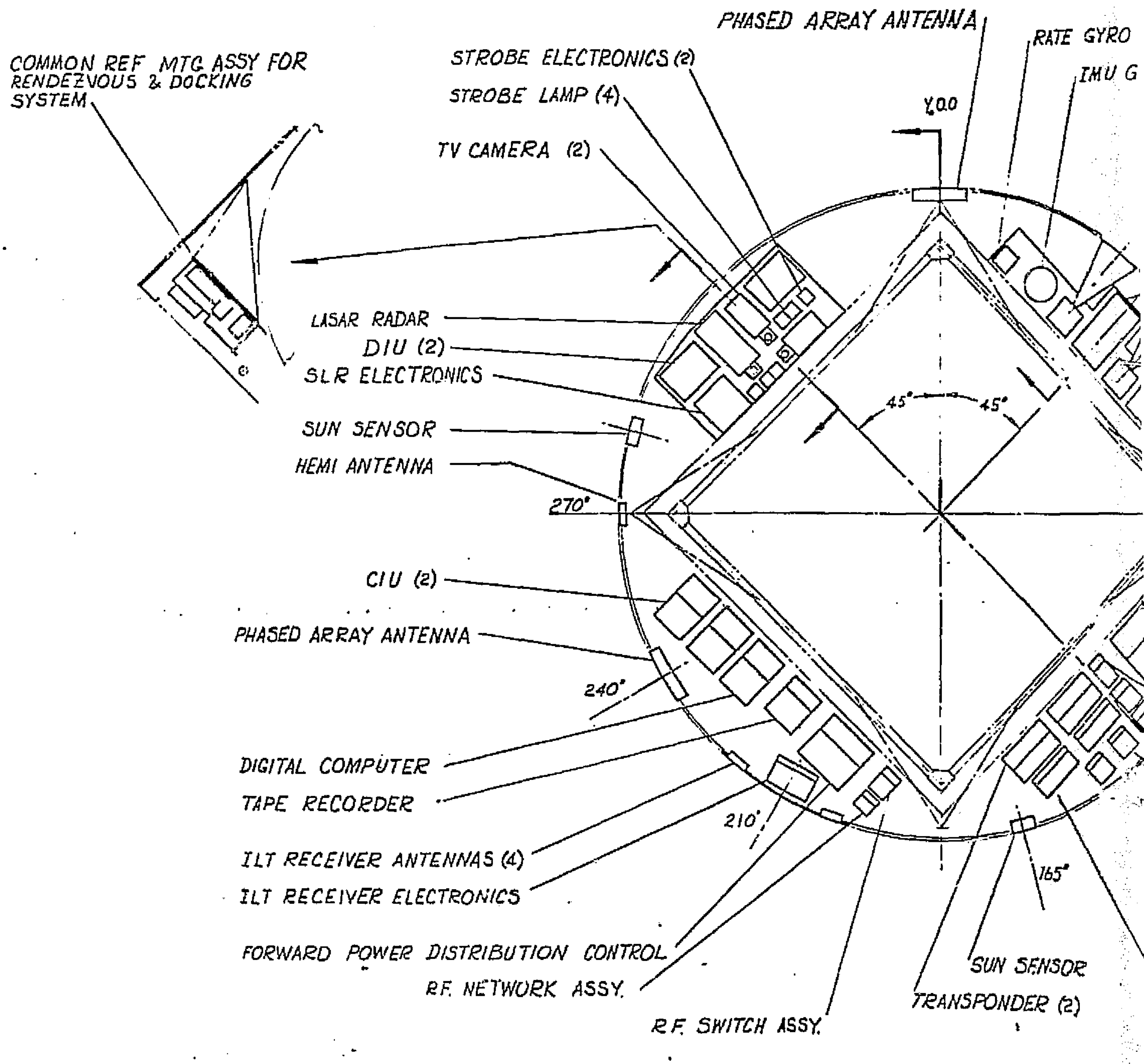
Table 8-3. Avionics Equipment Mounting Data – Supplement to Table 8-1 (for dimensions, power, and weights, see Table 8-1), Contd

Equipment	No. Required	Group No.	Base Area inch	Base Area (cm)	Location Angle	View	Comments
Communications							
Phased Array Antenna	3	F9	15 x 15	(38.1 x 38.1)	0, 120, 240 deg (0, 2.09, 4.18 rad)	Unobstructed	
Hemispherical Antenna	2	F9	4.6 x 6	(11.7 x 15.2)	90, 270 deg (1.57, 4.71 rad)	Unobstructed	
RF Network	1	F6	3.3 x 3.8	(8.4 x 9.7)	180 deg (3.14 rad)	None	
RF Switch	2	F6	5 x 5	(12.7 x 12.7)	180 deg (3.14 rad)		
Transponder	2	F6	15 x 7	(38.1 x 17.8)			
Signal Processor	2	F7	13.5 x 6	(34.3 x 15.2)			
Command Dist. Unit	1	F7	5 x 5	(12.7 x 12.7)			
Encrypter	2	F7	6 x 5	(15.2 x 12.7)			
Decrypter	2	F7	6 x 4	(15.2 x 10.2)			
Instrumentation							
Transducers	243	All					Scattered
Signal Conditioner/MUX, No. 1	1	F8	12 x 10	(30.5 x 25.4)			One 12 x 6 in. (30.5 x 15.2 cm) face has 480 wire connections
No. 2	1	A2	12 x 10	(30.5 x 25.4)			
No. 3	1	A2	12 x 10	(30.5 x 25.4)			
Electrical Power							
Fuel Cell Power Plant	2	A3	12 x 16	(30.5 x 40.6)			
Emergency Battery	1	A4	8 x 11	(20.3 x 27.9)			
Power Distribution and Control							
Forward Control Unit	1	F4	10 x 6	(25.4 x 15.2)			
Aft Control Unit	1	A4	12 x 15	(30.5 x 38.1)			
Power Processing Unit	2	A3	9 x 9	(22.9 x 22.9)			
Remote Power Controller	59	All	1 x 1	(2.5 x 2.5)			
Harnesses/Connectors		All					
Arm/Safe Switches	2	F4				None	

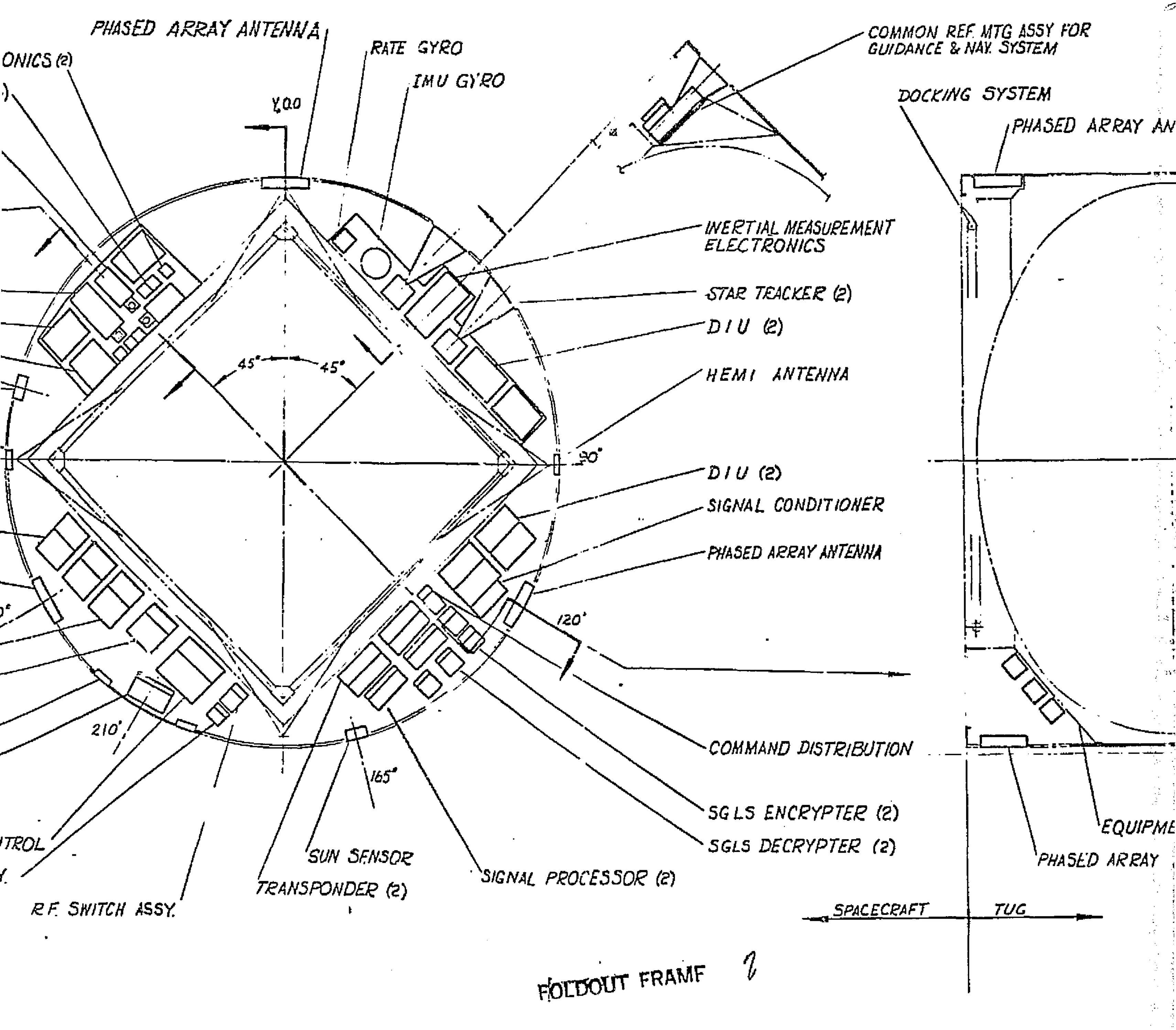


ORIGINAL PAGE IS
OF POOR QUALITY

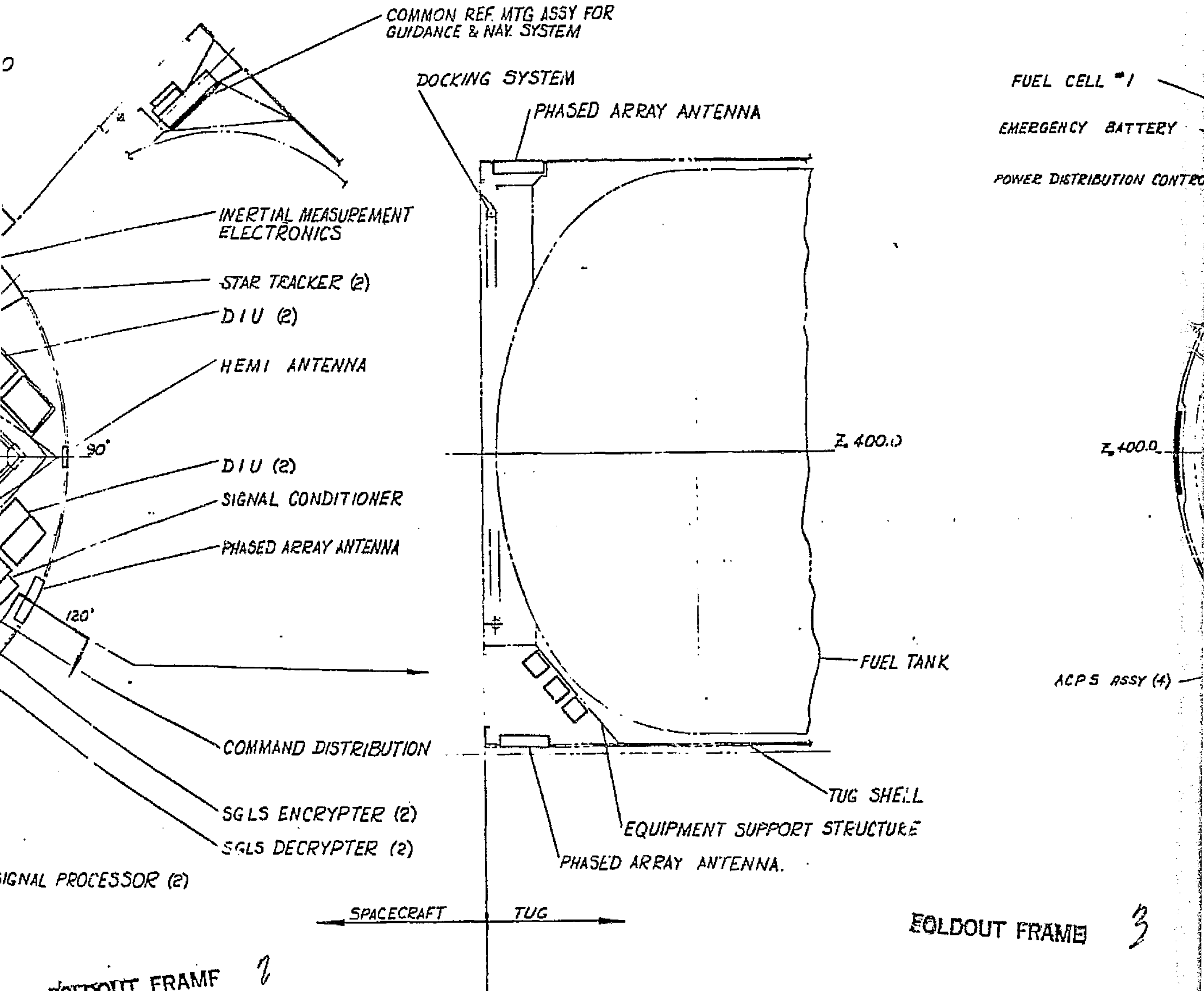
Figure 8-1. Space Tug Avionics Equipment Installation



FOLDOUT FRAME 1

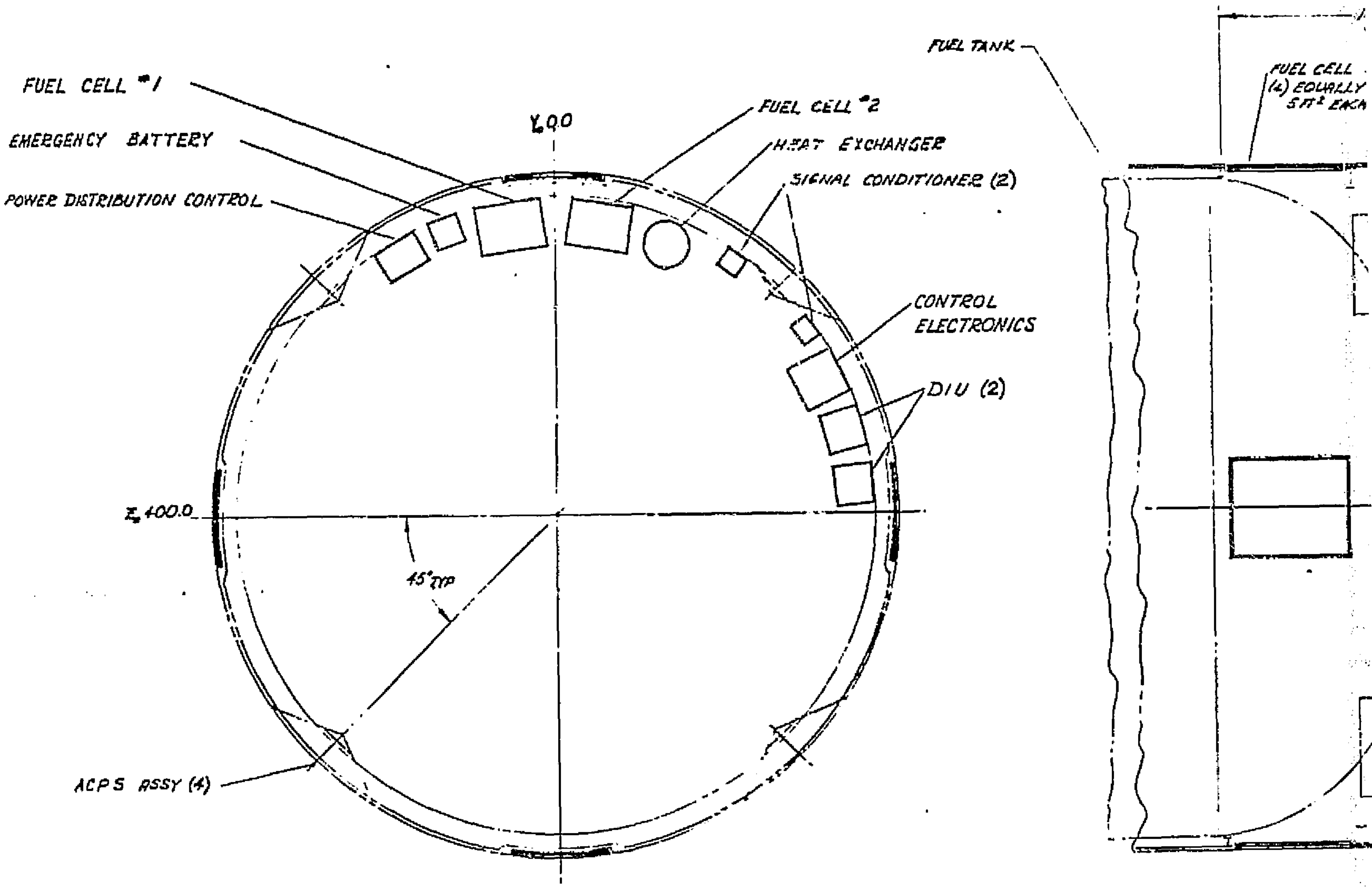


FOLDOUT FRAME 2



FOLDOUT FRAME 2

FOLDOUT FRAME 3



OUT FRAME 3

FOLDOUT FRAME 4

have been located 120 degrees (2.09 radians) apart around the outer shell, and two hemispherical antennas are provided for receiving and for nearby link transmissions.

Additional future effort to improve and optimize the avionics installation when component selection is "final" should be:

a. Further analysis of the proposed mounting arrangement and the rearrangement so as to minimize connector lengths, improve accessibility, and provide for the separation of sensitive components.

b. Perform a thermal control analysis and develop a thermal hardware design to maintain the system within specified temperatures. Provisions must be made to analyze the ground launch, on-orbit, and reentry environments.

SECTION 9

SENSITIVITY ANALYSES

9.1 MULTIPLE PAYLOAD ACCOMMODATION

Analysis of the impact of multiple payloads on the avionics requirements has been accomplished by the MacDonnell Douglas Company based upon payload data from the NASA Space Shuttle Payload Data (SSPD) Study with further analysis provided by the General Electric Company. The results of the analysis and the requirements appear in the December 1974 MidTerm Progress Review, Report MDC G5629, for the IUS/Tug Payload Requirements Compatibility Study — Contract NAS 8-31013.

Primary avionic effects of multiple payloads result from the probability of greater electrical power load, increased data handling/processing requirements, extra communications, and added caution and warning functions. Operational complexity increases and the total mission durations tend to lengthen. Checkout functions during the Shuttle orbital phase can be significantly greater, depending on the specific payload requirements. The particular concern of this trade study has been the sensitivities and effects on the Tug avionics system and interface for multiple payloads.

9.1.1 REQUIREMENTS. A comparison of anticipated Tug avionic requirements for single and multiple payloads is shown in Table 9-1. These impacts and requirements were defined in the MDAC Payload Requirements Compatibility Study, Contract NAS 8-31013. General conclusions of that study were that the effects of multiple payload operation on the design of the Tug avionics system are minimal. Two areas have been called out by MDAC for special attention:

a. Design of the Tug forward Digital Interface Unit (DIU), which must handle the increased command outputs and signal inputs.

b. Peak electrical power requirements while the spacecraft is attached, which may or may not be best supplied by the Tug fuel cells.

9.1.2 AVIONICS SYSTEM IMPLEMENTATION

Electrical Power. Neither the average power nor the peak power load requirements for multiple payloads appear to represent any special problem. The recommended baseline design for the electrical power source is adequate to satisfy the indicated needs. Dual fuel cells — each a nominal 2 kW — can supply a total of 3.5 kW each for eight hours as a peak load situation. Short duration peaks for the payloads above the average power load of 1 kW for the Tug proper can easily be handled without

Table 9-1. Tug Impact Summary Avionics

Services Required	Tug Single	Multiple	Impacts
Power (avg kW)			Avg power — no impact.
Post-Deployment/			Peak power — current limiter,
Pre-Separation	0.7	1.15	use of payload batteries as
Transfer	0.2	0.2	peaking supply
Power (peak kW)	2.4	3.4	
Energy	11.8	12.7	Requires use of propulsion system reactants or extra tanks
Telemetry (RT)			Requires multiple inputs to
Digital Rate (Kbps)	Variable to 10.024	Variable to 10.024	forward DIU variable rate interleave capability. Spacecraft data selection switching
Discrete Controls			
Orbiter/GSE Safing	4	12	Safing control cable assembly
Tug/Orbiter Discretes	32	64	Multiple DIU outputs.
Ordnance Initiation	4	18	Multiple PCU outputs
Discrete Talkbacks	20	50	Multiple DIU inputs
Processing Rate (Kops/sec)	0.1	0.3	None at levels assumed
Main Memory (K words)	1	2	
Command (serial) (Kbps)	Variable to 2	Variable to 2	DOD and NASA unique implementation require
C&W Signals			
Tug Processed	2	6	Inclusion in Tug data format and transfer to Orbiter.
Orbiter Transfer	35	35	C&W cable assembly

further redesign. Figure 9-1 illustrates the power available from the Tug fuel cells and shows the plots of several typical payloads. For the short duration peaks that may be encountered for multiple spacecraft up to 3.4 kW and at an energy level of 12.7 kW-hr, there are no new system requirements created except for the distribution and switching circuitry and the interface wiring.

Telemetry. Some minor provisions are needed for the multiple inputs and the interleaving of data. The requirements appear to be satisfied well within the present baseline implementation recommendations.

Discrete Controls and Talkbacks. The number of wires for safing control and the number of multiple DIU and PCU outputs/inputs increases, but they are readily incorporated within the baseline design with no appreciable impact.

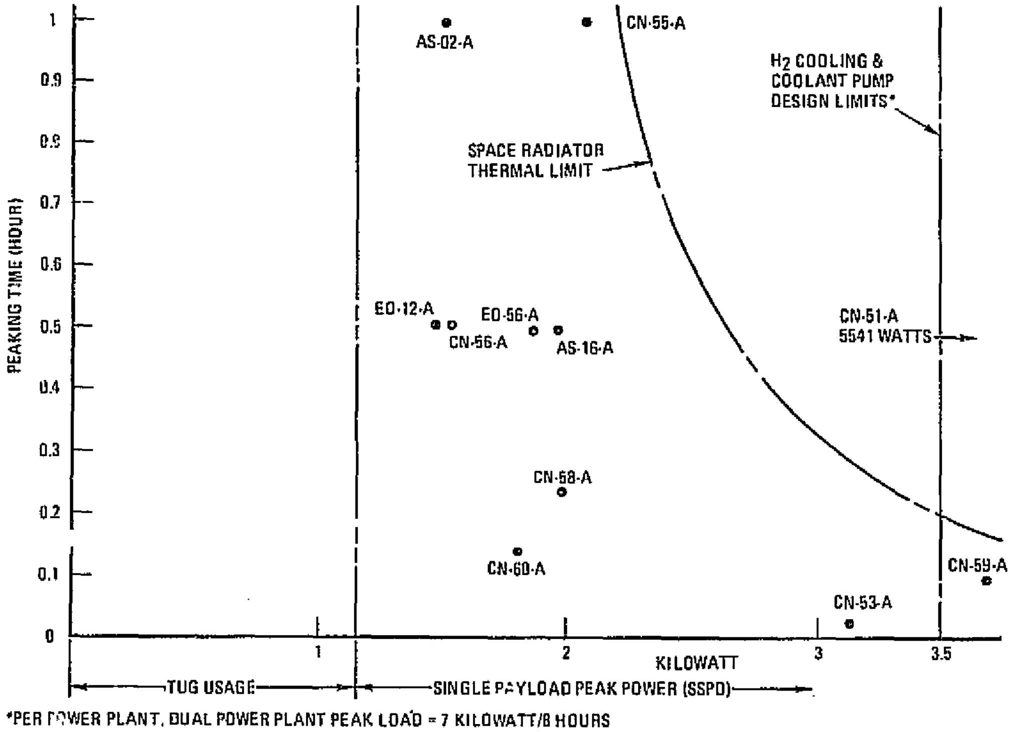


Figure 9-1. Two Kilowatt Electrical Peak Power Thermal Limits (During Spacecraft Checkout and In Orbit)

DMS Processing Rate and Main Memory Sizing. There are very minor effects on rate and/or main memory size, 0.2 Kops from 0.1 Kops and 2K words from 1K words, which will not require special accommodations.

Command Signals. This is not a significant change from the capacity needed for single payloads.

Caution and Warning. There is a slight increase for Tug processed signals, from 2 to 6, and no change for Orbiter transfer signals. No unique provisions are anticipated.

9.1.3 COST ELEMENTS. As can be deduced from the implementation discussion, there are no major or significant cost impacts to the recommended avionics system baseline arising from the addition of the requirement of multiple payload delivery missions. Some minor system design accommodations are needed, but no appreciable effects will take place for program costs as compared to single payload delivery.

9.1.4 CONCLUSIONS. Sensitivities of the Tug avionics system to multiple payload requirements are minimal, and the accommodations planned for single payloads can adequately handle the multiples. Sufficient peak electrical power is available from the recommended baseline dual fuel cell power plants. Increased interface provisions for the multiple payloads will be incorporated into the design of the forward digital interface unit (DIU). This will allow the increased command inputs, caution and warning signals, and discretes that result from transporting more than one payload. Generally, there are no major changes to the basic Tug avionics design.

9.2 AUTONOMY LEVEL SENSITIVITY

9.2.1 INTRODUCTION. Practical solutions to an operational Space Tug that represent cost effective configurations that can be realized at reasonable or low program risk tend to fall within the intermediate autonomy levels. The extremes of Level I, a fully independent vehicle, and of Level IV, with nearly total dependency on ground support, appear to be costly choices. Either the vehicle nonrecurring development costs are excessive (Level I), or the ground support nonrecurring and recurring operating costs are exorbitant for these extremes because of the dependence on designs, facilities, and large groups of personnel that do not yet exist.

Typically, the present study updated Tug avionics system baseline has gravitated to a Level II minus design configuration with a mixture of automatic and dependent externally controlled elements. The purpose of this trade study is the analysis and determination of the avionic system and subsystem sensitivities as the individual autonomies are varied consistent with the Tug vehicle operational autonomy levels of II and III. Impacts of shifts in the autonomy level of vehicle operation tend to be most pronounced for the Communications subsystem, the Data Management subsystem, and the Rendezvous and Docking subsystem with respect to the onboard avionics system. These are the subsystems where the involvement of command and control and support functions is greatest.

9.2.2 AUTONOMY DEFINITIONS. The original set of definitions for Tug autonomy were issued by MSFC for the NASA/Air Force Space Tug System Studies (STSS) and were translated by NASA for the MSFC documents MSFC 68M00039-1 and MSFC 68M00039-3 for application to this Avionics Definition Study.

Table 9-2 gives the basic STSS autonomy definitions for the baseline avionics system. For the 13 functions considered, these have been translated into functional requirements as they apply for Level II autonomy and Level III autonomy respectively. This autonomy requirements matrix is presented in Table 9-3.

9.2.3 SUBSYSTEM ANALYSIS. Each Tug avionic subsystem has been examined to determine how changes in the operational autonomy level can affect the design, performance, and cost factors. These sensitivities are presented in the following paragraphs by subsystem.

Communications. Command uplink and telemetry downlink are required for all levels of autonomy of the Tug. For Levels I and II, the uplink is needed only to provide a command override function. This is contrasted for Level III, with full state update requirements, and Level IV, where control of all phases would be from the ground. These communications requirements exhibit themselves more in the surrounding link elements, such as the number and location of ground stations and relay satellites (like TDRS), and the capabilities needed at each of the ground support facilities. Important, but less critical, are the detailed features of the onboard design such as

Table 9-2. Levels of Autonomy — Space Tug Definitions

<u>Level I Autonomy</u>
Completely independent of any manmade inputs after separation (such as beacons, Orbiter, and ground).
Onboard measurements and calculations enable mission to be completed in its entirety including all Tug and payload operations.
Final onboard rendezvous and docking capability.
Command uplink override capability and telemetry down link.
<u>Level II Autonomy</u>
Ground or navigation satellite beacons (either must serve multiple users) acceptable.
Level I autonomy will be required for those orbits where ground or satellite beacons do not provide satisfactory state determinations.
Final onboard rendezvous and docking capability.
Command uplink override capability including payload status, redirection, and retargeting of mission with telemetry downlink.
<u>Level III Autonomy</u>
Ground stations provide state update during entire mission.
Onboard calculations are performed for mission completion.
Final rendezvous is made by onboard capability.
Final docking with ground support.
Command and telemetry capability.
<u>Level IV Autonomy</u>
All phases are controlled from the ground.
Calculations are performed primarily on the ground (such as main burn and midcourse — duration and direction).
Ground will control final rendezvous and docking.
Command and telemetry capability.

Table 9-3. Autonomy Requirements Matrix for Levels II and III

Function	Requirements Level II	Requirements Level III
Boost, Initialization (C/O Primary System)	Shuttle control and monitor	Same as level II
Post Separation Activation Sequence	Automatic (after secure rf uplink from ground to initiate)	Same as level II
Orbital Tracking	No requirement	TDRS and DSN (beacon)
Attitude Update	Independent, on board	Same as level II
Position and Velocity Vector Determination	Independent, ground augmentation optional	State update during the entire mission from ground stations
Tug Telemetry (Downlink)	Secure events and analogue parameters when stored limits exceeded	Full capability needed during mission
Redundancy Management and System Reconfiguration	On board subsystem control, fault isolation, redundancy management; and switchover by on board checkout and fault isolation. Secure command override for burn, abort and alternate missions cancellation (single mode word commands. Status secure downlinked for ground monitor. Secure command to load memory modules, i.e., redundancy management diagnostic trouble shooting and overrides).	Partial diagnostics and reconfiguration control on ground.
Alternate Mission	Determination and selection by ground option (alternate missions determined and initiated from ground by secure uplink).	Onboard calculations are performed for mission completion
Guidance, Navigation, and Control (for main engine burns and mid-course APS correction burns)	GN&C automatic and independent (include pre-burn, burn, post-burn, targeting, and reconfiguration). Use of ground beacon acceptable.	On board calculations for mission completion. State update from ground
Shuttle Rendezvous	Automatically accomplished by on board computation of navigation and guidance except beacon optional for Shuttle contact. All rendezvous to be coplanar with Shuttle, including abort.	Ground tracking and ground computation of transfer trajectory. Onboard computation if ground agrees. Rendezvous coplanar including abort.
Payload Rendezvous	Automatically accomplished by terminal phase guidance with target passive. Event TM secure monitoring	Final rendezvous made by on board capability
Payload Deployment and Monitor	Location, initialization, spin-up and release performed automatically. Prior deployment, Tug monitors go/no-go payload status. Thruputs payload telemetry to ground	Location verified by ground. Event sequence automatic with ground mode optional. Thruput of payload telemetry to ground. Go/no-go from ground.
Payload Docking	Accomplished automatically with target passive or not actually evasive. Event TM secure monitoring	Final docking with ground support. Man-in-loop remote control.

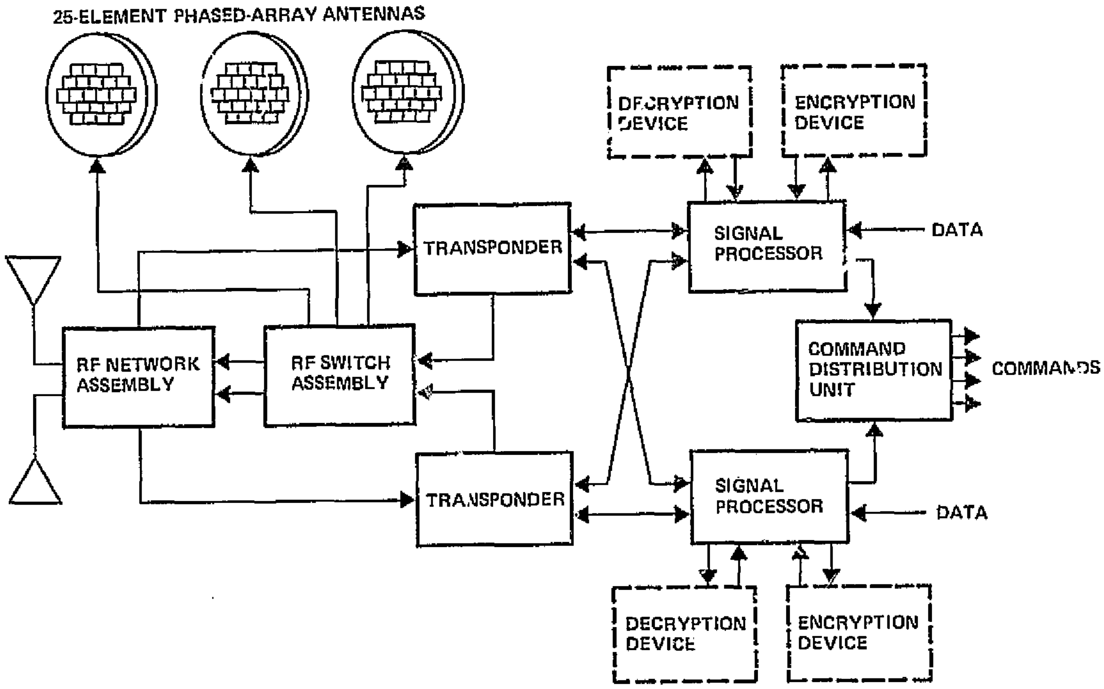


Figure 9-2. Baseline Communications Subsystem

Tug antenna coverage, gain, and directivity. This trade study has been primarily concerned with the sensitivities and impacts of the onboard avionics system to the autonomy level. Figure 9-2 is a diagram of the Tug baseline onboard Communications subsystem. Elements sensitive to the differences in requirements between autonomy Level II and Level III are heavily outlined. Implementation sensitivity effects for these are:

Telemetry Data Rate

Level II — this mode of vehicle operation demands automation of several functions that are related to state vector update and rendezvous and docking. Lower data rates and less TM processing are required. In particular, the higher rates (such as 50 Kbps) that have been identified for a remote TV docking system would not be needed.

Level III — ground control of final docking will require higher data rates. Preliminary design of the candidate for a manually controlled remote TV docking has shown that rates of the order of 50 Kbps are needed for the slow scan TV application, and 160 Kbps for maintenance data telemetry. This impacts both the data switching and the data processor provisions.

RF Coverage — Uplink/Downlink

Level II — continuous direct transmission and reception between the Tug and the ground network stations is not feasible for some mission orbits. This necessitates in addition to the ground stations, the utilization of relay satellites and/or satellite beacons. Storage of maintenance data in the DMS tape recorder of up to 800 bits is required, if no telemetry dump is made for the engine burn data midflight. For those situations where satisfactory state determination is not available from external sources, independent Level I onboard equipment with this capability must be provided.

Level III — demands even greater rf contact with the ground stations, and the use of TDRS, or the equivalent, is a necessity for the present Tug mission model. For some of the mission orbits there is doubt that the planned active STDN stations and TDRS can adequately support the requirements of Level III operation.

Level III operation will increase the communication link recurring operating costs for the program, but will not significantly influence the onboard avionics cost for the Communications subsystem.

Data Management Subsystem. Changing between Level II and Level III autonomy has no appreciable impact on the computer and data bus architecture of the DMS. However, the software programming of onboard control, state vector updates, onboard checkout, and the telemetry formatting are directly affected. These impact the computer sizing relative to throughput and main memory and can be a driver in the selection of available machines.

Figure 9-3 shows the recommended baseline DMS, which utilizes parallel dual data buses that are controlled by a SUMC digital computer that is redundant and fault tolerant. At the present degree of specification, there is no change between Levels II and III except that that the computer sizing will be affected, and the tape recorder would

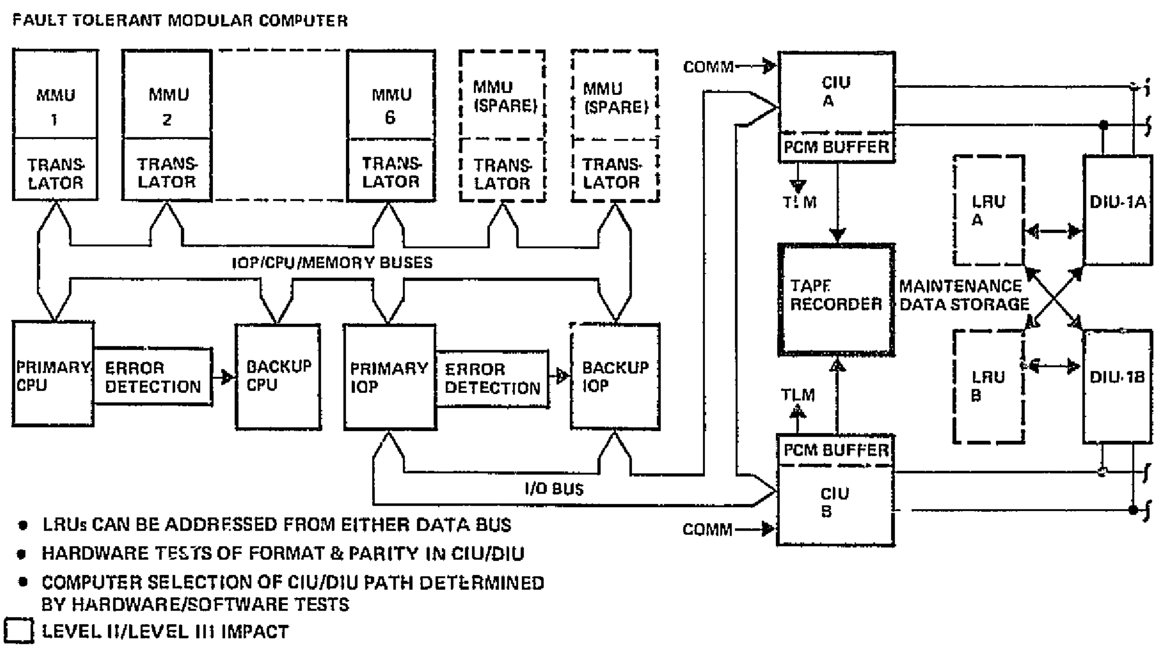


Figure 9-3. Baseline DMS Subsystem

need to be the NASA STD 10^9 bit recorder for Level II autonomy to avoid a wide bandwidth data dump, 160 Kbps or more, and to maintain independency with the ground. Recording of the engine data for maintenance and record purposes requires nearly 800 Mbits of recording. For Level III, a telemetry data dump is needed sometime after the second main engine burn; however, it is assumed for Level II that all the data would be returned on the tape at landing. Table 9-4 lists the software estimates as they have been projected for the recommended Tug avionics system (autonomy Level II minus) and the expected numbers of words for the extremes of Level II and Level III. For cost comparisons, see Table 9-5 in Section 9.2.4.

Table 9-4. Tug Software Estimate Comparison

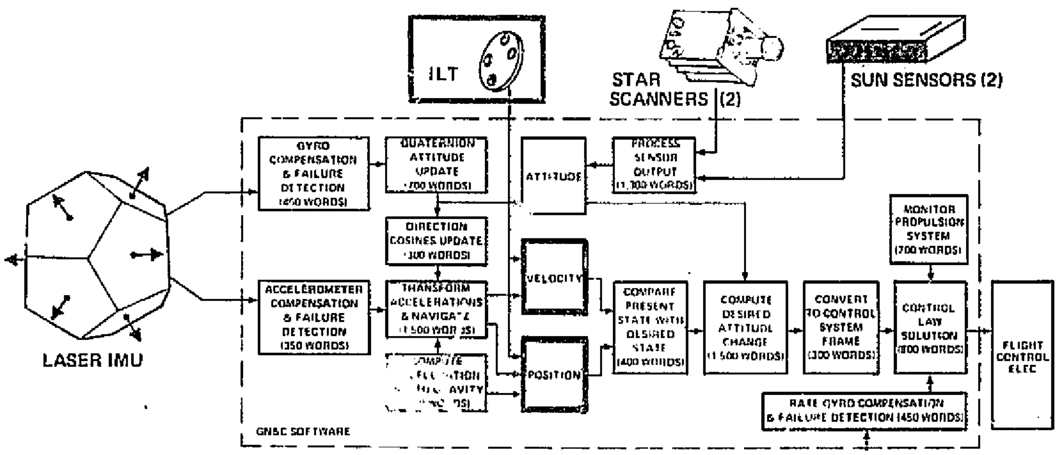
Function	Baseline	Level II	Level III
Executive	3,067	3,067	2,500
Data Pool	2,680	2,680	3,000
Navigation	4,371	4,371	2,500
Guidance	4,595	4,595	4,595
Flight Control	2,340	2,340	2,340
Telemetry	4,017	4,017	4,017
Sequencing	1,816	1,816	1,500
Rendezvous & Docking	3,500	3,500	1,000
Tank Press. Control	794	794	794
Phased Array Control	220	220	220
Checkout (Status)	6,200	6,200	3,500
Redundancy Management	3,126	3,126	2,000
Utility Modules	3,900	3,900	2,000
Total	40,626	40,626	31,949
Instruction Mix: Assume: 75% 16 bit 25% 32 bit	25,391	25,391	19,296
HOL Compiler Overhead Assume: 20%	5,078	5,078	3,859
Estimated Memory Required	30,469	30,469	23,155 words

Guidance, Navigation, and Control — Principal GN&C subsystem sensitivities to autonomy level derive from the basic question of how much capability is needed onboard the Tug for state determination. For the more dependent levels (III and IV), the need for onboard sensors and for onboard computational capacity is reduced. However, the problems of maintaining communications over the external rf links at the proper update times are significant for some mission orbit situations. These are of particular concern during polar orbits over the southern hemisphere where tracking and ground station communications can be troublesome.

Figure 9-4 shows the baseline GN&C subsystem, and the areas most sensitive to changes of the autonomy level between Levels II and III are indicated. Descriptions of the differences follow:

Position and Velocity Update

Level II — Tug state determination is completely onboard and independent. This requires the use of either Earth or celestial objects as a frame of reference. Level II is the same as a Level I requirement for this function. The baseline implementation recommendation for independent determination of position and velocity is the Interferometer Landmark Tracking method. Attitude determination for Level II and for Level III is onboard with star scanners and sun sensors and is not involved in this comparison.



- DUAL STAR TRACKERS & SUN SENSORS RETAINED FOR ATTITUDE UPDATE
- ILT FOR POSITION & VELOCITY UPDATE
- GN&C SOFTWARE = 8,500 WORDS
- ☐ LEVEL II/LEVEL III IMPACT

Figure 9-4. Baseline GN&C Subsystem

Level III — ground augmentation to provide update of the state vector can result in lower costs for the onboard avionics. The Interferometric Landmark Tracker would not be used. However, these costs will be offset by increased costs for ground tracking, ground computation, support software, and increased operational need for communication services. Orbital restrictions for direct link operation require either relay satellite support or mission orbits that permit needed state updates from the ground stations at practical mission times. Approximate cost data for the GN&C elements is compared for autonomy Levels II and III in Table 9-5 in Section 9.2.4.

Rendezvous and Docking Subsystem — Direct effects on this subsystem arise from the basic differences between a fully automatic and independent docking method required for Level II autonomy, as compared to the dependent remote control and monitoring of final docking, which is the candidate for Level III operation. Sensor requirements onboard and the communication link specifications are significantly different for the two cases.

Sensors and Docking Control

Level II — completely independent target search, acquisitions, lock-on, and terminal docking control capabilities are required. This necessitates the further development of sensors and controls specifically for this application. The requirements and problems associated with these elements have been a primary examination of Task B of this Avionics Definition Study and the details will not be repeated here. Figure 9-5 shows the scanning laser radar (LADAR) required.

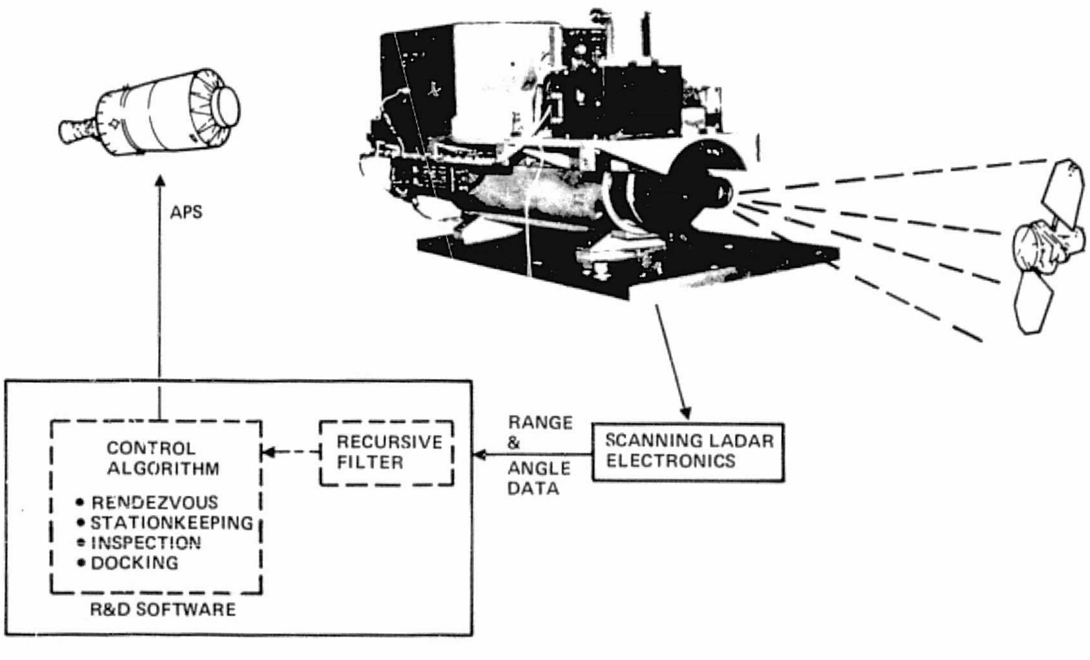


Figure 9-5. Rendezvous and Docking Subsystem Autonomous Candidate

Level III — where onboard fully automated sensors and control are not used, remote control from the ground is needed for the final docking. Prime candidate for that function has been a remote TV method using slow scan and requiring increased communication bit rates. This remotely controlled system with man-in-the-loop is illustrated in Figure 9-6.

Comparisons of the costs of the Level II and Level III solutions are given in Table 9-5 in Section 9.2.4.

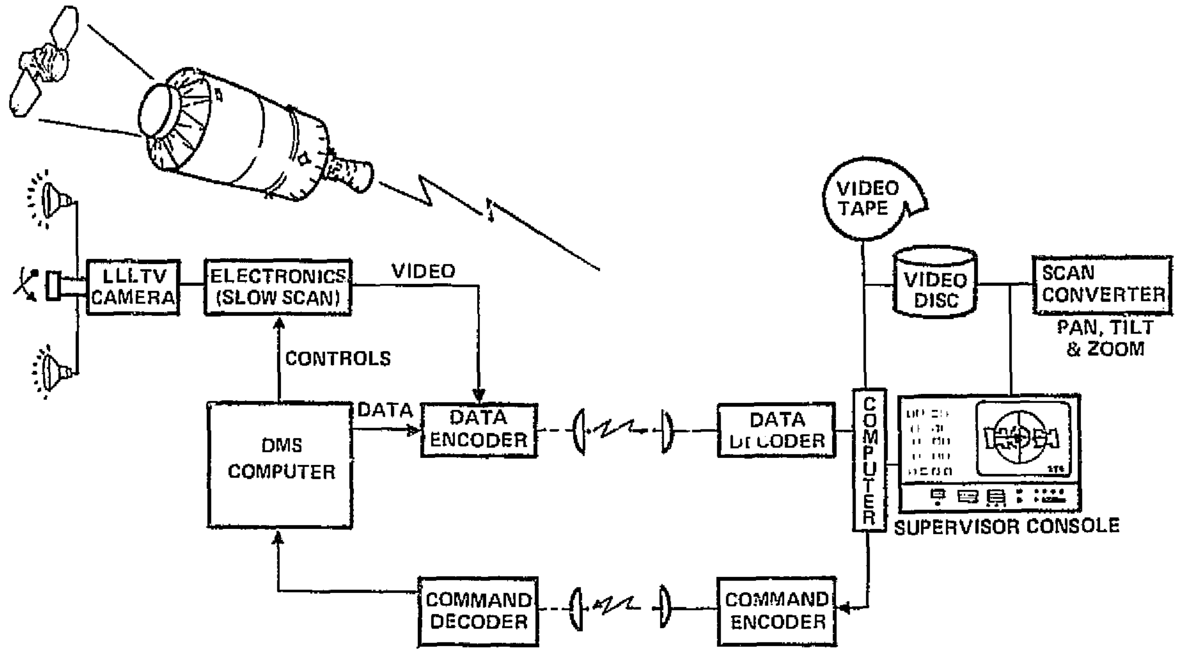


Figure 9-6. Rendezvous and Docking Subsystem Man-in-the-Loop Candidate

Electrical Power Subsystem — For the two autonomy levels being considered, there are differences in the electrical loads deriving from the GN&C subsystem, the Rendezvous and Docking subsystem, and the DMS sizing. Level III operation has a lesser demand for electrical power than Level II. This reduction is less than 5% of the total Level II Tug power requirement. Principal effect is the use of less reactants from the main propellant tanks. This is such a small amount as to be almost unnoticed against the normal boiloff losses, etc. The fuel cell stack design is not impacted, and no significant design changes are anticipated within the electrical power subsystem between Level II and Level III operation.

Instrumentation Subsystem — There will be some secondary impacts on the details of the Instrumentation subsystem. In magnitude and in cost impact they are relatively minor. Consequently, they have not been fully defined for the purposes of this trade due to the lack of data and because the differences between Level II and Level III autonomy are expected to be minimal.

9.2.4 COST COMPARISONS. Subsystem cost data for the onboard avionics system has been estimated for the Level II and Level III Tug operational autonomy conditions. These cost comparisons are shown in Table 9-5. Airborne associated costs only are shown, with no attempt to guess at the full impacts of the ground environment costs on program level decision.

Data Management Subsystem (DMS) differences are reflected in the DMS software costs. There is also the possibility of a small delta in the selection of the recording capacity of a NASA standard tape recorder, either a 10^8 or a 10^9 bit machine. This could increase the cost of Level II by approximately $25,000 and is negligible in the analysis. The decision in this area is not yet firm, and the cost element delta was omitted.

Table 9-5. Autonomy Cost Summary — Airborne Associated Costs Only

	Costs (in thousand dollars)					
	Level II			Level III		
Subsystem	DDT&E	Prod/ Oper	Total	DDT&E	Prod/ Oper	Total
GN&C	3,897	9,761	13,658	1,241	7,944	9,185
Rendezvous & Docking	11,755	16,223	27,978	1,614	5,368	6,932
DMS Software	4,663	5,666	10,329	3,637	4,419	8,056
Totals	20,315	31,650	51,965	6,492	17,731	24,173

9.2.5 <u>CONCLUSIONS AND RECOMMENDATIONS.</u> A best compromise between the autonomy Levels II and III appears to be "Level II minus" as originally called out for the baseline at the start of the Avionics Definition Study. The biggest driver on the exact definition of this intermediate state appears to be the final decisions on the operation of the Rendezvous and Docking Subsystem. Some of the questions about the details of the docking method are expected to be answered in later NASA studies that follow the outputs of Task B.

Sensitivities of the subsystem designs to autonomy level are greatest in the:

a. Rendezvous and Docking Subsystem

b. Guidance, Navigation, and Control Subsystem

c. Communications Subsystem

The DMS can also be noticeably impacted, primarily in software requirements and sizing, where it is involved in the management and computational services for the noted subsystems. Cost factors can exhibit appreciable differences for all of these subsystems, and the final choices can swing large elements of program costs between nonrecurring developmental and recurring operational costs.

As more complete trade studies are conducted, and as concept verification testing is accomplished, the definition of the baseline is expected to change. The precise level of autonomy may change within the avionics system elements, but the total vehicle operational level of autonomy is projected to remain at a level intermediate between the present definitions of Level II and Level III.