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DRL LINE ITEM 7

MSC-04482
SD 72-SA-0007

ORBITAL OPERATIONS STUDY
FINAL REPORT

APPENDIX A
INTERACTIVITY ANALYSES

MAY 1972

APPROVED BY

L. R. Hogan

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FOREWORD

This report contains the results of the analyses conducted by the Space Division of North American Rockwell during the Orbital Operations Study, Contract NAS9-12068, and is submitted in accordance with line item 7 of the Data Requirements List (DRL 7).

The data are presented in three volumes and three appendixes for ease of presentation, handling, and readability. The report format is primarily study product oriented. This study product format was selected to provide maximum accessibility of the study results to the potential users. Several of the designated study tasks resulted in analysis data across elements and interfacing activities (summary level); and also analysis data for one specific element and/or interfacing activity (detailed level). Therefore, the final report was structured to present the study task analysis results at a consistent level of detail within each separate volume.

The accompanying figure illustrates the product buildup of the study and the report breakdown. The documents that comprise the reports are described below:

Volume I - MISSION ANALYSES, contains the following data:

- o Generic mission models that identify the potential earth orbit mission events of all the elements considered in the study
- o Potential element pair interactions during on-orbit operations
- o Categorized element pair interactions into unique interfacing activities

Volume II - INTERFACING ACTIVITIES ANALYSIS, contains the following data:

- o Cross reference to the mission models presented in Volume I
- o Alternate approaches for the interfacing activities
- o Design concept models that are adequate to implement the approaches
- o Operational procedures to accomplish the approaches
- o Functional requirements to accomplish the approaches
- o Design influences and preferred approach selection by element pairs.

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This volume is subdivided into four books or parts which are:

Part 1. INTRODUCTION AND SUMMARY - Condensed presentation of the significant results of the analyses for all interfacing activities

Part 2. STRUCTURAL AND MECHANICAL ACTIVITY GROUP

- o Mating
- o Orbital Assembly
- o Separation
- o EOS Payload Deployment
- o EOS Payload Retraction and Stowage

Part 3. DATA MANAGEMENT ACTIVITY GROUP

- o Communications
- o Rendezvous
- o Stationkeeping
- o Detached Element Operations

Part 4. SUPPORT OPERATIONS ACTIVITY GROUP

- o Crew Transfer
- o Cargo Transfer
- o Propellant Transfer
- o Attached Element Operations
- o Attached Element Transport

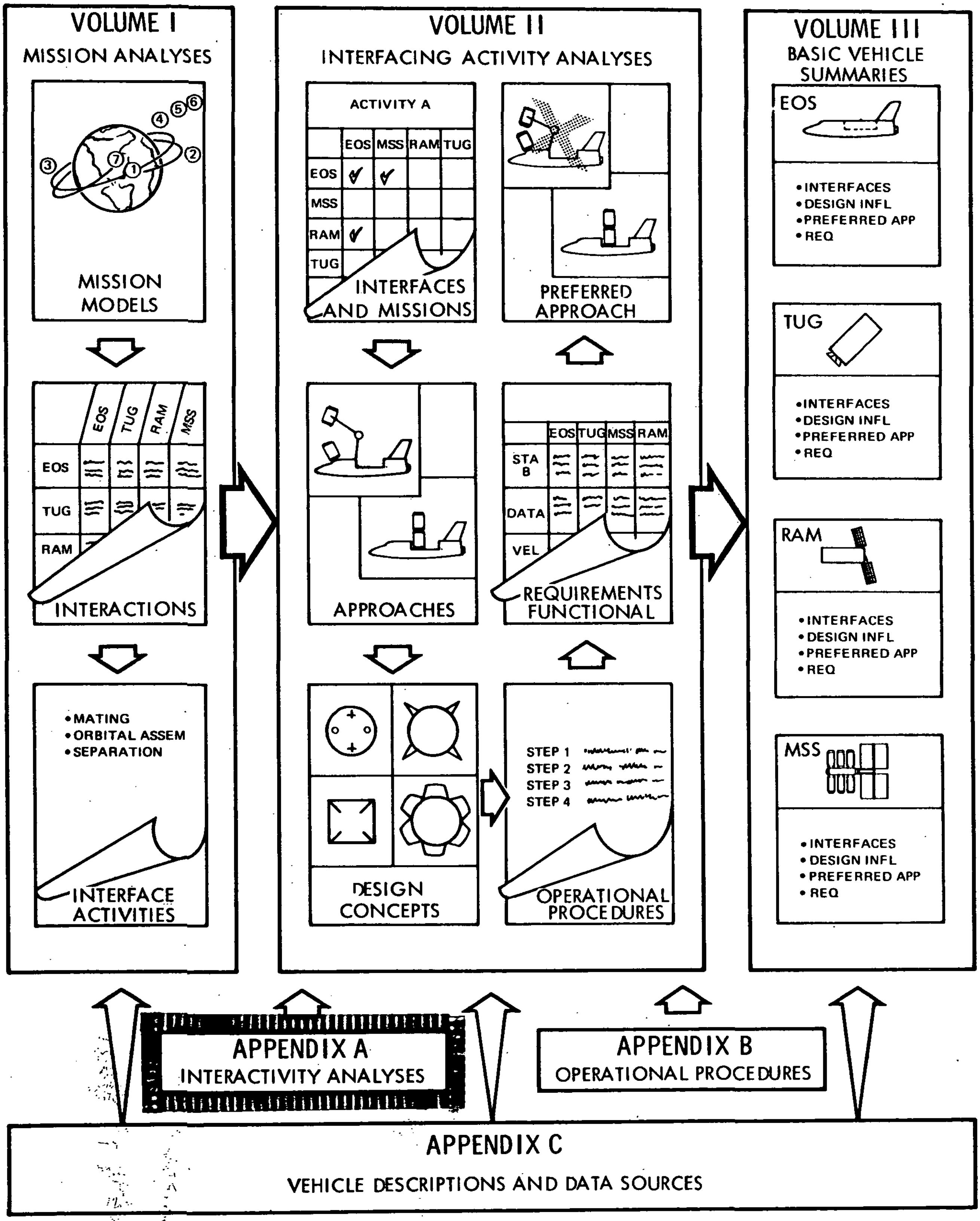
Volume III - BASIC VEHICLE SUMMARIES, contains a condensed summary of the study data pertaining to the following elements:

- o Earth Orbital Shuttle
- o Space Tug
- o Research and Applications Modules
- o Modular Space Station

Appendix A - INTERACTIVITY ANALYSES, contains many of the major trades and analyses conducted in support of the conclusions and recommendations of the study.

Appendix B - OPERATIONAL PROCEDURES, contains the detailed step-by-step sequence of events of each procedure developed during the analysis of an interfacing activity.

Appendix C - VEHICLE DESCRIPTIONS AND DATA SOURCES, presents a synopsis of the characteristics of the program elements that were included in the study (primarily an extraction of the data in Appendix I of the contract statement of work), and a bibliography of the published documentation used as reference material during the course of this study.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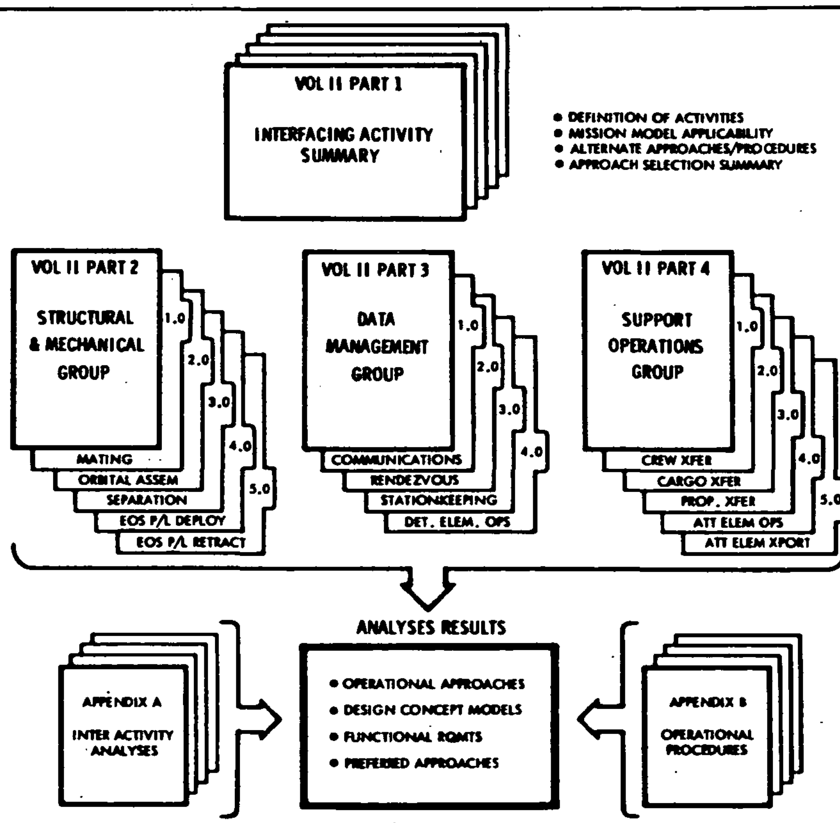
INTRODUCTION

The technical studies contained in this appendix comprise a portion of the supplemental analyses conducted to verify that safe, feasible, design concepts exist for accomplishing the attendant interface activities. Technical data contained in the separate sections of this appendix is primarily concerned with functions and concepts common to more than one of the interfacing activities or elements.

The relationship of this book to the remaining documents is illustrated by the following:

INTERFACING ACTIVITY ANALYSES DOCUMENTATION



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ILLUSTRATIONS
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ILLUSTRATIONS
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PROPELLANT TRANSFER ANALYSES (CONTINUED)

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(Continued)

TABLES

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TABLES
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A1. STATE VECTOR UPDATE, ATTITUDE DETERMINATION AND LOS TRACKING

INTRODUCTION AND SUMMARY

This study was conducted as a portion of subtask 2.4 (titled Design Influences) to support the interfacing activities of (1) rendezvous, (2) stationkeeping, and (3) detached element ops. The key function matrix of each activity identified (a) state vector update, (b) attitude determination, and (c) line-of-sight (LOS) tracking as requiring further analysis to assure safe, feasible design concepts exist to perform the functions. A secondary reason for this study is the potential commonality between the hardware required for each function. A tertiary benefit is also anticipated from the application of derived design concepts to other interfacing activities such as mating.

Several alternate design concepts were postulated for each of the three key functions. A review of the tables will reveal that the list is by no means complete, however the candidates were derived from consideration of the vehicle inventory. Three candidates from each function table were selected for detailed investigation. The rationale used for this selection is the technology status and/or the commonality potential.

It was concluded from this study that adequate design concepts exist for all cases requiring the functions of (a) state vector update, (b) attitude determination, and (c) LOS tracking. The procedures developed appear compatible with the spectrum of design concepts.

INTERFACE ACTIVITY APPLICABILITY

1. The stationkeeping operation potentially involves all three functions -- attitude determination, state vector update, and perhaps LOS tracking. For purposes of this discussion, it will be assumed that the stationkeeping is occuring between two orbital elements. Consider now the design options when all three functions are necessary. The three functions can be satisfied using either the laser radar and an attitude reference or a gimballed radar and an attitude reference. The state vector update function is satisfied by the range, range-rate determination capability of either the laser or MW radar.

2. Rendezvous operations require for the initial phases(s) both the state vector update and attitude determination function. For the terminal phase LOS tracking may be desirable. The state vector update function can be provided using star-horizon measurements to initialize an inertial navigation system such as either a gimballed platform or strapdown inertial measurement unit. The attitude reference would be provided by the IMU and updated as required using one of the star tracker mechanizations. Options for the terminal phase rendezvous sensor includes PRS or any of the LOS tracking alternatives.

3. Detached element operations when space controlled are essentially equivalent to either the terminal phase of rendezvous or stationkeeping as discussed above.

4. Ground controlled detached element operations entail the functions of state vector update and attitude determination. The attitude determination function is associated with the proper orientation of the element in order to execute propulsive positioning maneuvers. The state vector update function could be performed autonomously onboard using star-horizon measurements or from the ground using either the existing MSFN net or TDRS.

CONCLUSIONS AND RECOMMENDATIONS

The Scanning Laser Radar (SLR) system is recommended as the sensing system for determination of navigation parameters at close ranges (less than 75 nautical miles) between elements. This system provides precision measurements of range, range-rate, line-of-sight angle and angle rates between elements that are necessary for close range stationkeeping, terminal phase rendezvous and docking maneuvers. Other systems - rf ranging, microwave radar - cannot provide the measurement precision nor perform measurements down to zero range. The following table defines the accuracies projected for the Scanning Laser Radar system that use passive optical reflectors on the target element:

Range - ± 0.02 % of range or 4" whichever is greater
Range-rate - ± 1.0% of range-rate or 0.4 in/sec, whichever is greater
Line of Sight Angles - ± 0.02°
Relative Target Attitude - ± 1.0° (when range is less than 1000 ft)
Angular Rates - ± 1.0% or ± 0.01 deg/sec whichever is greater

The Scanning Laser Radar system provides direct readout of these parameters that can be used for stationkeeping, rendezvous and docking maneuvers. It provides a common system for these activities that performs with precision to effect optimum maneuvers with safety.

The Scanning Laser Radar is considered as a feasible concept for relative element position determination at ranges under 75 miles and for relative attitude determination at less than 1000 to 500 feet. For longer ranges, where stationkeeping and rendezvous operations are performed up to element separation of 2000 n miles, other techniques must be used to provide relative position data, as well as state vector, orbital parameter data and attitude orientation data for each of the elements.

For state vector updates of the individual elements, star-horizon measurements provide sufficient accuracies for rendezvous and stationkeeping at separation ranges greater than 75 n miles. Star tracker similar to the ITT Aerobee 150A with accuracies of 36 arc seconds is present state-of-the-art. Incorporation of horizon trackers provides increased reliability, allows use for attitude determination and provides the attitude reference necessary for star tracking measurements.

Existing state-of-the-art hardware such as the Quantic Mod IV horizon tracker provide local vertical measurement accuracies to within 0.1 degree. This combination can be used for most element pair autonomous operations at greater than 75 n.mile range. When ground tracking is available from the ground network, improved accuracies would result by using tracking and ranging updates from ground stations. The same combination of star tracker and horizon scanner can be used to provide three axis, local level reference. The hardware to perform within sufficient accuracies is developed and available for use.

The design concept model that results for rendezvous, stationkeeping and docking for the ranges required is displayed in the following table.

Table A1-1. Approach Selection

	Greater Than 75 Miles		Less Than 75 Miles
	Autonomous	Ground Control	
Attitude Determination			
1. Horizon-scanner and star-tracker	X	X	
2. Scanning laser radar using target passive reflectors			X
State Vector Determination			
1. Star trackers and horizon trackers	X		
2. Star tracker and horizon trackers with ground tracking/ranging update.		X	
Relative Position			
Scanning Laser Radar using target passive reflectors			X

STUDY APPROACH

The same approach was applied to the three functions of state vector, attitude determination, and LOS tracking. Briefly the approach is as follows:

1. Postulate alternative design concepts

2. Select concepts for detailed analysis

3. Prepare the following for each selected concept

   a. Short description of how it is mechanized (functional diagram with words), how it works, and technology status

   b. Evaluation of other functions that can be performed using the identified hardware elements.

   c. Evaluation of the constraints which limit the applicability of the concept for various missions.

4. Relate the results of the above analysis to the previously established operations and procedures. Consider at least stationkeeping, rendezvous, and detached element operations. Extract conclusions and formulate recommendations for commonality.

STATE VECTOR UPDATE

This function pertains to the process of navigation. It concerns the determination of the orbital position and velocity of the vehicle. Table A1-2 lists the candidate concepts postulated with an identification of those selected for detailed discussion in subsequent paragraphs.

Table A1-2. State Vector Update Concepts

Concepts	Descriptions
Star Horizon* Measurements	Uses star trackers and horizon trackers, no ground support required, low cost.
Ground Tracking MSFN	Existing capability, large scale ground support required, good performance
TDRS*	Excellent performance, some ground support
Space Track (Station Provdes)	No ground support, steerable dish on space station, transponder on user, computation on station
Unknown Landmark Track	No ground support, automatic optical tracker on spacecraft, partially developed, good performance
Precision Ranging* System	Ground transponders; transceiver on spacecraft; provides precise range and range rate measurements; established technology
	*Selected for more detailed analysis

Star Horizon Measurements

1. Concept Description

A typical star-horizon mechanization is shown in functional diagram form on Figure A1-1. This concept uses gimballed star trackers and a four head horizon edge tracker. The measurements from these instruments can be processed to determine the local level plane, star elevation angles, horizon depression angles, and apparent earth diameter. These data are then used to estimate the vehicle state vector.

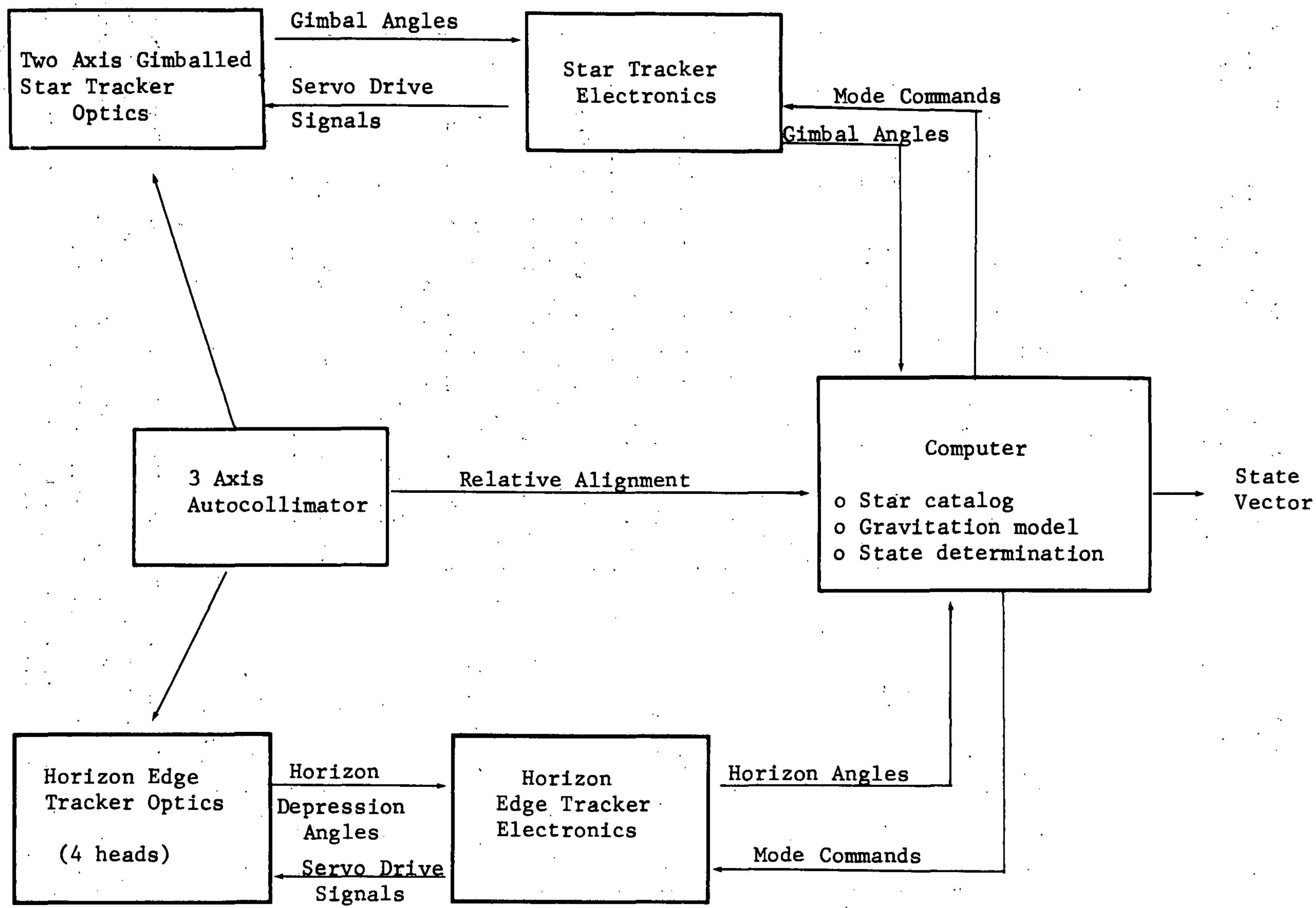

Figure A1-1. Star-Horizon Functional Diagram

The autocolimator shown on the diagram is used if it is necessary to mount the star tracker remote from the horizon tracker heads. The autocollimator enables a precise measurement of the relative alignment between the optics. This alignment is critical to the performance of the system.

The technology status for the star-horizon concept is considered excellent with potential application to mid- or late-1970's missions. With respect to the alternative concepts presented in Table A1-2, star-horizon measurements are considered second only to ground tracking and perhaps PRS measurements. The necessary hardware is essentially developed.

2. Multifunction Considerations

The hardware elements shown on the functional diagram can also be used as a source of the attitude reference for local level mode attitude control. The star trackers can be used as an inertial attitude reference (see Attitude Determination). They can also be adapted to track an optical beacon and serve as a rendezvous sensor. The autocollimator can be used as a means of transferring alignment from the attitude reference to an experiment which is alignment critical. The computer can serve other subsystems, experiments, flight operations, and other functions depending upon its capacity.

3. Multimission Considerations

The star-horizon concept for state vector update is perhaps best suited to missions which are performed under stable orbit conditions, i.e., the modular space station. Missions which frequently require propulsive maneuvers for orbital transfers would probably utilize inertial navigation during the propulsive phases. The star-horizon concept is of course, applicable during stable orbit phases of such missions and can be used to initialize the inertial navigation process.

The star-horizon concept is best performed in a local level flight mode. It is therefore best suited to a vehicle which operates continuously in the local level mode or at least can tolerate periodic local level operation to allow the sensors to perform the navigation measurements. Thus the concept is ideally suited to an earth survey mission and poorly suited to a mission such as solar astronomy.

The star-horizon concept is applicable to lunar orbit missions as well as earth orbit missions. For the lunar missions the sightings are taken on the lunar horizon. The horizon sensor for lunar orbit use would probably operate in the 40 micron band rather than the 14-16 micron band which is preferable for earth orbital operations.

TDRS

1. Concept Description

The tracking and data relay satellite (TDRS) concept employs a network of ground station(s) and synchronous satellites than can be used to provie state vector updates. This concept has the potential of providing continuous contact between an earth orbit user and the ground.

This discussion will be limited to the tracking function of the system. The measurements will probably consist of range and range rate. There are numerous ways of mechanizing the system. One for example would initiate the tracking at a ground station with the signal relayed through a TDRS to the user where it is turned around by a transponder. The return signal is relayed back to the ground station through the TDRS. The ground station which accurately knows the TDRS position then processes the data to determine the position of the user vehicle. The result is transmitted to the user. Other mechanizations might process the data onboard the user - a concept perhaps best suited to a sophisticated user with extensive computation capability such as the Space Station.

The tracking performance of the TDRS concept depends upon many factors including the required response time (time available for tracking), and the capability to simultaneously track a user from more than one TDRS. The performance is relatively insensitive to apriori data defining user position, user orbital eccentricity, and choice of 28.5 degrees or equatorial orbits for the TDRS.

2. Multifunction Considerations

The TDRS concept serves not only the state vector update function but also a communications or data relay function.

3. Multimission Considerations

A functioning TDRS network with deployed TDRS and ground stations is suitable for performing the state vector update function for many vehicle types or missions. TDRS can readily satisfy the requirements of a vehicle such as space station. A vehicle such as station which does not require fast response tracking (and actually can be tracked over more than one orbit) can probably be served by a single TDRS vehicle.

The fast response problem, perhaps associated with orbit insertion of a manned spacecraft, may require simultaneous tracking from two TDRS.

Ground Tracking - MSFN

This concept employs the existing MSFN tracking to determine the state vector of the user vehicle. The result is then transmitted to the user. The capability of the system is restricted by the number of ground stations available. Use of this concept would be heavily impacted by TDRS. If TDRS is available, then use of the ground net can be expected to be minimized.

Precision Ranging System

1. Concept Design

The precision ranging system (PRS) is mechanized using a transceiver on the spacecraft and a network of ground transponders. When the spacecraft passes within range of a ground transponder it transmits a coded signal that is received and processed by the transponder. If the code on the signal transmitted by the spacecraft agrees with the transponder code, a return signal is transmitted from the transponder. The spacecraft receives this signal and computes the phase difference between the transmitted and returned signals and thereby determines the slant range between the spacecraft and the transponder. Doppler velocity can also be determined. The precise location of the ground transponders is stored in the onboard computer. These data when processed with the slant range measurement can then be statistically filtered to estimate the vehicle state vector. The accuracy of the estimate improves as the spacecraft passes within range of other beacons and repeats the process.

The spacecraft portion of the PRS is the interrogator equipment which consists of the transceiver and an antenna assembly. The remainder of the system is composed of transponders located globally and a transponder onboard cooperative targets. The basic measurement is the range from the onboard interrogator to one of the transponders. In operation, the modulation generator in conjunction with the reference oscillator in the onboard interrogator supplies one or more range data modulation frequencies or tones to the transmitter. The transponder receiver detects these signals and passes them on to the modulator, where they are remodulated on an offset carrier. (The offset carrier approach allows the transponder to transmit and receive simultaneously).

The receiver in the onboard interrogator detects the offset carrier signals. These range signals are then demodulated and fed to a phase digitizer. Here, the phase of the received moduled tones are converted to digital words. The reference tones also are converted to digital words and then subtracted from the received digital words to yield a word representing the phase difference. The phase difference is then directly translated into a range measurement. A phase digitizer is provided for each frequency or tone that is generated.

Additional processing in the transceiver combines the range measurements for each tone to arrive at one unambiguous range signal that is transmitted to the computer. The number of tones required is dictated by the maximum unambiguous range measurement required and the accuracy required.

The system also measures range rate by measuring the Doppler offset of the carrier frequency caused by the relative motion between the interrogator and the transponder. Very accurate range-rate measurements are obtained by generating the signal in the interrogator, transmitting it to the transponder, and receiving the transmitted signal from the transponder in the interrogator. This roundtrip approach is used so that the frequency of the received signal can be compared directly with that of the generated signal thus ensuring a high degree of accuracy.

The technology for PRS is considered to be within today's state-of-the-art requiring only implementation.

2. Multifunction Considerations

The PRS concept is suitable not only for orbital state vector update, but also rendezvous, deorbit, aerodynamic maneuvering and landing approach. In the case of rendezvous, the PRS concept provides relative range and range rate data between the active vehicle and the target. The active vehicle interrogator works with a transponder on the target vehicle. The concept does not provide LOS angular data and therefore a supplementary sensor may be required.

The concept is suitable for the terminal phase of rendezvous and stationkeeping assuming the nominal range is on the order of 500 feet. Resolution of the range data does not permit PRS use for automatic docking. Estimated range accuracy at 500 feet is $\pm$ 50 feet with range rate determined to 0.1 FPS.

A vehicle capable of deorbit, entry, aerodynamic maneuvering and landing can utilize PRS for navigation during all of these phases. A cluster of ground transponders positioned with respect to the touchdown point can provide cross track and down range data for the aerodynamic phase. Altitude data from PRS may not be suitable due to the acute angle to the transponders.

3. Multimission Considerations

Given an established network of ground transponders, the PRS concept has the potential of satisfying a broad spectrum of users. The potential performance of the system will be heavily dependent upon the location of the transponders with respect to the ground trace of the user. A global network of transponders will produce better performance than a network restricted to the continental United States. Furthermore, a network of transponders placed for support of low inclination earth orbit missions would not provide comparable performance for polar orbit missions.

NR studies (Reference 4) have shown that star-horizon measurements can be used to supplement a concept such as PRS. The result is that the star-horizon measurements bound the error by preventing the large error buildup between transponder sightings. Figures A1-2 and A1-3 illustrate this. Figure A1-2 shows the navigation error that results when only range rate measurements are made and there are orbits with no transponder contact. Figure A1-3 shows the same situation with star-horizon measurements added.

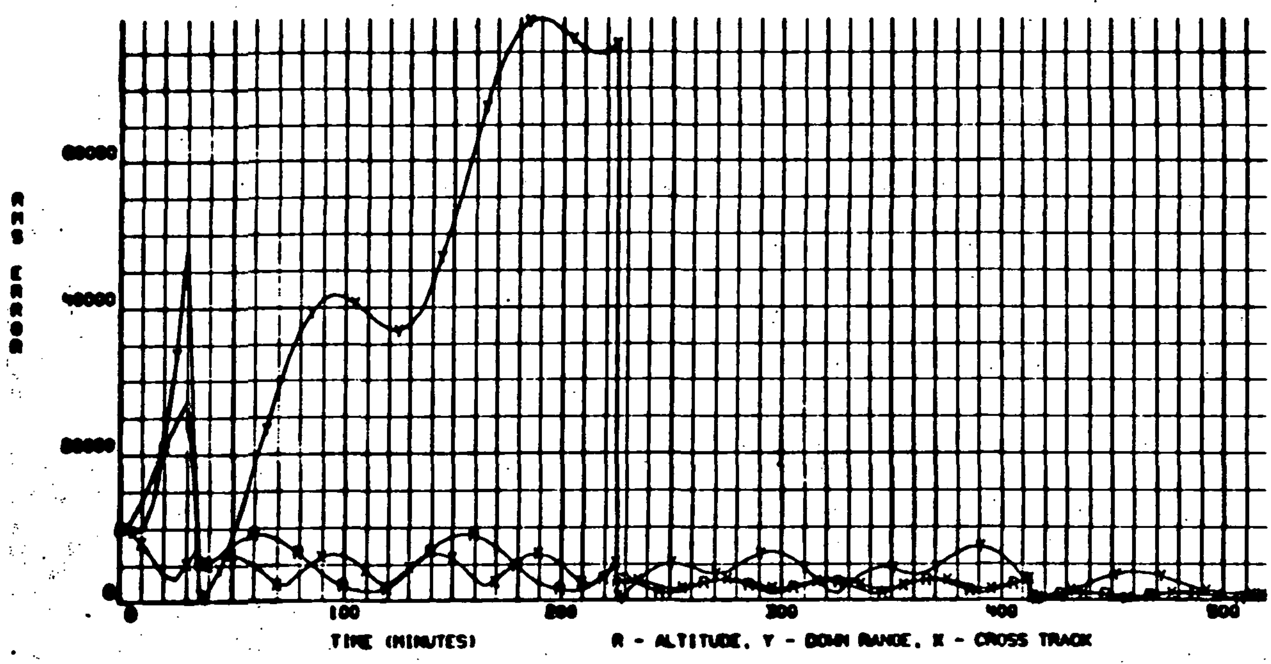

Figure A1-2. Beacon Measurement Performance Using
Range Rate Measurements Only

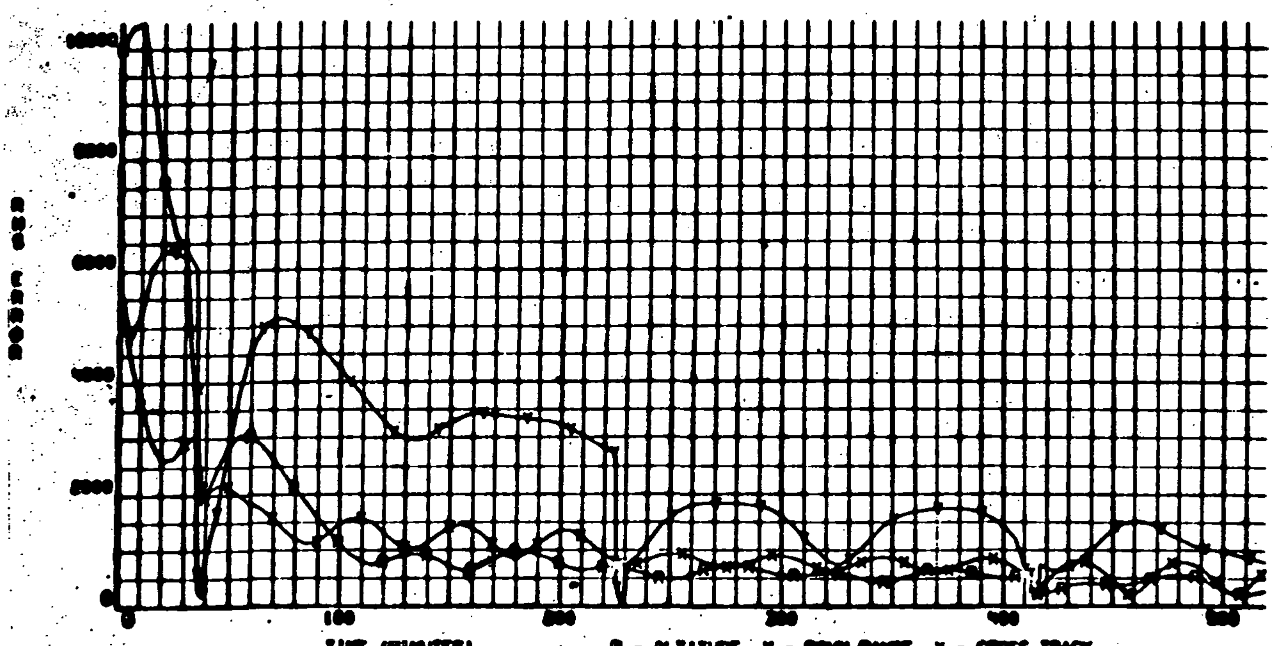

Figure A1-3. Beacon Measurement Performance Using Range
Rate Measurements Plus Star-Horizon Measurements

The concept of combining star-horizon measurements with PRS would significantly improve the performance obtainable from a limited ground transponder network.

PRS is applicable for lunar orbit missions but primarily from a rendezvous standpoint rather than a navigation standpoint. The navigation problem is hampered first by the problems of placing transponders on the surface of the moon and second either maintaining or replacing them when failures occur.

For rendezvous operations, PRS is equally applicable to earth or lunar orbit operations.

ATTITUDE DETERMINATION

The attitude determination function concerns the establishment of an attitude reference. The maintenance of the reference once established may be accomplished using the same hardware or perhaps additional equipment. Concepts for performing this function are contained in Table A1-3.

Table A1-3. Attitude Determination Concepts

Candidate	Descriptors
Horizon Scanner & Gyro Compassing	Three axis, local level, velocity vector reference, existing technology
Horizon Scanner & Ion sensor	Three axis, local level, velocity vector reference, limited life
Strapdown Star Tracker* & Star Pair Logic	Three axis, inertial reference, demonstrated as breadboard
Sextant	Two axis, inertial reference, requires operator, existing technology
Pair of Gimballed* star trackers	Three axis, inertial reference, existing technology
Horizon Scanner & * Star Tracker	Three axis, local level reference, hardware developed
Sun Sensor and Star Tracker	Three axis, solar reference, hardware developed
RASS	Three axis, measures altitude and velocity, multi-beam radar, partially developed
	*Selected for more detailed analysis

Star Pair Attitude Reference

1. Concept Description

This concept is an autonomous star mapping attitude reference technique (SMART). The concept uses one field of view and only two stars to provide three axes attitude information. To identify stars, the SMART concepts recognizes to some degree of resolution that the separation angle between any two stars is unique. The 180-degree ambiguity possibility is resolved by noting which of the two stars is brightest. The system solves for vehicle orientation in an inertial reference frame by utilizing the stars identified and stored star coordinates in that reference frame. The operation of the concept is illustrated in the diagram on Figure A1-4 and described in Reference 6.

The only hardware required for the SMART concept is an electronically scanned star mapper and a computer -- thus no moving parts. For most missions, it would be advantageous to utilize three star mappers mounted orthogonally to each other to provide for greater reliability and sufficient celestial sphere coverage.

Figure A1-5 is a map prepared from a search of the celestial sphere to determine stars of apparent visual magnitude +3 or brighter that satisfy a 2' 32" angular separation criteria assuming a 40 degree sensor FOV. The map shows that approximately 50 percent of the area of the celestial sphere satisfies these criteria. The map was prepared by a digital computer search of an extensive star catalog (approximately 260, 000 stars) prepared by the Smithsonian Institute.

The concept has been demonstrated at the breadboard level using a Raytheon 706 computer and a simulated optical system with a 40-degree field of view. In this configuration the system was capable of providing attitude updates in less than 50 milliseconds. The computation utilizes approximately 4K, 16 bit words and the computer has a 1.8 microsecond add time.

2. Multifunction Considerations

The computer, depending upon its characteristics, can be used for other vehicle computations.

The strapdown star mapper is less flexible than the gimballed type but may be suitable for some special purpose optical tracking functions. The tracker is restricted to operation within the field of view. This can be accomplished by maneuvering the tracking vehicle or restricting the operations of the target vehicle. If these constraints cannot be met, then the strapdown star tracker is not suitable and another form of tracking must be used.

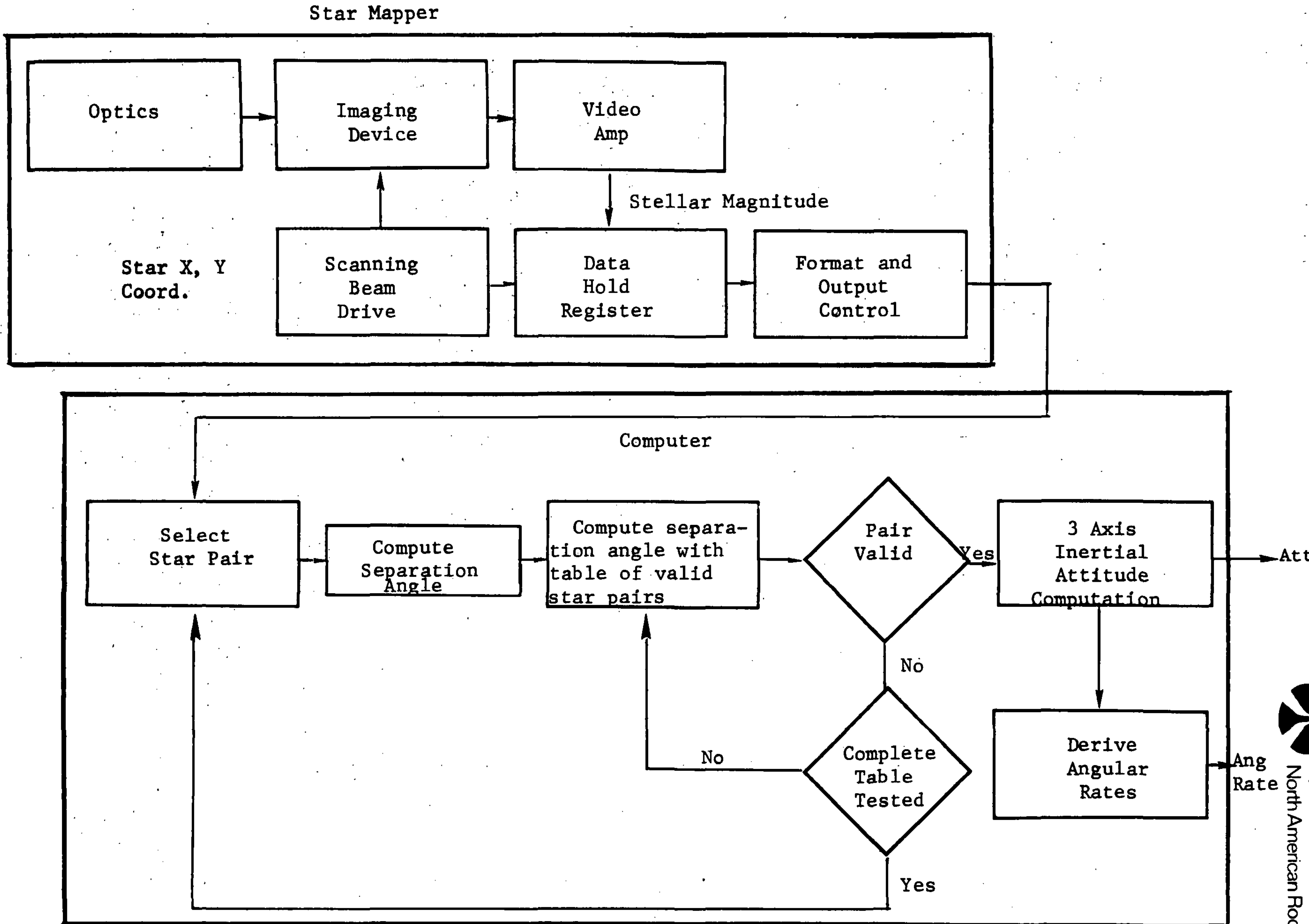

Figure A1-4. Functional Diagram - Star Pair Attitude Determination Concept

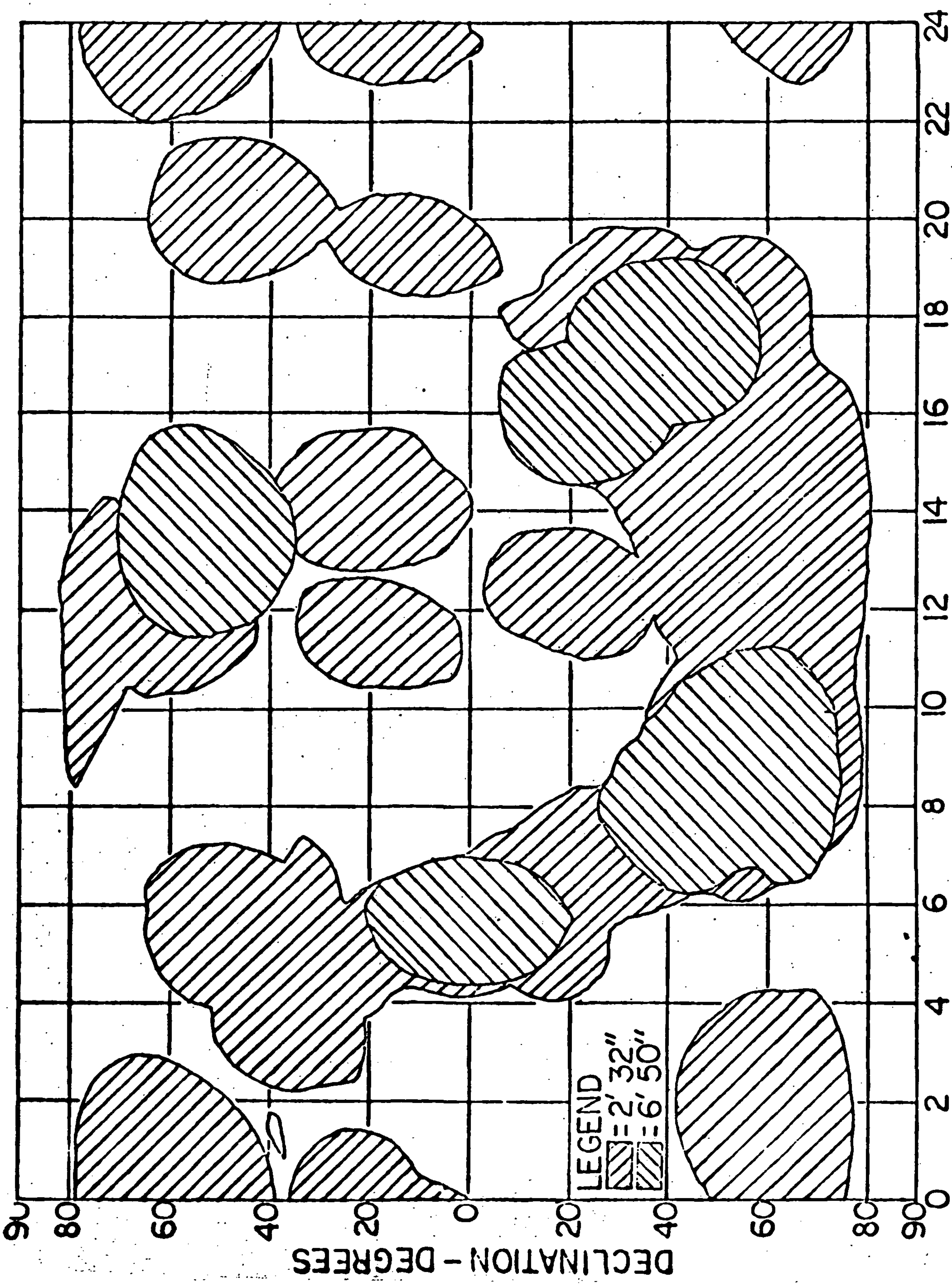

Figure A1-5. Areas With a Mag 3 Star Pair with a Unique Angular Separation of 2' 32" and 6' 50" in a 40 Degree Field of View

3. Multimission Considerations

This concept inherently provides a stellar inertial reference and therefore is best suited to missions conducted in an inertial flight mode. It is expected that the concept is also applicable to slowly spinning vehicles such as those which operate in a local level flight mode. The concept can be mechanized to update a conventional inertial measurement unit. SMART works without prior knowledge of position and is therefore applicable to the problem of initial acquisition or recovery from an arbitrary orientation.

Gimballed Star Trackers

1. Concept Description

This concept employs two gimballed star trackers to compute an inertial, 3 axis attitude reference. A functional diagram is shown on Figure A1-6. It may be necessary to mechanize the concept with an inertial measurement unit (shown dashed in the diagram). The star trackers are then used to update the IMU and the IMU provides the reference between updates.

The measurement data from the star trackers is processed in the computer using statistical filtering of redundant information. The effects of noise errors in the measurements can thereby be reduced. A math model of this concept is derived in Appendix A of Reference 3.

2. Multifunction Considerations

The gimballed star trackers are suitable for performing navigation measurements as discussed in Attitude Determination and optical line-of-sight tracking as discussed in LOS Tracking.

3. Multimission Considerations

This concept inherently provides an inertial reference and is therefore best suited to missions performed in an inertial flight mode.

The concept as presently envisioned does not include an arbitrary acquisition capability. In other words, some means on initial acquisition such as a manual telescope is required. With additional software complexity, an acquisition mode could probably be developed utilizing the star trackers in a search mode and pattern recognition techniques.

This technique is used on the Modular Space Station where a sextant/telescope is available for initialization of the system.

Horizon Sensor and Star Tracker

Refer to the material presented previously for the star-horizon state vector update concept. The functional diagram, discussion of operation, multifunction and multimission considerations are applicable in their entirety.

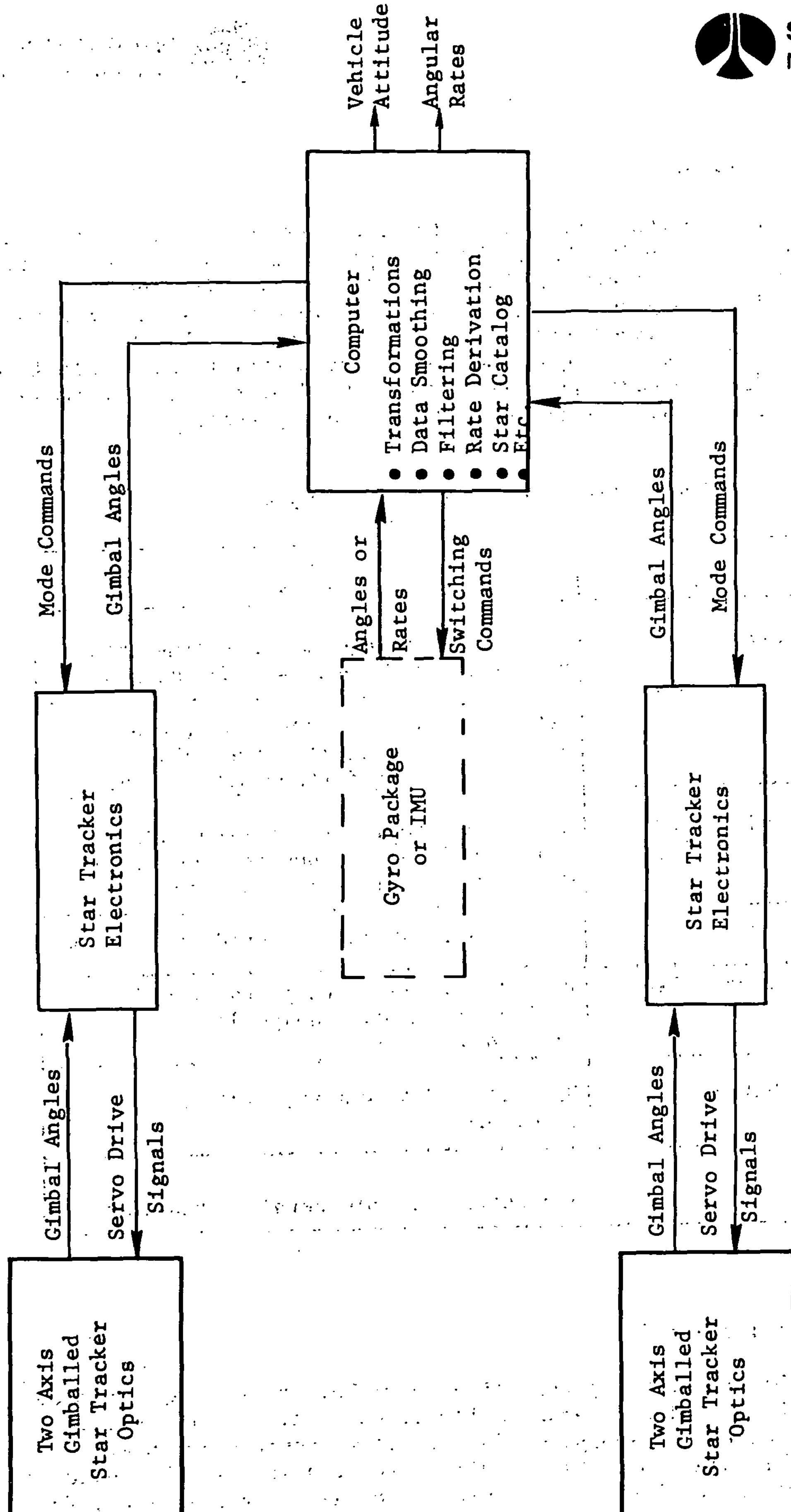

Figure A1-6. Gimballed Star Tracker Functional Diagram

LINE OF SIGHT TRACKING

The LOS tracking function is usually associated with a rendezvous or stationkeeping operation involving the trajectory or path control of one vehicle with respect to another. The tracking function involves the determination of two angles (e.g., azimuth and elevation) in a coordinate frame fixed within the tracking vehicle. These data may then be transformed as required depending upon the use of the information. Table A1-4 lists the candidate concepts postulated with an identification of those selected for detailed discussion in subsequent paragraphs.

Table A1-4. LOS Tracking Concepts

Candidate	Descriptors
Manual Sextant	Determine LOS angles, full time crew attention required, hardware exists
Laser Radar*	Determines range, range rate, LOS angles and rates, relative roll angle; requires further development
Gimballed MW Radar*	Determines range, range rate, LOS angles and rates; good for long range, high power, existing technology
Star Tracker*	Determines LOS angles, hardware exists
Interferometer Radar	Determines range, range rate, LOS angles; requires transponder; electronic scan
Gimballed Ultra Tracker	Determines LOS angles; partially developed
	*Selected for more detailed analysis

Star Tracker

1. Concept Description

This concept employs either a gimballed or strapdown star tracker to track an optical beacon on the target vehicle. A typical functional diagram of the equipment in the active vehicle is shown on Figure A1-7. This diagram shows gimballed star trackers which provide a relatively large field of view or angular coverage using relatively few sensors. The gimballed trackers have the disadvantages of higher weight, greater power requirements, and electromechanical complexity.

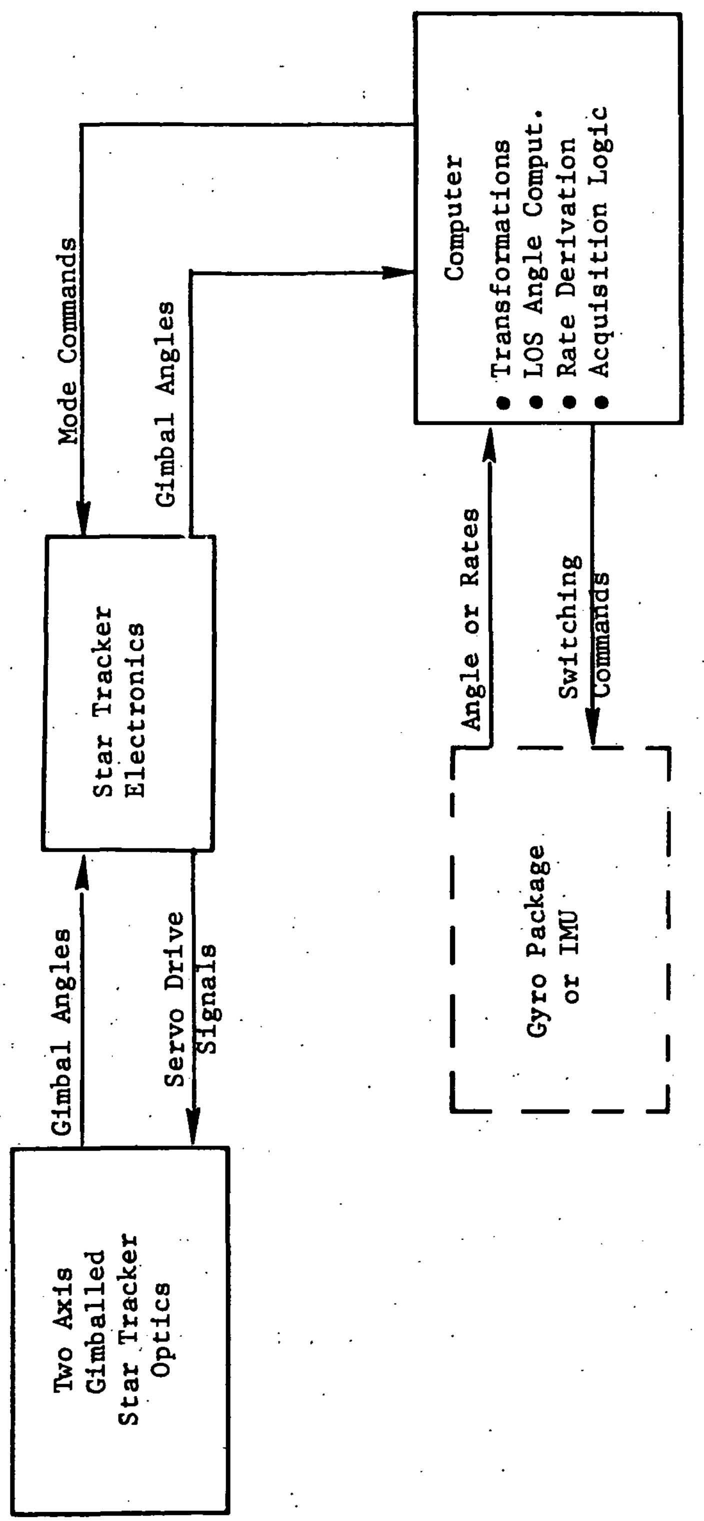

Figure A1-7. Functional Diagram - Star Tracker LOS Tracking

Optical tracking concepts in general can be hampered by operational limitations regarding the proximity of the sun to LOS, earth reflected sunlight, and reflected light from the spacecraft. Use of silicon detectors as opposed to photomultipliers can minimize these restrictions. The primary advantage of the star tracker concept is the potential commonality of the sensor for other functions.

Gimballed optical trackers are considered well within the current state-of-the-art.

2. Multifunction Utilization

See discussion under star-horizon state vector update.

3. Multimission Utilization

See discussion under star-horizon state vector update.

Scanning Laser Radar

1. Concept Description

A Scanning Laser Radar system is presently under development by the ITT Aerospace/Optical Division in San Fernando, California. Testing has been performed and the concept is considered feasible and practical. The data presented herein is derived from an ITT, NASA contract final report (reference DS-531).

The system utilizes an active Laser scanner as pictured in Figure A1-8. The major components are the laser transmitter, beamsteerer, receiver optics and scanning optical detector.

Four optical corner cube reflectors are uniquely spaced about the target vehicle docking boresight. The laser beam sequentially interrogates each reflector, obtaining range and angle (with respect to the boresight axis of the laser on the chaser vehicle). The corner reflectors, only a few inches in size, give almost a 100-to-1 gain in signal returned to the receiver over ordinary surface reflection (Lambertian).

Separate range and angle readings to each target reflector are made and the incoming beam angle of the laser radar is determined by the chaser vehicle with the radar transmitter-receiver. Using the known spacing distance of the reflector array with their ranges and angles, the chaser vehicle computes cross-axis distances and axial range. The unique (nonequidistant) spacing of the reflectors also provides relative roll index data.

A solid-state gallium arsenide diode is used as the laser transmitter because it maximizes life and reliability and requires minimum power. The beamsteerer contains a piezoelectric beam deflector to deflect the beam in two axes from the transmitter so that the narrow laser beam (<0.1 degree) can be pointed or scanned anywhere in the 30- by 30-degree

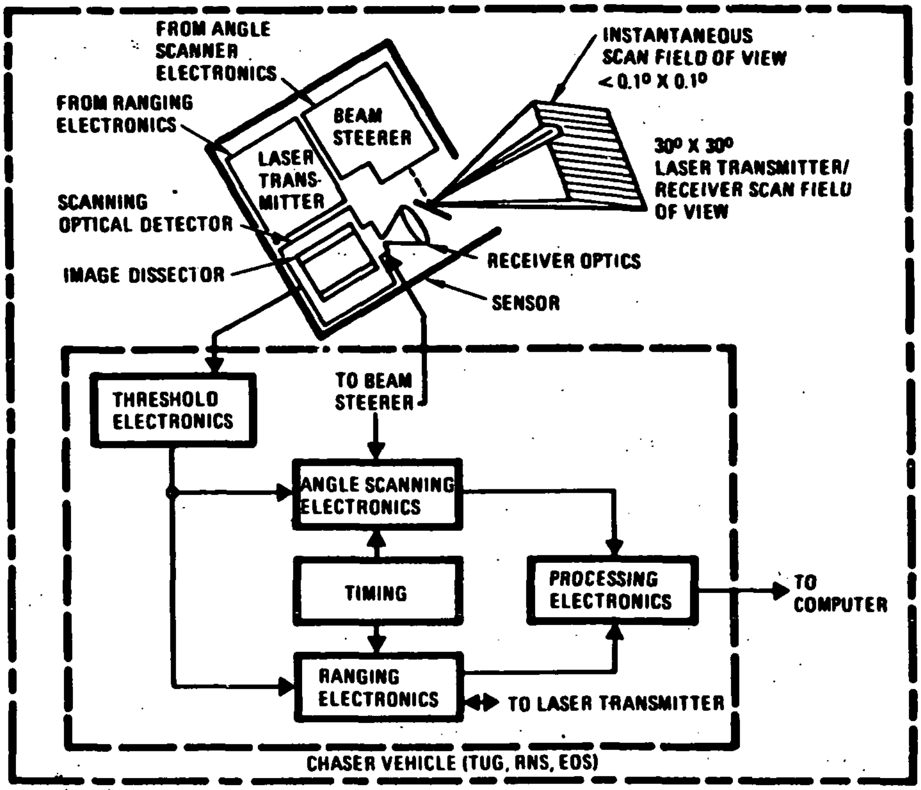

Figure A1-8. SLR Rendezvous System for Chaser Vehicle

field of view. When a control voltage from the angle scanner electronics is applied across the piezoelectric crystal, it bends proportional to the applied voltage and is deflected by the attached mirror. A special passive optical system will amplify the deflection to ± 15 degrees. Moving parts are therefore eliminated while retaining a 30- by 30-degree scanning capability for acquisition.

The receiver consists of a narrow-band optical filter and a multi-element lens assembly. The optical filter allows only the radiation centered around the laser wavelength to pass through to the scanning optical detector. A multi-element lens collects the energy from the returning laser radar signal and forcuses the photons of laser light to a small spot on the screen (photocathode of image dissector).

Electrons from the small spot on the image dissector are than emitted into an aperture and go to an electron-multiplier and then to the threshold electronics. By varying the electromagnetic field, the image dissector can effectively scan the surface of the photocathode and determine where the spot is located. The location of this spot is directly proportional to the pitch and yaw angle of the chaser.

The pulse ranging electronics determine range by measuring the laser pulse propagation time from transmitter to the target and back to the receiver. The threshold electronics determines the presence or absence of a receiver laser signal.

The data outputs from the chaser electronics are:

R = range

$\overline{R}$ = range-rate

θp = chaser pitch L.O.S. angle

θy = chaser yaw L.O.S. angle

$\dot{\theta}p$ = chaser pitch L.O.S. rate

$\dot{\theta}y$ = chaser yaw L.O.S. rate

ϕ = chaser-target relative roll index

L.O.S. = line of sight

A typical operation of the system would proceed after target acquisition at approximately 75 n miles.

Once the cooperative, passive, corner cube reflector targets have been acquired, the scanning laser beam of the chaser will interrogate a reflector and obtain range and angle information (chaser's pitch and yaw with respect to the laser beam boresight axis) as shown in Figure A1-9. The chaser's laser radar electronics output data updates the chaser vehicle's guidance and control computer which commands the necessary flight maneuvers.

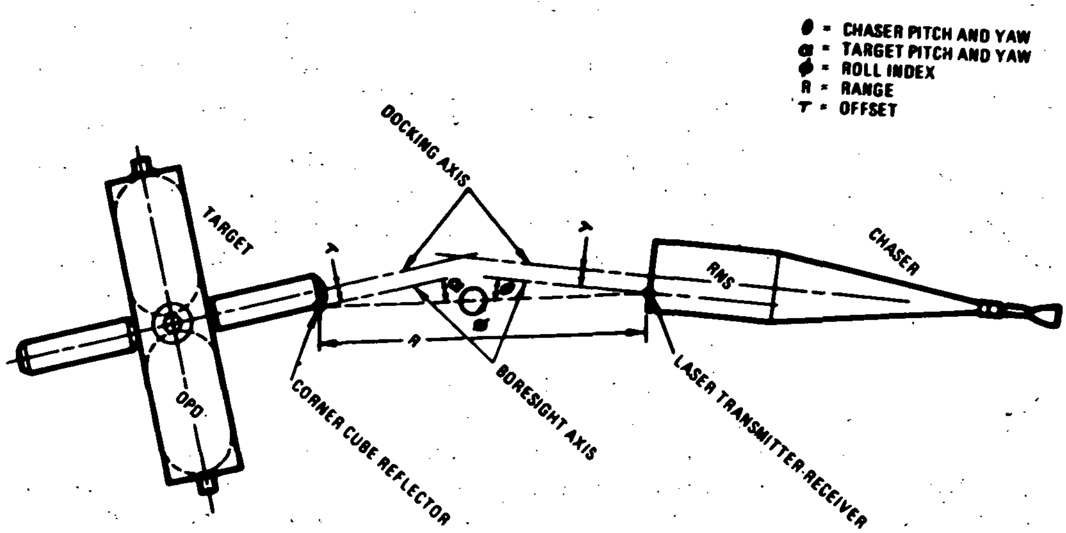

Figure A1-9. Rendezvous Technique

At a range of 1000 feet, pitch, yaw, and roll index angle information will be available to the chaser computer for correction of the docking vehicle's orientation to permit boresight alignment. If the offsets (cross distance between laser boresight axis and docking axis of each vehicle) are not the same, further orientation corrections will be made as the vehicles close the range.

The target pitch and yaw angle is calculated indirectly by the chaser's guidance and control computer. The chaser's guidance and control system can then compute and command the necessary flight maneuvers.

From a technology standpoint the SLR concept is in the development stage. ITT has worked the problem extensively including field tests of prototype equipment.

2. Multifunction Considerations

The device in addition to providing LOS angle tracking provides a direct measure or range, range rate, and roll index. Thus it provides all the guidance measurements necessary to perform rendezvous, stationkeeping, and even automatic docking.

The potential exists to combine a communication function with the radar function. Additional electronics for pulse modulation and demodulation would be required at the transmitter and receiver respectively. With a pulse rate of 1 KHz for the radar function, PCM communication rates of 7000 bits/sec are reasonable.

A further possibility is to use the optical detector section of the receiver as a star mapper or tracker. This portion of the unit is an image disector similar to that used in strapdown star trackers. Using the optics for this purpose, the device could perhaps perform the sensor function of the star pair attitude reference concept discussed in Section III. It is estimated that the angle accuracy on any +3 magnitude star (or brighter) would be o.03° RMS. Note that this device provides relatively wide FOV (30° x 30°) although not the 40° x 40° discussed for the star pair concept.

3. Multimission Considerations

The SLR concept is suitable for any cooperative rendezvous problem in either earth orbit or lunar orbit.

Used as a docking sensor, the device would facilitate shuttle docking or berthing operations. The primary advantage in using a precise docking sensor is that the closing velocity at contact can be accurately controlled. This minimizes the energy attenuation requirements and should facilitate the use of a common docking system over a wide range of vehicles.

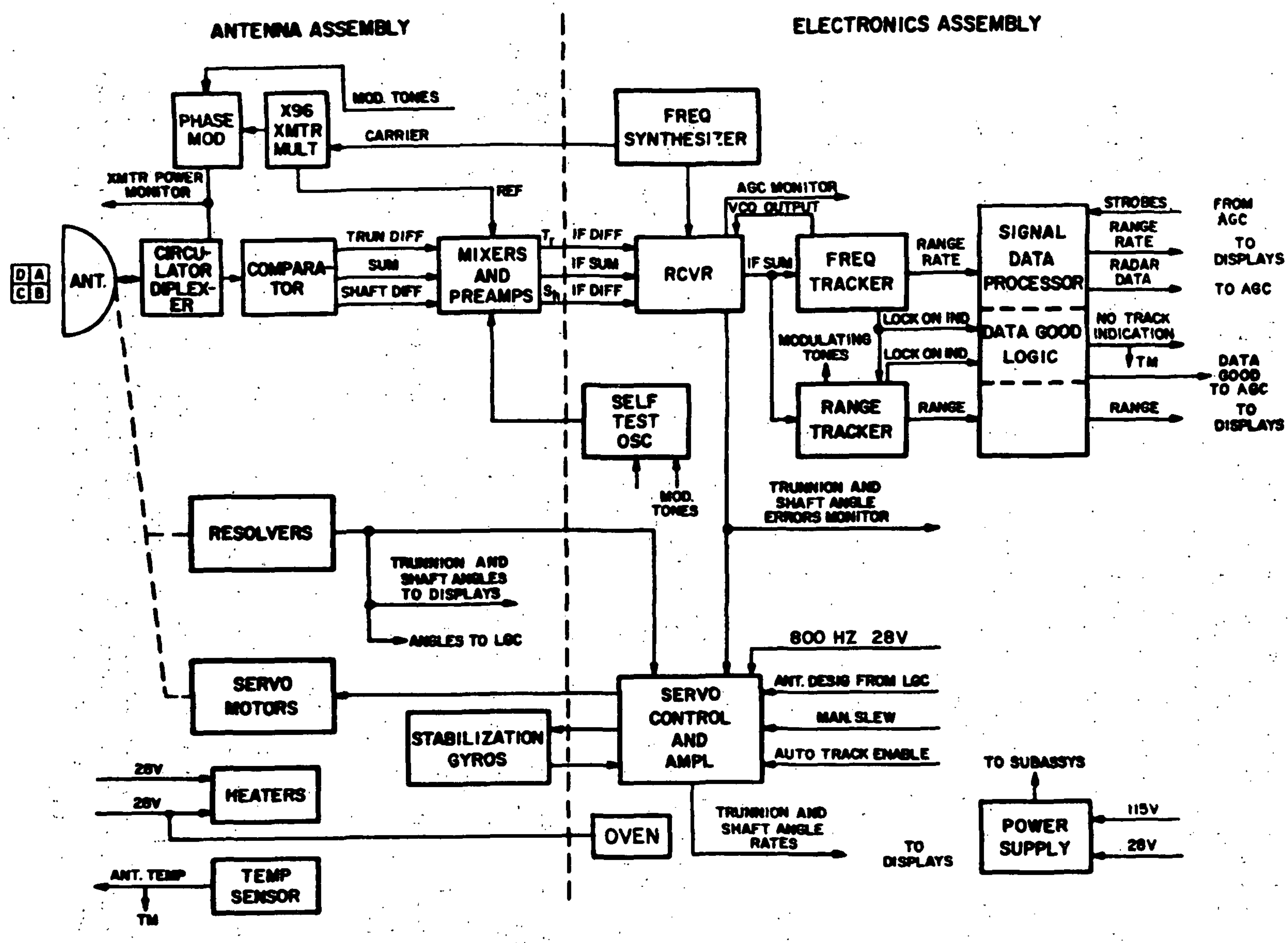

Figure A1-10. Rendezvous Radar Block Diagram

An SLR system would enable a vehicle such as the Modular Space Station to perform the active role in docking operations as may be desirable, for example, to rescue a disabled RAM. The SLR provides the data necessary for a pilot to execute control commands. The problem with using standard visual cues (out-the-window) is that the acceleration levels are not large enough to adequately correct errors once they become large enough to be detected visually. The SLR provides adequate lead for the necessary corrections to be made even with low acceleration levels. MSC man in the loop simulation studies have demonstrated this capability.

Gimballed MW Radar

1. Concept Description

A functional diagram of a typical gimballed microwave radar is shown on Figure A1-10. The radar provides a direct measure of line-of-sight angles and rates. The radar furthermore measures range and range rate.

The concept illustrated in the diagram is the CW rendezvous radar produced by RCA. This radar is an X-band, tone modulated CW, monopulse amplitude comparison type that operates in conjunction with a coherent cooperative transponder located in the target vehicle.

In operation, the radar signal is received by the transponder, where it is coherently shifted in frequency by a ratio of 240/241. The translated carrier is remodulated with the received tones and is retransmitted to the active vehicle where comparison of the relative CW radio frequency (RF) carrier energy received by the radar four-horn feed provides for measurement of the LOS angles. The range from radar to transponder is determined by measuring the phase difference of the transmitted and received tones. LOS relative velocity (range rate) is extracted from the two-way doppler shift between transmitted and received carrier frequencies.

The antenna assembly for the radar includes a 24-inch parabolic Cassegrain reflector, a four-horn X-band feed, shaft axis and trunion axis servo motors, and gimbal elements. In addition, there are internally mounted stabilization gyros, resolvers, a transmitter frequency multiplier chain, modulator, mixer preamplifiers, heaters, and temperature sensor.

The radar can be operated in three modes. A manual mode permits a crewman to slew the dish using the associated controls and displays. A second mode allows the computer to automatically position the radar. The last mode is a closed loop automatic tracking mode using the radar electronics.

The radar is designed to acquire and track the associated transponder at ranges up to 400 nautical miles and as close as 80 feet. Range rate is determined with a tolerance of $\pm$ 1 FPS over a range of $\pm$ 4900 FPS.

Power requirements are relatively high compared with other concepts. DC power depending upon the antenna motor activity ranges from a minimum of 170 watts to a maximum of 280 watts. Heater power requires an additional 111 watts (peak). AC power requirements are less than 25 watts (peak).

This concept has been demonstrated on Apollo missions.

2. Multifunction Considerations

The gimballed radar can be used for stationkeeping as well as rendezvous. However, the stationkeeping mode must be designed accounting for the radar bias and random errors. Comparing the MW radar with the laser for short range stationkeeping -- the superior performance of the laser would make it preferable.

The radar would be suitable for use in tracking free-flying RAM's from a vehicle such as Space Station. The radar could provide all the measurements necessary for the station to monitor the position of the RAM and perform guidance computation as required for repositioning.

3. Multimission Considerations

The microwave radar concept is a suitable rendezvous sensor for either earth or lunar orbit missions. The device is suitable for stationkeeping but will not permit automatic docking.

Application of the concept requires a cooperative transponder on the target vehicle. Multiple transponders may be required to provide spherical coverage.

The concept is applicable to any rendezvous mission considered within this study.

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4. IL 192-100-CWR-71-437, Final Study Report - Autonomous Navigation Techniques for Space Station, November 16, 1971, M. F. Madden

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A2 EOS PAYLOAD DEPLOYMENT/RETRACTION AND STOWAGE COMMONALITY

Three distinct types of commonality analyses will be reported in this report. The first commonality is between EOS payload deployment and EOS payload retraction and stowage. The majority of EOS flights will perform both deployment and retraction on the same mission. A second commonality analysis was also conducted for both the manipulator and the pivot mechanism approaches. Some operational advantages were uncovered for each approach. The third commonality analysis to be reported is the applicability of the procedure(s) to the specific payload elements identified in the matrices. The combined deployment/retraction and stowage procedure developed as part of the first commonality analysis was used for this task.

EOS PAYLOAD DEPLOYMENT/RETRACTION AND STOWAGE COMMONALITY

Both scheduled and unscheduled activities were investigated for examples where the deployment and retraction of EOS payload is required as part of the same mission. The following briefly shows some of the substantiating examples that drove the combination of the two interfacing activity procedures:

<u>Scheduled Activities</u>

a. Module Exchange
   <u>Examples</u> (1) MSS cargo module exchange
   (2) MSS power module exchange
   (3) OPD propellant module exchange

b. Element Replacement
   <u>Examples</u> (1) TDRS replacement
   (2) Space based tug replacement

c. Routine Service

   <u>Examples</u> (1) Detached RAM resupply
   (2) Tug resupply

d. Multiple Elements

   <u>Example</u> (1) Deploy MSS module and retrieve a satellite

<u>Unscheduled Activities</u>

a. Rescue

   <u>Examples</u> (1) EOS rescue
   (2) MSS rescue

b. Maintenance

Examples (1) Detached RAM
(2) Space tug

The examples identified create the need for a common deployment and retraction procedure. However, there are many additional examples where only a deployment or a retraction activity is required. Therefore, a common procedure must include the following five separate operational options:

1. Deploy payload only
2. Retract and stow payload only
3. Deploy one payload, then retract and stow a second payload
4. Retract and stow one payload, then deploy a second payload
5. Retract a payload (service) then redeploy the same payload

All options except (4) can be performed with either a manipulator or a pivot mechanism. Option (4) would not be practical with the current design concepts of the pivot mechanism. Options 3 and 4 cover two separate payloads while option 5 is for a single payload.

Figure A2-1 shows the procedural steps for each of the operational options. A procedure flow diagram was prepared for both the manipulator approach and the pivot mechanism approach.

Manipulator Approach (refer to Figure A2-1)

The first three steps are basic to either a deployment or retraction activity. At step 4, either a deployment path (4D) or a retraction path can be followed (4R). Steps A1 through A6 represent a side path for either deployment or retraction where a payload adapter is required and the payload is to be placed at the external airlock. Steps 5D through 8D cover the preparation of the payload for removal from the cargo bay. Steps 5R through 8R cover the operations associated with attaching the manipulator to the payload. Steps B1 through B7 include the operations required during service at an external airlock for the following: (1) Prior to initial deployment, (2) Prior to redeployment, or (3) Prior to retraction and stowage in the cargo bay. If no service is required at the external airlock, a direct path can be taken from step 8 to step 9. Steps 9D through 12D cover the operations for both the deployment and the redeployment options. A recycle path from step 12D to step 4D is provided to include the deployment of subsequent payloads. Steps 9R through 12R covers the positioning of a retrieval in the cargo bay. A recycle path back to step 4R is provided for the retraction of subsequent payloads. Steps A7 through A12 covers the operations with the removal of the payload adapter from the external airlock and the stowage in the cargo bay. These steps are companion to A1 through A6 and only exist if an adapter is required. The remaining two steps (13 and 14) are the reverse of steps 1 and 3, and cover the manipulator stowage and deactivation which is common to either deployment or retraction.

Figure A2-1. Operational Procedure Flow Diagram

Figure A2-2 illustrates the five major operational options represented by the basic procedure. Each of the numbered circles corresponds to an operational block of the procedure. This figure illustrates how the procedure can be used for each option that the various payloads may require. The options 1 through 5 will be used in subsequent matrices to show applicability to the EOS payloads as a result of the third phase of the commonality analyses.

Pivot Mechanism Approach (refer to Figure A2-1)

This procedure is structured in a format similar to the manipulator approch previously discussed. The operational blocks with the letter "D" are for deployment with the "R" designator for retraction and stowage. Steps A1, A2, and A3 are provided for the case where manned ingress is required as part of the service prior to either deployment or retraction with the payload pivoted out of the cargo bay.

Operational Option number 4, retract and stow one payload, then deploy a second payload, is not possible with the current pivot mechanism and cargo bay configuration. The remaining four options are applicable as follows:

Option 1 - Deploy payload only

Sequence 1, 2D, 3D, 4D, 8D, A1, A2, A3a, 9D, 10D, 11D

Option 2 - Retract and stow payload only

Sequence 1, 2R, 3R, 4R, 5R, 6R, 7R, 8R, A1, A2, A3b, 9R, 10R

Option 3 - Deploy one payload, then retract and stow a second payload

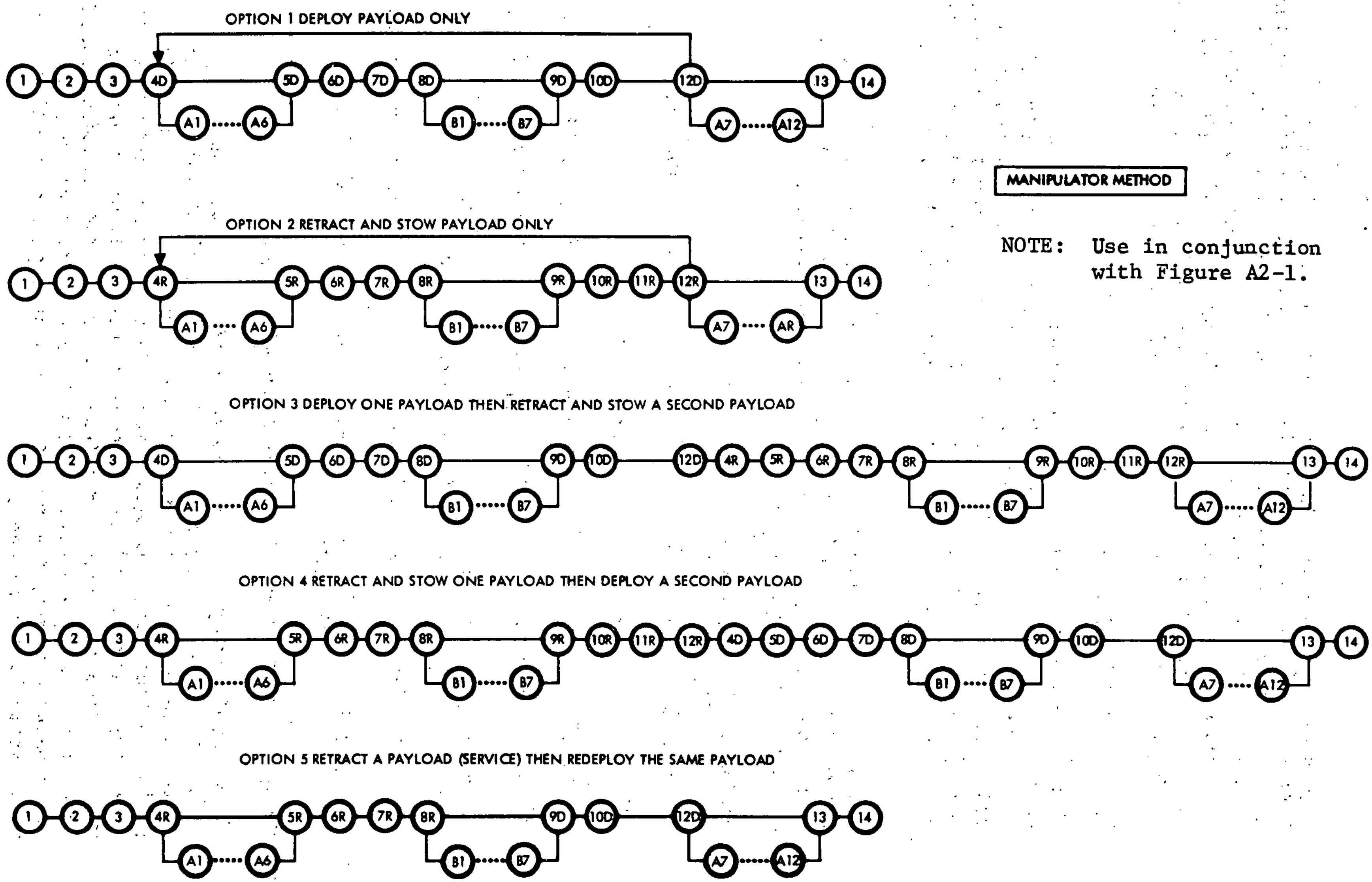
Sequence 1, 2D, 3D, 4D, 8D, A1, A2, A3a, 9D, 10D, 2R, 3R, 4R, 5R, 6R, 7R, 8R, A1, A2, A3b, 9R, 10R

Option 5 - Retract a payload (service), then redeploy same payload

Sequence 1, 2R, 3R, 4R, 5R, 6R, 7R, 8R, A1, A2, A3a, 9D, 10D, 11D

B. Manipulator and Pivot Mechanism Approach Commonality

The manipulator approach has an apparent operational advantage associated with the deployment and/or retrieval of multiple payload on the same mission. No current requirement for this multiple payload option has been identified. This study is constructed to considered pairs of orbital elements; therefore, no penetration was conducted to derive a requirement. The pivot mechanism could be augmented with a special device to permit multiple payload deployment; however, the retraction and stowage case would be the most difficult design case.



MANIPULATOR METHOD

NOTE: Use in conjunction with Figure A2-1.

Figure A2-2. EOS Payload Deployment--Retraction and Storage Operational Options

The pivot mechanism has an apparent operational advantage for the case where crew ingress and/or checkout of a payload must be conducted with the payload out of the cargo bay. The interface remains intact throughout the transfer of the payload from the cargo bay to the 90-degree deployment position. The manipulator approach requires a complete separation and reconfiguration of the interface at an external airlock.

The pivot mechanism does not readily permit the operational option of retracting and stowing one payload prior to deployment of a second payload during the same mission. This apparent operational disadvantage may not be significant as no known requirement for this option has been uncovered during this study.

The following commonality exists between selected operational steps of the manipulator and pivot mechanism procedures:

Table A2-1. Operational Commonality

Common Function (Refer to Figure A2-1)	Manipulator Operation	Pivot Mech. Operation
Illuminate cargo bay	2	1
Transmit payload conditioning commands	5R	3R
Verify payload retrieval condition	6R	4R
Prepare for crew ingress	B4	A1
Ingress and perform service	B5	A2
Condition payload for deployment	B6a	A3a
Condition payload for earth return	B6b	A3b
Verify payload operational status	12D	10D

Table A2-1 shows that only a small degree of commonality exists between the two procedural methods at the detail function level.

C. Operational Procedure Applicability

The five separate operational options combined with both manipulator and pivot mechanism approaches combine for a total of 10 possible procedure alternates to the payload elements. The matrix presented in Figure A2-3 has two distinct parts with the abscissas displaying manipulator applicability and the ordinate displaying the pivot mechanism applicability. A summation of both ordinate and abscissas for identical coded blocks will indicate how many of the procedure alternates are possible for each payload element. The returnable tug for example has all five manipulator procedure possibilities and four pivot mechanism procedure possibilities for a total of nine.

		SPACE PROGRAM ELEMENTS																
		(PIVOT MECH) EOS	TUG		RAM				SATELLITE			EO RESUP. MODS	LOW EO MSS ***	CPS		RNS CLS ***	OLS ***	OPD ***
			RTN	SPACE BASED	ATT. EOS *	DET. EOS	ATT. MSS	DET. MSS	EOS DELIV.	EOS + 3RD ST DELIV.	RETR, RESUP.			OIS	CLS			
	EOS (MANIPULATOR)	NA	1,2,3, 4,5	1,2,3, 4,5	1,2,5	1,2,5	1,2,3,4	1,2,5	1,2,3 4,5	1	1,2,5	1,2,3, 4,5	1,2		NA **	1,2	1,2	1,2
TUG	RETURNABLE	1,2,3,5								NA								
	SPACE BASED	1,2,3,5		NA			NA	NA		NA	NA	NA	NA	NA	NA	NA	NA	NA
RAM	ATT. EOS *	1,2,5																
	DET. EOS	1,2,5																
	ATT. MSS	1,2,3		NA									NA					
	DET. MSS	1,2,5		NA									NA					
SATEL	EOS DELIV.	1,2,3,5																
	EOS + 3RD ST	1	NA	NA														
	RETR, RESUP.	1,2,3,5		NA														
	EO RESUP. MODS	1,2,3,5		NA								NA	NA		NA	NA		NA
	LOW EO MSS	1,2		NA			NA	NA				NA	NA					
CPS	OIS			NA											NA	NA	NA	NA
	CLS	NA **		NA								NA		NA	NA		NA	NA
	RNS-CLS	1,2		NA								NA		NA		NA	NA	NA
	OLS	1,2		NA										NA	NA	NA	NA	
	OPD	1,2		NA								NA		NA	NA	NA		NA

LEGEND

Operational Options:

1 Deploy payload only
2 Retract and stow payload only
3 Deploy one payload, then retract and stow a second payload
4 Retract and stow one payload, then deploy a second payload
5 Retract a payload (service), then redeploy same payload
NA Not Applicable

NOTES:

* Refer to Attached Element Operations (2.12)
** 33-ft dia not compatible with EOS cargo bay
*** Refer to Orbital Assembly (2.2) for all modules except initial

Figure A2-3. Operational Procedure Applicability

A3. COMM. LINKS, MODULATION/DEMODULATION AND MICROWAVE PREAMPS.

INTRODUCTION AND SUMMARY

Three alternate approaches were considered for commonality of system requirements. Since each of these approaches - element to element, element to ground and element to TDRS - will be implemented during mission operations, commonality of equipment to provide for these approaches is a goal. The following discussion indicates the choice of frequency bands to satisfy the ground and TDRS requirements - i.e., S-band, VHF and Ku-band. These frequency bands are usable for element-to-element links (refer to Tables A3-1 and A3-2). A discussion of digital modulation techniques indicates that PSK/PM is the most efficient technique. It also provides compatibility with the ground network.

Operation at Ku-band requires transmitter power levels in the order of 25 watts RF and receiver system noise temperatures approaching 1000°K. A summary of a technology study follows indicating that present state-of-the-art equipment can meet these requirements. Tunnel diode amplifiers (TDA) can be used for most receiver front ends where $\simeq$ 1200°K noise temperature is needed. Traveling wave tubes (TWT) will satisfy the power output requirements of 25 watts.

Table A3-1. Proposed Alternatives

ALTERNATE APPROACHES	ORBITAL ELEMENTS								
	EOS	MSS	TUG	RAM	CPS	RNS	OPD	OLS	TDRS
RF LINKS									
• VHF	V,D	V,D	V,D	V,D,T/R		V,D	V	V,D	V,D
• UHF	V		V	V,D,T/R		V,D			
• S-BAND	ALL	ALL	ALL	ALL	ALL	ALL	V,D,TV	ALL	
• C-BAND			T/R			ALL			
• X-BAND						T/R			
• Ku-BAND		ALL	ALL	ALL		ALL			ALL
LASER BEAM									
• LASER SCANNING * RADAR	T/R	T/R	T/R	CC	T/R	T/R	T/R	T/R	
• LASER DATA XFER							D		

V - VOICE, T/R - TRACKING & RANGING, D- DATA TRANSFER, TV - TELEVISION, CC - CORNER CUBES
*TRACKING & RANGING FOR RANGES UP TO 75 NAUTICAL MILES

Table A3-2 Element-to-Element Design Concepts

CONCEPT	ADVANTAGES	DISADVANTAGES
UHF	OMNI ANTENNAS FOR ELM/GND INTERFACE	LO BW FOR DATA; HI USE; GALACTIC NOISE
VHF	MSFN/TDRS; COMPATIBILITY; XMIT TO GND VIA OMNI; HI REL.	LO BW FOR DATA; HI USE; GALACTIC NOISE; T/R - RNG & RNG RATE ONLY (AM ONLY)
S-BAND (2-4 GHz)	MSFN COMPATIBILITY; XMIT TO GND VIA OMNI, HI REL. PRN T/R	LIMITED BW FOR DATA; LG HI GAIN ANTN FOR T/R & DATA XFER; ANTN SIDELOBE PROB FOR T/R
C-BAND (4-6 GHz)	XMIT TO GND VIA OMNI; HI REL DUE TO MIL A/C USE MORE ACCURATE T/R THAN S-BAND	LIMITED BW FOR DATA; SAME ANTN REQMTS AS S-BAND; NOT COMPATIBLE WITH MSFN OR TDRS; HI AMATEUR, COMMERCIAL & MIL. USE
X-BAND (8-12 GHz)	LIGHTER EQUIP AND ANTENNAS; EXISTING TECH; MORE ACCURATE T/R THAN C-BAND	NOT COMPATIBLE WITH MSFN OR TDRS; ALLOCATED FOR RADIO NAV & ASTRONOMY USE
Ku-BAND (12-16 GHz)	TDRS COMPATIBILITY; SMALLER AND LIGHTER EQUIP & ANTENNAS; EXISTING TECH; HI BW	SIGNAL ACQUISITION DIFF; UNPROVEN REL; HI PROPAGATION LOSS
MMW (40 GHz & UP)	NO ALLOCATIONS; SM & LIGHT EQUIP & ANTENNAS	HI PROPAGATION LOSS; TECHNOLOGY DEVELOPMENT REQUIRED
LASER BEAM	HI BW CAPABILITY; HI ANTN GAINS; NO FREQ ALLOCATIONS; ACCURATE T/R; NEGL ANTN SIDELOBES	LO DETECTION RNG (<100 NM); HI PWR REQMT; TECH DEVELOPMENT REQUIRED; POINTING & ACQUISITION PROBLEMS

SIGNAL MODULATION/DEMODULATION

For digital signal transmission, signal modulation/demodulation schemes are a second driving function to the need for compatible communications techniques among orbital elements. Table A3-3 identifies the advantages and disadvantages of conventional AM, FM, and PM modulation techniques and their specialized digital nomenclature. Examination of this table provides a basis for determining the preferred technique to be used for this aspect of the communications activity. For the applications studies, where relatively high digital data rates, PRN ranging signals and the transmission of analog TV are to be used in various combinations, the digital data must be transmitted via a subcarrier to allow simultaneous operation with other signals. Amplitude modulation, as noted, is not feasible because of the low data rate capability and its lack of immunity to noise. Also, it should be noted that AM is not compatible with TDRS due to the amplitude limiting techniques employed. Although FM (FSK) provides noise immunity, it lacks the data rate capability and ease of simultaneous data transmission. As indicated, direct PM (PSK) on the carrier provides an efficient transmission technique and better noise immunity than FM (FSK). Figure A3-1 illustrates the improvement in noise immunity with PM (PSK) modulation. A bit error rate (BER) of 1×10^{-5}, for instance, shows the need for signal-to-noise ratio [energy per bit (E_b) to noise spectral density (N_o)] of:

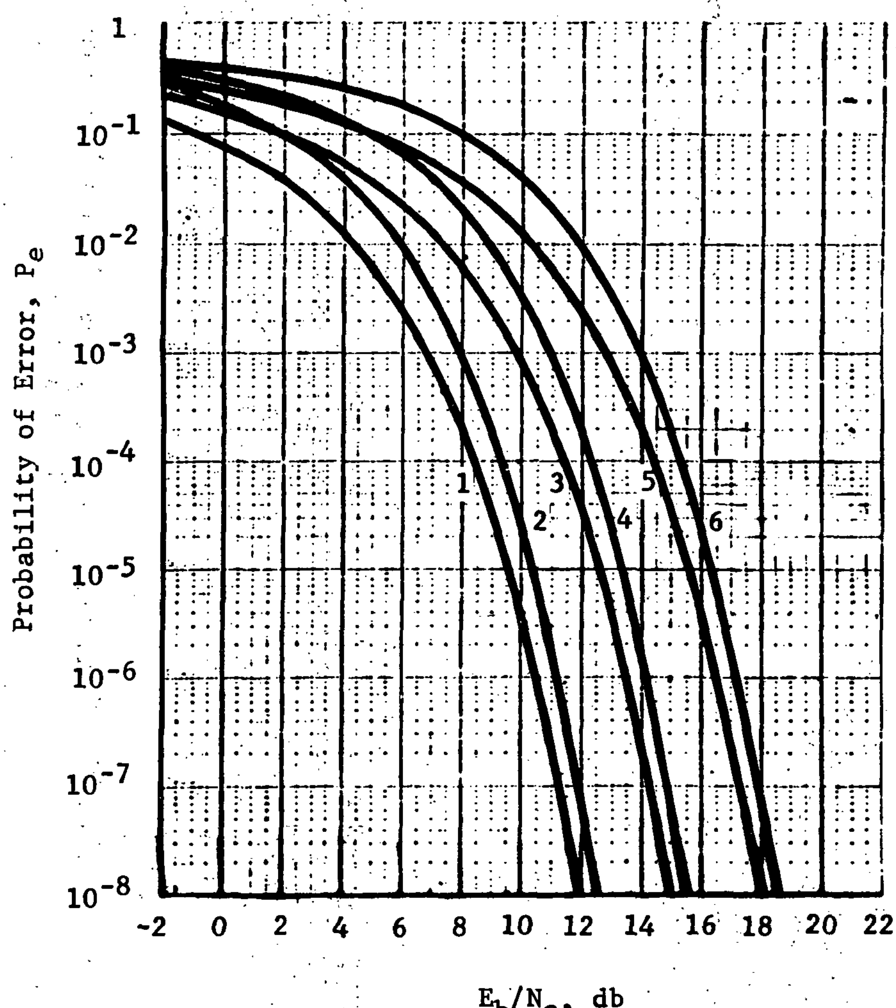
Coherent PM (PSK) = 9.6 dB
Coherent FM (FSK) = 12.6 dB
Coherent AM (ASK) = 15.6 dB

Coherent PM (PSK), therefore, has 3 dB (1/2 power) improvement over FM (FSK) and 6 dB (1/4 power) over AM. For very high data rates ($\simeq$50 Mbps), direct carrier PM (PSK) may be necessary. For the normal operations, however, where simultaneous signals must be transmitted and data rates up to 5 Mbps communicated, PCM/PSK/PM is the most efficient technique. This technique applies the PCM (digital) data on a subcarrier by PSK modulation (biphase) and then phase modulates (PM) the carrier with the modulated subcarrier.

These techniques are used in the ground network system where PRN ranging directly PM (PSK) modulates the carrier, and digital data is transmitted on a subcarrier of 1.25 MHz with PM (PSK) modulation. In summary, direct carrier PM (PSK) modulation may be necessary for very high rates ($\simeq$50 Mbps) and the PCM/PSK/PM technique for simultaneous operations and up to medium data rates. These are the only two techniques that will be considered further in this study.

Table A3-3. Signal Modulation Concepts

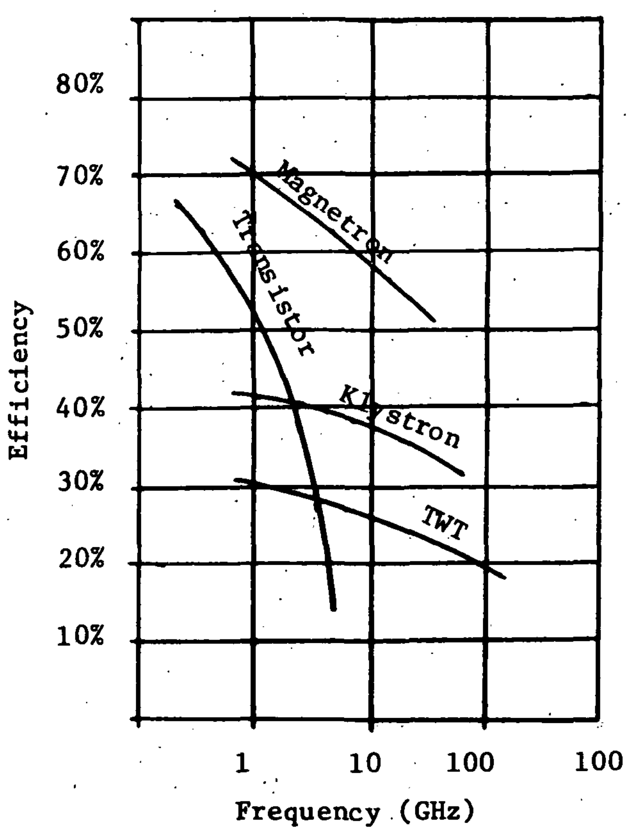
APPROACH	ADVANTAGES	DISADVANTAGES
AM (ASK)	INSENSITIVE TO FREQUENCY VARIATIONS, SIMPLE IMPLEMENTATION AND CONTROL CIRCUITRY, LOW EQUIPMENT COSTS	VERY LOW DATA CAPABILITY, NOISE SENSITIVE
FM (FSK)	2-10 TIMES IMPROVEMENT IN BIT ERROR RATE (NOISE PERFORMANCE) THAN AM, CONTINUOUS ENERGY TRANSMISSION	LOW DATA CAPABILITY, SENSITIVE TO FREQUENCY SHIFTS CAUSED BY DOPPLER, OSC. DRIFT, ETC.
PM (PSK)	SUPERIOR NOISE PERFORMANCE, HIGH DATA CAPABILITY, BINARY AND MULTILEVEL MODULATION FORMATS, 50% LESS POWER THAN ASK OR FSK SYSTEMS	MODIFIED Ø-LOCK LOOP TECHNIQUES, HIGHER EQUIPMENT COMPLEXITY
PCM/PSK/PM	SIMPLE Ø-LOCK LOOP, SQUARE-WAVE COHERENT WITH DATA RATE, SUBCARRIERS YIELD NO LOSS IN CARRIER TRACK, DETECTOR EFF. WITH ANY % CARRIER POWER	LESS EFF. THAN PSK ON CARRIER, APPROX. ONE-HALF POWER MUST REMAIN IN CARRIER TO ALLOW Ø-LOCK LOOP TRACKING WITH SINUSOIDAL SUBCARRIERS, SQUARE-WAVE SUBCARRIER REQUIRES TWICE XMITTED WAVEFORM BANDWIDTH
PCM/FSK/PM	SAME ADVANTAGES FOR CARRIER DETECTION AS PSK/PM	APPROX. TWICE XMITTER POWER REQ'D OVER PSK/PM



(E_b = Energy/bit
N_o = Noise Spectral Density)

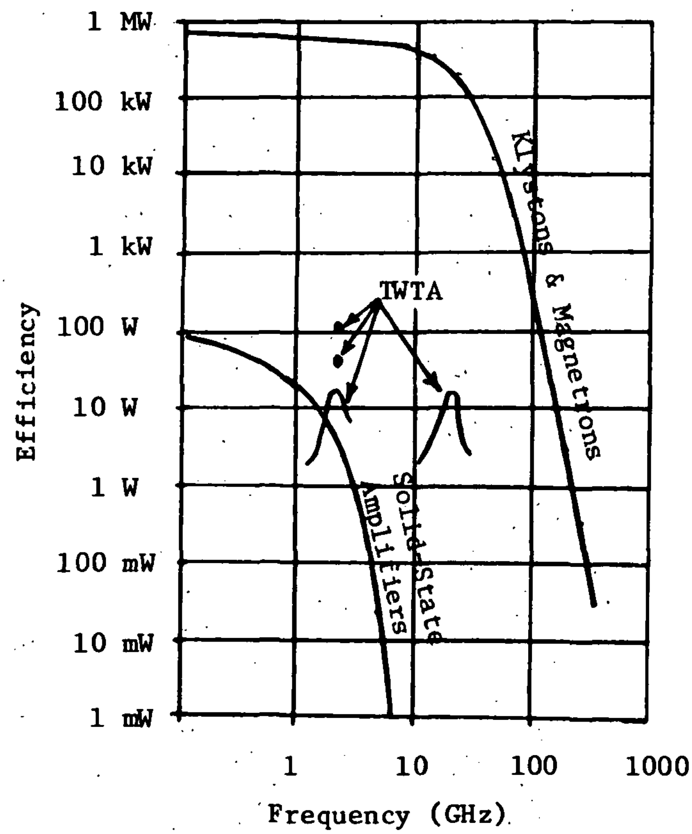
1. Coherent PSK
2. Differentially Coherent PSK (DPSK) & FM
3. Coherent FSK
4. Noncoherent FSK
5. Coherent On-Off Keying (ASK)
6. Noncoherent On-Off Keying

Figure A3-1. Improvement in Noise Immunity



Efficiency-versus-frequency characteristics for microwave tube and solid-state devices.

Figure A3-2



Summary of power-versus-frequency characteristics of microwave tubes and solid-state amplifiers.

Figure A3-3

MICROWAVE POWER AMPLIFIERS

Figures A3-2 and A3-3 summarize the results of the microwave power amplifier survey. Presented are the efficiency-versus-frequency and power-versus-frequency characteristics for microwave tubes and solid-state devices. As seen by the graphs, for low power amplifiers (less than 1 watt) through the L-band and S-band range, a transistor low-noise amplifier is the best choice. A solid-state amplifier will offer somewhat better RF performance with respect to linearity, efficiency, gain, and phase matching - as well as better life and reliability - than does its traveling-wave-tube amplifier (TWTA) counterpart. However, for medium power amplifiers (greater than 10 watts) the TWT is still the primary energy converter for frequencies above 2 gHz.

Currently available continuous wave spaceborne TWTA's typically generate 10 watts RF with a 45-db gain and 30 percent efficiency. Higher output power can be achieved but at the expense of decreased gain.

Bandwidths of 10 percent or more are attainable depending upon the gain-flatness specification. For example, at S-band carrier frequencies, TWT's make an RF bandwidth of greater than 200 MHz a possibility. The broadband nature of these devices also is a liability in that they require RF filters.

The TWT units themselves typically weigh 18 ounces and occupy a space of 2 by 2 by 10 inches. Each tube, however, must be accompanied by a power supply that delivers up to 5000 vdc. These units are typically 80 to 85 percent efficient, weigh 3 pounds, and occupy a space 4 by 6 by 12 inches. Their weight and size increase with the power level handled, the voltage required, and the number of telemetry points taken within the unit.

TWTA's have domonstrated excellent reliability in both actual space missions and laboratory life tests. Hughes Electron Dynamics Division reports that after more than 8 years, Syncom 2.5 watt L-band tubes are still operating. Their only degradation appears to be a 0.1-db drop in power level every 3 years.

The TWTA curves appearing in Figure A3-3 are the results of a survey of now available or prototype S-band and Ku-band units from Hughes, Watkins-Johnson, and Varian Associates. Of the S-band TWT's available, two tubes built by Watkins-Johnson for JPL should be singled out. The tubes have been designed for high output levels (50 and 100 watts) and high efficiencies (approximately 45 percent). Furthermore, their output levels can be adjusted over a 10-db range by simply programming the power supply voltages.

MICROWAVE PREAMPLIFIERS

A survey of recent literature concerning noise temperature performance of state-of-the-art microwave preamplifiers obtained the results delineated below. These data were used in support of the calculations of the functional requirements section.

Figure A3-4 shows the receiver system temperatures of receivers with a "worst case" tunnel diode preamplifier (TDA) and galactic background noise. Also illustrated is the system noise temperatures of 500°K for S-band which seem reasonable for tunnel diode preamplifiers, while uncooled paramps exhibit a 130°K system noise temperature.

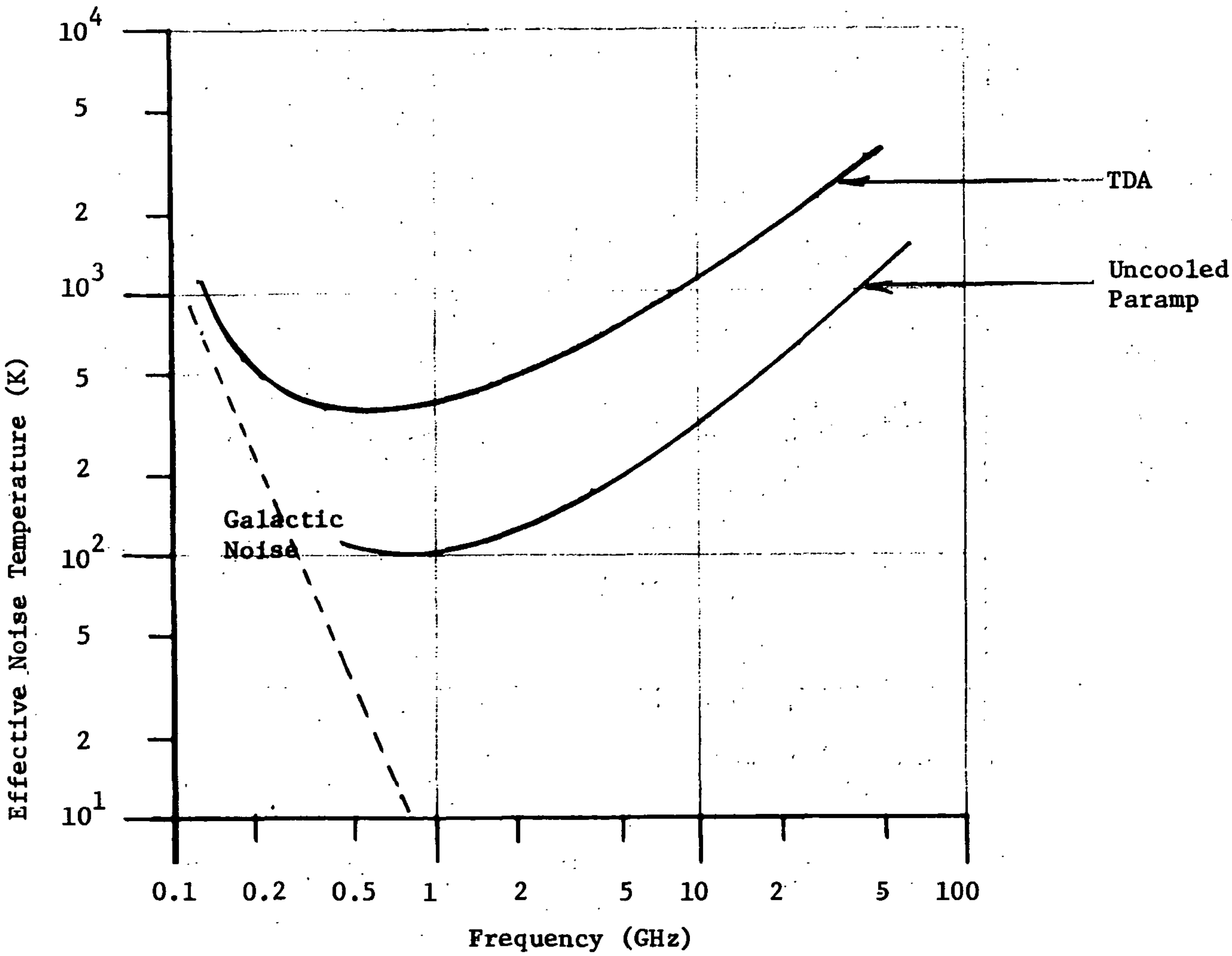


Figure A3-4. Effective System Noise Temperature with Galactic Background

Figure A3-5 shows the system temperature when a TDA or uncooled paramp is used and a 290°K earth is seen by the receive antenna main beam. At S-band TDA system temperature is 800°K while the uncooled paramp is approximately 400°K. System noise temperatures for S-band receivers not seeing the earth in the antenna main beam (i.e., element-to-element communications) would be somewhere between 130°K to 400°K for uncooled paramps and 500°K to 800°K for TDA's.

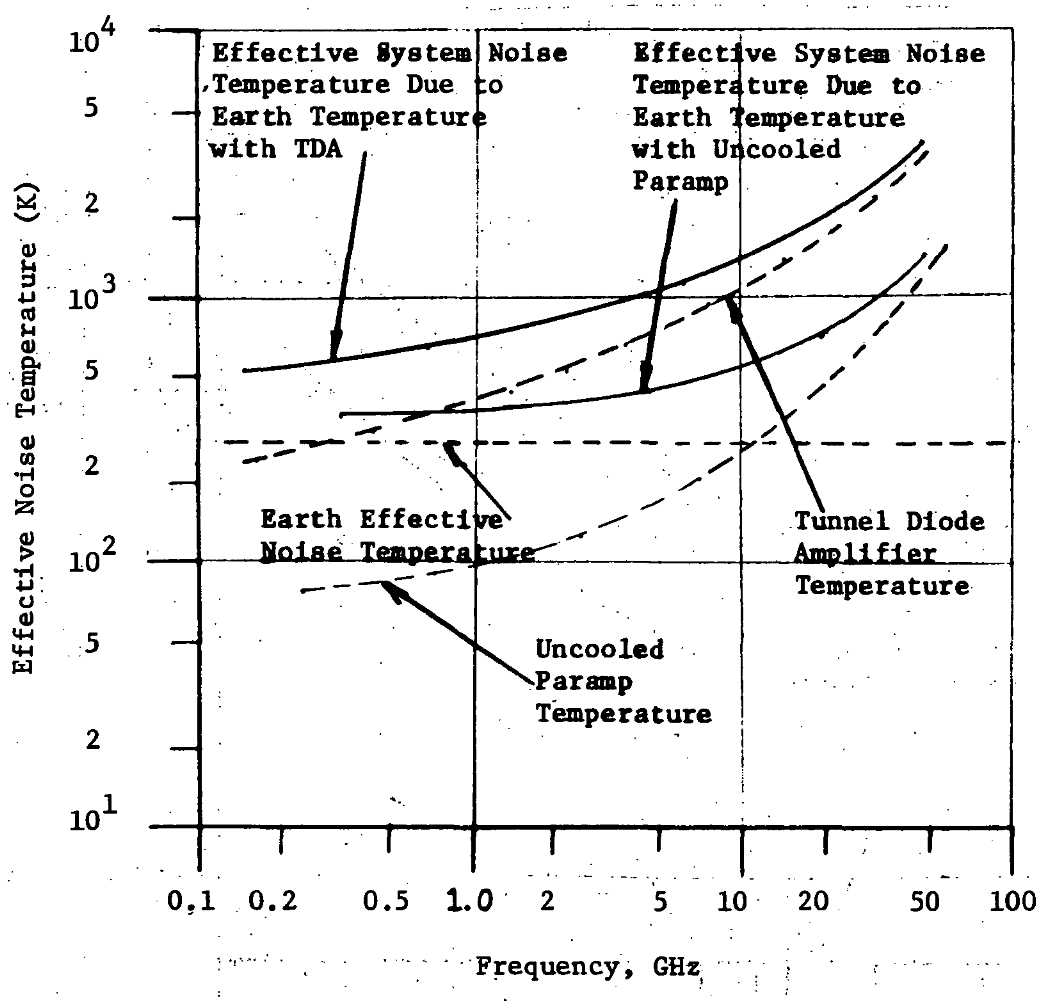


Figure A3-4. Effective System Noise Temperature with Earth Background

A4. JET PLUME IMPINGEMENT

INTRODUCTION

The effects of rocket engine exhaust plume impingement on a spacecraft may be conveniently divided into three general areas. These are:

1. Aerothermodynamics. This phenomenon relates to convective heat transfer and dynamic pressure generated forces experienced by a spacecraft surface due to plume impingement.

2. Contaminants. Contaminants in the form of unburned propellant, particulate matter, and compounds formed during pulse mode operation have been shown to degrade the performance of spacecraft functional surfaces such as solar cells, optical windows, and thermal protective coatings.

3. Electromagnetic Interference. This factor relates to the effect on electromagnetic radiation transmitted through an exhaust plume to a ground station or an adjacent spacecraft.

Only convective heat transfer can be attenuated by distance to an acceptable level. The other effects can only be spread over a larger area, but the net effect on sensitive hardware remains the same. Dynamic pressure, to some extent, defeats the desired action of the thruster--in the extreme case, impulse reversal may result. Such loss of impulse depends mainly on the percentage of the plume intercepted and the relative positions of the center of gravity, the thrusters, and the affected hardware (attached to the thrusting element only); it is relatively independent of distance.

These potential problem areas have been extensively investigated by both governmental agencies and industrial sources. The bulk of the work has been directed toward earth storable bipropellant combinations, but the increasing use of hydrazine thrusters in vehicles has resulted in the implementation of hydrazine-related exhaust plume analytical and experimental studies.

DISCUSSION

Aerothermodynamics

The key element in the analysis of exhaust plume phenomena is the ability to define plume boundaries and characteristics. Figures A4-1 and A4-2 are representative flow characteristics for an RCS monopropoellant (N_2H_4) engine firing at an altitude of 400K feet. The exhaust plume shown was produced by an engine developing 1000 pounds of thrust at a chamber pressure of 150 psia. The parameters presented in the figures are dimensionless, thereby permitting use of the information over a wide range of engine dimensions and element geometry. A bipropellant thruster, operating at the same chamber pressure,

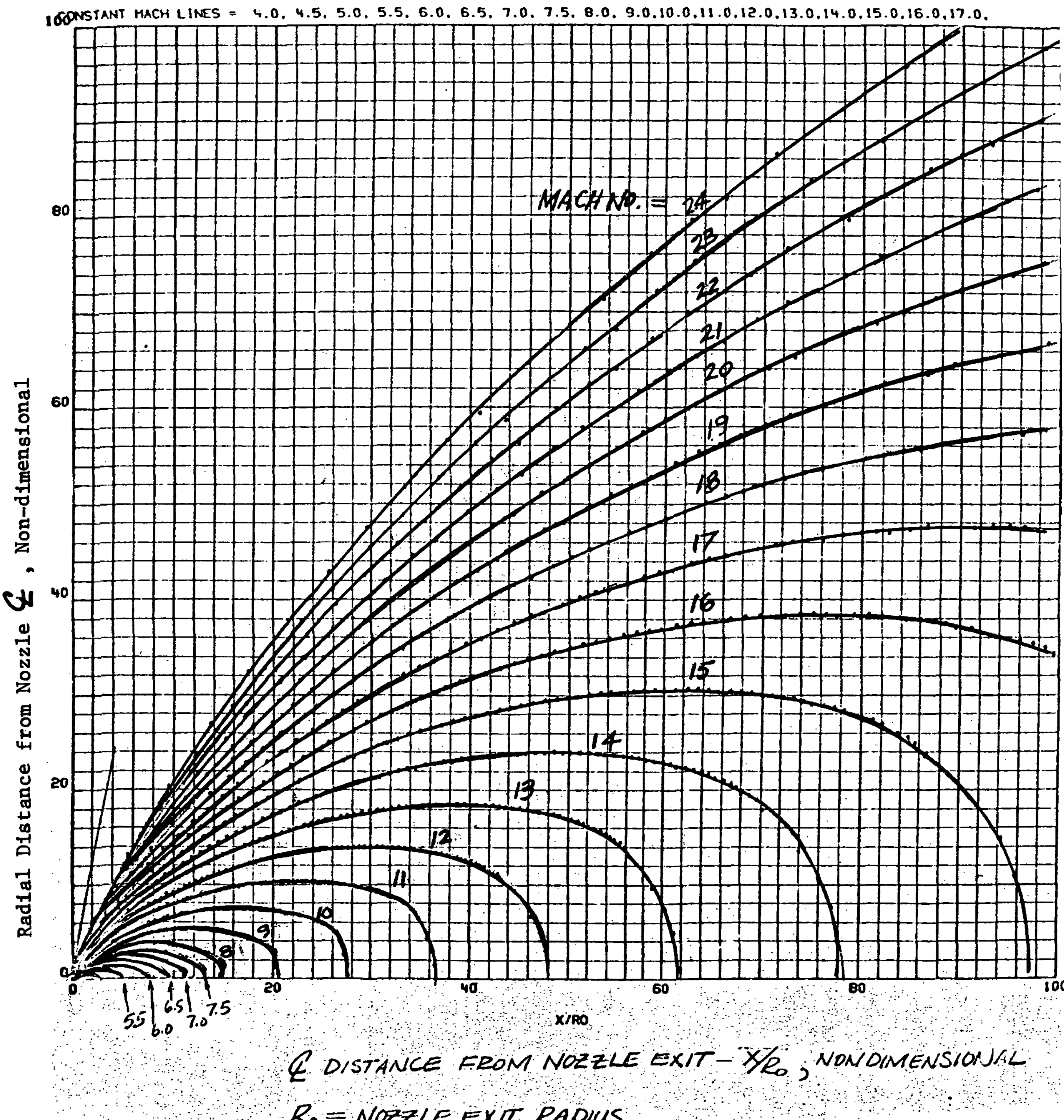


Figure A4-1. RCS Monopropellant (N_2H_4) Engine Plume Isomach Contours at Altitude = 400K Feet

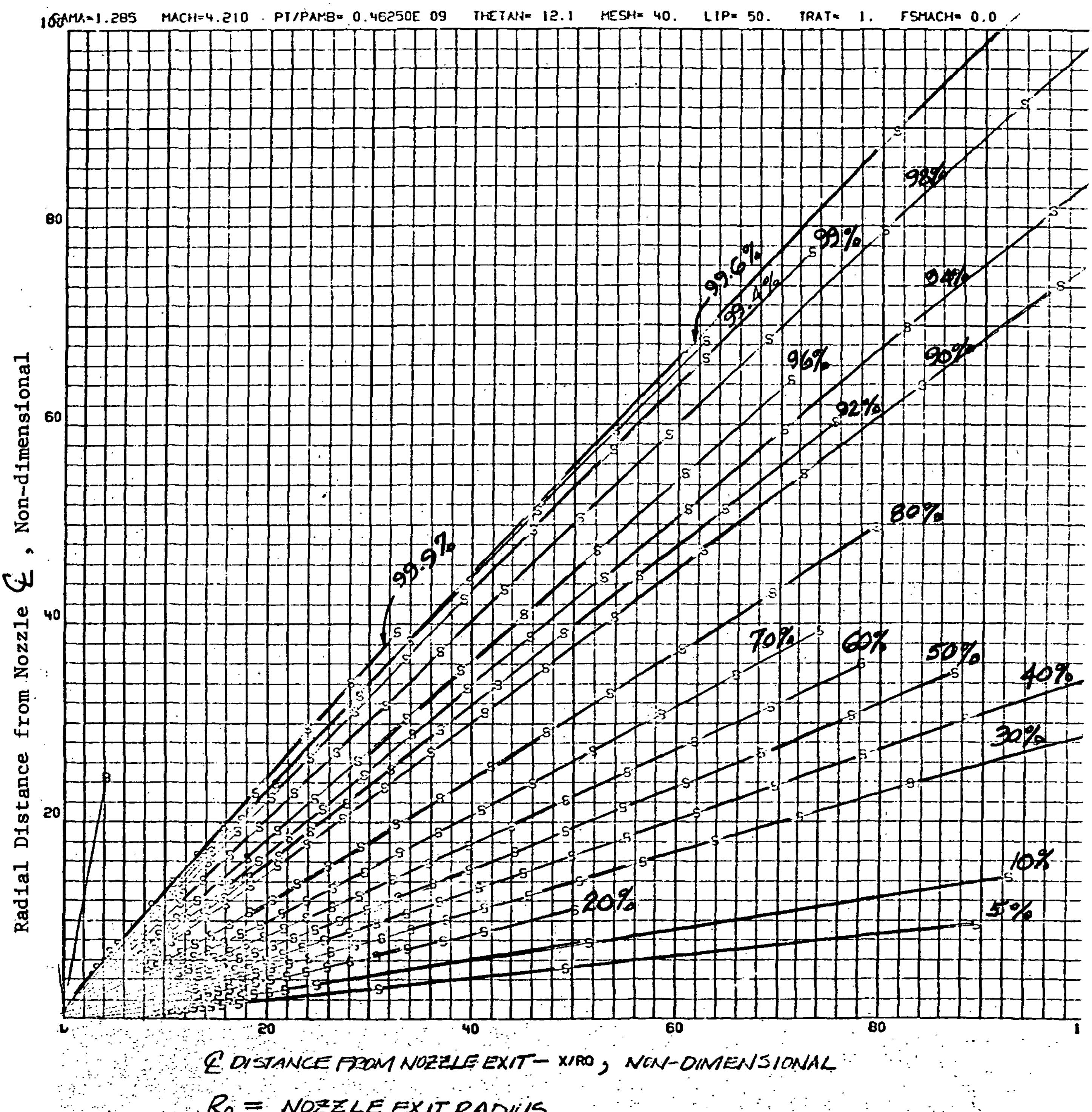


Figure A4-2. RCS Monopropellant (N_2H_4) Engine Plume Streamlines
(Percent of Total Mass Flow)
Altitude = 400K Feet

would produce approximately the same impingement pressure. The thermal effects, however, would be substantially greater due to the higher combustion chamber temperature of the bipropellant engine. The monopropellant thruster operates at approximately 2100°R, whereas the corresponding temperature of a bipropellant engine would be on the order to 5500°R.

Contaminants

Contamination can most effectively be reduced by controlling plume impingement and selecting low contaminant propellants. A bipropellant attitude control engine, using the monomethyl hydrazine-nitrogen tetrioxide propellant combination, generates a large number of chemically complex exhaust products. Pulse mode operation, utilizing minimum pulse widths on the order of 15 ms, introduces an additional fuel nitrate exhaust product. A comprehensive chemical analysis of the propellants also reveals the presence of "tramp" metals. This material is accumulated during the manufacture and storage of the propellants and is allowable under the terms of the propellant specifications.

Hydrogen peroxide and hydrogen/oxygen systems are generally quite good from a contamination standpoint except for the presence of water as a major constituent in the exhaust. The condensible species in the plume pose the potential problem of depositing on low temperature surfaces. Water, because of its relatively high freezing point, is prone to deposition and to condensation into prismatic droplets or crystals.

Hydrazine RCS will not seriously contribute to the contamination of the environment. The exhaust is typically composed of 20 mole percent ammonia, 30-percent nitrogen and 50-percent hydrogen with trace quantities of water and less than 0.1 percent of low-molecular-weight hydrocarbons, principally methane. Catalyst particle loss will introduce a trace amount of alumina/iridium particles in the size of 1 to 500 microns. The rate of introduction of these particles should be consistent with Class 100,000 requirements; but, at precise points within the plume, specifically within the divergent half-angle of the nozzle, this level may be exceeded during firing. To preclude impingement of particles on sensors, the nozzles should be canted to at least this half angle (~15 degrees) away from sensitive areas. Deposition of hydrazine plume products does not occur on materials in space above a temperature of 0°F, and detrimental plume contaminants are not formed with properly treated dechlorinated Shell 405 catalyst. Ammonia and traces of water in the plume condense on materials at temperatures below 0°F, but no residue remains after these volatiles evaporate. Thus, monopropellant hydrazine presents relatively few plume/contamination problems, and is considered the best propellant choice from this standpoint.

Electromagnetic Interference

The USAF, Navy, Army, and NASA have conducted extensive investigations into the effect of missile and booster exhaust plumes on radar attenuation and communications. A substantial portion of the effort expended during the program was devoted to evaluating the effect of transmitting electromagnetic radiation through an attitude control rocket exhaust plume (22-pound thrust).

An Apollo "C" band antenna and a specially designed and fabricated "C" band slot antenna were tested. The avionics experiments conducted and the results are presented in the following table.

Experiment	Results
Antenna breakdown	1700 watts required to achieve breakdown for no-flow condition; 126 watts during engine firing
Signal transmission	Minimal losses - engine mass flow too low
Plasma noise	No significant result
Antenna mismatch	No significant result
Antenna coupling	Some increase consistently observed + 4 db
Antenna depositions	No deterimental effects detected

The data indicate that exhaust plume interference should be minimal. Signal transmission is an area, however, which should be further examined because the mass flow from a vehicle such as the EOS may be on the order of 70 times greater than that of the engine used in the above evaluation.

SUSCEPTIBLE ELEMENT PAIRS

Although all elements are susceptible to jet plume contamination (radiatords, hatch windows, optics, etc.) the ones most susceptible are the MSS, RAMs and satellites because of their scientific sensors that are exposed to the environment. It is assumed that all free-flying elements are designed to preclude damage to their sensors from their own jets. Therefore, only element pair operations must be evaluated.

In the case of the MSS, the elements that interface with it and may cause plume impingement problems are the EOS, tug and DRAMs. The MSS contamination problem can be avoided to some extent in the case of the EOS and tug by performing mating and separation maneuvers at an isolated port such as at the end of the core module. DRAMs may dock and separate at side ports on a core module and contamination of adjacent modules could occur. If this is a serious problem, the service ports for RAMs must be carefully designated to preclude damage to adjacent modules. That is, RAM ports on the MSS would be adjacent to MSS modules that do not contain exposed sensors.

RAMs and satellites that interface with tugs must be designed to be compatible with the tug jet plume during transport and attitude control operations. The singular logistics vehicle that can provide plume protection to payloads is the EOS because it can retain the payloads within its cargo bay during transport and orbital operations, such that the only time a payload will be susceptible to contamination is when it is erected out of the payload bay, is being separated from the EOS, or when a payload is mating to the EOS. Because of the potential long duration that attached RAM's may remain affixed to the EOS in an erected state for general operation, it is necessary that protection be first afforded these elements. Separating payloads would be the next most critical event in that these payloads are being released to begin operations. More than likely, mating to payloads is the least problem in that these payloads will probably be returning to earth.

Plume Geometry Relative to Jet Pod Location

Figure A4-3 is the representative model of an RCS plume developed from the curves shown in Figures A4-1 and A4-2. From this model the subsequent exhaust cone geometric patterns were developed.

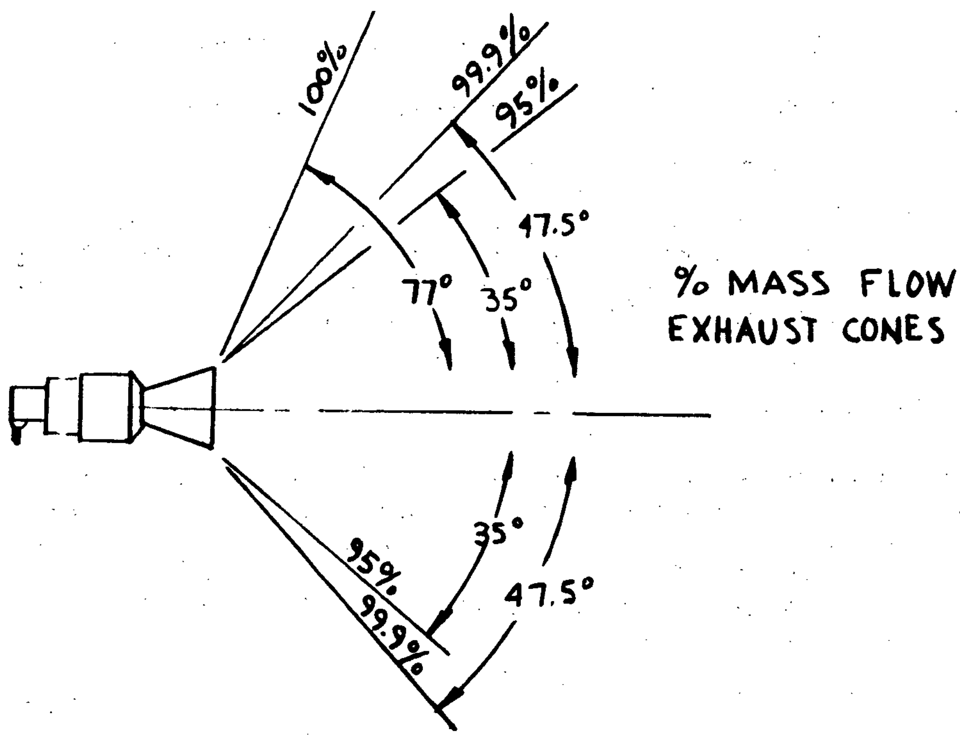


Figure A4-3. RCS Monopropellant Percent Flow Exhaust Cone

Tests were conducted using a 25 pound hydazine engine to determine the potential contamination on various surfaces at a distance of approximately 10 feet from the engine. Only a slight degradation was evidenced. Extrapolation of the data from the 25 pound jets to the 1000 pount jets proposed for the EOS indicated that comparable effects would occur at a distance of about 60 feet from the EOS jets.

Figure A4-4 depicts an EOS configuration that provides a volume above the cargo bay that is essentially free of jet exhaust contamination. As long as the payload is extended directly above the cargo bay by the manipulator it is not subjected to jet exhaust.

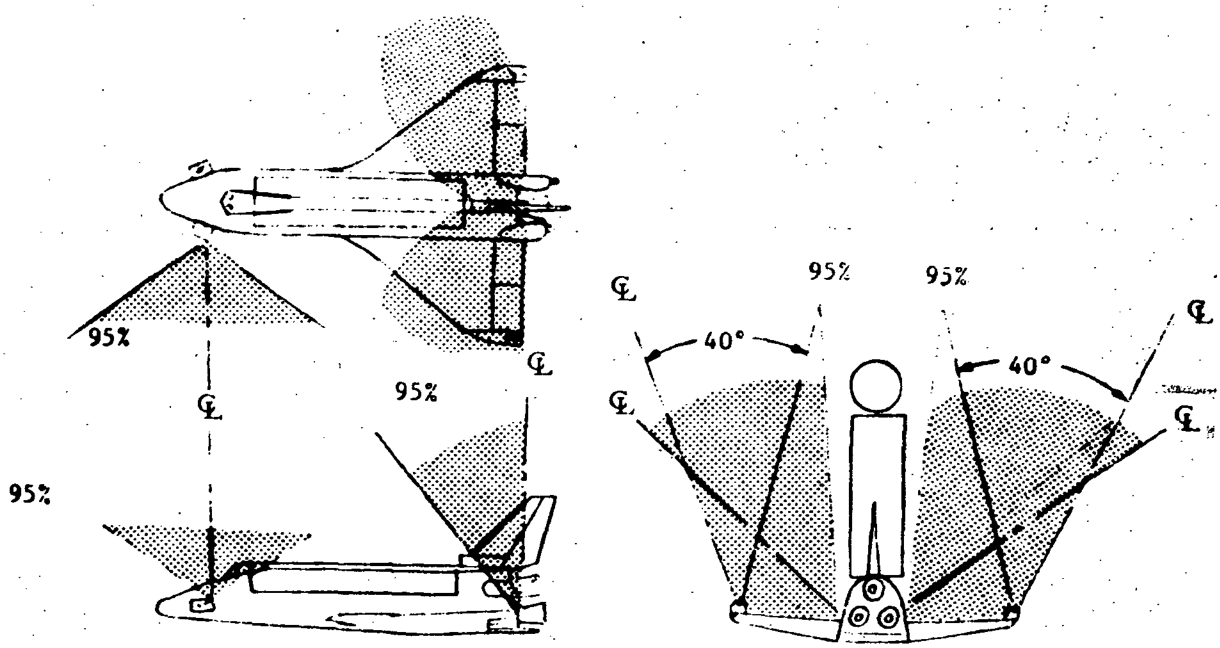


Figure A4-4. EOS Jet Configuration A

Figure A4-5 depicts the exhaust plume geometry for one proposed configuration of the EOS. The true distance between the jets and a deployed payload is approximately 100 feet. Although contamination effects would be minor at this distance some degradation would still occur. Note that the payload is not in the exhaust flow until it is about 50 feet from the cargo bay.

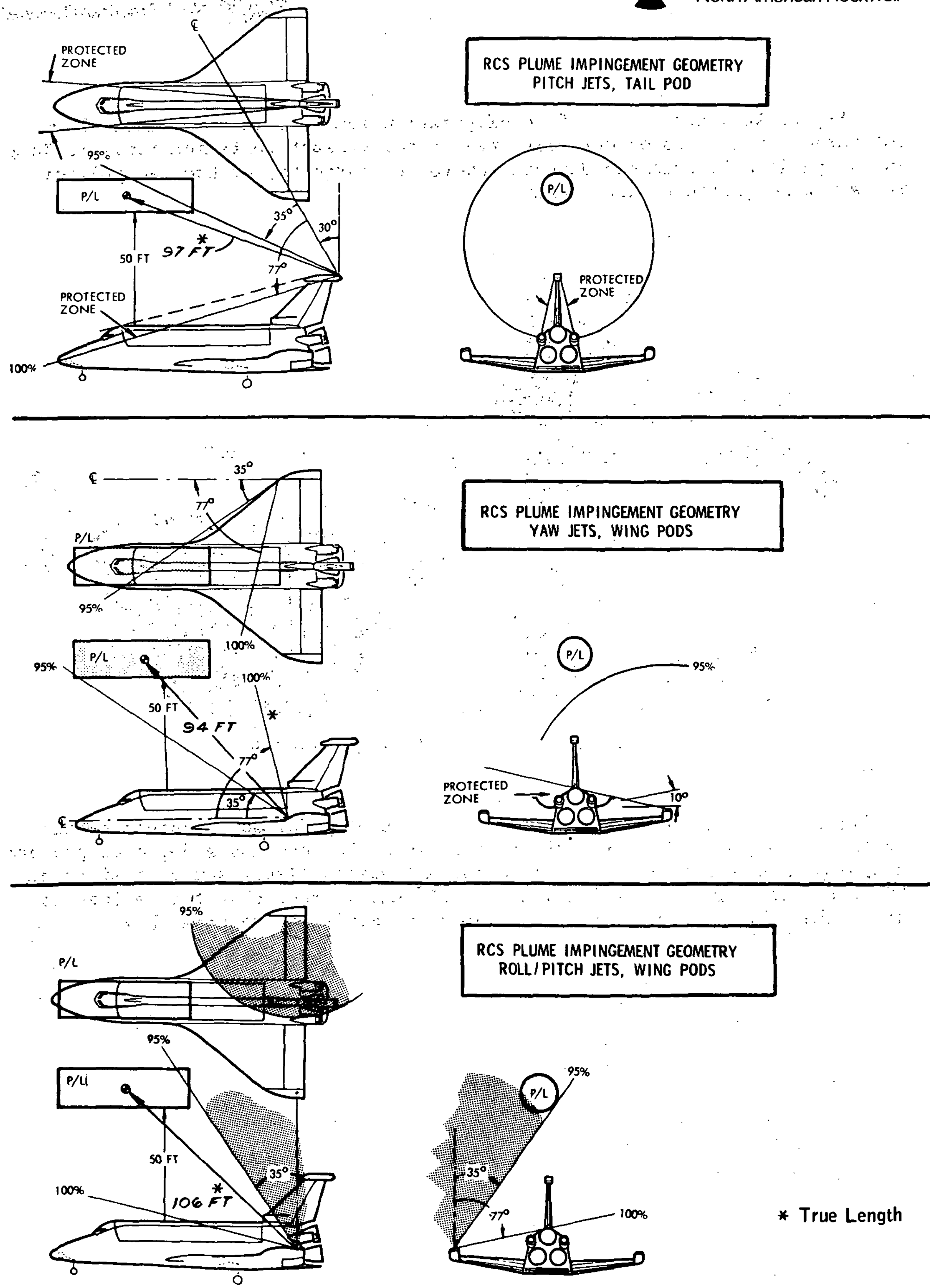


Figure A4-5. EOS Jet Configuration B

CONCLUSION

The introduction of numerous sensors that must be exposed to the space environment and the potential combinations of element pairs proposed during the next 15 to 20 years will require a more detailed analysis of plume impingement characteristics. Certain recommendations that can be made based upon the analyses conduct thus far are:

1. Select propellants that have "clear" exhaust products. Hydrazine is one of the leading candidates.

2. Arrange thrusters on all elements, where feasible, to preclude plume impingement on both the parent element and elements operating in close proximity to the parent element.

3. Select operational duty cycles that will minimize jet thrustings during docking and separation operations.

4. Do not rely solely upon a separation distance (e.g., manipulator or extension/retraction device) to preclude plume impingement.

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MANIPULATOR INTERACTIVITY ANALYSES

A-5

A5 MANIPULATOR INTERACTIVITY ANALYSES

The purpose of these analyses is to determine what synergistic benefits result if a manipulator is used in the next 20 years of space activity. A review of the individual interfacing activities (Volume II, Parts 2, 3 and 4) has indicated that another design of less complexity and lower cost can be used in lieu of a manipulator. However, these same activities also indicate that certain advantages can be gained by employing a manipulator. This study will attempt to assess a manipulator's worth as it applies across all activities. The criteria used will be the same comparison factors as presented by the separate activities and will add the integration necessary to evaluate commonality between activities and elements.

SUMMARY

These analyses contained in subsequent paragraphs point out that a manipulator can provide synergistic benefits which will enhance some operations, in particular the Orbital Assembly and Payload Deployment activities. At the same time, however, it has been shown that where a manipulator is preferred, the preference is based on operations that occur very infrequently. Developmental costs for a manipulator definitely exceed those for developing the other alternatives, whereas operational costs can be competitive. The weight increase by selecting a manipulator is dependent on the activity involved and the type of payload being delivered. For orbital assembly involving a mating activity, the weight penalty to direct dock was in some cases higher than manipulator hardware. For general payload delivery that does not involve a mating operation, the weight penalty for a manipulator can be on the otder of 1500 pounds.

All conceptual alternates, manipulator, direct docking, pivoting device, jet translation can provide commonality to some extent, but none offer total commonality. Perhaps a manipulator's greatest quality is that of extending an element designers limitations which could result in synergistic benefits that have yet to be identified. With a manipulator available, new orbital assembly techniques could be developed (space welding), experiment activities could be extended (fabrication outside of the protective environment of the orbiting element), and operations presently not considered feasible could become routine (minute inspection and resupply of satellites without first performing a hard dock.

Results of this analysis indicate that if a trade must be made between a direct docking concept with a pivotal mechanism and a manipulator, then the direct docking concept should be selected. This concept is the least cost, is capable of performing all of the identified operations with some kit requirements, and can be universally implemented by all of the study elements.

The recommended alternative method is to develop both concepts with manipulator development placed on a lower priority. As a manipulator becomes available and confidence in its capabilities is assured, it could be phased into more operational activities. This developmental process should be particularly advantageous in that it spreads out the initial costs and at the same time will introduce the manipulator into the program when its synergistic benefits can be more programmatically realized.

INDEPENDENT INTERFACING ACTIVITY ANALYSES

Four of the fourteen interfacing activities evaluated the application of a manipulator to accomplish the identified operations. Each of these activities is defined in Volume II, Part 2. Table A5-1 shows the results of the independent analyses for the following interfacing activities:

- Mating
- Orbital Assembly
- Payload Deployment
- Separation

Mating

Mating selected direct dock primarily for commonality, least cost, and minimum maintenance. The primary driver for a manipulator was the need to capture satellites which were too small to incorporate the common mating port in the design. This handicap was overcome by using a special docking adapter that fits within the common mating port hardware.

Approximately 6 percent (62 missions) of the EOS orbit missions in the first ten years involve the deployment of satellites too small for the standard mating port. If these satellites must be retrieved than a special adapter must be used. Because the adapter is relatively simple to install, there is really no driver for a manipulator. If a manipulator were available, it would not eliminate direct docking.

Orbital Assembly

Orbital assembly is a two-step operation. The first stage involves the mating of a module to an element and the second stage is configuring the interfaces. The first stage is the only stage that is concerned with manipulator selection. A teleoperator concept was eliminated because it required that a new program element be introduced and a teleoperator could not add any real benefits that would not be available with a reasonably flexible manipulator or by using IVA-EVA techniques. Because the first step is only a mating task, the problem was to determine if a manipulator for the assembly tasks provided enough advantages to override the direct docking concept selected by the mating activity or if there were some required operations that direct docking would not be able to accomplish. The only case where a manipulator approach offered superior advantages was for a module interchange at the same port. For direct docking, at least four separate dockings are required: (1) dock the new delivered module to a holding port, (2) separate and dock to the module being replaced, (3) separate the module and dock it to the new module stored on the space port, and (4) separate the new module and dock it to the cleared port. Figure A5-1 depicts these operations.

Table A5-1. Independent Activity Analyses Results

Activity	Selection	Rejection	Advantages	Disadvantages
Mating	Direct Dock	Manipulator	1. Commonality with element pairs	1. Requires special adapter to mate with small satellites
Orbital Assembly	Direct Dock	Manipulator Teleoperator	1. Commonality with mating	1. Numerous dockings to exchange modules on a single port 2. Limits module interchange for cargo or propellant transfer unless redundant ports are available 3. Requires that rigidizing techniques for element assembly on boosters be complex, or the element stack disassembled
Payload Deployment/ Retraction	Pivoting Mechanism	Manipulator	1. Retains interfaces between payload and EOS orbiter during deployment. 2. Commonality with mating.	1. Difficult multiple payload deployment task. 2. Payload deployed close of the EOS orbiter 3. Loss of selective payload positioning in cargo bay.
Separation	Jet Translation	Manipulator	1. Plume impingement	1. Commonality with element pairs

With a manipulator, this can be handled with a single berthing of the delivery element and then two berthings of modules as shown in Figure A5-2.

It can be seen that this interchange of modules is a rather complex operation, particularly for direct docking, and is somewhat expensive fuel wise to accomplish (approximately 350 lbs of fuel for EOS orbiter per dock). The manipulator operations after the initial berth are relatively safe .

The issue is how often will this type operation occur. Assuming that the study element model designs are essentially fixed, then the only elements that have available holding ports are the MSS, OLS, and modular OPD. The propellant transfer activity does not recommend an OPD in the near term program. Therefore, how often will the MSS and OLS require this operation ? The design of the common modules for these elements is such that no replacement is anticipated in the first ten years of the elements existence with probability estimates before replacement ranging out as great as 25 years. Even if two modules were to be replaced in ten years, this low figure would not drive a selection for manipulators. In fact, if this were the only use for a manipulator it would be cheaper to go up, remove the module, return it to earth and return with the new module.

If an exchange of modules between logistics vehicles is required, the simplest operation with a single mating port is to employ both docking and manipulator techniques. Figure A5-3 shows this type operation by having a Tug and EOS orbiter interchange modules. With a manipulator alone, the reach requirement for the manipulator can be in the range of 120 feet if the same operation is performed using just manipulation. Figure A5-4 illustrates the operation. If this module interchange at the same port is required, and the EOS orbiter or Tug is not equipped with a holding port, then as can be seen a manipulator is required. But, the only time this operation appears to be valid is for propellant tank exchange and cargo module exchange. The propellant tank exchange concept is applied in the "Propellant Transfer" activity. The operation is an alternate mode of resupplying an element (CPS or RNS) with propellant. The concept has the EOS orbiter deliver a full propellant tank to the CPS or RNS, remove the empty tank from the CPS or RNS and install the full tank. With direct docking, this concept requires a holding port and four separate dockings or the tank must be attached to another parallel feed point. Though this concept is viable, the Propellant Transfer activity did not select it from its alternatives. Rather it selected performing the operation using direct fuel transfers which requires only a single tank with the tank design common for all of these type missions. With a manipulator available, the Propellant Transfer preference would not change. Therefore, for this particular activity, the manipulator would not be an influencer, however, the lack of a manipulator will tend to invalidate the tank exchange option or require extensive docking maneuvers between a holding port with loaded propellant tanks or two feed systems from parallel tanks will be required. Cargo module

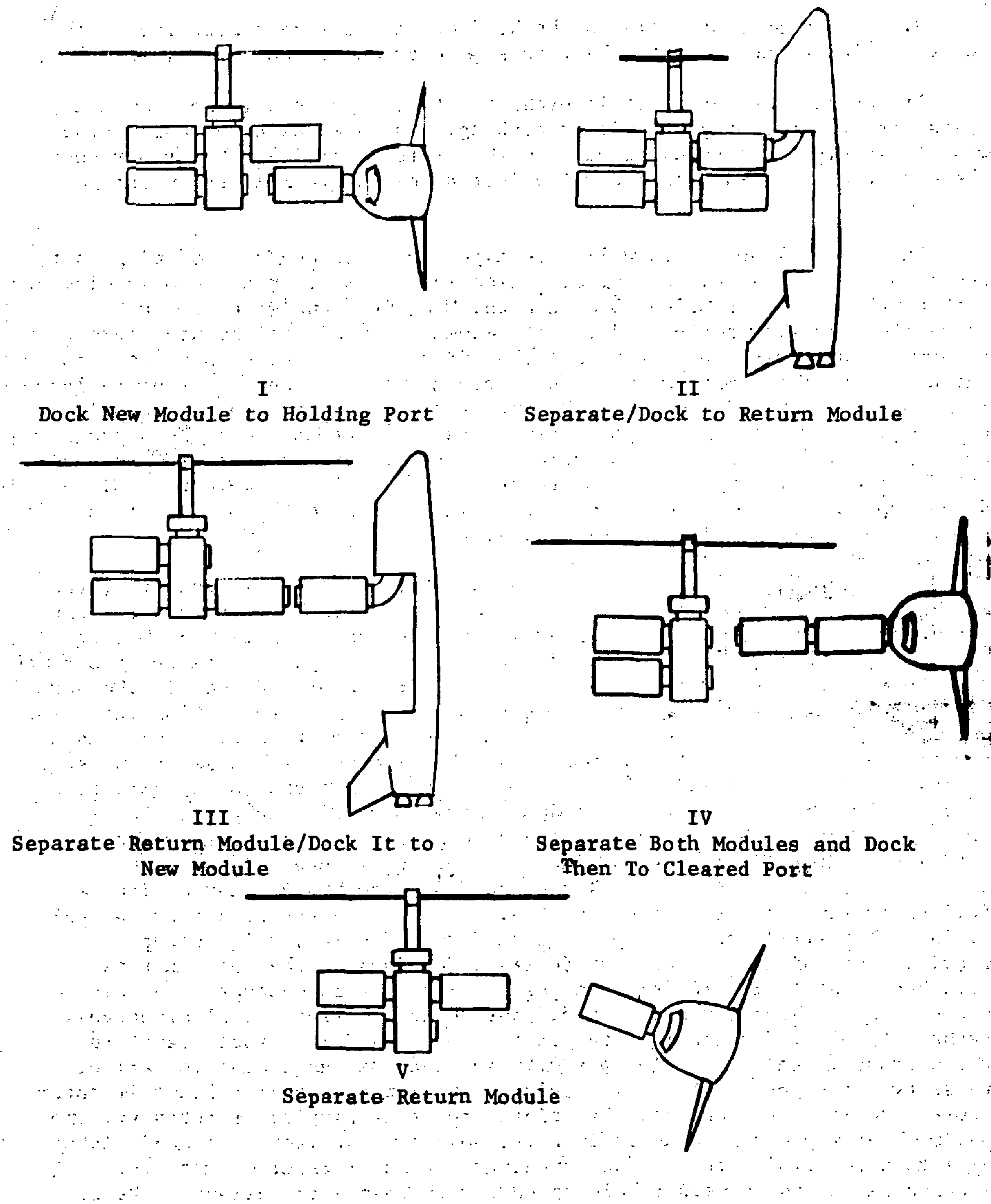
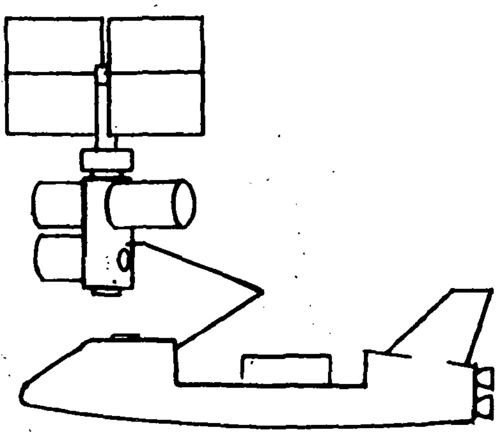
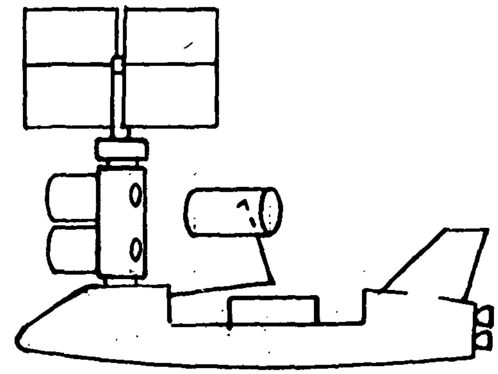


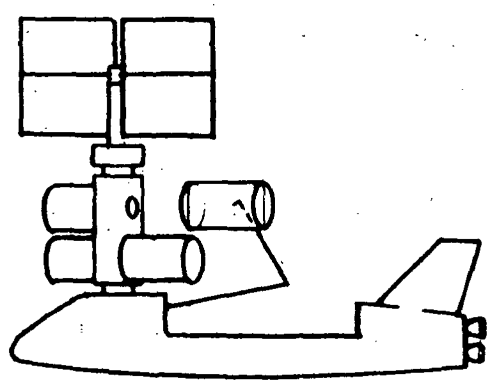
Figure A5-1. Direct Docking-Module Interchange



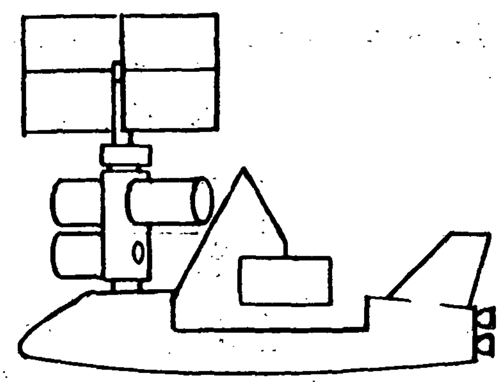
I
Berth EOS to MSS



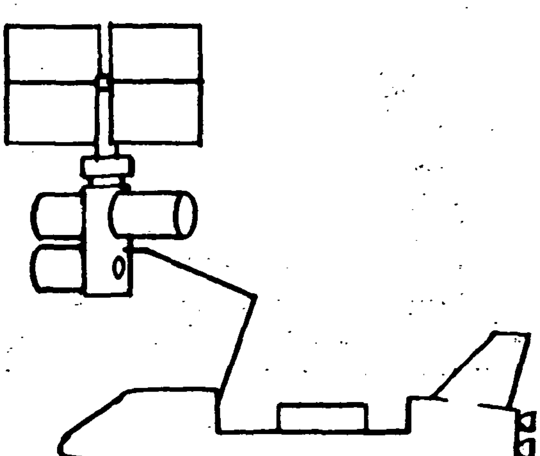
II
Separate Return Module/Berth it to Holding Port



III
Remove New Module from Cargo Bay/
Berth It to Cleared Port



IV
Remove Return Module/Stow in Cargo Bay



V
Separate EOS from MSS

Figure A5-2. Manipulator-Module Interchange

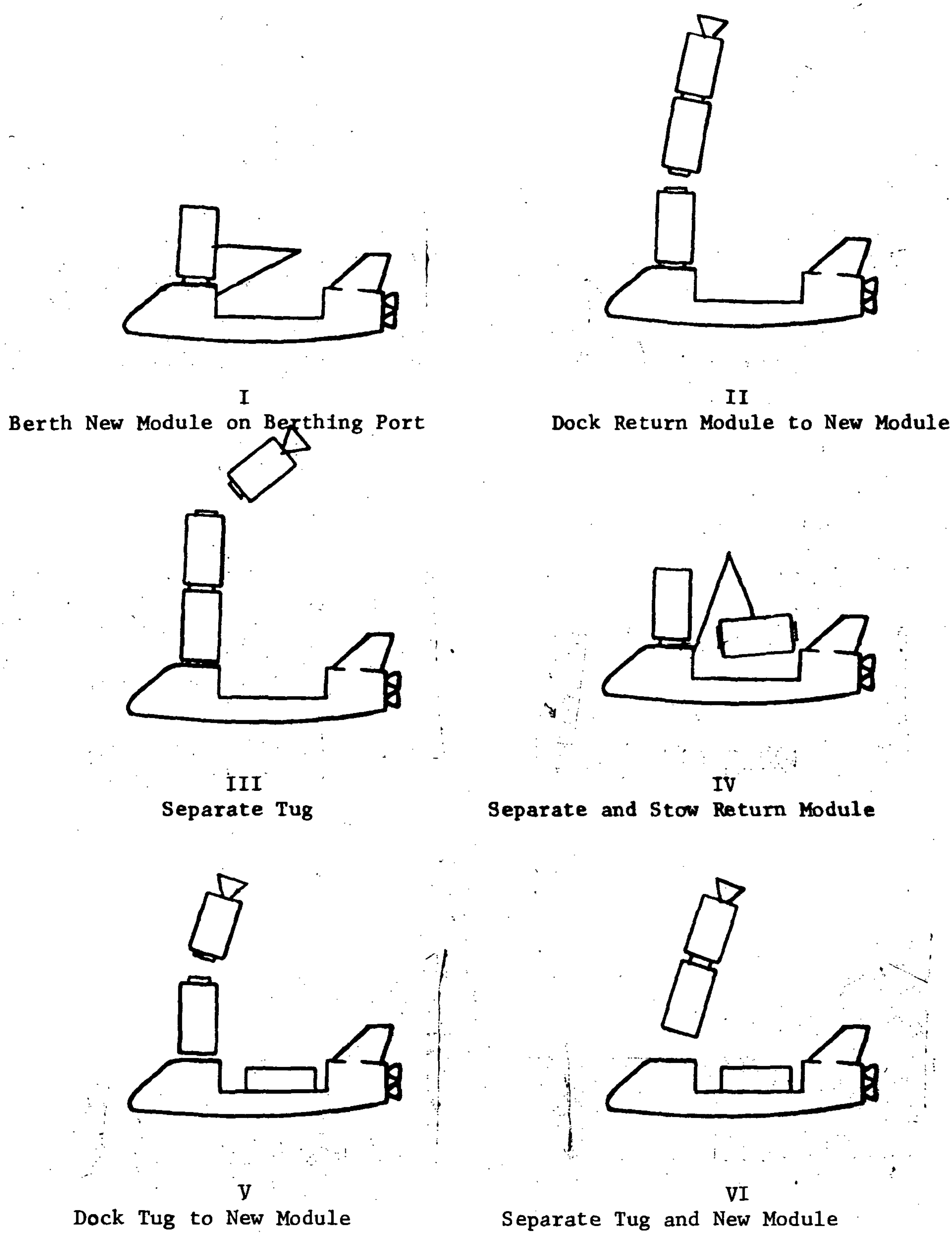


Figure A5-3. Logistics Vehicle Module Exchange-Manipulator/Direct Dock

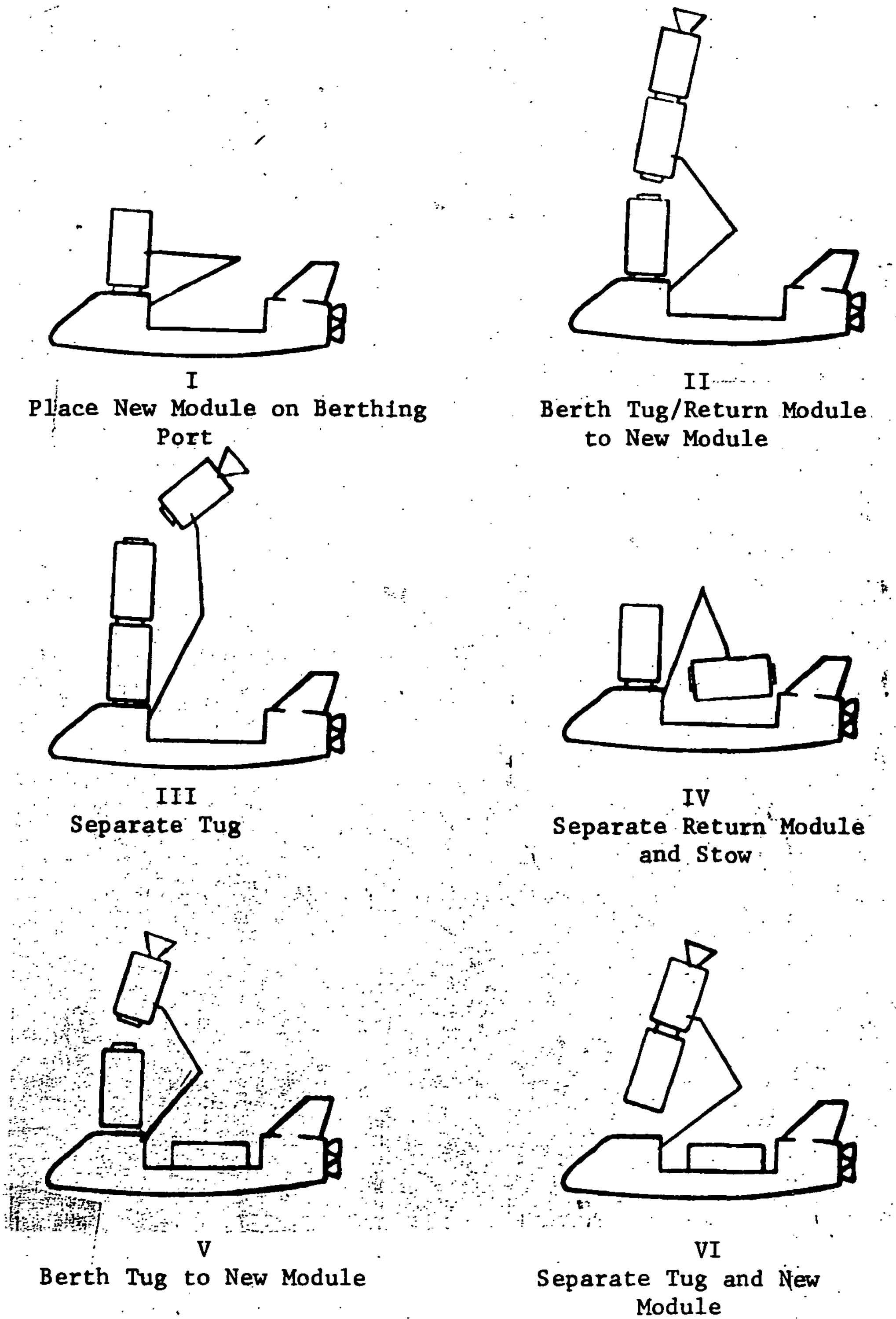


Figure A5-4. Logistics Vehicle Module Exchange-Manipulator transfer is a regular function that occurs between the MSS and the logistics elements. If the cargo module is interchanged at the same port, then the problem becomes one of four direct dockings or a manipulator will have to be available. But, the designs of the MSS's that have been investigated use dual cargo modules in the program that mate to different ports such that when a new module is delivered it is mated to an open port and remains there. The return cargo module is then picked up and returned. The open port is now available for the next routine resupply module. Because of this design, there is no driver for a manipulator, however, like the propellant transfer option, the lack of a manipulator will force this redundant cargo module port design.

The final consideration for a manipulator is that of assembling an element on a orbiting logistics element (RNS or CPS) for transport to a higher energy orbit. The RNS is of lower concern because it does not impose very high acceleration loads on an attached element. The CPS on the other hand will generate acceleration loads on the order of 2.5 g's. The worst case elements for transport to the higher energy orbits are geosynchronous MSS and the OLS. Both of these elements are of similar design, therefore we will only consider the worst case OLS. Two options for stacking the OLS on the booster element are available. The OLS can be placed on the booster element in a fully assembled configuration with some sensitive exceptions (i.e., solar arrays retracted) or the OLS can be stacked on the booster in a disassembled state. The decision is one of compatibility with the acceleration loads. If the fully configured element is placed on the booster element, the stack will appear somewhat like that shown in Figure A5-5.

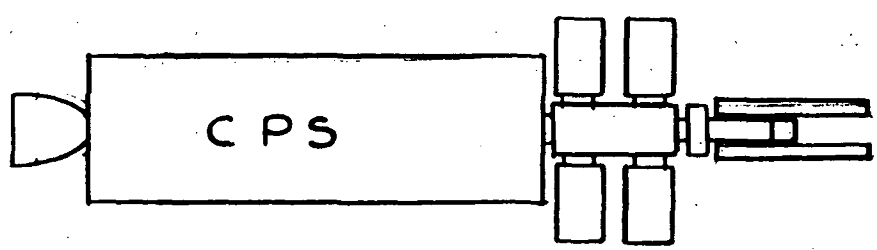


Figure A5-5. Fully Assembled OLS Stacked on CPS

The problem with this design is the load applied on the OLS appendages and at appendage mating ports. If the modules and mating ports are not designed to withstand the bending load, then supplemental structure or rigidizing techniques must be employed that will support the loads. Supplemental structural will be built into the MSS, but rigidizing techniques will be added on. Figure A5-6 shows two designs that could be employed.

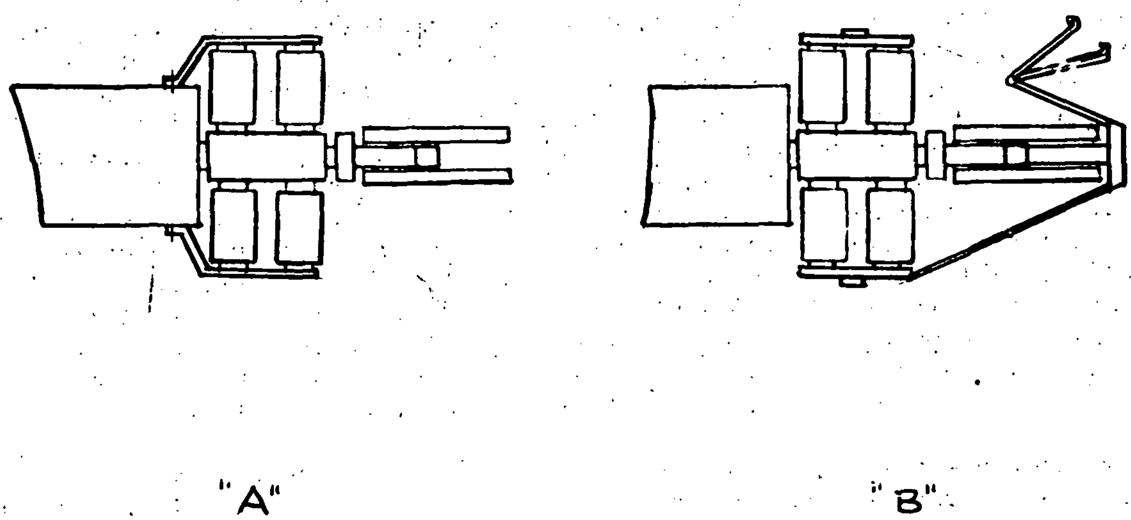


Figure A5-6. Add-On Module Transport Rigidization

Each of these concepts can be installed with relative ease if a manipulator is in the program, however, only "B" is applicable to direct docking and will require not only five additional dockings, but a complex tension strap or EVA techniques employed.

By stacking the modules in an unassembled manner, the task can be performed using only direct dock (see Figure A5-7).

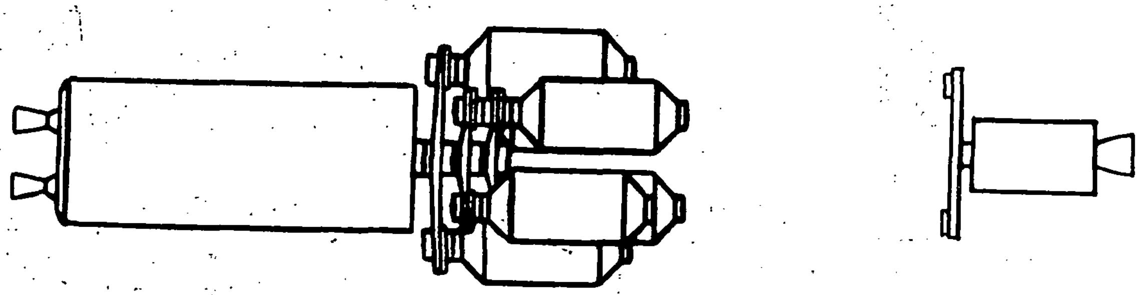
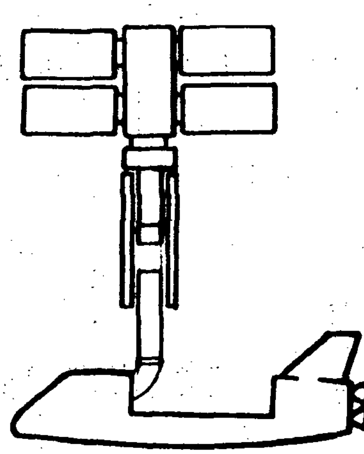


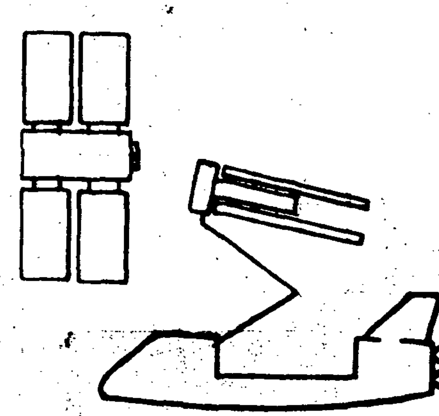
Figure A5-7. Direct Docking Assembly on CPS

If the OLS is required to be assembled and checked out in low earth orbit, then the preferred mode would be to transport it in this assembled configuration. Disassembling it for boost to a higher energy orbit is inefficient, it adds numerous tasks to the operation, reduces reliability, and adds assembly and checkout time at the new orbital position. It can be argued that lunar bases if checked out in low earth orbit must be disassembled for reassembly on the lunar surface and therefore a precedence is established, but this is necessary because of the landing operations and structural arrangement of the complex on the lunar surface. More than likely, the lunar base will never be fully assembled in an earth orbit. The OLS study (DS-350) recommends that the OLS be assembled, checked-out, and verified in low earth orbit before launch to lunar orbit. However, no detailed trade to determine the ramifications of first assembly and checkout in lunar orbit was performed such that a firm requirement for low earth orbital assembly and checkout can be fully justified. Because both the OLS and the geosynchronous MSS elements are not considered to be near term programs, a manipulator for this operation, if indeed the assembly and checkout will be performed before transport to the higher energy orbit, would not tend to be a driver but rather a synergistic benefit.

Another assembly operation that would definitely benefit by a manipulator in the program is that of assembling peripheral equipment on an orbiting element. The MSS is equipped with two five foot dish antennas mounted on mating ports. These antennas can be designed to be fitted through a mating port passage, installed and erected using EVA-IVA techniques, or it can be equipped with two mating ports such that it could be docked in place, or it could be designed to be an integral part of the module it is installed on and automatically erected when the module is properly mated. With a manipulator, the antenna could be berthed in position which would probably permit the simplest antenna design. Solar array replacement on an MSS requires that a long adapter be utilized to effect a direct dock retrieval of the array. With a manipulator the retrieval is a much simpler task. Figure A5-8 illustrates these two manipulator retrieval concepts.



Direct Dock Concept



Manipulator Concept

Figure A5-8. Solar Array Retrieval

EOS Payload Deployment

This activity selected the pivoted mechanism as opposed to the manipulator. The primary reasons for this selection were the commonality it provided between activities and the low cost. The pivotal device is deficient in three areas and will require supplemental hardware for support. The three areas of decifiency are:

1. The manipulator is able to deploy multiple payloads.
2. The manipulator is able to deploy payloads farther outside of the orbiter moldline than the pivotal mechanism.
3. With a manipulator, orbiter c.g. can be more effectively controlled because placement of small elements can be more selective.

The need to deploy multiple payloads can occur in about six percent of the first ten-year mission or about 60 missions. Though many of the multiple payload missions will be similar, they are by no means identical such that a common cradle could be developed for holding and independently deploying the payload. Even though the number of these type missions is relatively small, they tend to drive the manipulator concept. Methods for deploying multiple payload other than by manipulator will result in a far less flexible concept which in turn may perturbate payload designs.

The manipulator is able to deploy payloads outside of the EOS orbiter moldline. The pivot mechanism is also capable of deploying payloads outside of the EOS orbiter moldline, but for only a short distance. With a manipulator the payload can be extended far enough away from the EOS orbiter that interference problems associated with appendages on either the EOS orbiter or payload are considerably reduced or totally eliminated. This can be particularly beneficial for separation. If an element is sufficient distance from the EOS orbiter at time of release, recontact possibilities become less of a hazard.

Payloads must be placed in the EOS orbiter cargo bay such that the c.g. of the payload fits within a prescribed position which is determined by the payload weight. This particular problem should not be a driver in that only a few of the payloads fall outside of the parameter. The payloads that are problems are those elements that are very massive and compact. Of the 976 missions only 17 appear to require special consideration such as an adapter to move the element into an acceptable location.

A disadvantage of a manipulator over the pivoting device is that during the deployment of an element by a manipulator, electrical or fluid ties between the EOS orbiter and the payload must be severed. With a pivoting device the connections can be maintained.

Separation

Separation benefits primarily from a manipulator in that an element can be separated in such a manner that jet exhaust plume impingement can be greatly reduced. Because several satellites and RAM's are sensitive to jet plume impingement, this particular advantage can be a strong driver. The separation activity, however, rejected this option in that, where available, jet selection logic could be utilized, or sensitive elements could safely use their jets to perform the separation, or a design more economical than a manipulator could be employed if these first two options were not satisfactory.

Table A5-2 shows the programmatic impacts of selecting either a manipulator or a direct docking pivoting mechanism concept.

Table A5-2. Alternate Selection Impact Summary

CONSIDERATIONS	IMPACT OF SELECTION	
	MANIPULATOR	DIRECT DOCKING
Change in mode	Not possible	Will accommodate either
Manipulator design	Reach dependent	Independent*
Payload impact	1 mating port, 1 mating receptacle	2 mating ports (reduced length)
Mating port design	Simple - lightweight	Requires attenuation
Retention in EOS	Requires all loads be reacted by side-mounted devices	Part of the loads reacted at forwardmating port when landing

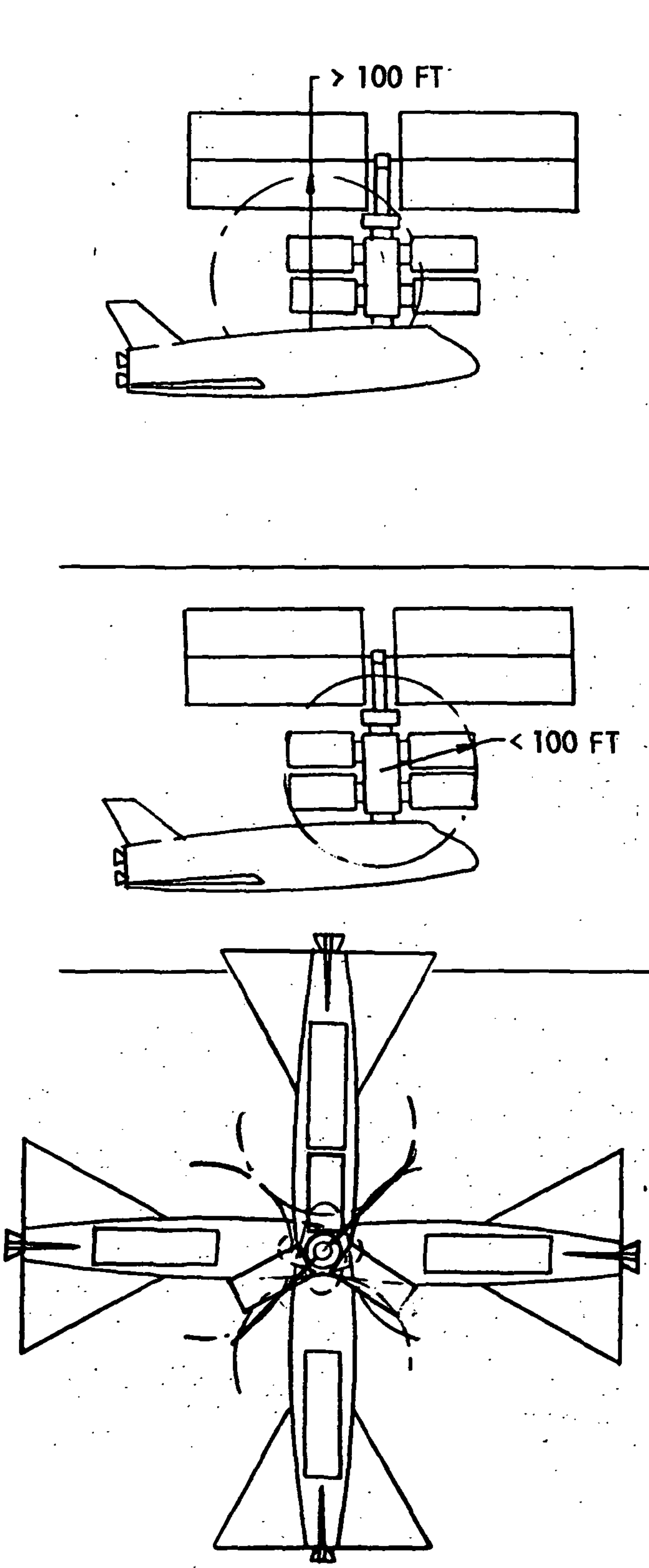
*when docking a short module parallel to a long module, an adapter must be used to provide clearance

Change in Mode

With a manipulator, elements will not be equipped with dynamic attentuation system or with mating ports that can accept relatively large misalignments. Whereas with direct docking, if a manipulator is introduced into the program at some future date, the direct docking systems are fully compatible.

Manipulator Design

Manipulator reach is relative to module length, location of mating port, assembly operations involved, manipulator location, manipulator end effector receptacle location, number of manipulators involved in the activity, mating ports available and their location, and capability of mating ports to accept variable orientation matings. Some of these variables are shown in Figure A5-9 using an EOS orbiter and MSS for a model. General advantages and disadvantages of each option are also pointed out.

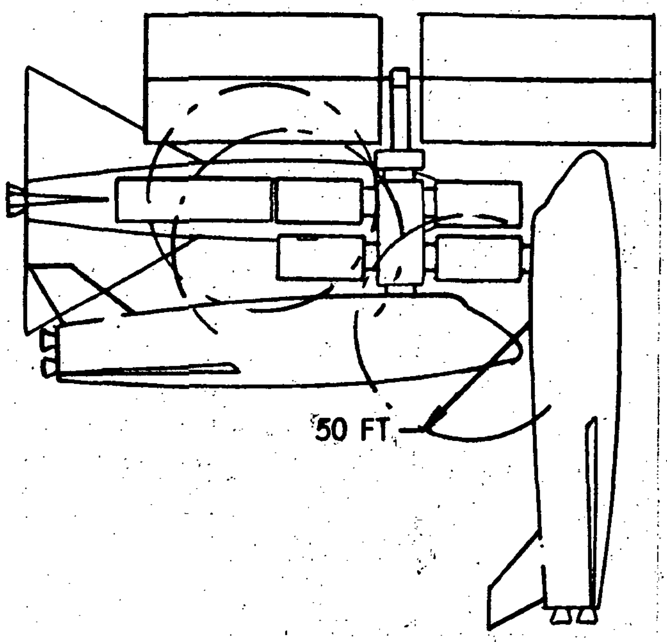


- SINGLE MANIPULATOR
- SINGLE MANIPULATOR ELEMENT BERTHING LOCATION
  1. DYNAMIC STABILITY IS QUESTIONABLE
  2. HEAVIEST
  3. REQUIRES MULTIDEGREES OF FREEDOM TO ACCOMPLISH ALL TASKS
  4. SINGLE UNIT, GROUND-BASED, ALLOWS MAXIMUM MAINTAINABILITY
  5. SINGLE UNIT ALLOWS MAXIMUM EXPERIENCE

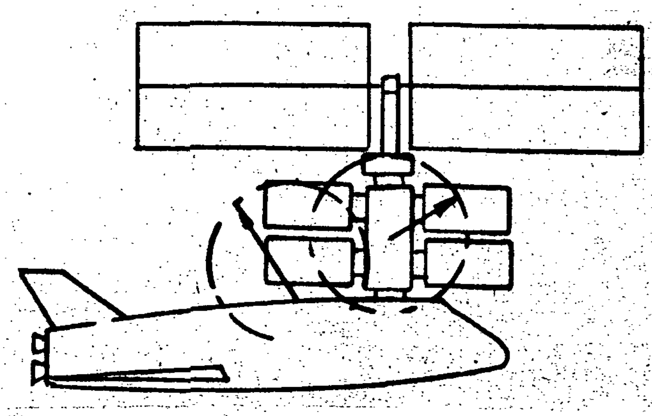
- SINGLE MANIPULATOR, DIFFERENT LOCATION
  1. SHORTER REACH REQUIREMENT
  2. REQUIRES ON-ORBIT MAINTENANCE
  3. MSS FLEXIBILITY
  4. EOS MAY STILL REQUIRE MANIPULATOR FOR OTHER OPERATIONS

- SINGLE MANIPULATOR
- SINGLE MANIPULATOR ELEMENT BERTHING LOCATION, BUT PORT ALLOWS VARIABLE POSITION BERTHING
  1. MINIMUM COMPUTER INTERFACE PROBLEMS FOR MSS INTERFACES
  2. SINGLE UNIT, GROUND-BASED, ALLOWS MAXIMUM MAINTAINABILITY
  3. SINGLE UNIT ALLOWS MAXIMUM EXPERIENCE TO BE GAINED

Figure A5-9. Manipulator Concept Options (Sheet 1 of 2)



- SINGLE MANIPULATOR
- MULTIPLE MANIPULATOR ELEMENT BERTHING LOCATION
  1. MANIPULATION MAY BE ACCOMPLISHED IN A SINGLE PLANE
  2. SINGLE UNIT, GROUND-BASED, ALLOWS MAXIMUM MAINTAINABILITY
  3. SINGLE UNIT ALLOWS MAXIMUM EXPERIENCE TO BE GAINED



- MULTIPLE MANIPULATORS
  1. MINIMUM REACH LENGTH
  2. PROVIDES OPERATIONAL FLEXIBILITY
  3. REQUIRES ON-ORBIT MAINTENANCE
  4. REQUIRE TWO OPERATIONS TO PERFORM BERTHING (I.E., BERTH ELEMENT TO ONE PORT THEN MOVE IT TO ANOTHER OR PASS THE ELEMENT BETWEEN MANIPULATORS)

Figure A5-9. Manipulator Concept Options (Sheet 2 of 2)

Payload Impact

Direct docking requires that a payload which is to be mated to another element be equipped with a mating port on each end of the element. This naturally reduces the effective length of the module and increases its weight. The reduction in length will vary from a few inches up to three feet depending on design and whether the port is active or passive. The resultant weight increase could be as much as 200 pounds. However, the additional weight for manipulators will be higher such that direct docking will result in an actual net weight savings. If we consider that a manipulator is only on an EOS orbiter and a space based tug is in the program, (interfacing port with the EOS and a port for the tug to engage the payload). For small satellites, the mating ports may be manipulator interface capture receptacles. If a tug is not in the program, particular payloads can be manufactured with none or only one mating port. Those elements that can be retrieved and directly stowed in the cargo bay such as statellites and EOS supported free flying RAM's could be designed without mating ports. Modules such as those of the MSS that attach to the core module could be manufactured with only a single mating port. But, when we consider a geosynchronous MSS or an OLS, a tug is required, therefore, these elements which are practically identical to the low earth orbital MSS would require mating ports on both ends.

Mating Port Design

If mating ports can be designed without impact attenuation, there are certain advantages gained. The mating port will in essence be maintenance free except for seals and latches. The latches will be clearly exposed such that they can easily be replaced in orbit. Utility interconnects will also be clearly exposed with straight across passages without any interfering mating port hardware. This is highly beneficial for elements like the MSS with its 19 plumbing lines, two air distribution ducts and 690 square inches of electrical feed throughs at some ports (DS225).

Retention in EOS Cargo Bay

Transmission of loads between the EOS orbiter and a payload in the cargo bay is dependent on the design and location of the physical attachments. With direct docking, the forward end of the module will be attached to the mating port on the pivoting device. Whereas this mating port attachment is not a necessity for boost loads, the landing loads (on the order of 3 g's) applied along the X-axis could possibly be damped using the attenuation of the docking port. This synergistic benefit is not available with a manipulation design. However, there are designs for side attachments to a payload that can effectively react the applied loads.

GENERAL CONSIDERATION FACTORS

Technology

Manipulators are new to space. They are made up of numerous mechanisms that will be exposed to the space environment. Because manipulators are in existence for earth applications does not make them state-of-the-art for space application. Several problems must be overcome before a manipulator is qualified for operations in a space environment. Some of these are listed as follows:

Weight to Strength Ratio

As the weight of a manipulator increases, payload capability is reduced. Materials that have a low weight to strength ratio will have to be utilized.

Environmental Protection

The arm will be exposed to both high and low temperatures simultaneously. Mechanisms and lubricants that cannot be exposed to the variable space temperatures will have to be enclosed and insulated. At the same time electrical equipment (i.e., torquing motors) will create heat that must be dissipated.

Zero-G Adaptable

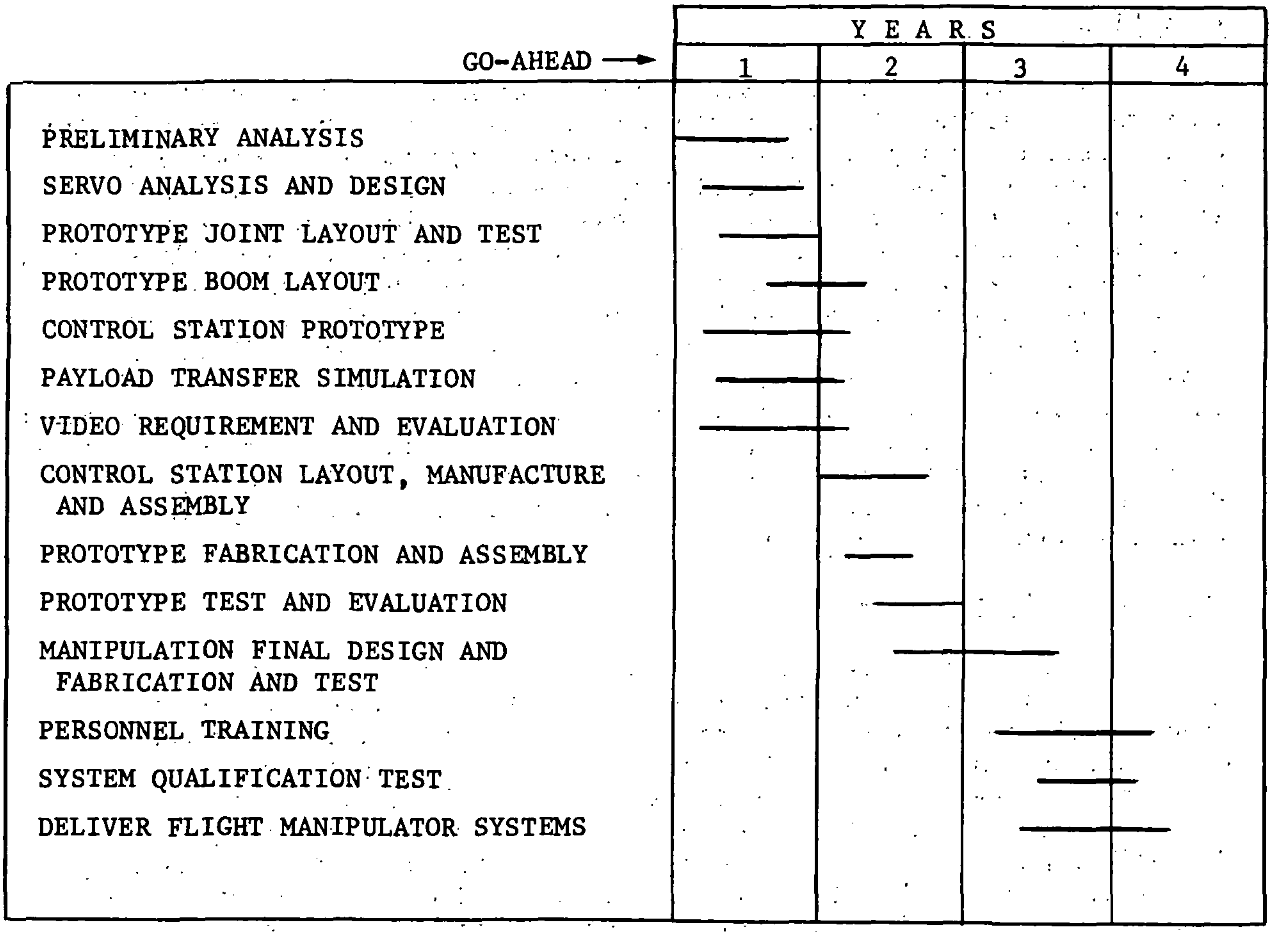
In a zero-g environment, manipulators not only maneuver the payloads, but at the same time it maneuvers the base it is attached to. If payloads must be strategically located in a particular position, then the ACS of the base must be integrated with the manipulator system or the manipulator torque must never exceed that of the ACS. Another problem is that of developing a ground test mechanism that can evaluate zero-g manipulator characteristics before flight.

Sensitivity

To be synergestically beneficial, the manipulator must be adaptable to many operations. Therefore, it will be expected to be relatively rugged (i.e., capture large mass elements of complex configuration and control the element even though the capture point is fairly distant from that element's c.g.) and it will also have to have a fine touch for performing intricate tasks (i.e., disengaging a payload retention device). If bilateral systems are used, then sensors must be developed to feed pressure impulses to the manipulator operator and the operator must be able to detect and react to the variety of forces which must be made sensitive to the controller's zero-g environment.

Development Schedule

One contractor estimates that the development time schedule for a manipulator is on the order of three and half years with a schedule somewhat as follows (DS 571):



The other alternatives, direct docking and pivot mechanisms are all well within present technology and would not put a strain on development schedule.

Checkout and Maintenance

If we consider that the manipulator will be only on the EOS orbiter (mating activity recommendation), then maintenance will always be performed on the ground and would not add any complexity to the system. However, if a manipulator is on an element that is not periodically returned to ground, then maintenance is a complex operation requiring EVA techniques which may be beyond present state-of-the-art. Sensor replacement or simple plug-in devices are feasible, but torquing motors, clutch mechanisms, cables and structural arms do not lend themselves to designs that are EVA repairable and still be suitable to the space environment.

Maintenance of a berthing port is considerably less than required for docking systems with attenuation. In fact, berthing port maintenance is almost nil except for seal replacement. However, with selective placement of the active mating ports (those ports with attenuation systems) the maintenance could be performed on the ground similar to manipulator maintenance.

Reliability

When comparing a direct dock concept and a manipulator berth concept in terms of reliability, the factors to be considered are the reliability of the direct docking capture and attenuation system and the manipulator arm mechanisms. Impact attenuation systems and capture systems will be exposed to a space environment in a passive state for long periods of time and still must be ready to perform multiple dockings when called upon. If the manipulator is also going to be exposed to these conditions, there is no doubt that the direct docking concept would be the more reliable simply due to the larger number of failure possibilities exhibited by a manipulator. If two manipulators are available for use, the reliability naturally increases. However, it is questionable if it would ever increase to that of a direct docking system. On the other hand, if the manipulator is periodically returned to earth for inspection and maintenance, its reliability factor could surpass that of impact attenuation systems that remain unattended in earth orbit.

Operational Costs

The operational costs are maintenance and payload weight. If a manipulator must be maintained in earth orbit or must be returned to ground periodically for maintenance, then the costs will far exceed that of direct docking systems. If the manipulator is only on the EOS orbiter, then maintenance costs will be low and may even be less than direct docking costs because of the low maintenance costs for berthing ports. Payload weights are shown in Table A5-3 for various missions. The weights represent those weights that are allocated to direct docking or manipulator berthing only. Weights used for the analyses are as follows:

Berthing port	200 lbs.
Active direct docking port	520 lbs.
Passive direct docking port	330 lbs.
Pivoting device:	560 lbs.
Manipulator system (1 arm and includes berthing port on EOS orbiter):	2400 lbs.
Propellant (includes back off hold, reorientation and recontract):	350 lbs.

Table A5-3. Payload Weight Delta for Particular EOS Missions

TYPE MISSION	DIRECT DOCK/ PIVOTING DEVICE	MANIPULATOR
Deliver core module to MSS (10 ports)	5570	4850
Deliver common module to MSS	1740	2600
Deliver EOS DRAM to operational orbit	1410	2600
Exchange MSS cargo module	2090	2600
Exchange module with Tug	3120	2800

Element Design Influence

Table A5-4 identifies the allocation of major hardware for various elements relative to whether a manipulator is utilized or not and what element is equipped with the manipulator. It can be seen that minimum mating ports results when a manipulator is available for use, particularly when the MSS is in the program. This is because appendage modules can be equipped with only one mating port.

Table A5-4. Major Hardware Allocation by Element

	Pivot Device	Manipulator	Berthing Port	Active Port	Passive Port	Manipulator Receptacle
MANIPULATOR ON EOS ORBITER ONLY						
EOS Orbiter		2	1			1
Space Based Tug				1		1
MSS			8	2		1
Satellite - EOS Supported						1
Satellite - Tug Supported					1	1
Satellite - EOS/Tug Supported					1	1
DRAM - EOS Supported			0 - 1			1
DRAM - EOS/MSS Supported			1			1
DRAM - EOS/Tug/MSS Supported					2	1
DRAM - MSS Supported					1	
CPS/RNS					1	1
MANIPULATOR ON SPACE BASED TUG ONLY						
EOS Orbiter	1			1		
Space Based Tug		1		1		
MSS				2	16	1
Satellite - EOS Supported					1	
Satellite - Tug Supported						1
Satellite - EOS/Tug Supported					1	1
DRAM - EOS Supported					1	
DRAM - EOS/MSS Supported					1	
DRAM - EOS/Tug/MSS Supported					1	1
DRAM - MSS Supported					1	
CPS/RNS					1	1

Table A5-4. Major Hardware Allocation by Element
(Continued)

	Pivot Device	Manipulator	Berthing Port	Active Port	Passive Port	Manipulator Receptacle
MANIPULATOR ON EOS ORBITER AND TUG						
EOS Orbiter		2	1			1
Space Based Tug		1		1		1
MSS			9	1		
Satellite - EOS Supported						1
Satellite - Tug Supported						1
Satellite - EOS/Tug Supported						1
DRAM - EOS Supported			1			1
DRAM - EOS/MSS Supported			1			1
DRAM - EOS/Tug/MSS Supported			1			1
DRAM - MSS Supported					1	
CPS/RNS					1	1
MANIPULATOR ON MSS						
EOS Orbiter*	1			1		1
Space Based Tug*				1		1
MSS		1	9		1	
Satellite - EOS Supported					1	
Satellite - Tug Supported					1	
Satellite - EOS/Tug Supported					2	
DRAM - EOS Supported					1	
DRAM - EOS/MSS Supported					1	1
DRAM - EOS/Tug/MSS Supported					1	1
DRAM - MSS Supported					1	1
CPS/RNS						

*No interchange of modules between EOS orbiter/space based tug

Table A5-4. Major Hardware Allocation by Element (Continued)

	Pivot Device	Manipulator	Berthing Port	Active Port	Passive Port	Manipulator Receptacle
DIRECT DOCK EOS ORBITER - SPACE TUG						
EOS Orbiter*	1			1		
Space Based Tug*				1		
MSS				2	16	
Satellite - EOS Supported					1	
Satellite - Tug Supported					1	
Satellite - EOS/Tug Supported					2	
DRAM - EOS Supported					1	
DRAM - EOS/MSS Supported					2	
DRAM - EOS/Tug/MSS Supported					2	
DRAM - MSS Supported					1	
CPS/RNS					1	
MANIPULATOR/DIRECT DOCKING ON EOS ORBITER						
EOS Orbiter		1		1		
Space Based Tug				1		1
MSS			8	2		
Satellite - EOS Supported						1
Satellite - Tug Supported					1	
Satellite - EOS/Tug Supported					1	1
DRAM - EOS Supported		0 - 1				1
DRAM - EOS/MSS Supported		1				1
DRAM - EOS/Tug/MSS Supported					2	1
DRAM - MSS Supported					1	
CPS/RNS					1	1

*No interchange of modules between EOS orbiter/space based tug

A6. ECLSS APPROACH FOR RAM SUPPORT

INTRODUCTION AND SUMMARY

This study was conducted to determine the preferred allocation of ECLSS functions between the EOS and a RAM support module (RSM). The results of this study affect the Attached Element Operations Interfacing Activity. Six alternate concepts were synthesized considering the prime factors of EOS dependency and hatch state (open or closed). Figure A6-1 shows each of the concepts. Concepts A1, A2 and A3 are all completely dependent upon the EOS while Concepts B and C are partially dependent. Concept D is totally independent of the EOS operating in a closed hatch configuration, however an open hatch remains as an optional mode.

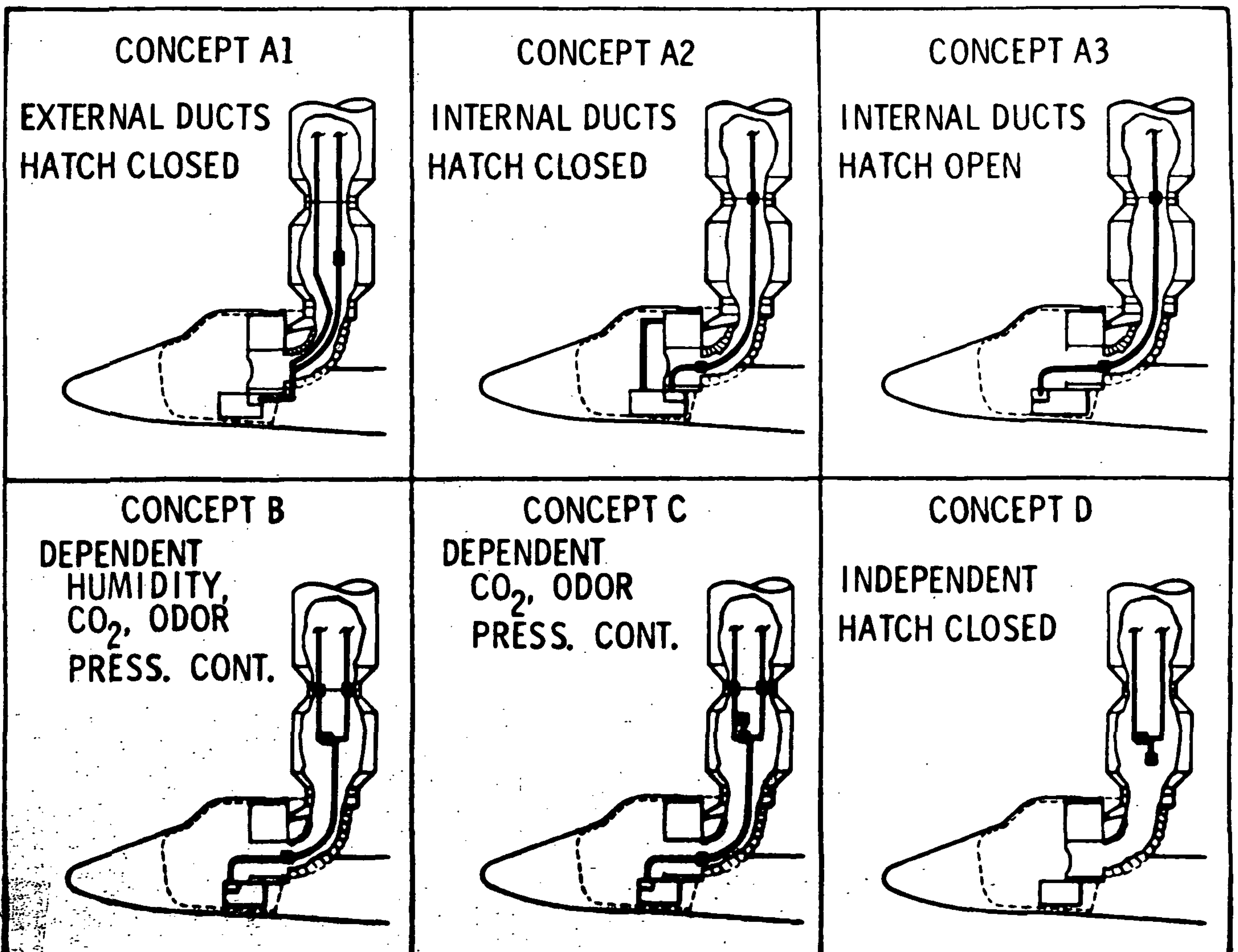


Figure A6-1. RAM ECLSS Concepts

Comparison data were developed for each concept option considering (1) safety, (2) program costs, (3) performance, and (4) physical characteristics (weight, volume, power). Based on these data, Concept D is the recommended approach for all RAM missions with a closed airlock hatch. The prime rationale for this selection is the lack of shuttle scars coupled with the low sensitivity to EOS or RSM crew size variations.

COMPLETELY EOS-DEPENDENT CONCEPT DESCRIPTIONS

Three completely dependent concepts are shown on Figure A6-2. Concepts that are less dependent on the shuttle including a fully independent concept are discussed subsequently.

Concepts A1, A2 and A3 are totally dependent on the shuttle ECLSS. Concept A1 provides atmospheric circulation to the RAM by supply and return ducts located external to the shuttle pressurized volume. This installation does not effect the integrity of the shuttle airlock. The ducts enter the RAM through the shuttle/payload adapter. Fans are installed in each duct to overcome the pressure drop in each duct. Duct shutoff valves are placed in the cockpit to isolate the cockpit from the RAM in case of an emergency. Apollo post landing vent valves (5-in. Ø) were selected for this application since sealing is required only in one direction and the valves are already developed. Concept A2 runs a single supply duct internal to the pressure hull from the shuttle to the payload and a sub duct from the airlock bulkhead to the shuttle ECLSS. Return air from the RAM passes through the shuttle tunnel and airlock into the shuttle sub duct. The shutoff valves installed in the ducts of this concept are a new development since they must withstand a full 14.7 psia pressure head from either side. Since ducting is internal to the pressure hull, flex ducting can be utilized which is lighter than that required for A1. An airlock hatch open concept is shown as A3. This concept simply lays a supply duct from the shuttle ECLSS to the RAM. Return air from the RAM again passes through the shuttle tunnel into the shuttle cockpit and ECLSS.

A generic characteristic of all dependent concepts is that their capacity is limited by the capacity of the shuttle ECLSS. The primary limitation is the cabin heat exchanger capacity. Table A6-1 shows the heat load breakdown for the NR 040 shuttle cabin heat exchanger. The design heat load for the cabin heat exchanger is 7200 Btu/hour. When the shuttle is powered down a sensible load of approximately 1100 Btu/hour can be absorbed from the RAM while maintaining a maximum air temperature of 75°F. It is estimated that 3500 Btu/hour could be absorbed if the air temperature were allowed to rise to 80°F. The maximum average power load is 4.6 Kw for the two life science experiment packages. Utilizing the rule-of-thumb that 20 percent of this electrical load heats the air, the RAM air load would be just under one kilowatt thermal or approximately 3100 Btu/hour. The alternatives to absorb this heat are: (1) increase the capacity of the shuttle heat exchanger, (2) accept the expected higher temperature environment, or (3) place an additional cabin heat exchanger in the RAM. The latter alternative is discussed later under the semi-dependent concepts. Note that selectable temperature control in the RAM cannot be attained by any totally dependent concept.

CONCEPT A1

External Ducts

Hatch Closed

2 + 2

CONCEPT A2

Internal Ducts

Hatch Closed

2 + 2

CONCEPT A3

Internal Duct

Hatch Open

2 + 2

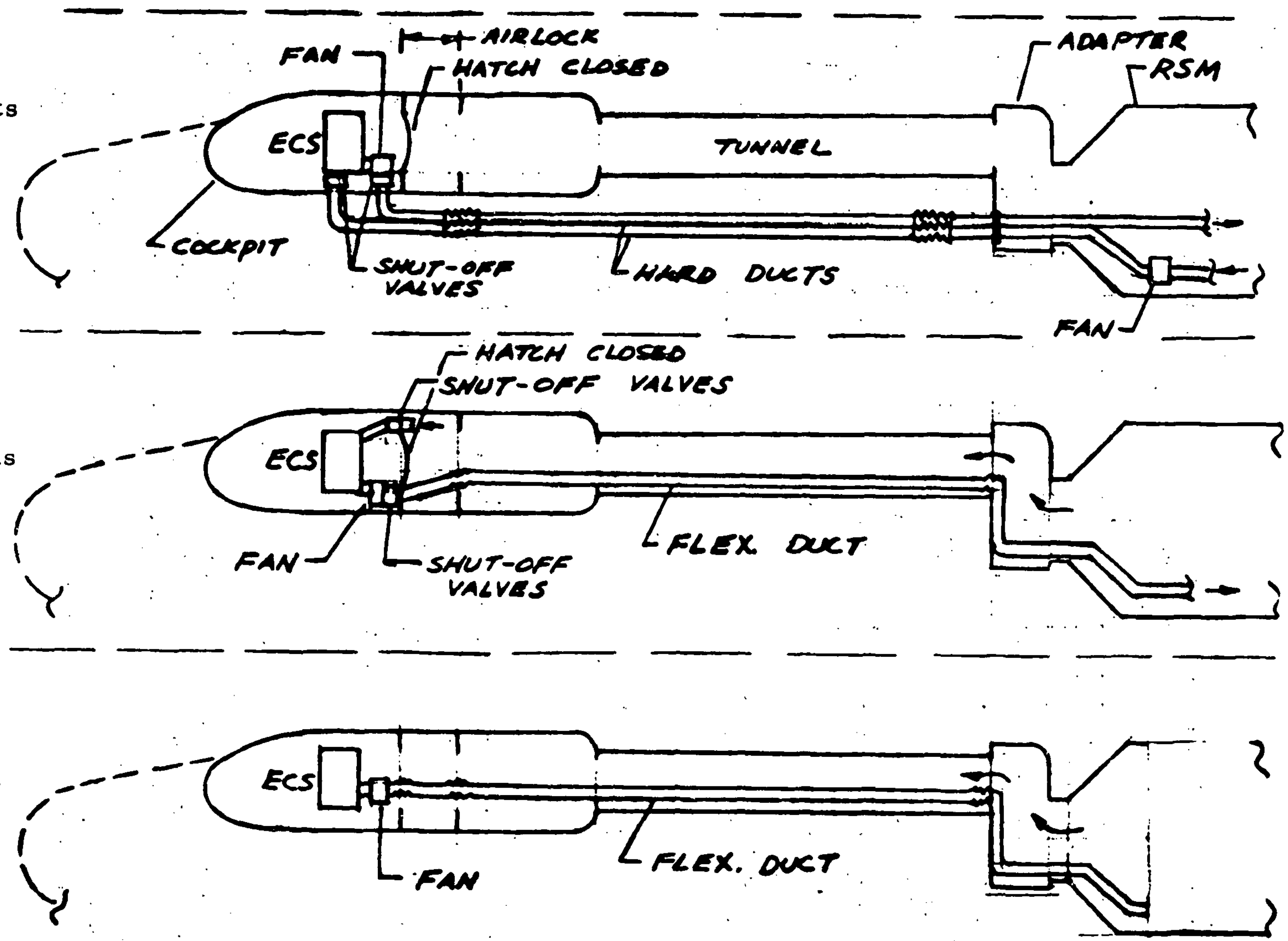


Figure A6-2. Completely Dependent Concepts

Table A6-1. Heat Loads on Shuttle Orbiter (H_2O) Coolant Loop (Btu/hr)

Heat Source	Rendezvous and Dock	Station Keeping	Dock-Undock
Display and control	1185	135	1185
ECLSS	2185	2849	2185
Metabolic	2133	896*	2133
Wall and window	1700	1000	1700
Total orbiter budget	7203	4882	7203
Available for RAM operations		2318	
Experimenters metabolic		1200	
Capacity for RAM heat load		1118	
*Orbiter Crew (2) only			

Humidity processing of the shuttle ECLSS is sized for a total crew compliment of four although this is a function of the latent/sensible heat load mix. With a high sensible/latent heat load ratio larger crew sizes could be accommodated. However, the limited sensible cooling capability and the resultant high air temperatures precludes a favorable sensible/latent heat load ratio. CO_2 processing is a function of LiOH element replacement frequency and therefore does not limit crew size capacity within the ranges being considered.

Concept A1 has operational advantages over other dependent concepts but penalizes the shuttle design by causing structural scars. This concept has hard ducting that passes external to the pressure hull allowing use of the shuttle airlock without shuting down atmosphere circulation to the RAM. Such ducts are heavier than an internal flex duct because they must withstand a 14.7 psia pressure differential. These ducts and their associated pressure hull penetrations would be permanent scars to the shuttle design requiring costs to be assessed to all shuttle missions instead of those scheduled only for RAM.

Shutoff valves are placed in the duct to isolate the cockpit from the RAM module should the RAM atmosphere become contaminated or should it become decompressed. A decompression requires sealing only in one direction, i.e., the shuttle cockpit must sustain a pressurized environment if the RAM is decompressed; if the shuttle cockpit is decompressed the crew can take refuge in the shuttle airlock. An Apollo post-landing ventilation valve was selected for duct shutoff. It is a five-inch valve that was developed for the Apollo program and can seal in one direction. These valves would also scar the shuttle design to assure cockpit pressure integrity on all shuttle missions.

Concept A2 reduces the scarring to the shuttle design. Ducting is installed through the airlock wall in this concept. Since only a supply duct is necessary and flexible ducting can be employed, duct weights are significantly reduced. Airlock operations require circulation to the RAM to be stopped since the duct must be disconnected to shut the airlock door. This limitation could be eliminated by scarring both sides of the airlock with duct penetrations. This scar does not appear to be justified when the infrequent use of the airlock is considered. The flexible duct can be removed from the shuttle when it is not committed to a RAM mission thus reducing the shuttle scar weight.

Shutoff valve design for the ducts of this concept must seal in both directions. The design must account for a depressurized airlock or a depressurized shuttle cockpit. Such a valve has not been developed in a manned space program for the size required by this application. Duct shutoff at other hatches is accomplished by removing a separable portion of the duct.

The open hatch concept, A3, reduces the shuttle scarring to a minimum for a dependent concept. Only the supply duct adapter that attaches to the shuttle ECLSS would be scar weight. Since the duct is passed through the airlock with both hatches open, airlock operation can only be accomplished by scarificing circulation to the RAM. Removable sections in the duct are installed to allow hatch closure.

SEMI-DEPENDENT AND INDEPENDENT CONCEPTS

The remaining concepts shown in Figure A6-3 are less dependent on the shuttle. Concepts B and C could be supplementary modifications to any of the "A" type concepts although pictorially they are shown with the ducting approach of A3. Concept B adds a sensible heat exchanger and fan to the RAM module. Shuttle support is then reduced to metabolic processing. Concept C utilizes a condensing heat exchanger instead of a sensible heat exchanger in the RAM. Selectable temperature control can be accomplished over the full 65 to 85°F range in Concept C. This function cannot be accomplished with Concept B because condensation will occur in the heat exchanger when trying to achieve the low end of the selectable temperature range. Shuttle support in Concept C is limited to CO_2 scrubbing and O_2 partial pressure maintenance. Concept D is totally independent of the shuttle ECLSS with total atmospheric processing occuring in both habitable volumes.

Concept B can overcome the limitations of the fully dependent concepts in sensible heat absorption and has a potential to accommodate larger crew sizes than 2 + 2 if the latent/sensible metabolic mix is favorable. The margin of this concept for unfavorable metabolic load mixes is greater than the "A" type concept wince the air temperature can be reduced in the RAM compartment reducing the propensity of the crew to perspire. Shuttle dehumidification processing is presently sized for a 2 + 2 crew and therefore RAM operational flexibility is limited by the constraints of heat load management. For this study, Concept B is assumed to be limited to a 2 + 2 crew size.

CONCEPT B

Dependent
Humidity
CO_2 + Odor
Pres. Cont.

Hatch Open

2 + 2

CONCEPT C

Dependent
CO_2 + Odor
Pres. Cont.

Hatch Open

2 + 2

CONCEPT D

Independent

Hatch Closed
(Option - open)

2 + 4

No Scar

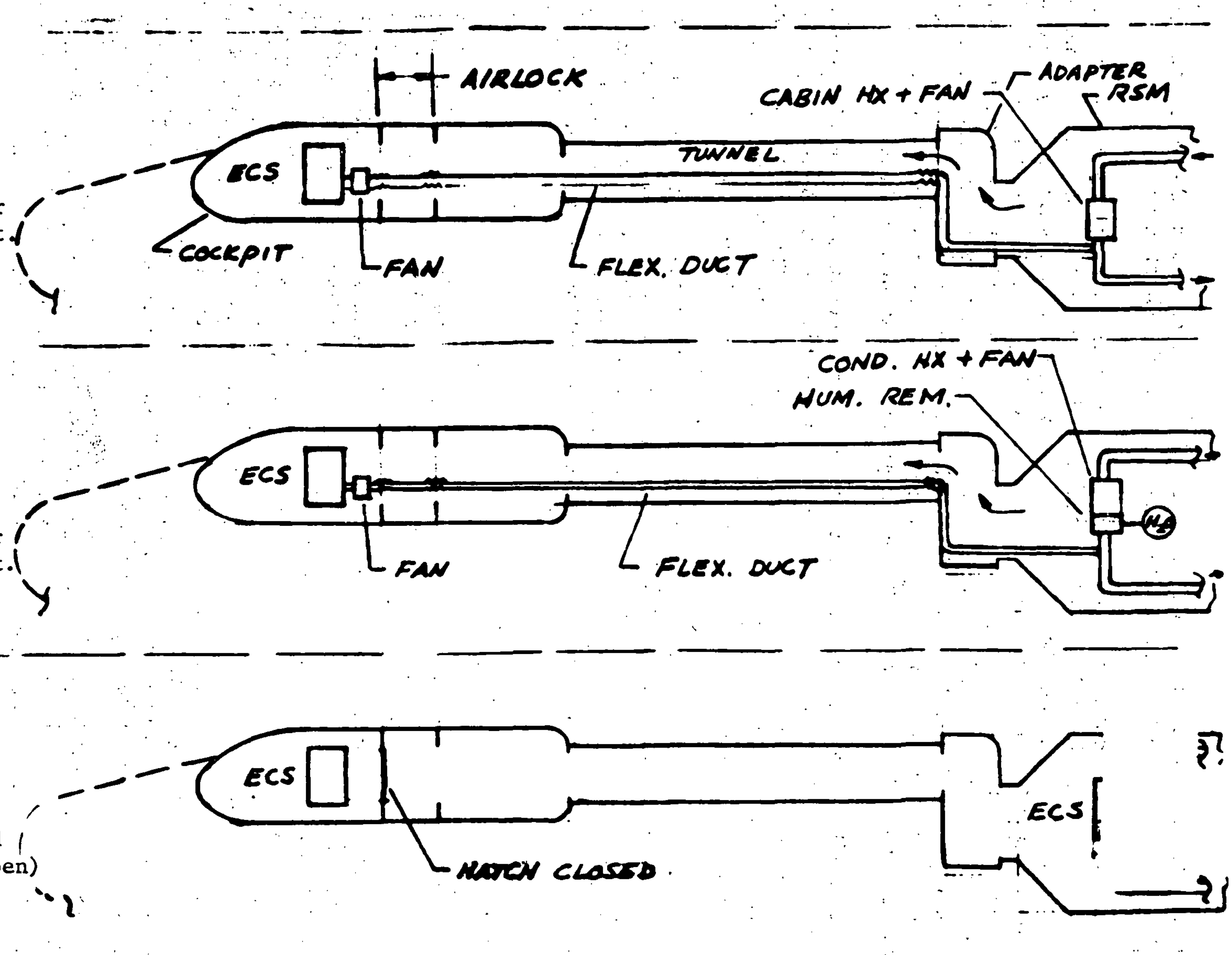


Figure A6-3. Partially EOS Dependent and Independent Concepts

Concept C can meet all the requirements defined for the RAM ECLSS. With humidity processing in the RAM, a total crew compliment of eight (orbiter crew plus experimenters) can be accommodated. This concept has the same capacity as two independent ECLSS assemblies but does not present the potential interaction problems associated with dual atmosphere controls. LiOH elements are replaced more frequently in the shuttle canister assembly to accommodate larger crew sizes than 2 + 2. When two canister locations exist, such as in Concept D, unequal LiOH element consumption will occur because of unequal crew residence or metabolic CO_2 production. As a result, optimum LiOH element utilization can be assured only by a single CO_2 processing location. Centralized partial pressure control is also preferable to separate controls in the shuttle and the RAM. Since pressure regulators seldom control at the same pressure, the valve with the lower crack point will control the atmosphere makeup for the entire vehicle. If the O_2 and N_2 stores feeding these regulators are separate, an imbalance of consumption will occur unless the two atmospheres are separated (by a closed hatch, for instance) or each regulator is periodically inhibited to equalized consumption. If the O_2 and N_2 stores feeding both regulators are common it will not matter which regulator controls.

Concept C with centralized CO_2 processing and O_2 partial pressure control is a better system operationally than the independent concept, Concept D, if the shuttle hatch can remain open. If the hatch must remain closed for safety reasons, Concept C requires the addition of scar weight to the shuttle like all other dependent concepts. However, the scar weight can be minimized because the duct size for only CO_2 control is smaller than that required for sensible and latent heat load control. The most significant benefit of having two independent ECLSS installations is that scars to the shuttle design are eliminated.

CONCEPT COST ANALYSIS

Costing of ECLSS assemblies was based on costs defined by NR on the shuttle program. Hardware was separated into hardware previously developed but applied to the RAM design and hardware requiring development for the RAM application. Hardware previously developed were charged the shuttle recurring cost plus 15 percent of the shuttle ECLSS nonrecurring cost to account for the development required to install this hardware into the RAM. The costs of newly developed hardware were determined by relating this hardware to the cost of the total subsystem. Cost scaling fact~~ors were not~~ available below the subsystem level.

The weights of all hardware were first defined so that relative costs could be evaluated. The independent concept D was used as a baseline since all the hardware for the RAM ECLSS is identified in this concept. The recurring cost for Concept D was determined by utilizing the subcontractor estimated recurring costs for the shuttle ECLSS assemblies utilized on RAM and applying an add-on multiplier to it. The multiplier (2.32) includes such factors as subcontractor cost growth, 70 percent; procurement costs, 15 percent; and in-house engineering and manufacturing costs. In-house development supporting this hardware was assessed at the same rate as that assessed on shuttle but scaled down by the ratio of the weight of the RAM ECLSS to the weight of the shuttle ECLSS.

Nonrecurring costs for the RSM ECLSS and TCS were previously reported as one lump sum in Reference (1). It was necessary to separate the ECLSS and TCS cost portions to evaluate the alternative concepts properly. Nonrecurring cost for the RSM ECLSS plus TCS was estimated previously at $25.2M (15 percent of the total shuttle ECLSS development plus system engineering costs). The assemblies identified for RAM make up $11.56M of the $35.8M of the shuttle ECLSS and TCS subcontractor nonrecurring costs, a ratio of .324 to 1.0. Utilizing the ratio of these shuttle nonrecurring costs, $8.15M was split out of the $25.2M lumped nonrecurring cost to be representative of the ECLSS portion.

Assessment of new development hardware required the establishment of subsystem CER's (cost equivalence ratios). The nonrecurring cost CER was determined by taking the total shuttle ECLSS development cost and multiplying it by the subsystem nonrecurring cost ratio. This value was in turn divided by the hardware weight to arrive at a nonrecurring CER of $100,000/pound of hardware. The recurring cost CER was determined by taking the recurring cost defined for Concept D and dividing that value by the weight of Concept D.

The cost of each alternative could now be evaluated. The weight of each new component was multiplied by the appropriate subsystem CER. The cost of each new component was then defined by multiplying this value by a factor relating estimated complexity of this component to the average subsystem complexity. Hardware items previously developed or included in Concept D were treated as costing the same as the hardware of that concept. Therefore, these costs were determined by scaling the Concept D costs by the ratio of the weight of these hardware items to the total weight of Concept D.

CONCLUSIONS

Table A6-1 summarizes the tradeoff data for each alternative. Hatch open conclusions are straightforward. Dependent Option A3 should be selected for 2+2 missions and upgraded with Concept C delta hardware. These options result in the lowest cost, lightest weight and best subsystem operational characteristics. The issue is not as clear cut for hatch closed operation. Concept A1 can be rejected because of a higher cost and shuttle scar weight than A2. Concept A2 can be selected if a $3.7M dollar savings in early year funding is significant to the RAM program and the cost of scarring the shuttle is negligible. Concept D is the preferred technical solution. It costs more initially although it is cheaper over the entire program.

Concept A2 has significant technical problems relative to both shuttle and RAM designs. The concept adds at least 117 pounds of scar weight to the shuttle. Cost of this car to the shuttle program could not be identified, but the modifications to the airlock wall and the shuttle ECLSS ducting must increase the cost of the shuttle although they do not increase the cost of the RAM. Residence in the RAM is precluded by shuttle airlock operation. This constraint prevents the use of the shuttle airlock as an alternative egress path for experiments requiring experiment control in the sortie module with EVA operations. Finally, RAM air temperature control is limited by shuttle heat exchanger capacity. Depending on shuttle cockpit conditions the RAM air temperature may be as high as 80°F. Moving to Concept B to overcome this

Table A6-1. Concept Comparison

	Dependent Concepts					Independent Concept "D"
	Hatch Closed		Hatch Open	Delta	Delta	
	"A1"	"A2"	"A3"	for "B"	for "C"	
RAM crew capability	2	2	2	2	6	6
Weight - lb	330	330	205	80	145	484
Power - watts	442	221	221	49	-16	205
Volume - cu ft	70	61	54	2.6	4.6	40
Cost - Millions						
Nonrecurring	7.0	7.1	3.4	1.4	2.6	8.2
Recurring - 1st unit	2.9	3.1	1.8	1.0	1.6	6.0
Total	9.9	10.2	5.2	2.4	4.2	14.2
Delta for 2+2 or 2+6	4.2	4.2	4.2	Upgrade to C		-
Program Total	14.1	14.4	9.4			14.2
Shuttle scar	EOS ECLSS Duct Adapter					None
	Penetrate cockpit and RSM wall	Penetrate airlock wall				
	Ducting full length	Ducting stub				
	172 lb	117 lb	15 lb			
Safety	Isolation hardware	Duct/hatch interference	RAM degrades shuttle			Backup
Performance	Limited RAM heat pickup			RAM heat pickup	No Hx condensation	Hatch closed eliminated CO_2 and $O_2/N_2\,\partial P$ interaction
	No selectable temperature control					
	Crew size limited to 2+2					
	No CO_2 or $O_2/N_2\,\partial P$ control interaction					

problem reduces the cost savings of independent operation. Therefore A2, if selected, must either depend on increased shuttle heat exchanger capacity or allow the air temperature to increase. Furthermore, if the air/coldplate electrical heat load ratio should increase from the assumed 20/80 level the problem would be further compounded. (NR space station studies generally had difficulty improving this ratio below 50/50.)

Concept D is the recommended approach for all RAM missions with a closed airlock hatch. In NR's judgment the indicated cost savings of $3.7 million with Concept A2 does not warrant the compromises that must be made in the RAM atmosphere control. Furthermore, this savings will clearly be reduced to an even less significant value when the shuttle scar costs are added to the RAM subsystems costs.

If Concept A2 is selected for the sortie RAM because of RAM funding constraints, upgrading of the ECLSS processing capacity to eight-man crew sizes should be accomplished by conversion to Concept C. Since the shuttle design has already been scarred by ducting for the sortie RAM, it is not cost effective to place an independent ECLSS in the RSM. It costs $10.1 million to convert Concept A2 into an independent ECLSS installation in the RSM where it will only cost $4.2 million to upgrade Concept A2 to Concept C. Furthermore, Concept C has performance advantages over Concept D as previously noted.

A7 ATTACHED ELEMENT OPS SUPPLEMENTAL ANALYSES

INTRODUCTION AND SUMMARY

Two separate analyses are detailed herein and summarized in the Attached Element Ops section of the report (Vol. II, Part 4, sec. 4). The analyses are (1) monitor and control and (2) payload capabilities.

The monitor and control topic represents an extensive review of the SOAR payloads to derive their respective monitor and control requirements.

The second topic presents the EOS payload capability as a function of (a) altitude, (b) orbit inclination, (c) up-down payload, and (d) ABES in or out. Subsequent to this presentation, there is a suggested method for conducting trade-offs considering payload weights, attitude hold, orbit variations, and propellant considerations.

MONITOR AND CONTROL

The monitor and control function is associated with a number of activities; detached element operations, communications, attached element transport, etc. Monitor and control, being an integral part of checkout operations, is an important part of Attached Element Operations. In order to establish a definition of the types of monitoring and control requirements, an examination of the inventory of SOAR payloads was conducted. This examination resulted in the definition of the basic hardware concepts for the power, data and communications, and guidance systems. Table A7-1 presents a definition of the hardware concepts for the various SOAR payloads. These hardware concepts provide a basis for defining subsequently the types of monitoring, checkout and control functions for Attached Element Operations, where the functions cut across all three alternate modes--RAM Operations, Service and Checkout, and Quiescent Storage. Table A7-1 summarizes the monitor and control requirements for the list of SOAR payloads.

Table A7-2 now presents a narrative discussion of the monitor and control requirements for thirty-seven different NASA payloads. These requirements are presented in a generalized manner and represent the level of detail as presently available in the literature.

In order to narrow down the total spectrum of monitor and control requirements into a manageable number to be considered, certain reouirements that have a low probability of occurrence will need to be eliminated. To lend some credibility to this selection process, Table A7-3 has been developed. This Table defines the status, priority and prognosis of certain payloads in the NASA unmanned inventory.

TABLE 1 - SOAR PAYLOADS

POWER | DATA & COMM. | GUIDANCE

	Shuttle	Fuel Cells	Batteries	Solar Array	Chem. A.P.U.	Pwr. Condit.	Pwr. Dist. & Con.	Radio-Iso. (RTG)	Shuttle Prov.	Data Proc. Ass'y	Data Storage	Antennas	Transmitters	Receivers	Hardline		Shuttle Prov.	Computation	CMG's	RCS	Stable Platform	Optics	Kick Stage	Inertial Ref.	Cryo.	Gas Jets	Elec. Prop.
HEAO Series			x	x		x	x			√	√	x	x	x					√	√	√			√	√		
Adv. Stellar			√	√		√	√		x	√	√	x	x	x					√	√	√			√	√		
Large Solar			√	√		√	√			√	√	√	√	√					√	√	√			√	√		
IR Astronomy			√	√		√	√			√	√	√	√	√					√	√	√			√	x		
OSO			x	x		x	x			√	√	√	√	√					x	√	√			√		x^1	
Interferometer			x	x		√	√			√	√	x	√	√					x	√	√		x	√		x^2	
Explorer Ser.			√	√		√	√			√	√	√	√	√					√	√	√		x	√	√		
Km Wave Tele.			√	√		√	√			√	√	x	x	x					√	√	√		√	√	√		
Plasma Physics			√	√		√	√			√	√	√	√	√					√	√	√		x	√	√		
Cosmic Ray Physics			√	√		√	√			√	√	√	√	√					√	√	√		√	√	√		
Mars Viking				x		√	√			√	√	x	x	x						√	√		x	√		√	x
Venus Series				x		√	√			√	√	√	√	√						√	√		x	√		√	
Jupiter Series						√	√	x		√	√	√	√	√						√	√		√	√		√	
Uranus Series						√	√	x		√	√	√	√	√						√	√		√	√		√	
Asteroid Sur.						√	√	√		√	√	√	√	√						√	√		√	√		√	x
Comet Rend.						√	√	√		√	√	√	√	√						√	√		√	√		√	
Comm/Nav. Ser.			√	√		√	√			√		x	x	x						√	√		√	√		√	
Educational			x	x		x	x			√		√	√	√						√	√		√	√		√	
Auto Lunar		√								√	√	√	√	√							√		x	√	√		
Solar Orbital Pr.				x		√	√			√	√	x	x	x						√	√		√	√		√	
Gravity-Relativ.				√		√	√			√	√	√	√	√						√	√		√	√	√		
Solar Sys. Escape						√	√	√		√	√	√	√	√				√		√	√	√	√	√			√

1. Pressurized Nitrogen
2. Hydrazine Thrusters

Key -

x - From SPPA Book

√ - No Documentation on Req'ds. Judgement Only

Table A7-2. NASA Shuttle Payloads*

Experiment	Mission Mode	Monitor and Control Requirements
1. X-ray stellar astronomy (HEAO-C)	Automated revisit	Man will be required during flight for checkout functions. During flight, man will be required for setup, operation and film changes. Consumables logistics every 180 days.
2. High energy stellar astronomy (HEAO-D)	Automated revisit	Module may be manned or remotely operated from ground. If manned, this would impose a constant station-keeping requirement upon the orbiter. Man will be used to set up, operate, monitor, and analyze the results of experiments.
3. Advanced stellar telescope (LST)	Automated revisit	Real time control of target selection and mission programming is important to the objectives of the large telescope facility. Space shuttle crew will service and maintain all subsystems, resupply propellants and other expendables, prepare the module for operation, conduct module launch operations, establish initial station-keeping trajectory and, if automatic tracking fails, manually dock the module from a remote control console on the shuttle. Shuttle revisits 6-12 month intervals provides film, maintenance, repair, sensor updating. It would also allow extra-high resolution imaging during shuttle stay by using the backup film plate camera.
4. Large solar observatory	Automated revisit	Checkout while attached to shuttle. If electronic imaging is used, logistics flights about once a year. If film option used, frequent flights required. Man performs periodic servicing-maintenance, and periodic remote monitoring and control of operations. Man needed to install new equipment and perform some calibrations that could not be performed remotely.
5. IR astronomy	Sortie SS or automated revisit	Cryogenics resupply every six months. Removal of protective covers and supports required for preparation. Examine optics for damage. The telescope will be delivered to orbit prechilled with liquid neon at 27.6^{o} K. Contaminant monitors and IR background sensors of the detached vehicle will be activated and must show acceptable levels before telescope aperture cover is opened. Optical alignment is checked in the red portion of the visible spectrum. An IR astronomical source of known size and spectral distribution is used for testing the interferometer portion of the instrumentation. A number of artificial

Table A7-2. NASA Shuttle Payloads*

Experiment	Mission Mode	Monitor and Control Requirements
		IR sources, supplemented by stars, are used for calibration of the instrumentation. The built-in instrumentation consists of reference sources, radiometer, interferometer, and solid-state detector matrix test sets. Test data output is in digital format and is routed via normal monitoring circuits. Also, an additional calibration and check-out test set will be available during maintenance periods.
6. OSO-K	Automated	These spacecraft are operated automatically from ground control. Satellite checkout and launch preparation in the shuttle bay. Revisits only for servicing and repair.
7. Radio inter-ferometer telescope	Automated	EVA support of unfurlable antenna may be required. On-board checkout of payload and kickstage required.
8. Solar orbital pair	Automated	Checkout of payload and kickstage.
9. Optical inter-ferometer	Automated	Checkout of payloads (2) and kickstage. Pointing of the two spacecraft and the variation of the distances between them will be commanded directly from the ground. Revisits required once a year, primarily for maintenance, replacement of faulty components and refurbishment of detector instruments.
10. Km wave tele-scope (KWOT)	Automated	Checkout KWOT subsystems prior to releasing it. Monitor KWOT antenna deployment sequence.
11. Astronomy explorer	Automated	Deploy and checkout of the payload packages.
12. Radio astronomy explorer	Automated	Perform initial deployment and checkout of the experiment equipment. Kickstage used to boost into lunar vicinity.
13. Large radio observatory	Automated revisit	Control of checkout and deployment of large 30 foot diameter parabolic dish. Revisit for maintenance, repair, and specialized operation.

Table A7-2. NASA Shuttle Payloads*

Experiment	Mission Mode	Monitor and Control Requirements
14. Low magnetosphere	Automated	Astronauts will check out, calibrate, and replace experiments prior to release from shuttle. Probable kickstage involved.
15. Mid-magnetosphere explorer	Automated	Same as 14.
16. High magnetosphere explorer	Automated	Same as 14. Experiments 14, 15, and 16 may be launched in one shuttle flight.
17. Plasma physics lab	Automated	Same as 14. Logistics resupply as often as every 30 days may be required for propellants.
18. Gravity-relativity satellites	Automated	Satellite experiments should be checked out and calibrated prior to release from shuttle. Gyroscope satellite may require no kickstage, but redshift satellite will require a kickstage.
19. Solar system escape satellite	Automated	Astronaut will check out, calibrate, and replace experiments and experiment components prior to release from shuttle. Kickstage required.
20. Cosmic ray physics lab	Automated	The initial setup, checkout, and calibration of the equipment will require several astronauts including at least one physicist for a period of two to three days. Once the experiment is running, it will operate in an automatic mode for as long as desired. If man is physically present, daily monitoring, checkout maintenance, and calibration would be desirable, although these functions will require no more than two or three hours per day. Any malfunctioning equipment would be replaced by the astronaut. Logistics resupply of LHe and experiments update every 30 days.
21. Mars viking	Automated	Spacecraft and propulsive kickstage will be deployed, checked out.
22. Venus explorer	Automated	Same as 21.

Table A7-2. NASA Shuttle Payloads*

Experiment	Mission Mode	Monitor and Control Requirements
23. Venus radar mapping	Automated	Two days in earth orbit for checkout of spacecraft and kickstage.
24. Venus explorer/ lander	Automated	Same as 21.
25. Grand tour	Automated	Space shuttle will transport the spacecraft with a propulsion stage to a 100 nm circular orbit. As extensive a survey as possible of all critical systems will be made just prior to launch from orbit. Those functioning improperly will be replaced or repaired. Critical system checks will be made prior to the spacecraft/propulsion deployment.
26. Jupiter Pioneer orbiter	Automated	Same as 25.
27. Jupiter tops orbiter/probe	Automated	Same as 25.
28. Uranus tops orbiter/probe	Automated	Same as 25.
29. Asteroid survey	Automated	Same as 25.
30. Comet rendezvous	Automated	Same as 25.
31. Automated lunar exploration	Automated	Same as 25.
32. Communications/ navigation medical network satellite	Automated	Checkout, calibrate, maintenance on payload and kickstage prior to deployment.

Table A7-2. NASA Shuttle Payloads*

Experiment	Mission Mode	Monitor and Control Requirements
33. Educational broadcast satellite	Automated	Same as 32.
34. Communications/ navigation follow on systems demonstration	Automated	Same as 32.
35. Communications/ navigation applications and technology satellite	Automated	Same as 32.
36. Communications/ navigation small applications satellite	Automated	Same as 32.
37. Communications/ navigation cooperative application satellite	Automated	Same as 32.

Table A7-3 Payload Program Status Priority

Payload	Status & Prognosis
1. X-Ray Stellar Astronomy (HEAO-C)	Considered to be well defined. High priority assigned by OSSA.
2. High Energy Stellar Astronomy (HEAO-D)	Considered to be well defined.
3. Advanced Stellar Telescope (LST)	Experiment is well defined and is highest priority within OSSA.
4. Large Solar Observatory (LSO)	Experiment is well defined and is given high priority by OSSA.
5. IR Astronomy	Experiment is well defined and is high priority by OSSA.
6. OSO-K	The spacecraft is well defined, but the experiment is undetermined. Mission is considered high priority by OSSA.
7. Radio Interferometer Telescope	Experiment is very poorly defined and requirements are beyond state-of-art technology. Low priority within OSSA.
8. Solar Orbital Pair	Considered to be a valuable scientific endeavor, but is currently beyond state-of-the-art, and is poorly defined, consequently, no priority has been established.
9. MAST	Experiment is well defined and considered a good candidate for early missions.
10. 0.3m Schmidt U.V. Telescope	Relatively well defined. Considered a good experiment but priority has not been established.
11. Small U.V. Survey Telescopes	Well defined. Detail design completed. Very high priority within OSSA; also for early missions.
12. Solar U.V. Spectrometer	Well defined. Considered a good experiment for early mission. Priority not established.
13. Coronagraphs	The 1 to 6 and 5 to 30 solar Radii Coronagraph experiments are considered to be well defined. Priority not established.
14. Optical Interferometer	Poorly defined and beyond state-of-the-art in laser technology. Considered a good experiment but priority not established.

Table A7-3 Payload Program Status Priority (Cont.)

Payload	Status & Prognosis
15. Kilometer Wave Orbiting Telescope	Poorly defined. Considered a good experiment, but priority is not established.
16. Astronomy Explorer	Spacecraft well defined. Experiments not defined. Priority for mission is high. Experiment priority not established.
17. Radio Astronomy Explorer	Same as 16.
18. Large Radio Observatory (LRO)	Experiment is poorly defined. High priority experiment within OSSA but not for early mission.
19. Mars Viking	This well-defined program will be continuation of the present funded program.
20. Mars Sample	This program is in a very preliminary planning phase, requiring technology beyond the present state-of-art.
21. Venus Explorers	While the general program is defined, due to similarity to other programs, it is yet to be seriously funded.
22. Venus Radar Mapping	No strong support has developed yet for this poorly defined program.
23. Venus Explorer/ Lander	This program is in the tentative proposal phase and is poorly defined.
24. Grand Tour	Program no longer funded.
25. Jupiter Pioneer Orbiter	This program is presently poorly defined.
26. Jupiter TOPS Orbiter/Probe	Poorly defined and uses concepts presently beyond the state-of-the-art.
27. Uranus TOPS Orbiter/Probe	Payload is poorly defined.
28. Asteroid Survey	Program is in the early concept stage and is not defined.
29. Comet Rendezvous	Poorly defined payload has not been approved.
30. Automated Lunar Exploration	This program cannot develop support although the definition phase is continuing.
31. Lunar Base Program	No strong support is evidenced for this program.

PAYLOAD CAPABILITY

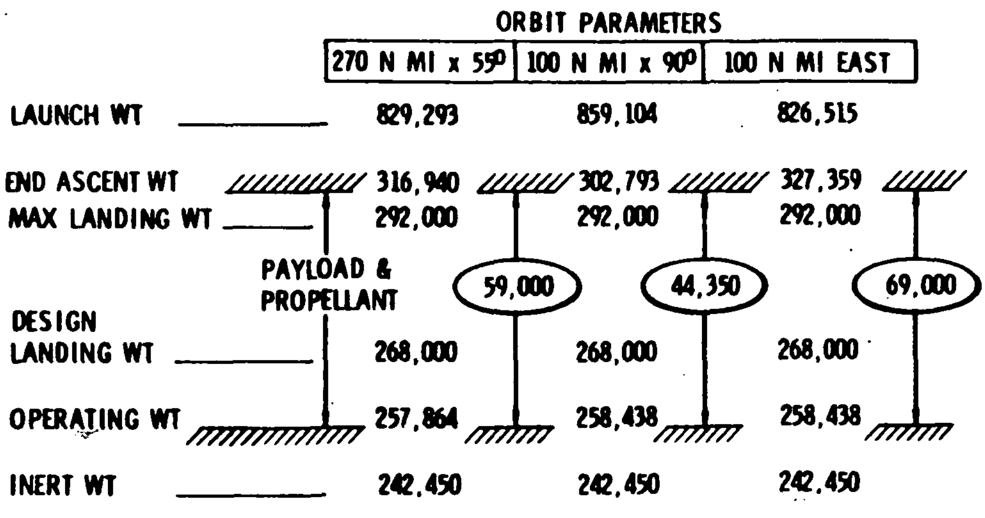
As previously mentioned, RAM operations involve the delivery to orbit and operation of a set of experiment packages, either in conjunction with the MSS or the orbiter itself, the later missions being defined as sortie missions. The development of a sortie mission payload package involves the integration of mission requirements (altitude, inclination), experiment requirements (weight, consumables) and shuttle payload capability. Because of the need to utilize as much of the shuttle payload capacity as possible while at the same time maintaining a compatible set of experiments, the integration of payload packages into a single shuttle sortie involves an extensive tradeoff around mission requirements and payload package combinations. The payload selection and tradeoff analysis is most effectively accomplished by trial and error selection and combination of payloads.

The payload selection analysis initiates with the shuttle orbiter mass properties characteristics. From the previous payload definition illustrations it is possible to compare the total payload weight requirements (experiments, subsystems, consumables, structure, etc.) with the shuttle orbiter payload capability. Should the requirement exceed the capability, then either an adjustment in mission requirements is needed (i.e., lower altitude) or experiment packages will need to be off-traded from the payload considered.

Figures A7-6 and A7-7 present a further illustration of a propellant/mission/payload trade analysis. For example, Figure A7-6 indicates that the total deliverable payload (payload up only) to 100 nm is 37,200 lbs. In achieving that altitude there remains 24,700 lbs of OMS propellant tank capacity available for meeting attitude hold requirements of an experiment program. Suppose the experiment program required a 0.5 degree attitude hold deadband, then there exists the capability for 64 hours of attitude hold. Now, the propellant required to support the attitude hold must be charged against the payload capability of 37,200, thus reducing the payload capability to 12,500 lbs. Should the actual payload weight be greater than the allotted 12,600 lbs, then a reduction in attitude hold time (with a corresponding decrease in OMS propellant) will result in added payload weight capability. Figure A7-7 illustrates a similar trade analysis for a shuttle orbiter attitude of 270 nm. Here the maximum payload capability is 26,800 (with no attitude hold requirement) and a minimum payload capability of 12,200 lbs (with 32 hours of 0.5 degree attitude hold capability).

Figure A7-1 summarizes the shuttle's mass properties characteristics for each of the three reference missions (28.5, 55, and 90-degree orbit inclinations) and indicates the allowable weight for payload and propellant in each case.

The propellant needed for each mission is that required to achieve the final circular orbit altitude above the 50- by 100-nautical mile orbit. In addition, since the shuttle orbiter attitude control propulsion system (ACPS) utilizes the same propellant tanks as the (orbital maneuvering system OMS) the total propellant weight requirement must also include the propellant needed to maintain vehicle attitude. Since the propellant requirement varies with each sortie mission, the amount of propellant needed must be calculated for each mission. However, there are a certain number of maneuvers that are common to all sortie missions and these are defined in Table A7-4 together with their propellant requirements.



INERT WT = DRY WT + FLIGHT CREW (2) + RESIDUALS
OPERATING WT = INERT WT + RESERVES + IN-FLT LOSSES & ABES PROPELLANT

Figure A7-1 Mass Properties Characteristics

Table A7-4. Common Mission Maneuvers and Propellant Requirements

Event	Propellant Weight (lb.)
Orbit injection (50 x 100 n. mi.)	400
Deorbit (from 100 n. mi.)	360
Preentry	150
Entry	1200
Total	2110

The shuttle's payload capability as a function of attitude for each of the three reference missions is presented in Figures A7-2 through A7-4 Accompanying the payload capability is a definition of the corresponding propellant required to achieve the altitude desired. There are constraints imposed on the shuttle that place certain restrictions on its payload capability which are included in the curves of Figures A7-2 through A7-4. These are defined as:

1. In order to have the capability of a "once-around" orbiter abort, there must be at least 1000 feet per second ΔV propulsion available. This requires 20, 000 pounds of OMS propellant.

2. The orbiter bay has a structural limit of 65, 000 pounds of payload. Therefore, no payloads above 65, 000 pounds are permissible, regardless of the ΔV capability.

3. The orbiter has a design landing rate of sink requirement of 10 feet per second. This, in turn, imposes a landing weight limit on the total orbiter vehicle which results in a limitation on the "down" payload weight for certain cases (particularly the due east launch, 28. 5 degrees).

As previously mentioned, in order to maintain attitude, the orbiter utilizes propellant from the OMS/ACPS propulsive ΔV tanks. Figure A7-5 provides a means of computing the amount of propellant needed to achieve various attitude deadbands used in this analysis.

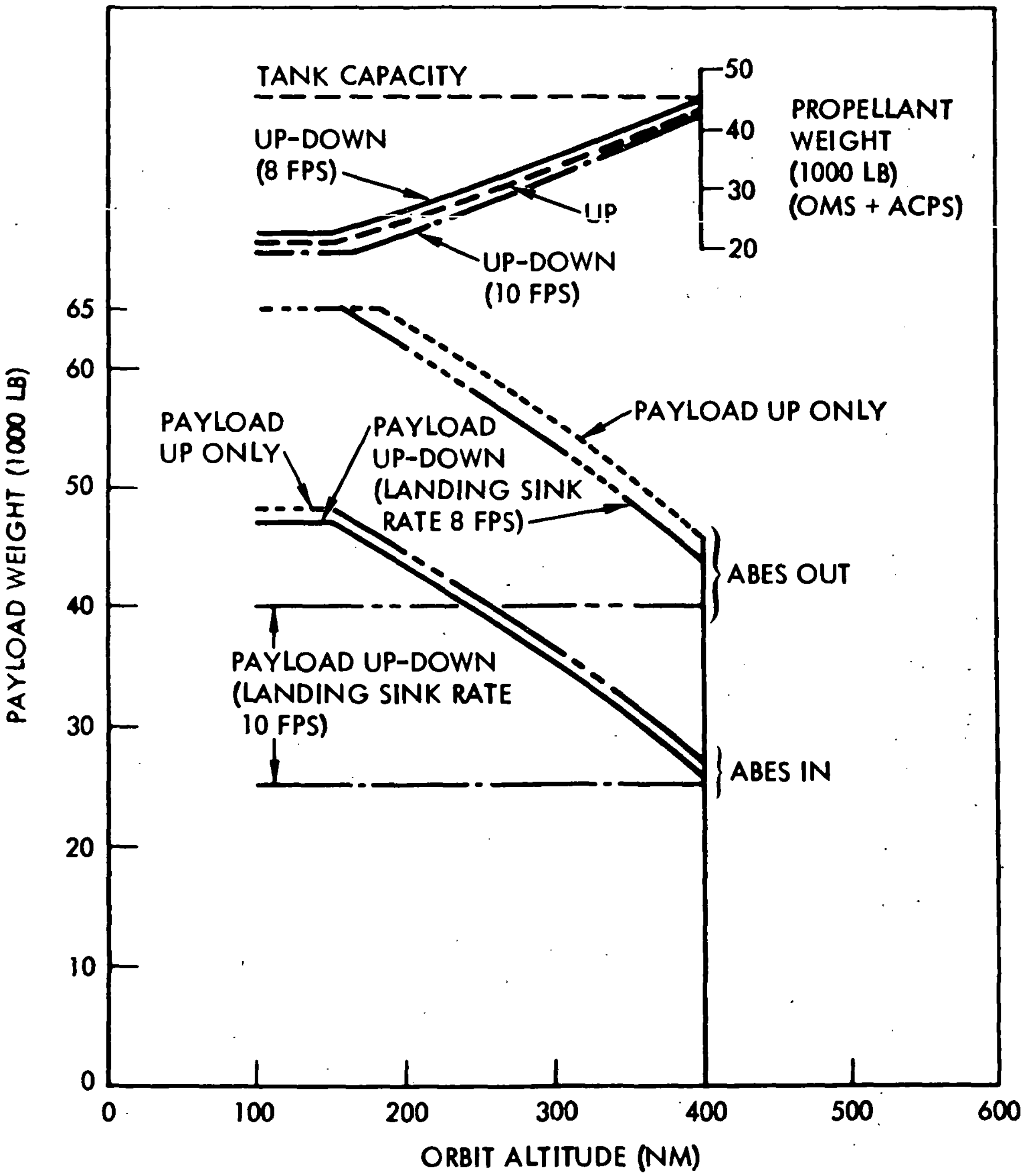


Figure A7-2. Payload Capability-Propellant Requirements:
28.5° Inclination Angle

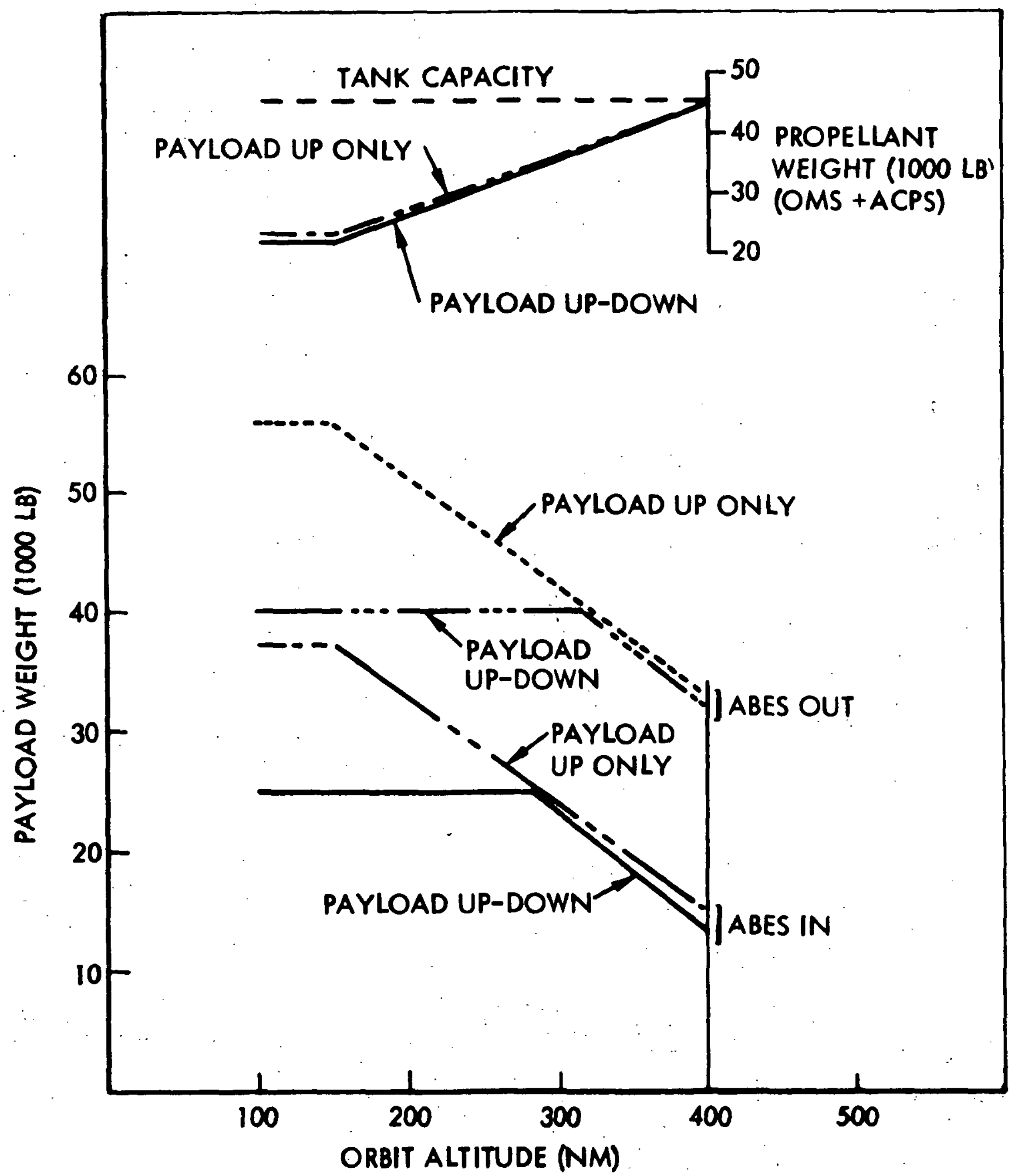


Figure A7-3. Payload Capability-Propellant Requirements: 55° Inclination Angle

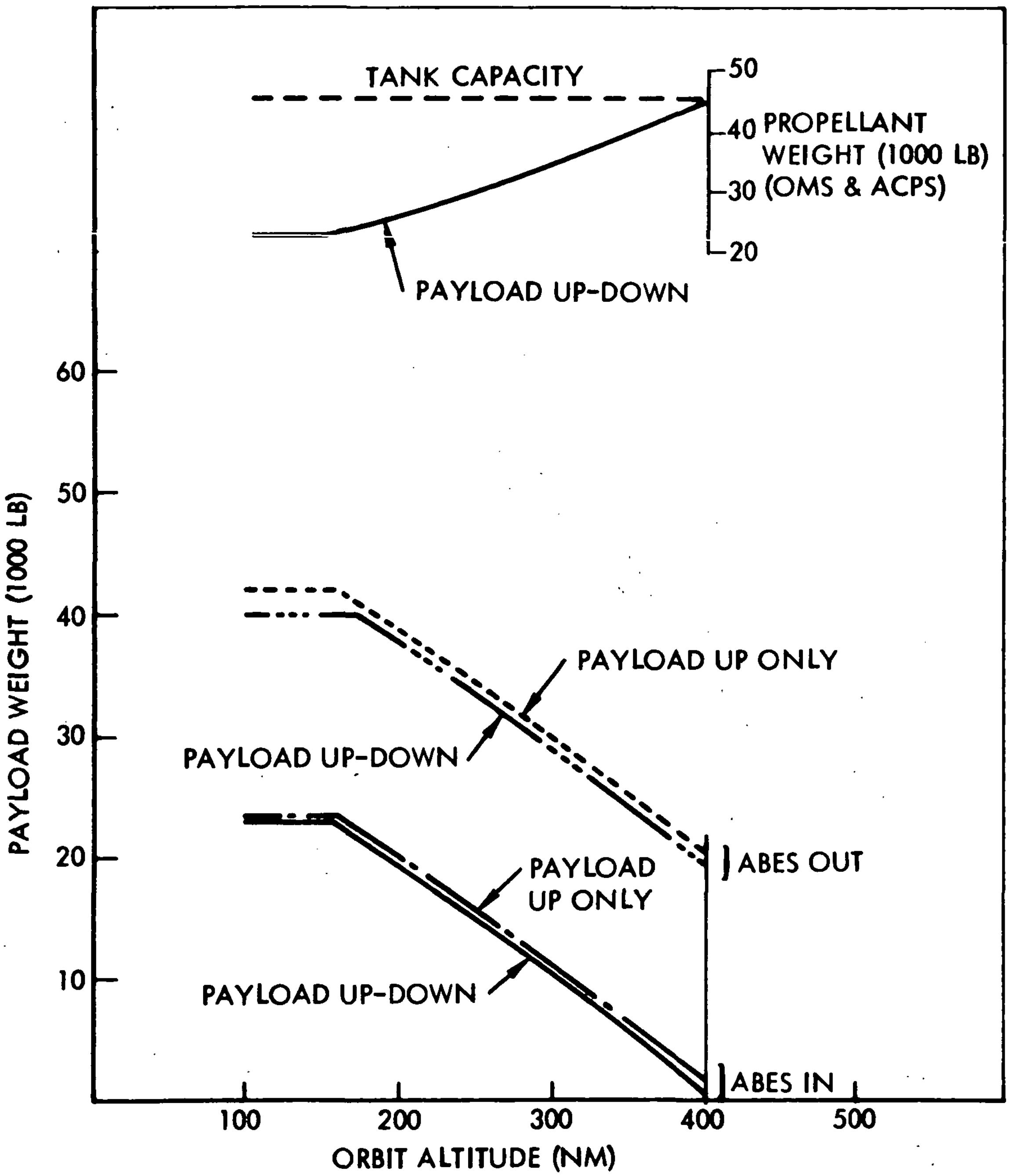


Figure A7-4. Payload Capability-Propellant Requirements:
90° Inclination Angle

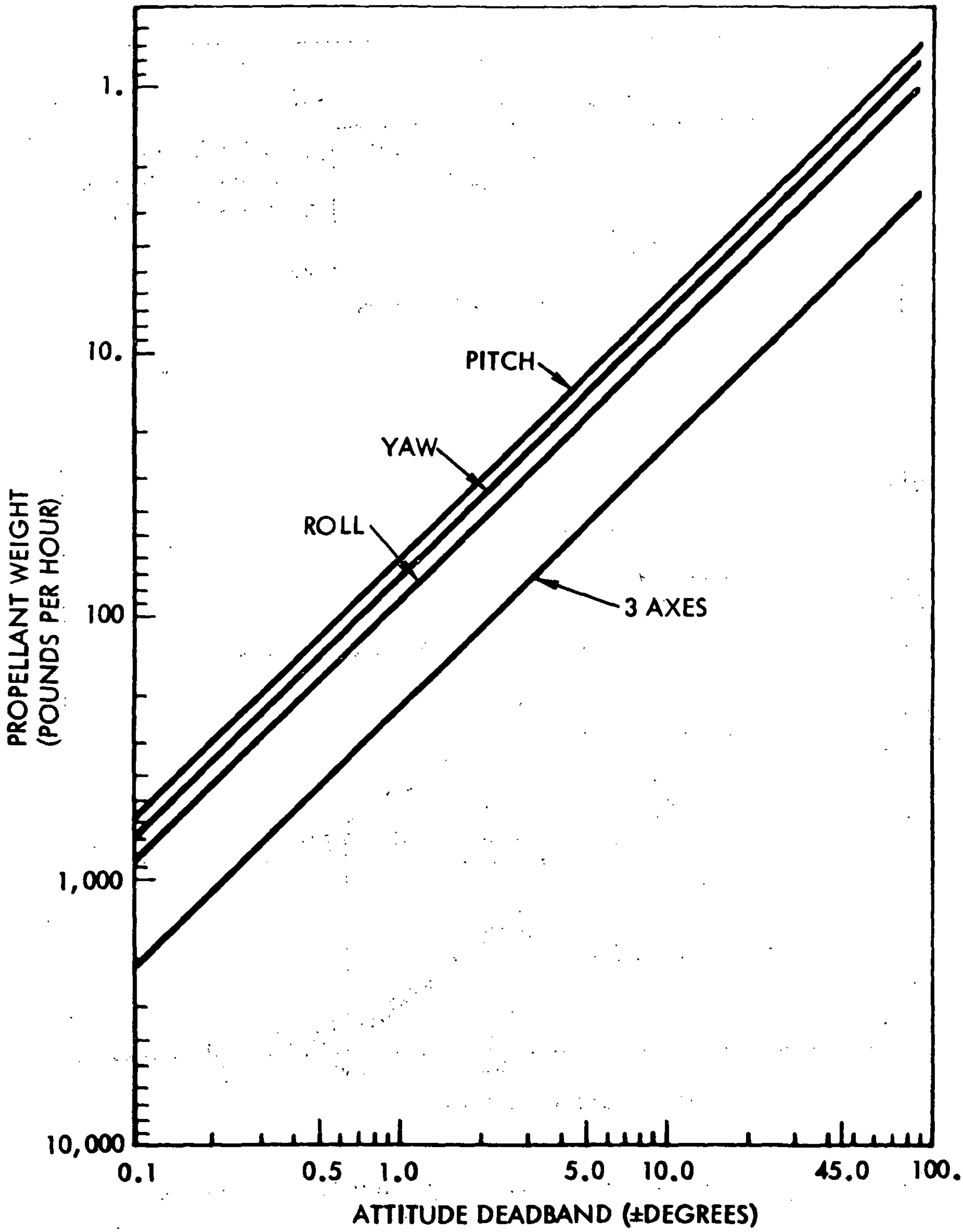
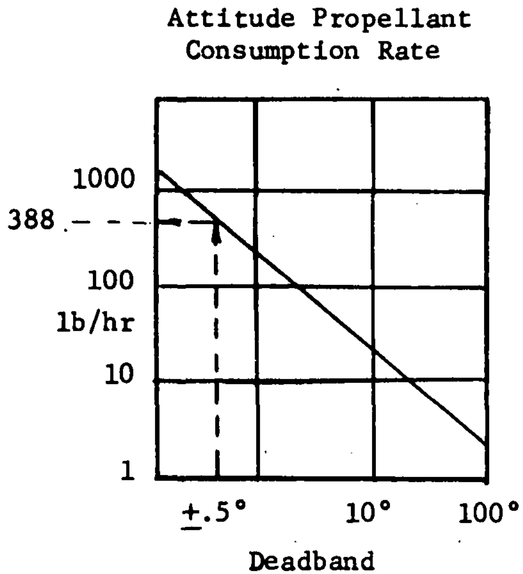
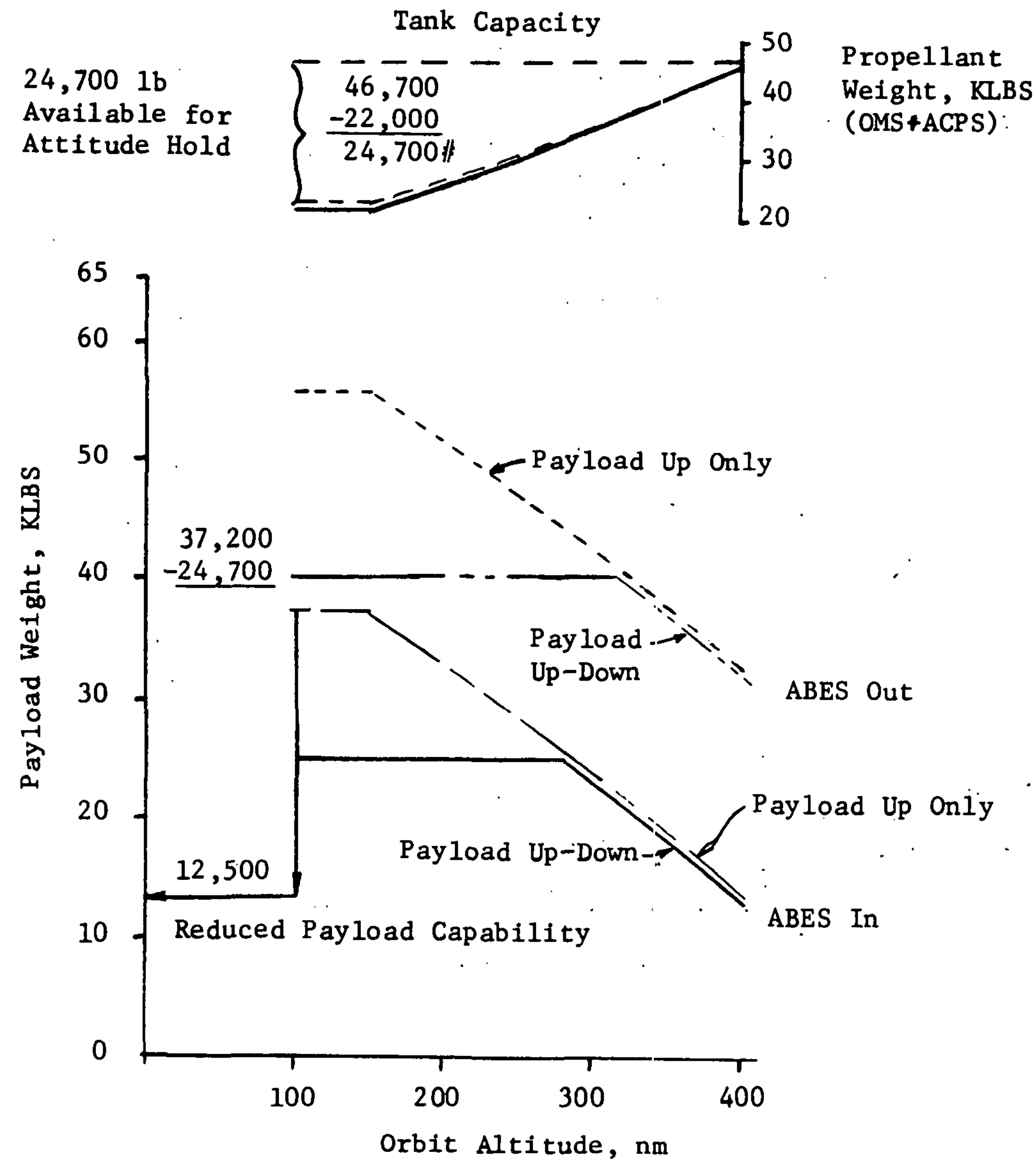


Figure Attitude Hold, ACPS Propellant Requirement





$$\frac{24,700}{388} = 64^{*} \text{ Hours of } 0.5° \text{ Attitude Hold}$$

*Unused abort propellant (2300 lb) provides an additional 6 hours of attitude hold capability

Figure A7-6. Payload/Propellant Trade (100 nm, 55°)

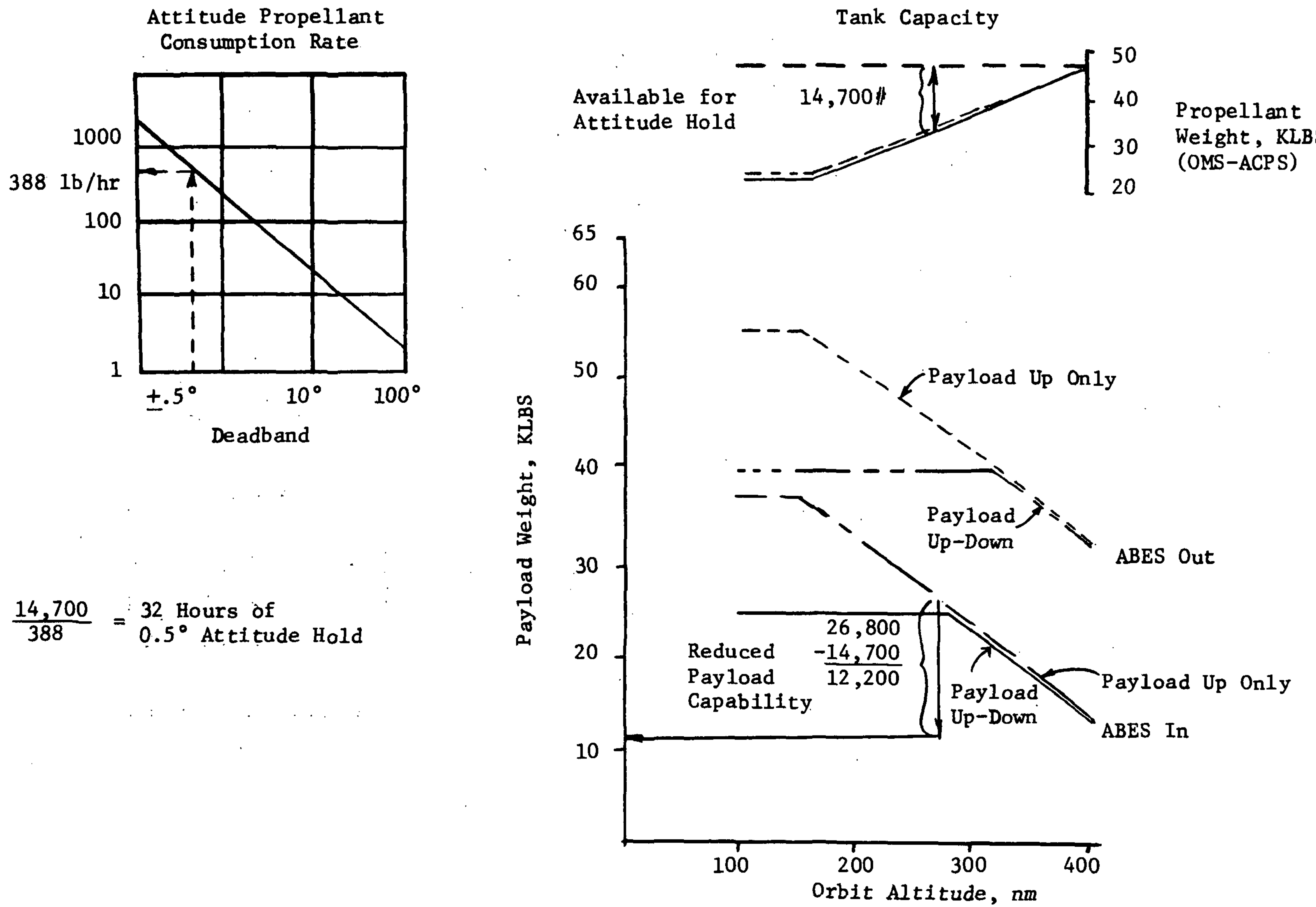


Figure A7-7. Payload/Propellant Trade (270 nm, 55°)

A8 DOCKING AND STRUCTURAL INTERFACE ASSESSMENT

This analysis study was conducted to support the Attached Element Transport interfacing activity (Volume II, Part 4, Section 5.0). Four transport vehicles (EOS, TUG, CPS and RNS) and their associated payloads were analysed for the following:

1. Determine masses, dimensions, c.g. locations, and dynamic loads of various potential payloads that impose requirements upon the four transporting vehicles.

2. Determine the need, if any for additional rigidization to supplement docking port structural interface.

3. Investigate the feasibility of a universal docking concept for all element pairs. Determine if any supplemental rigidization is required to the docking port structural interface as a result of the transport operation.

SUMMARY

Both docking and structural interface loads were estimated for all mating configurations to establish the requirements for structural attachment. The docking force time history plots as a function of stiffness and damping of the docking mechanism were developed to approximate the design loads for various docking concepts. The relatively low magnitude of these docking forces are considered to be well within the capability of the candidate docking concepts and therefore do not influence concept selection.

Whenever a transport vehicle is used to propel a payload between orbital positions, the linking interface experiences an added load due to the applied thrust. To estimate such added interface load, it was assumed that the maximum thrust is applied through the c.g. of the combined vehicles and that the vehicles are essentially rigid bodies. The results of this analysis shows the CPS/OLS case to exceed the practical design loads of the candidate docking concepts. The TUG and RNS used as delivery vehicle impose loads well within docking port design capabilities.

Four docking concepts were selected as potential candidates from a cursory evaluation of many concepts. These concepts are (1) international docking adapter, (2) multiple probe and drogue, (3) ring and cone, and (4) square frame and guidance latch. All four concepts can be designed to meet the defined requirements. The results of this assessment show that the square frame docking concept is the most desirable. This selection is based primarily on the ability to meet all requirements, permit light weight design and achieve high reliability with the lowest cost. However, since

the international concept is being studied in detail and is acceptable for the universal application, final concept selection should be based on the results of that study. This assessment was performed by comparing trade factors in the areas of technical performance, producibility, operations and relative costs.

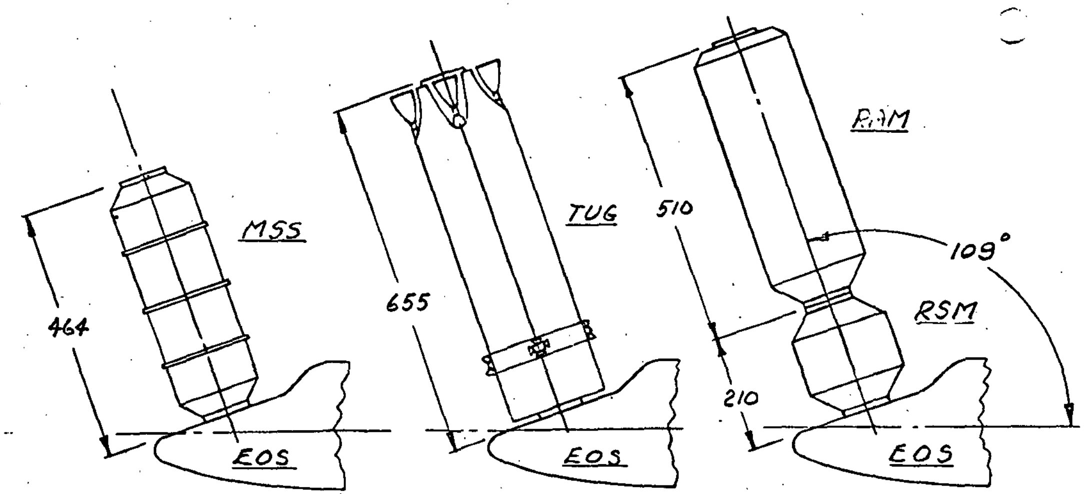
MATING CONFIGURATIONS

The transporter and mating payloads considered in this evaluation are listed in the following table A8-1.

Table A8-1. Element Pairs for Transportation

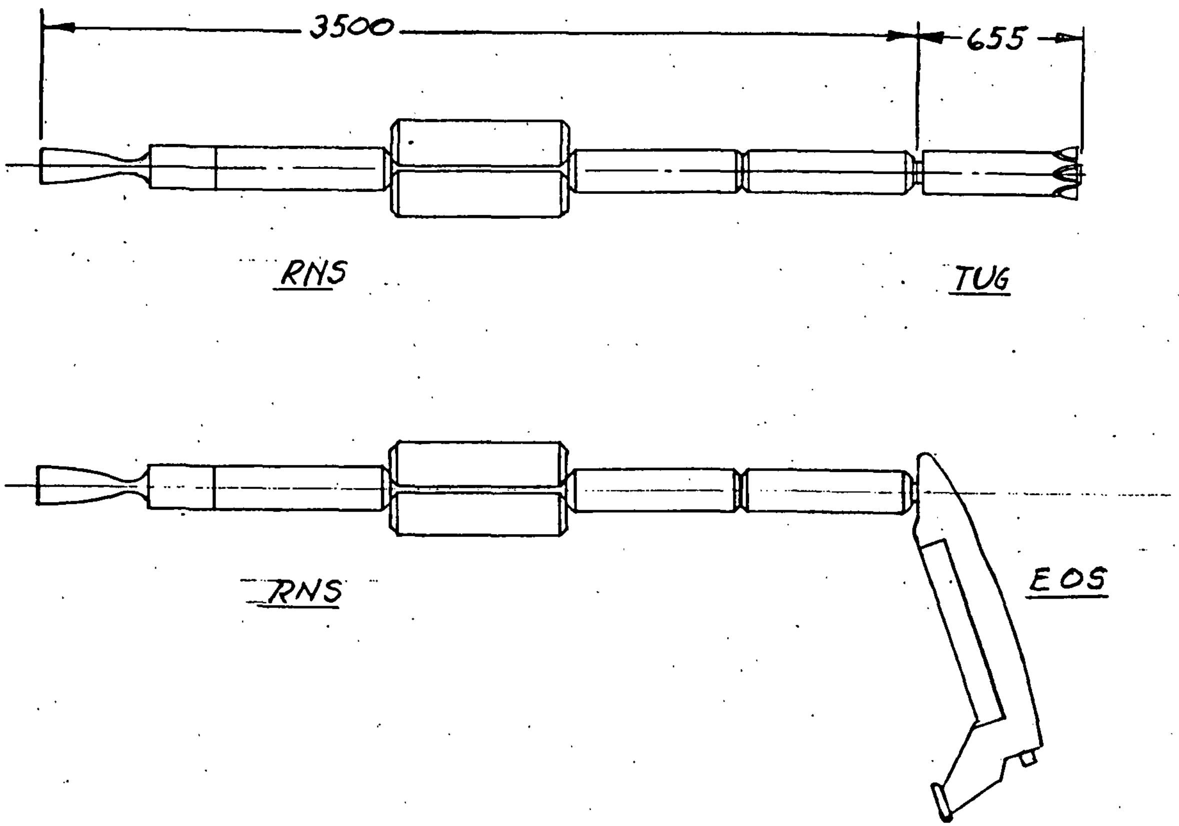
Transporter	Payload
Earth Orbital Shuttle (EOS)	Research Application Module (RAM) Space Tug Satellites Modular Space Station (MSS) Reusable Nuclear Stage (RNS) Orbital Propellant Depot (OPD) EOS Tug and Module
Space Tug	RAM MSS RNS OPD Space Tug
Chemical Propulsion Stage (CPS)	Orbital Lunar Station (OLS)
Reusable Nuclear Stage (RNS)	OLS

Weights, centers of gravity, and moments of inertia along with pictorial descriptions for the above vehicle combinations are presented in Figures A8-1 through A8-8. A typical small satellite (Intelsat) together with its upper stage booster is shown in Figure A8-11. This type of satellite can be delivered or retrieved from orbit without the need for an androgynous docking system. The upper stage booster could be equipped with a small docking fixture for use with small satellites only. This special docking fixture could be in the form of a kit that would temporarily replace or attach to the universal fixture. Therefore, no attempt has been made to consider the small satellite in the selection of the universal docking concept.



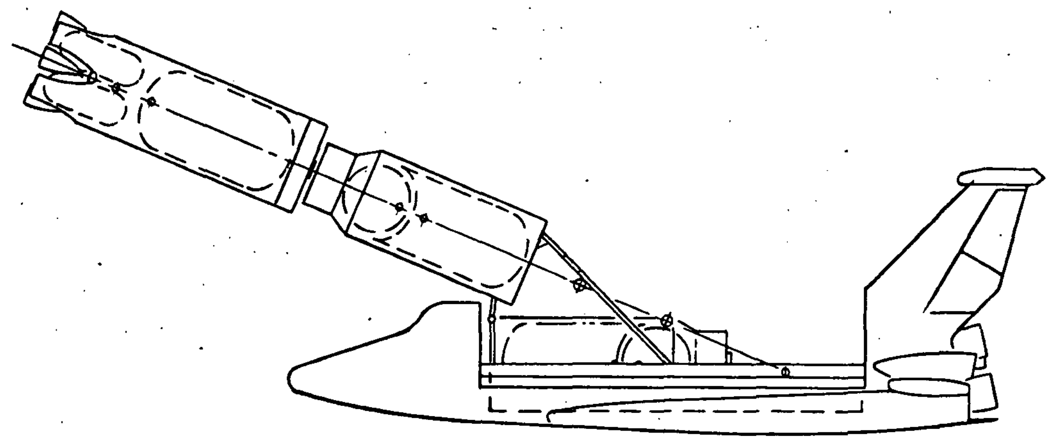
CONFIGURATION	WEIGHT LB	C.G. IN INCHES			MOMENT OF INERTIA - SL FT^{2}X 10^{-6}		
		X LONG	Y LAT	Z VERT	I_{XX} ROLL	I_{YY} PITCH	I_{ZZ} YAW
EOS - ORBITER ONLY	176370	1127	0	-23	1.05	5.72	6.13
RAM/RSM	32000	221	0	345			
EOS/RAM/RSM	208370	988	0	33	2.14	11.65	10.99
TUG	98550	184	0	452			
EOS/TUG	274920	789	0	147	4.78	21.66	18.38
MSS	20000	252	0	252			
EOS/MSS	196370	1038	0	5	1.44	9.09	9.13

Figure A8-1. EOS Docked to MSS Tug and RSM



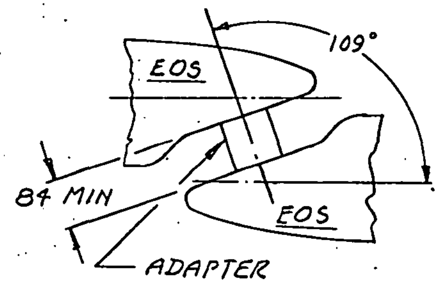
CONFIGURATION	WEIGHT LBS.	C.G. IN INCHES			INERTIA IN SLUG FT² x 10^{-6}		
		X LONG	Y LAT	Z VERT	I_{xx} ROLL	I_{yy} PITCH	I_{zz} YAW
RNS/TUG	485325.0	1828.2	0.0	0.0	0.72	198.05	198.05
RNS/EOS	563148.0	1969.0	0.0	0.0	1.63	211.92	212.33
RNS	386775.0	1361.1	0.0	0.0			
TUG	98550.0	3661.6	0.0	0.0			
EOS	176373.0	3302.0	0.0	0.0			

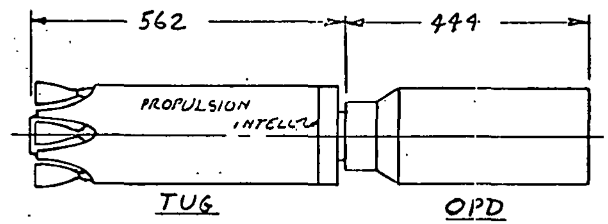
Figure A8-2. RNS Docked to Tug and EOS



CONFIGURATION	WEIGHT LBS.	CENTER OF GRAVITY			MOMENTS OF INERTIA		
		X LONG INCHES	Y LAT INCHES	Z VERT INCHES	I_{xx} ROLL	I_{yy} PITCH	I_{zz} YAW
					SLUG-FT^2 X 10^{-3}		
ORBITER/OPD STOWED	248000	1095	0	-17	1154	6080	6445
ORBITER/OPD DEPLOYED	248000	938	0	89	3091	14618	13087
ORBITER/OPD FULL/TUG EMPTY	257000	900	0	110	3786	17539	15335
ORBITER/OPD 1/2 PROP/TUG 1/2 PROP	257000	835	0	149	5662	25189	21106
ORBITER/OPD EMPTY/TUG FULL	257000	789	0	177	6880	30189	24894

Figure A8-3. Orbital Propellant Transfer





CONFIGURATION	WEIGHT LBS	C.G. INCHES			INERTIA - SLUG FT x 10^-6		
		X LONG	Y LAT	Z VERT	Ixx ROLL	Iyy PITCH	Izz YAW
EOS/ADAPTER/EOS	357746.0	172.0	0.0	60.6	2.64	81.75	82.03
EOS ORBITER (T)	176373.0	-787.0	0.0	145.0			
EOS ORBITER (B)	176373.0	1127.2	0.0	-23.0			
ADAPTER	5000.0	320.0	0.0	42.0			

EOS

CONFIGURATION	WEIGHT LBS	C.G. INCHES			INERTIA - SLUG FT² x 10^-6		
		X LONG	Y LAT	Z VERT	Ixx ROLL	Iyy PITCH	Izz YAW
TUG/OPD	159183.0	494.7	0.0	0.0	0.14	3.70	3.70
TUG	84556.0	223.4	0.0	0.0			
OPD	74627.0	802.0	0.0	0.0			

TUG/OPD

Figure A8-4. Tug Docked to OPD-EOS Docked to EOS

CONFIGURATION	WEIGHT LB	C.G. INCHES			INERTIA - SLUG FT^2 X 10^{-6}		
		X LONG	Y LAT	Z VERT	I_{XX} ROLL	I_{YY} PITCH	I_{ZZ} YAW
EOS/RSM/RAM/TUG	306920	656	0.0	381	2.73	6.19	3.67
EOS	176370	1127	0.0	-23	1.05	5.72	6.13
RSM	12000	295	0.0	132			
RAM	20000	177	0.0	473			
TUG	98550	-46	0.0	1116			

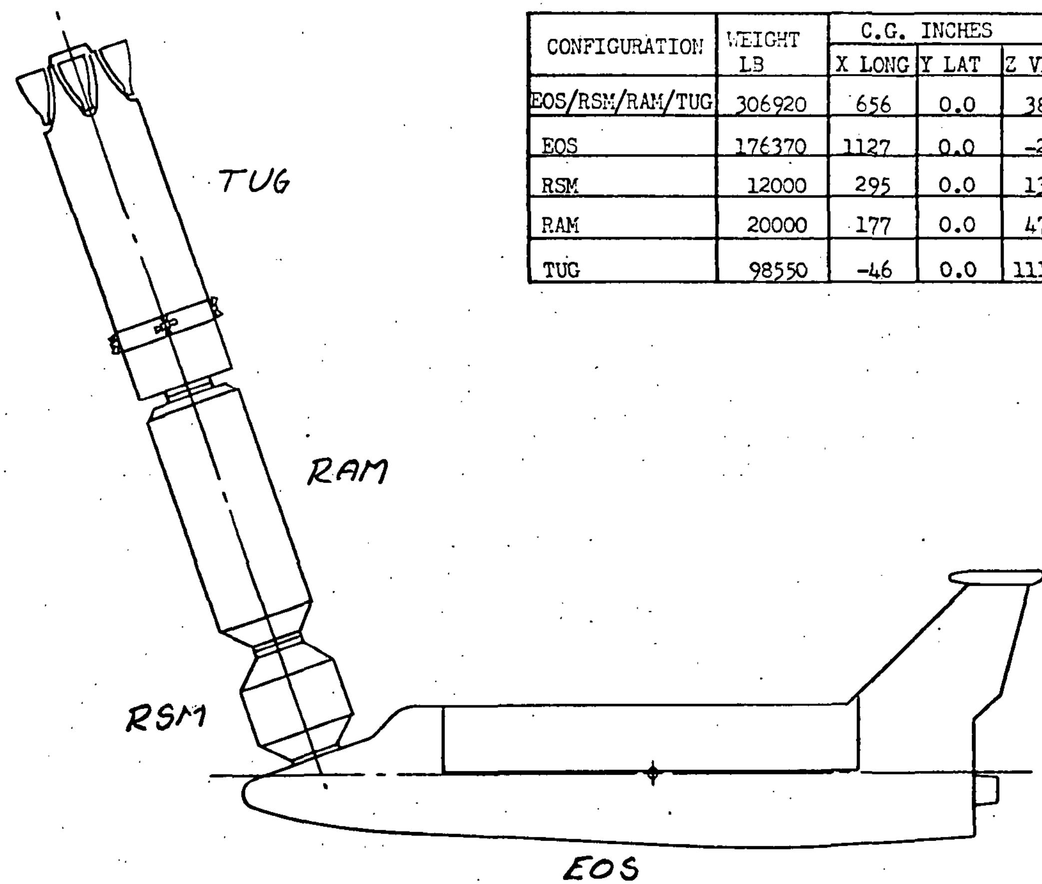
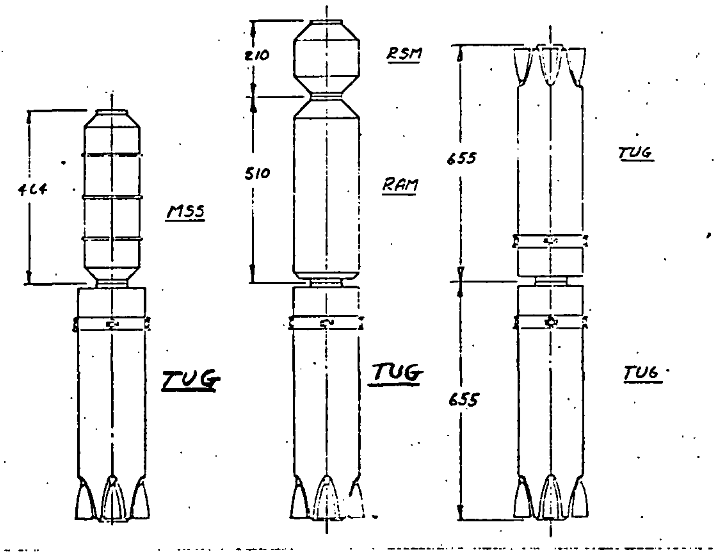
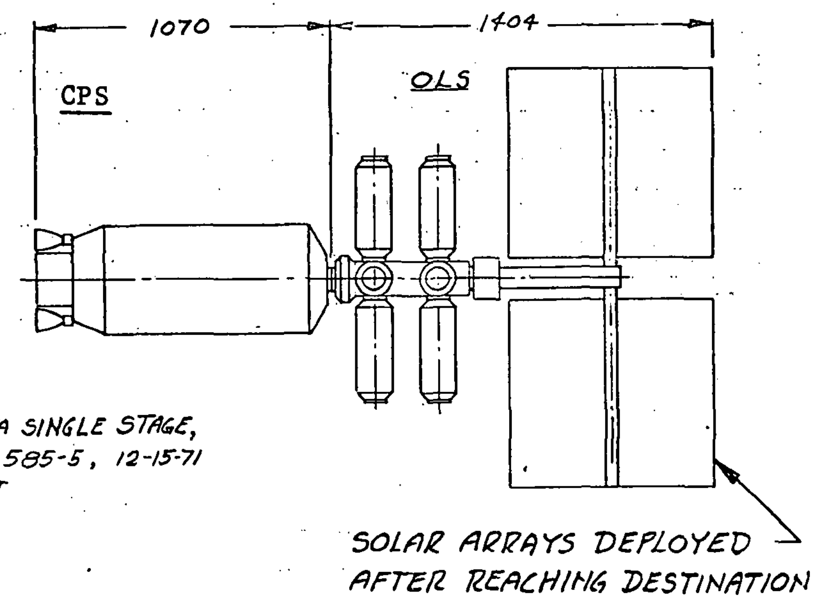


Figure A8-5. EOS Docked to Tug and RAM



CONFIGURATION	WEIGHT lbs	C.G. INCHES			INERTIA - SLUG $FT^2 \times 10^{-6}$		
		X LONG	Y LAT	Z VERT	I_{XX} ROLL	I_{YY} PITCH	I_{ZZ} YAW
TUG/TUG	197100.0	655.0	0.0	0.0	0.17	7.89	7.89
TUG/RAM/RSM	130550.0	452.7	0.0	0.0	0.19	4.18	4.09
TUG/MSS	118550.0	369.5	0.0	0.0	0.18	2.24	2.16
TUG	98550.0	260.4	0.0	0.0			
RAM	20000.0	910.0	0.0	0.0			
RSM	12000.0	1270.0	0.0	0.0			
MSS	20000.0	907.0	0.0	0.0			

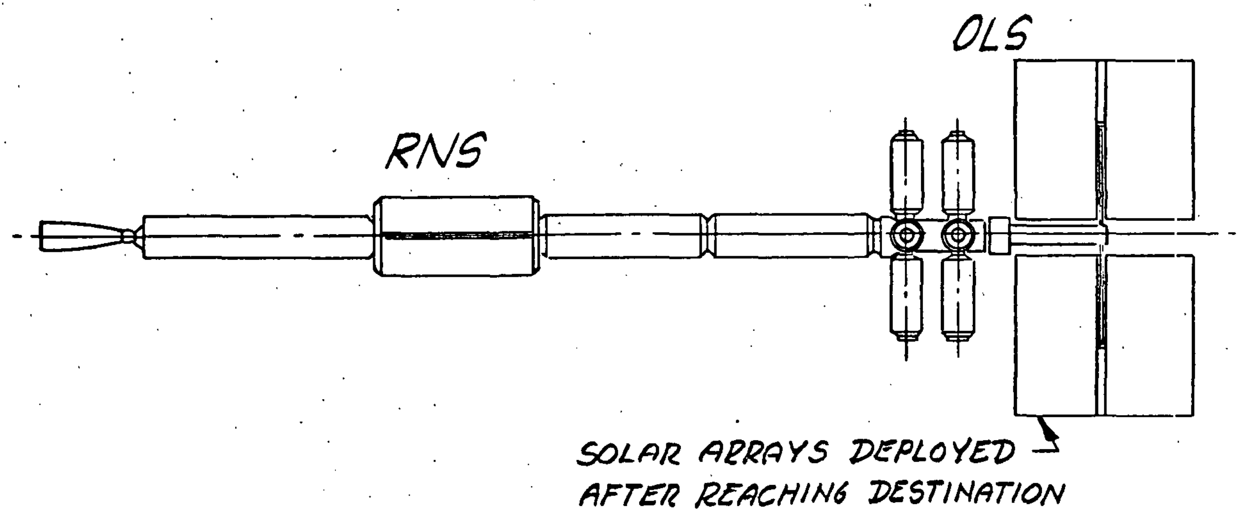
Figure A8-6. Tug Docked to MSS RAM and Tug



REFERENCE :
CIS DWG 2315-16, TTA SINGLE STAGE,
PHASE A STUDY SD 71 585-5, 12-15-71
5th MONTHLY REPORT

	CONFIGURATION	WEIGHT LBS	C.G. INCHES			INERTIA - SLUG FT^2 x 10^{-6}		
			X LONG	Y LAT	Z VERT	Ixx ROLL	Iyy PITCH	Izz YAW
	CIS/OLS-IGNITION	1202050	473	0	4.8	4.05	44.02	43.65
	CIS/OLS-BURNOUT	272050	986	1	2.4	4.02	19.72	19.37
	CIS - FULL	1034850	329	0	0	0.65	6.89	6.92
	CIS - BURNOUT	104850	379	3	-2			
	OLS	167200	1366	0	36			

Figure A8-7. CPS Docked to OLS



CONDITION	WEIGHT	C.G. IN INCHES			INERTIA IN MILLIONS SLUG FEET^2		
		X LONG	Y LAT	Z VERT	I_{XX} ROLL	I_{YY} PITCH	I_{ZZ} YAW
GROSS - RNS IGNITION	553975	2034	0	11	4.03	236.49	236.10
GROSS - FWD TK DRY	518150	1978	0	12	4.04	226.19	225.80
GROSS - 2 FWD TKS DRY	482325	1968	0	13	4.05	221.04	220.64
GROSS - 6 TKS DRY	339020	2210	0	18	3.63	186.68	186.29
GROSS - END BOOST	266400	2700	0	23	3.62	115.03	114.66
OLS ONLY	167200	3589	0	36			

Figure A8-8. RNS Docked to OLS

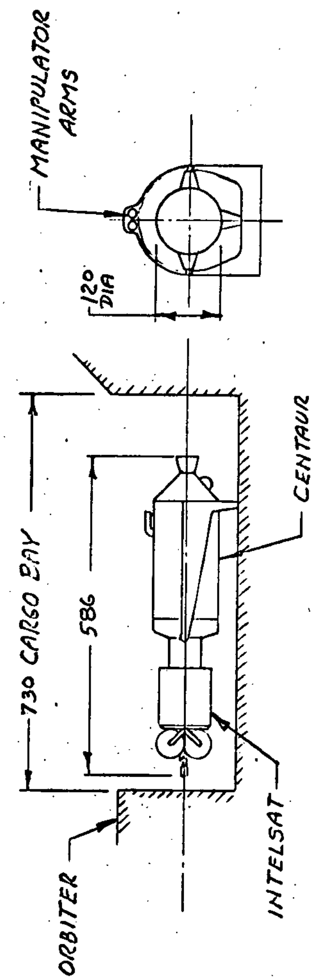


Figure A8-9. Satellite in EOS Cargo Bay

DOCKING LOADS

The dynamic loads for the docking mechanism can be obtained from an analysis which includes the dynamic equations of the docking mechanism as well as the dynamic equations of the associated vehicles. However, for the purpose of preliminary design, it is more desirable to estimate the docking load by modeling the active and inactive vehicles as two masses and the docking mechanism as spring and damper system. Accordingly, the approximate force equation has the following form:

$$\ddot{X}_T = e^{\alpha t} \ddot{X}_{TO} \left(\text{Cos } wt + \frac{\alpha^2 - w^2}{2\alpha \quad w} \sin wt\right)$$

where

$$\alpha \pm iw = \frac{1}{2M_T M_S}\left[- D(M_T+M_S) \pm \left(-D^2(M_T+M_S)^2 - 4K\ M_T M_S (M_T+M_S)\right)^{1/2}\right]$$

D = Damping constant in lb-sec/ft

K = Spring constant in lb/ft

$$\ddot{X}_{TO} = \frac{D(\dot{X}_{SO} - \dot{X}_{LO})}{M_T}$$

M_T = Mass of active vehicle

M_S = Mass of inactive vehicle

$\dot{X}_{SO} - \dot{X}_{LO}$ = Initial closing velocity

Analyses were conducted among the four transport vehicles and their associated payloads and representative results are shown in Figures A8-10, A8-11, A8-12, and A8-13. These figures show the docking force time history as a function of stiffness and damping of the docking mechanism and may be used for approximate design loads for various docking concepts. The relatively low magnitude of these docking forces are considered to be well within the capability of the candidate docking concepts and do not influence the concept selection

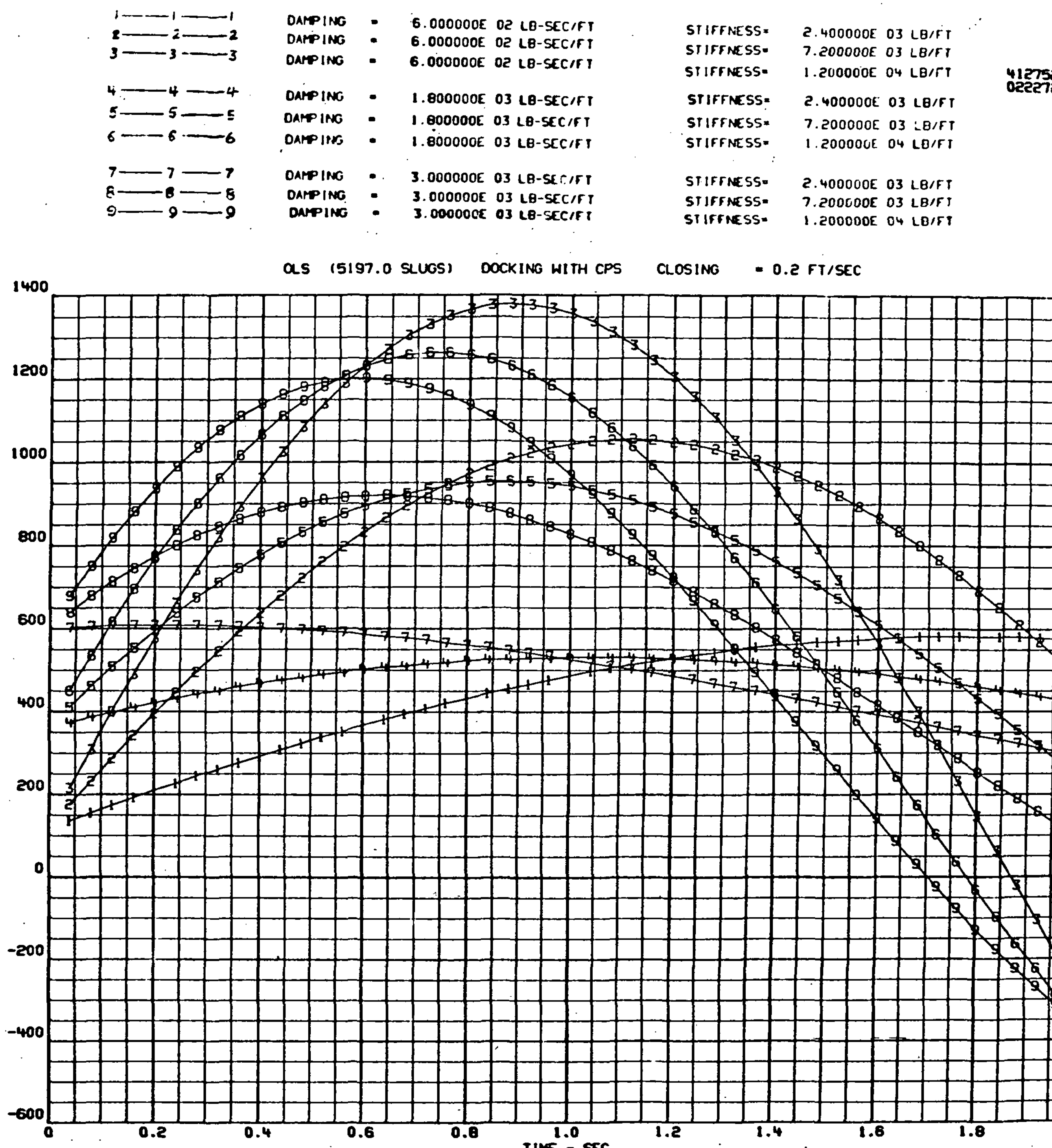


Figure A8-10. CPS/OLS Docking Loads

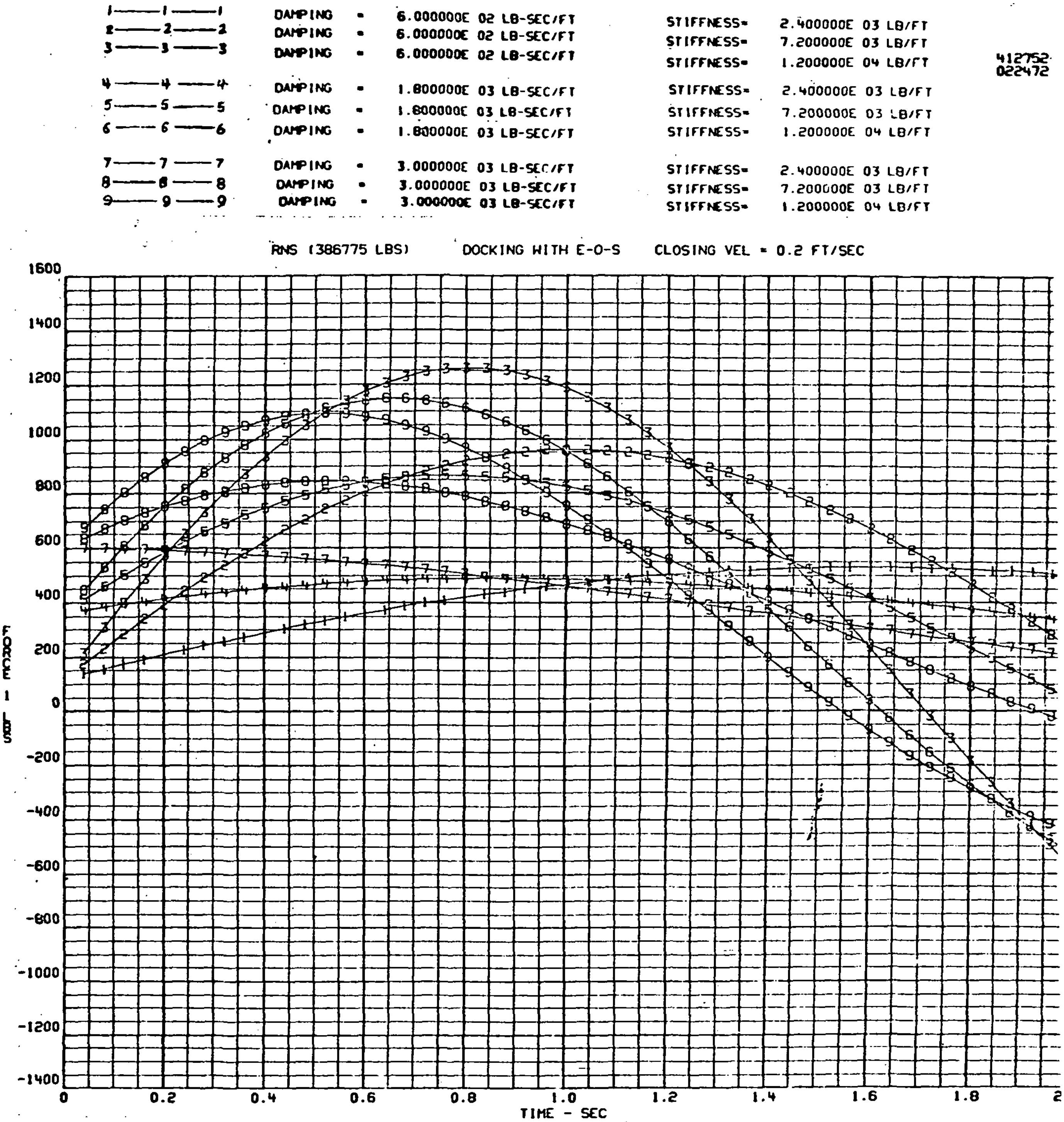


Figure A8-11. RNS/OLS Docking Loads

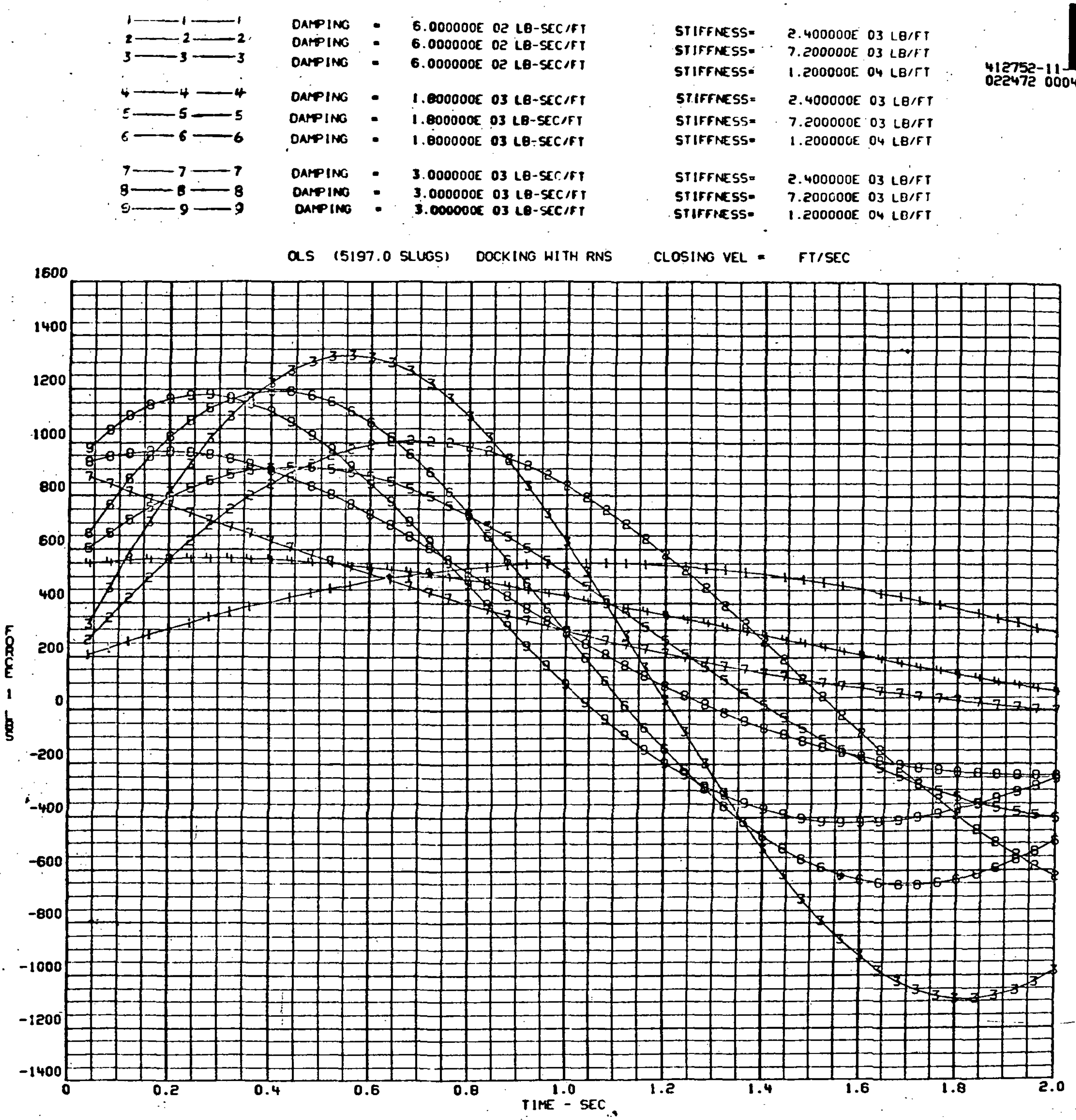


Figure A8-12. EOS/RNS Docking Loads

1——1——1	DAMPING = 6.000000E 02 LB-SEC/FT	STIFFNESS= 2.400000E 03 LB/FT	
2——2——2	DAMPING = 6.000000E 02 LB-SEC/FT	STIFFNESS= 7.200000E 03 LB/FT	
3——3——3	DAMPING = 6.000000E 02 LB-SEC/FT	STIFFNESS= 1.200000E 04 LB/FT	412752-11 022472 0003
4——4——4	DAMPING = 1.800000E 03 LB-SEC/FT	STIFFNESS= 2.400000E 03 LB/FT	
5——5——5	DAMPING = 1.800000E 03 LB-SEC/FT	STIFFNESS= 7.200000E 03 LB/FT	
6——6——6	DAMPING = 1.800000E 03 LB-SEC/FT	STIFFNESS= 1.200000E 04 LB/FT	
7——7——7	DAMPING = 3.000000E 03 LB-SEC/FT	STIFFNESS= 2.400000E 03 LB/FT	
8——8——8	DAMPING = 3.000000E 03 LB-SEC/FT	STIFFNESS= 7.200000E 03 LB/FT	
9——9——9	DAMPING = 3.000000E 03 LB-SEC/FT	STIFFNESS= 1.200000E 04 LB/FT	

RNS (3083.0 SLUGS) DOCKING WITH TUG CLOSING VEL = FT/SEC

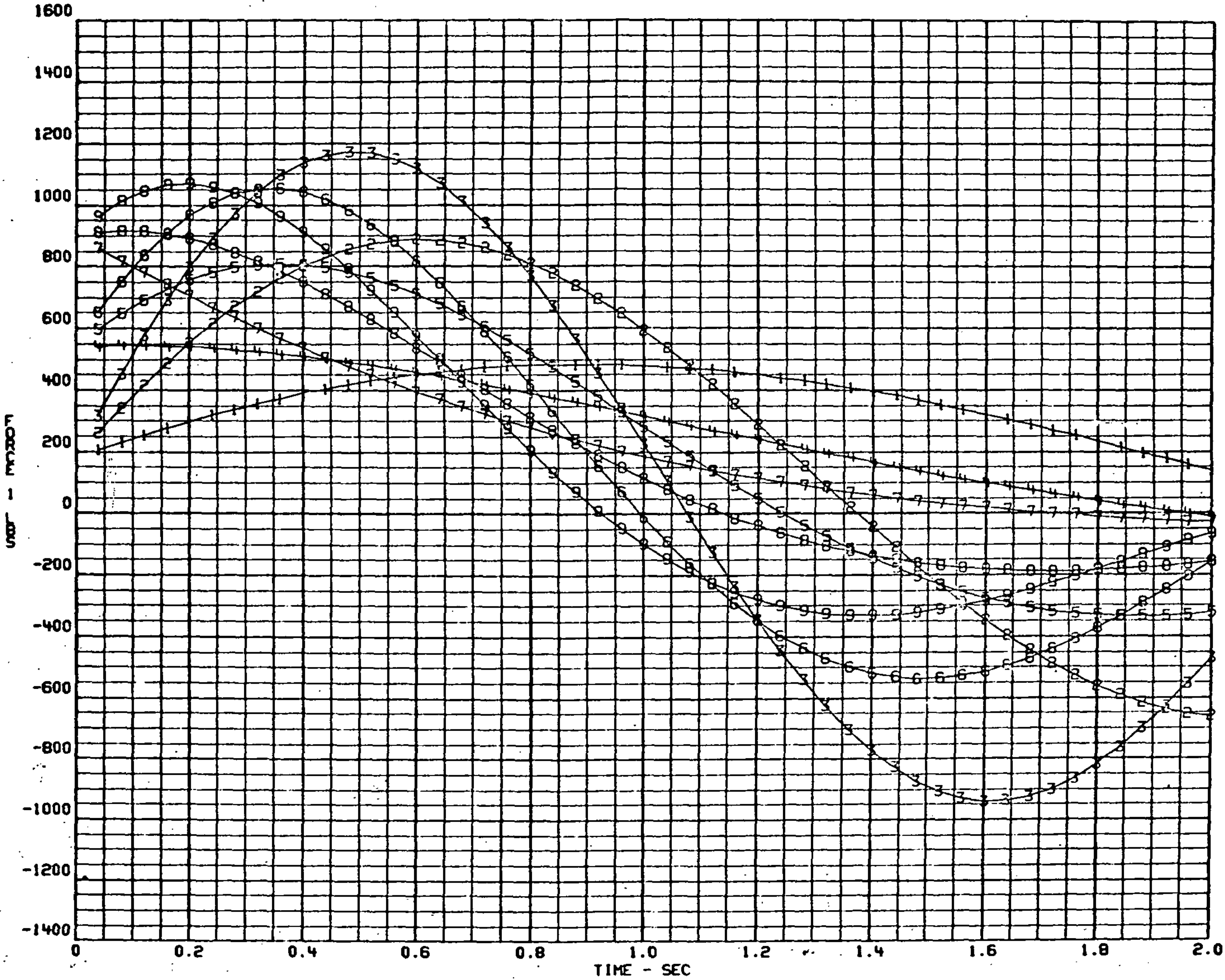


Figure A8-13. Tug/RNS Docking Loads

INTERFACE STRUCTURAL LOADS

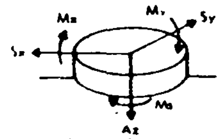
Whenever a transport vehicle is used to propel a payload from one position to another while in orbit, the linking interface experiences an added loading due to the applied thrust. To estimate such added interface load, it was assumed that the maximum thrust of a given transport is applied through the center of gravity of the combined vehicle and that the vehicles are essentially rigid bodies. The result is summarized in Table A8-2.

Table A8-2. Interface Loads

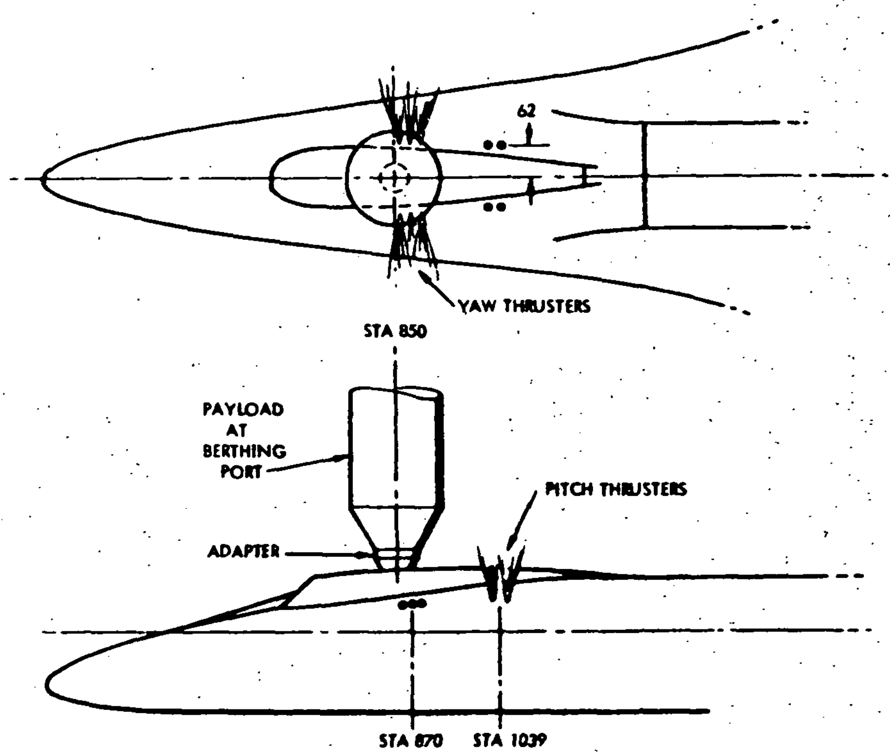
Configuration	Thrust (Lbs x 10^{-3})	Axial Load at Interface (Lbs x 10^{-3})
Tug/Tug	70.2	35.1
Tug/RNS		55.9
Tug/MSS		11.8
Tug/RAM		11.8
CPS/OLS (Fully Fueled)	960.0	133.6
CPS/OLS (Nearly Empty)		590.0
RNS/OLS (Fully Fueled)	75.0	22.6
RNS/OLS (Nearly Empty)		47.2

EOS on orbit transport is expected to occur only when payloads are securely attached in the cargo bay and not when a payload is erected and mated to the EOS mating port. Therefore, the only loads that must be considered are from activation of the EOS attitude control propulsion system (ACPS) engines. Table A8-3 shows the interfacing loads that result from firing two 2100 pound thrust EOS ACPS engines located as shown on the accompanying figure. These interface loads induce a 20,000 pound ultimate load in the mating port latches that retain the errected payload to the EOS. An additional 1,000 pounds per latch occurs due to internal pressure to bring the total load per latch to 31,000 pounds (DS 244).

Table A8-3. Mating Port Loads



Load Condition	Force x 10^{-3} (lb)			Moment x 10^{-6} (in.-lb)		
	Sx	Sy	Az	Mx	My	Mz
±X translation	∓2.33		±1.07		±1.03	
±Y translation		∓2.60		±2.62		±0.64
-Z translation	-1.05		2.45		1.16	
+Z translation	-0.46		-3.88		0.51	
±Roll rotation		∓0.50		±0.11		±0.09
+Pitch rotation	1.5		1.45		-1.66	
±Yaw rotation		∓1.08		±1.32		∓0.87



Reference Payload--Orbiter ACPS Configuration

CANDIDATE DOCKING CONCEPTS

A review of many different docking concepts was performed to select candidate configuration for detailed assessment. The concepts shown in Figure A8-14 were reviewed to establish compatibility with the following requirements:

1. Concept must be androgynous
2. Provisions for the transfer of a 40 x 40 x 50 inch package
3. Concept to be permanently installed in all mating vehicles
4. Capable of mating with all defined configurations

Three of the concepts shown in Figure A8-14 were selected for further evaluation: (1) multiple probe and drogue, (2) ring cone, and (3) the square frame. In addition to these selections, the design being considered for international docking was selected for evaluation.

The multiple probe and drogue concept as shown in Figure A8-15 employs three probes and three drogues on each vehicle. During the docking operation these probes on one vehicle would be activated while the probes on the mating vehicle would remain passive or act as a redundant system. Only the drogue cones are required where the mating payloads are passive. Capture latches are contained in the probe fingers and are engaged when the probe finger is seated in the drogue. The actuator contained within the shock mounted struts pulls the two mating vehicles together. Structural interface loads are transferred by latches located either on the transfer tunnel or the vehicle outer shell structure.

The ring cone concept shown in Figure A8-16 was proposed for hard docking to the NR Space Station. The mating operation is performed by fitting together two identical six-foot diameter machined rings, one on each of the mating vehicles. Alignment is achieved by matching notches and tapers on the rings. Shock struts and actuators located on the inside perimeter of the docking enclosure provide attenuation and actuation for the capture and locking latches. Locking latches for load transfer are located on a 6 1/2-foot diameter.

The square frame concept is shown in Figure A8-17. This concept employs a square frame mating fixture containing guide arms and capture latches and eight shock absorber actuators. The square frame serves to align the modules in the roll axis because the engaging guide arms are camed into the corners of the frame. Load transfer latches are located on the outer shell structure. Package transfer is accomplished through hatches in the hemispherical airlocks.

Figure A8-18 illustrates a modification to the concept being considered for the international docking study. This concept employs a mating fixture consisting of a six-foot diameter tubular ring with three tubular nodes that extend approximately four feet beyond the ring. These nodes fit between the nodes on an identical fixture during the docking operation. Shock attenuators located around the ring can be activated to extend the ring and capture latches to mate within the ring on the mating vehicle. The attenuators are also used to draw the two vehicles together. Locking latches are located around the inside diameter of the ring or on the outer shell structure.

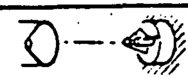
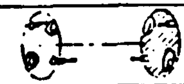
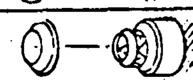
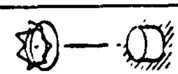
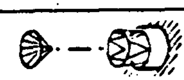
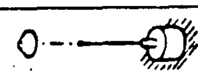
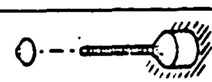
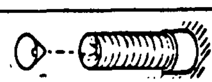
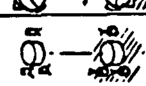
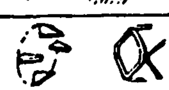
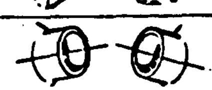
CONCEPTS		60 IN. DIA OPENING	NO LOOSE PARTS	SHIRT SLEEVE MAINT	REDUNDANT COMPONENTS	WTS	DYNAMIC STABILITY	REMARKS
APOLLO PROBE & DROGUE		MOUNTED ON THE HATCHES		100%	LATCHES & SEALS	PROBE 260 DROGUE 60	YES	QUALIFIED EFFICIENT
MULTIPLE PROBE & DROGUE		✓	✓	LATCHES ONLY	ATTENUATORS LATCHES & SEALS	—	YES	EVA MAINT
CYLINDER AND CONE		✓	✓	100%	ATTENUATORS LATCHES & SEALS	CYLINDER 400 CONE 300	YES	APOLLO DERIVATIVE
GEMINI TUNNEL AND CONE		✓	✓	LATCHES ONLY	LATCHES AND SEALS	—	YES	EVA MAINT
CONE AND RING		MOUNTED ON THE HATCH		CONE AND LATCHES	LATCHES AND SEALS	—	YES	BULKY
STEM AND CABLE		MOUNTED ON THE HATCH		100%	LATCHES AND SEALS	—	NO	DELICATE LOW RELIABILITY
INFLATABLE PROBE		MOUNTED ON THE HATCH		100%	LATCHES AND SEALS	—	NO	VULNERABLE TO PUNCTURES
INFLATABLE TUNNEL		✓	✓	100%	LATCHES AND SEALS	—	NO	VULNERABLE TO PUNCTURES
NASA RING AND RING		✓	✓	LATCHES	ATTENUATORS LATCHES & SEALS	RING 420 RING 420	YES	EVA MAINT
MULTIPLE FORKS		✓	✓	LATCHES ONLY	ATTENUATORS LATCHES AND SEALS	FORKS 460 FORKS 460	YES	EVA MAINT THERMAL
MULTIPLE PROBE/RING		✓	✓	LATCHES ATTENU-ATORS	ATTENUATORS LATCHES SEALS	—	YES	EVA MAINT THERMAL
CYLINDER TO CYLINDER		✓	✓	100%	LATCHES SEALS	—	YES	MODIFY 3A CYL/CONE NO ATTENUATION
RING AND CONE		✓	✓	ALL BUT SEALS	ATTENUATORS LATCHES AND SEALS	ACTIVE 520 PASSIVE 280	YES	SIMILAR TO CYLINDER AND CONE
SQUARE FRAME		✓	✓	—	—	—	YES	—

Figure A8-14. Docking Concept Options

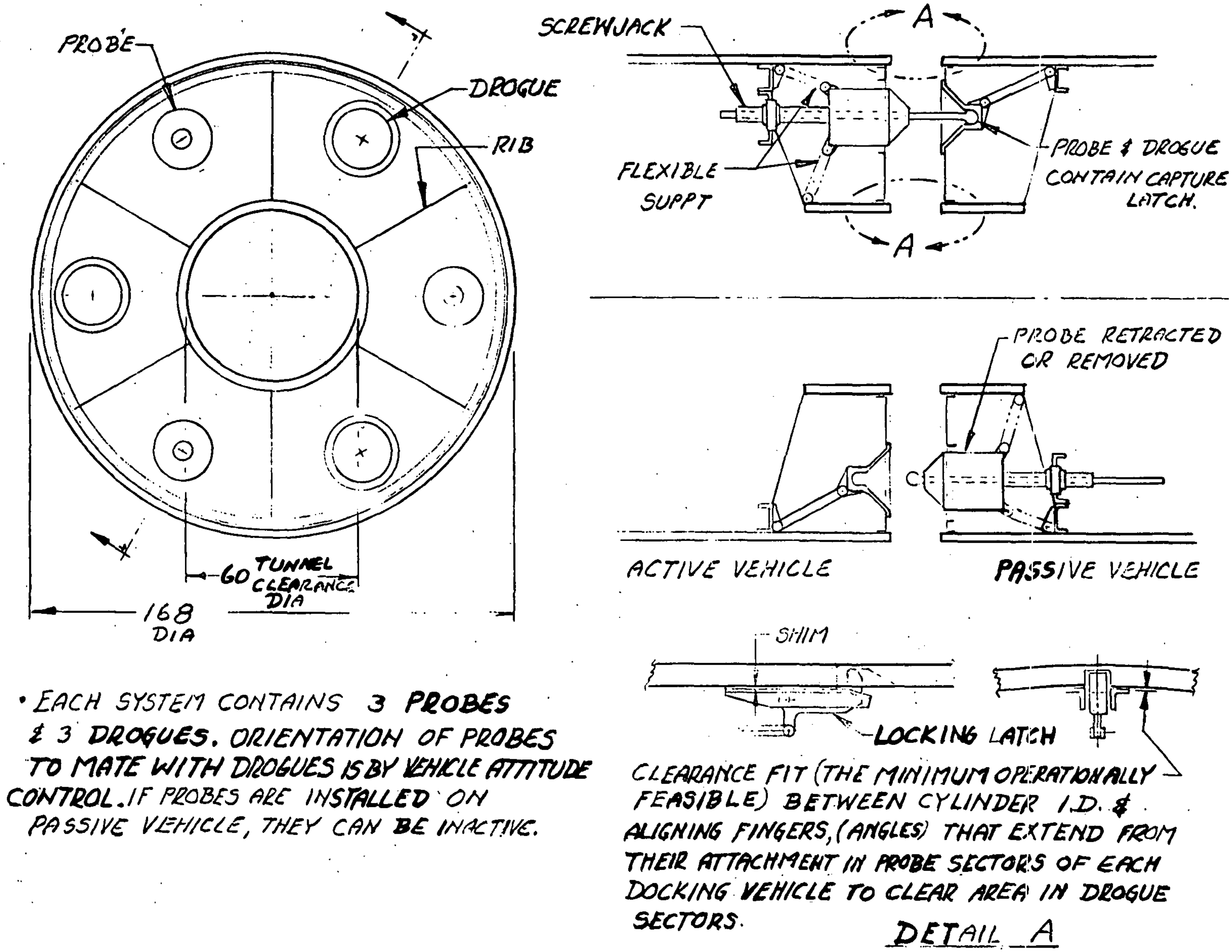


Figure A8-15. Multiple Probe and Drogue Docking Concept

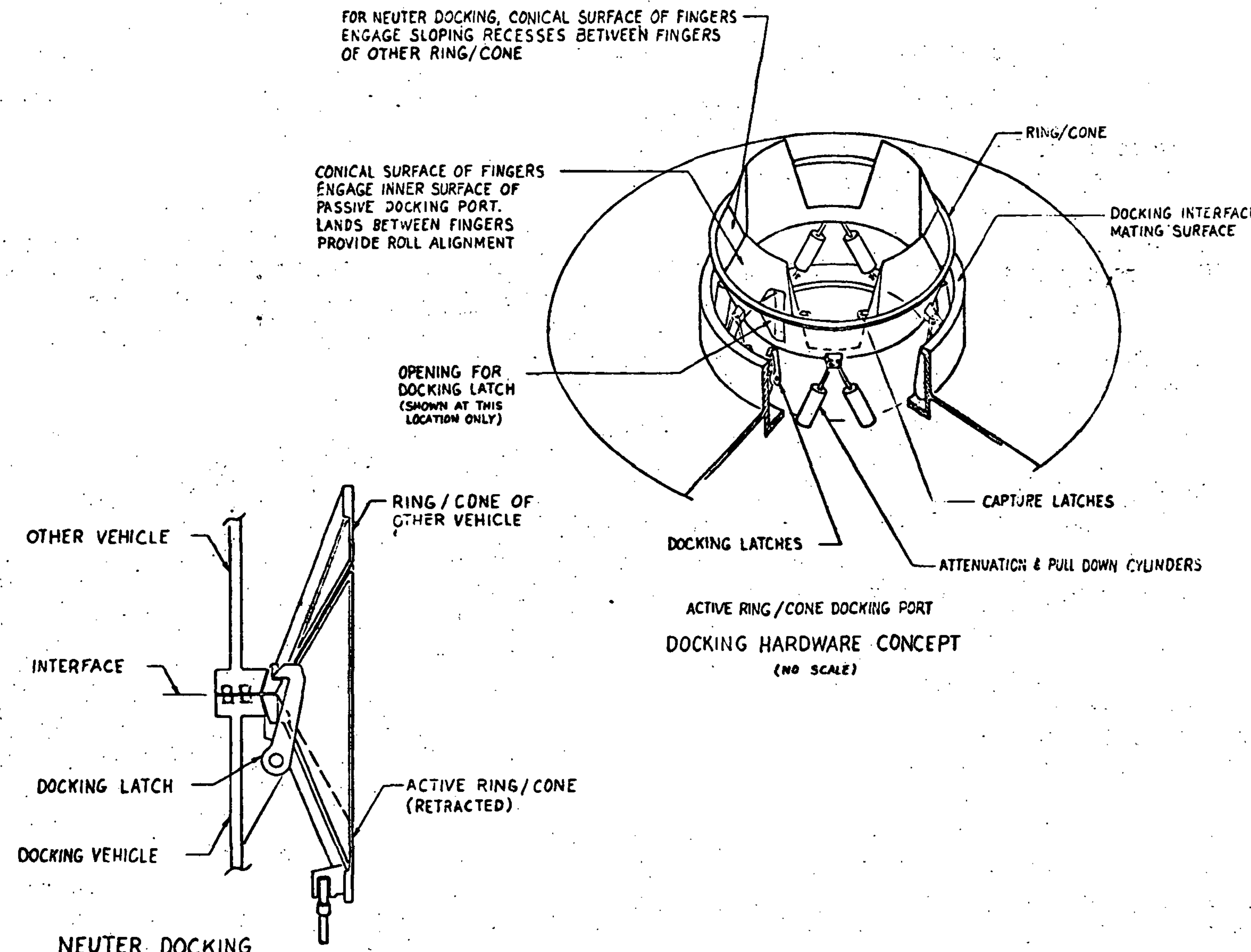


Figure A8-16. Ring Cone Docking Concept

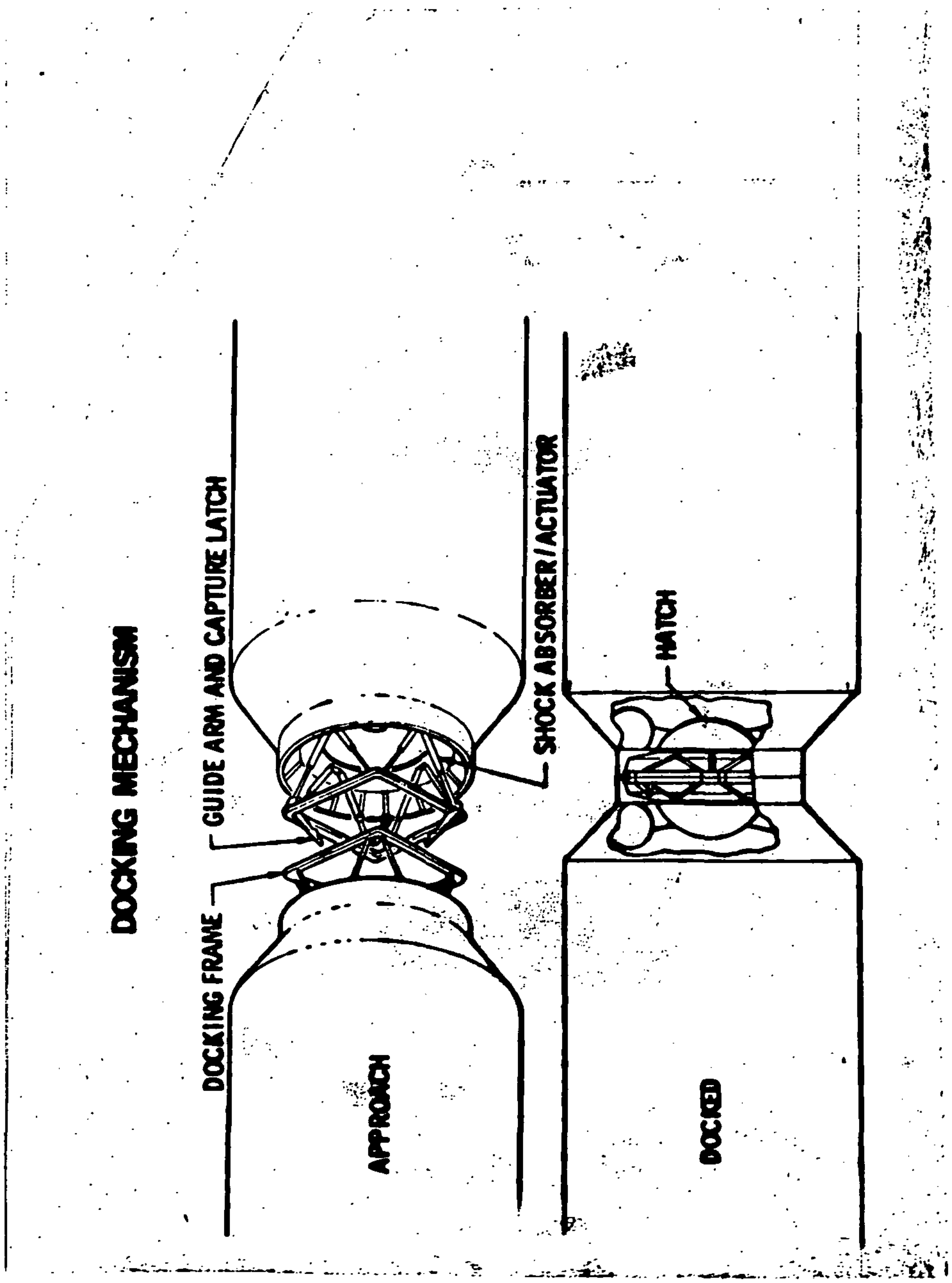
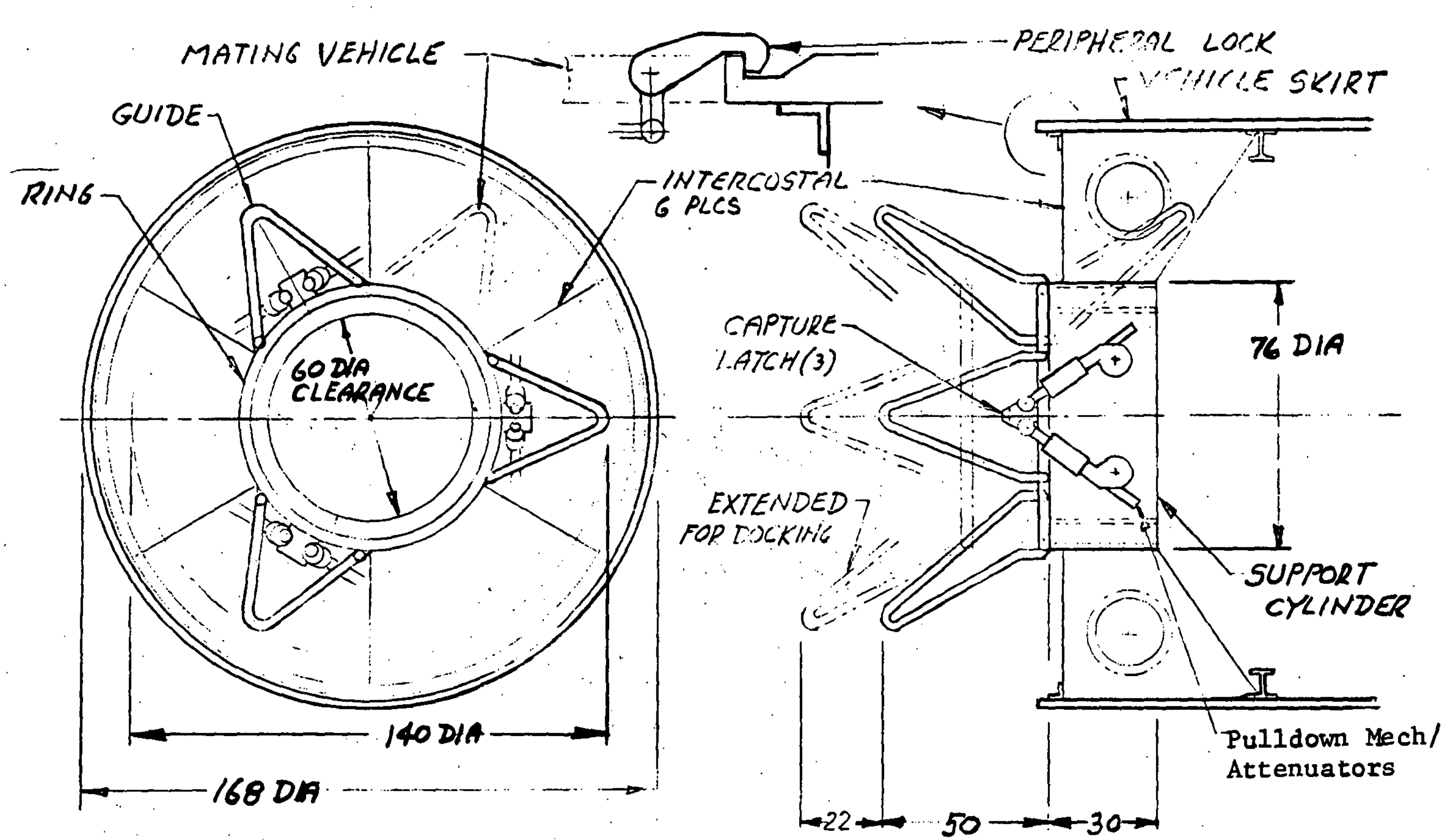


Figure A8-17. Square Frame Docking Concept



- DOCKING SYSTEMS FOR MATING VEHICLES ARE IDENTICAL
- KIT PROVISIONS CONSIDERED FOR SPECIAL MISSIONS
- ONE OR BOTH MATING VEHICLES CAN BE ACTIVE
- GUIDES CAN BE FOLDED BACK FOR CLEARANCE, & UNFOLDED FOR DOCKING.
- PRESENT ORBITER CARGO BAY DESIGN PROVIDES TIE-DOWNS INDEPENDENT OF DOCKING FIXTURE.

NR proposes a hydraulic/ pneumatic system that is much simpler than the Soviet screw jack system.

Figure A8-18. International Docking Concept

Concept Selection

Concept selection was performed by comparing design consideration factors for the four candidate concepts in an evaluation trade tables. Factors considered in this assessment were technical performance, producibility, operations, and relative costs. Results of this trade study are shown in Table A8-3.

Technical Assessment

The trade factors considered in the assessment of technical performance were system complexity, reliability, development state-of-the-art weight, size and impact on vehicle length. Complexity is measured by the number of parts making up the device along with the number of moving parts. The ring and cone appears to be the most complex. Complex detailed design and assembly problems are required to develop this concept because of the complex machined ring. The square frame concept appears to be the least complex. It is anticipated that the transfer hatches would increase complexity to approximately that equal to the international and probe/drogue concepts. All four candidate concepts have inherent redundancy provided both mating vehicles can be activated. The neuter condition facilitates docking from either mating vehicle.

All four concepts are considered within current state-of-the-art from a design and materials standpoint. However, the actuation system, flexible points, and spring supports of the probe/drogue desing will require some development. Little or no development is expected with the international ring/cone concept. The cylinder that provides both attenuation and actuation for the square frame concept could require development; however, considerable effort has been expended in definition of this concept. The literature describing the square frame indicates an unconventional hydraulic/cylinder is purely theoretical or if working models have been tested. A review of the mechanics of operation shows the design to be feasible. Although the state of development is unclear, the square frame concept is rated as current state-of-the-art.

The docking weight assessment shows that three of the four concepts can be installed on a vehicle for approximately 350 pounds. This estimate includes all mechanisms and supporting structure for each system. The ring cone design weight estimate of 600 pounds was obtained from the Space Station program.

The size assessment was based on the area enclosed by each concept. The most compact design is the ring cone while the square frame encloses the largest area. The international docking conept is rated as moderately

Table A8-3. Docking Concept Trades

	International	Multiple Probe and Drogue	Ring Cone	Square Frame
Technical				
Complexity	Medium	Medium	High	Medium
Reliability	Good	Good	Good	Good
State of Art	Current	Development Required	Current	Current
Weight	350 Lbs.	350 Lbs.	600 Lbs.	350 Lbs.
Size	Mod. Compact	Mod. Compact	Most Compact	Least Compact
Impact on Vehicle Length	3 Feet	1 Foot	3 Feet	3 Feet
Producibility				
Ease of Mfg.	Two	Three	Four	One
Fab. State of Art	Current	Current	Current	Current
Inspection Capability	Accessible	Accessible	Accessible	Accessible
Operations				
Maintainability	IVA	IVA	Shirtsleeve	IVA
Adaptability to All Vehicle Combinations	Good	Good	Good	Good
Gender	Neuter	Neuter	Neuter	Neuter
Relative Cost	Medium	Medium	High	Medium

compact by virtue of the capability of folding back the node. The probe/drogue concept was also rated moderately compact since the probes and drogues can be located at any diameter beyond the transfer tunnel.

The final consideration in the technical assessment was the effect of the docking installation on the overall length of the vehicle. This trade factor is considered important because of added vehicle weight and increased body loadings during maneuvering. The probe/drogue design would require about one foot of additional body structure while the ring cone, international and square frame concepts require approximately three additional feet for installation. The international system with nodes extended would extend an additional four feet but this can be overcome by folding back the nodes as previously stated.

Producibility Assessment

The assessment for producibility of the candidate docking concepts is comprised of ease of manufacturing, fabrication state-of-the-art, and inspection capability.

Ease of manufacturing is a measurement of the number of parts and complexity of fabrication and assembly of each candidate concept. The concepts are ranked in numerical order with the simplest being rated as number one. The square frame concept appears to be the simplest with a machined square frame and eight actuators and is therefore rated number one.

The international docking concept is ranked as number two after comparison with the alignment problems of the probe/drogue concept and the complex design of the ring cone. The multiple probe/drogue design requires very close alignment of the components and must be controlled for both mating vehicles. This concept is ranked third in order of preference from a manufacturing standpoint. The ring cone design requires a complex machined ring approximately six feet in diameter. The fabrication of this ring leads to the most complex manufacturing sequence.

All four design concepts were reviewed to establish fabrication state-of-the-art. Each design appears to be within current manufacturing technology and is therefore rated equal. Inspection capability is considered as a measurement of system accessibility for inspection purposes. No major inspection problems are projected for any of the candidate concepts and all provide adequate access; therefore the concepts are rated equal.

Operations Assessment

The operations assessment is divided into three measurement categories: maintainability, adaptability to all vehicle combinations and gender.

Maintainability is measured by the number of items requiring maintenance on each design and the ability to service these items in deep space environment. The square frame and the international designs appear to have lower maintenance requirements than the other two concepts. However, any required maintenance on either the international concept or square frame could be performed by intravehicular activity (IVA) since all components are within easy reach of the transfer tunnel. The multiple probe/drogue concept will require periodic maintenance of the actuators and latch mechanisms. These components appear to be accessible using IVA either from the transfer tunnel or under the closeout structure. The ring cone concept would require maintenance; however, all moving parts are accessible from within a pressurized tunnel. Therefore, the maintenance environment is considered shirtsleeve.

Each candidate docking concept has been compared to the mated vehicle combinations to determine adaptability from both installation and loads transfer considerations. The international docking concept can be used with all vehicle combinations defined in this study. Structural interface loads are low for all vehicle combinations except the Chemical Propulsion Stage coupled with the Orbital Lunar Station. The docking configuration for mated vehicles with low interface loadings would utilize load latches located on the transfer tunnel ring. The CPS/OLS vehicle combination will require latch mechanisms located on the outer shell structure. Additionally, a structural conical shaped transition is required for primary load transfer between the diametrically different vehicles. The international design candidate is rated good for adaptability assessment rationale for the adaptability of the probe/drogue concept is identical to the international system rationale and is also rated as good. The ring cone concept is rated equal to the international and probe/drogue concepts; however, the advantageous feature of a shirtsleeve environment for maintenance would be lost for the CPS/OLS vehicle combination. The square frame is limited to structural attachment at the outer shell only. By employing transition cones for the structural attachment of the docking, this concept is compatible with all vehicle combinations.

Since a basic requirement for the selected concept is that of being androgynous, the gender is of importance. All of the four candidates are neuter type and can be docked to identical concepts.

Relative Cost

Relative cost is assessed by considering development requirements, complexity and weight. The ring cone design is heavier and more complex than the other candidate concepts and is rated as the highest cost. The international, probe drogue and square frame concepts require development but are lightweight and moderately complex and are therefore rated to cost somewhat less than the ring/cone.

Trade Table Summary

An evaluation of the trade table shown in Table A8-3 shows the ring cone concept to be the least desirable. The weight penalty associated with the ring cone design makes it unacceptable for small, weight critical vehicles such as the Space Tug or Cargo and Propellant modules. The design complexity and high relative cost of the ring cone also add to its undesirability.

Both the international and multiple probe and drogue concepts are equally acceptable for all trade factors considered. Although these concepts are rated equal in complexity and state-of-the-art, the problems involved with the international concept appear nearer a solution. Assuming the current international docking concept is developed, no specific features in the probe/drogue concept warrant its separate development. The square frame concept is equal to both the international and probe/drogue designs for all technical considerations, however, this concept appears to be the lowest in cost and easiest to manufacture. Therefore, the square frame is the most desirable design. Further analysis is recommended to verify both the cost and manufacturing assessment of the square frame and international concepts. This assessment shows that the international concept is acceptable for a universal application and should be used if the concept is developed.

ELECTRICAL AND PRESSURE INTERFACE CONCEPTS

A review of the mating vehicles indicates a requirement for both electrical and pressure interfaces across the mating planes. One design concept for each interface connector has been established. These connectors are designed to be compatible with any docking structure and also retain the androgynous interface requirement.

The proposed electrical connector is shown in Figure A8-19. It is designed for automatic or remote operation. The connector has a spring loaded cap over the contact pins for protection prior to engagement and guides the pins during engagement. The non-conductive cap has a heavy base allowing the pins to extend about halfway through the cap base. Two probes of different diameter are employed to insure radial and angular alignment. Completion of the contact operation forces the pins through the cap holes and drives them into the recepticle. The connector is then locked in place by a simple state-of-the-art locking device.

Figure A8-20 presents a remotely or automatically controlled device designed to accommodate interface connections across a docking interface. The design is basically a sleeve connector supported by a pivoted, extendable arm. The sleeve contains two drogue portions of a line connector. Although each mating vehicle would have this sleeve to achieve the androgynous characteristic, only one is used during the mated sequence. The activated sleeve is extended and rotated into alignment with the mating fluid probes. Alignment of the line probes extending from the two docking vehicles can be assured by establishing the proper angular dimension between the probes and an index point on the docking concept. For example, the multiple probe/drogue docking concept would have the line probe located midway between a drogue and a probe.

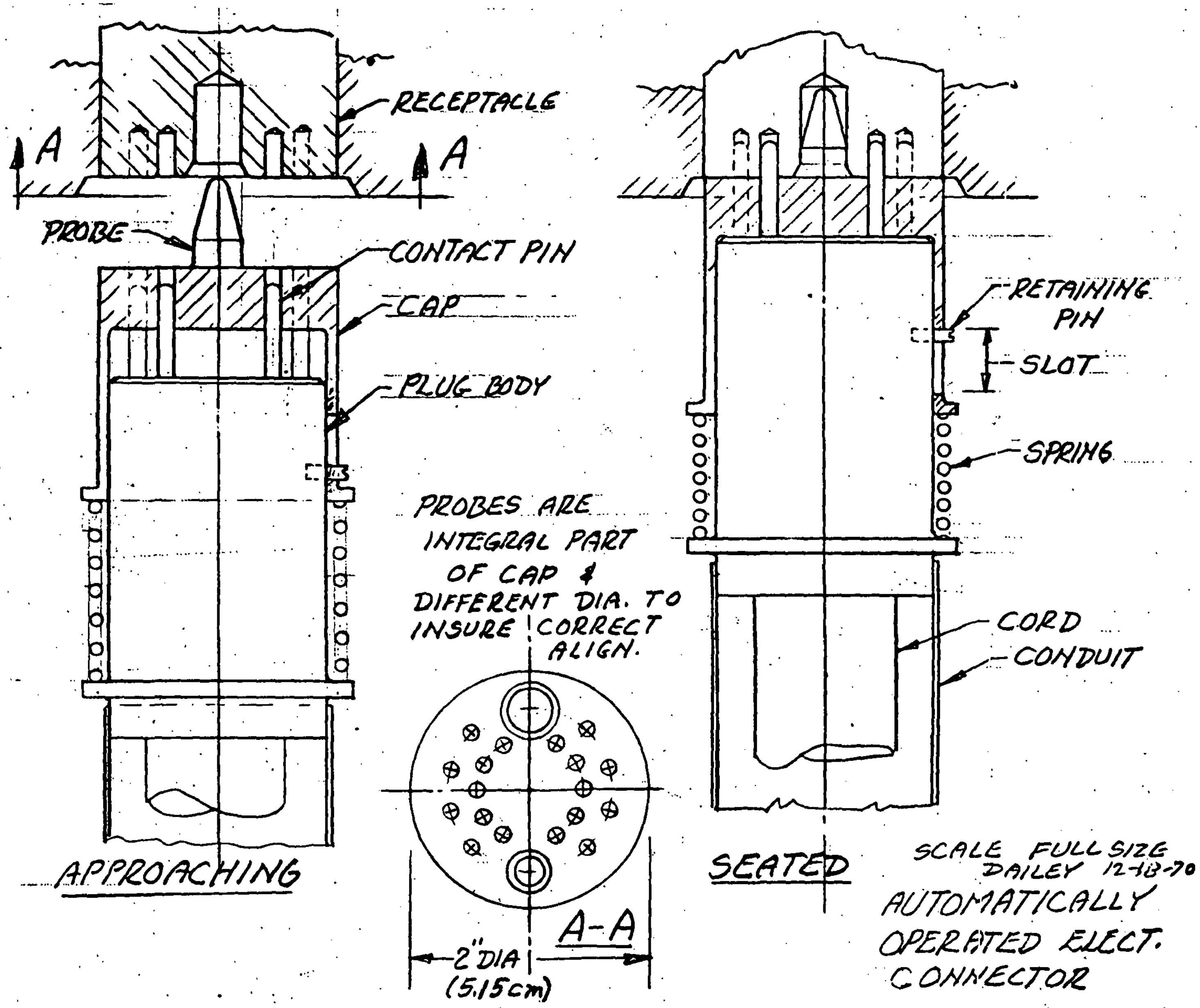


Figure A8-19. Electrical Connector

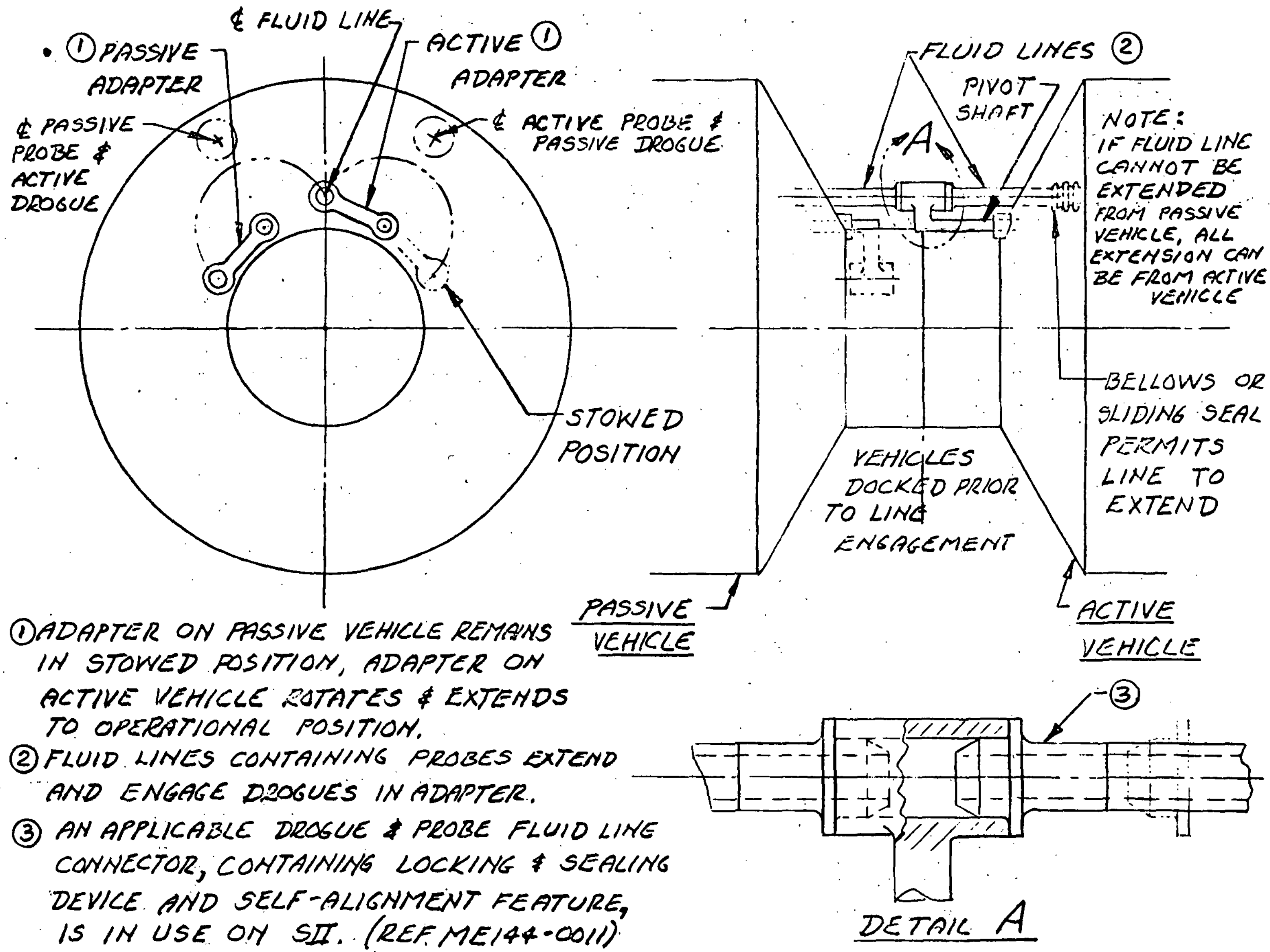


Figure A8-20. Androgynous Line Interface

A9 PROPELLANT TRANSFER ANALYSES

1.0 INTRODUCTION AND SUMMARY

During the course of this study, certain elements of the potential space vehicle inventory were identified as possible recipients or suppliers of propellants in orbit. The initial supplier in all cases is the Earth Orbital Shuttle (EOS) and the potential recipients are various versions of the TUG, Chemical Propulsion Shuttle (CPS) and Reusable Nuclear Shuttle (RNS). Delivery of propellants to elements in earth orbit may be accomplished by modular transfer or fluid transfer, delivered directly to a user or through a depot where it is stored and then transferred to a user. The depot could be an Orbital Propellant Depot (OPD) capable of supporting the CPS or RNS or a smaller mini-depot capable of supporting the TUG. This portion of the Appendix will provide detailed backup data relative to propellant logistics hardware concepts, transfer system analyses, physical interface definition and number of flights to support the spaced based vehicles. The data were generated in support of the Propellant Transfer Interfacility Activity contained in Vol. II, Part 4, Section 3.0.

Propellant logistic hardware concepts evaluated during the course of the study included logistic propellant tanks, mini-depots, and OPD configurations for use with both linear and rotational acceleration modes of propellant settling. Logistics tanks were defined that would be used for direct fluid transfer of propellants. The tanks included transfer line interconnects, compressors, thrusters, and other transfer system components. Mini-depot and OPD configurations were defined to support the TUG, CPS, and RNS. Descriptions of the mini-depot and OPD include transfer and gaging system requirements.

An analysis of fluid transfer systems resulted in the selection of the four major subsystem concepts necessary for in-orbit fluid transfer. Linear acceleration was selected as the baseline for liquid/vapor interface control for direct transfer. It was determined that separation of the logistic tank from the orbiter after delivery to the user, and use of linear acceleration was the best method of transfer even though some penalty in propellant usage was involved. The thermodynamic control subsystem was baselined to the connected ullage concept, and for expulsion, the gas pump subsystem was selected. Net Positive Suction Pressure (NPSP) control by an active pressurization system was selected on the basis of development risk.

Paragraph 4.0 of this appendix deals with the logistic tank to Shuttle and logistic tank to user vehicle physical interfaces. Interfaces under both conditions were previously identified with final definition dependent on further vehicle and tank requirements. Design concept docking and line interconnect fixtures are conceptually presented; however, no system selection is made or required to support this study.

Modular propellant transfer is addressed, identifying some potential problems using the modular approach. This analysis combined with the considerations of Vol. II, Part 4, Section 3.0 has established that significant weight penalties would be imposed on the TUG, CPS, or RNS if they were modular oriented.

An additional supportive task was the development of the number of flights required for supply of the TUG, CPS, or RNS. Delivery and loss criteria are defined and typical data is presented. These data are used in selection of element-to-element propellant transfer methods of Vol. II, Part 4, Section 3.0.

2.0 PROPELLANT LOGISTICS HARDWARE CONCEPTS

In most cases, it has been considered adequate to present one viable design concept for each interfacing activity included in the Orbital Operations Study. However, because of the lack of any design precedent (or generally accepted design concept) for propellant transfer in orbit, it was felt desirable to present design analysis and trade data for various potential design concepts. This enhances the capability of selecting a transfer design approach that is viable.

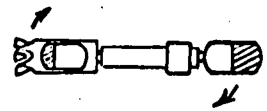
2.1 DESIGN CONCEPTS

Various design concept options to support the space-based propulsive elements have been investigated. For the space-based TUG the investigation has led to four options utilizing a mini-depot for short term orbital storage, two options without orbital storage where propellant is transferred directly to the TUG from the logistics resupply tank and two options using a second TUG for storage. These options are summarized in Figure A9-1. The first four options (1a through 2b) employ a mini-depot to store propellant in orbit. The next two options (3a and 3b) involve direct transfer from the logistics tank to the TUG and do not provide for propellant storage in orbit. The last two options (4a and 4b) utilize a second TUG in orbit allowing the TUGs to alternate as temporary depots.

The modular mini-depots (1a and 1b) store propellant in the logistics tank as delivered by the Shuttle (through a modular propellant transfer to the depot) and have an equipment module providing the orbital capability for fluid transfer of propellant to the TUG. The 1a and 1b versions differ in the propellant settling technique. The rotational depot requires the transfer equipment module for a counterweighted boom to prevent the combined vehicle center-of-gravity (cg) from falling within any of the tanks during transfer. The linear depot has a smaller, lighter transfer equipment module that can share the initial launch with the logistic tank, but the settling technique uses more propellant because thrust must be applied continually during transfer.

Options 2a and 2b are also mini-depots but the storage tanks are a permanent part of the depot. This approach offers the lowest potential boiloff rate in tank design. Since the mini-depot stage tank weight does not reduce the amount of propellant delivered (as in a logistics tank), additional thermal control systems and insulation can be included. Also, since the module can be launched with tanks empty, the tank supports can be minimized thus reducing heat conduction paths between exposed structure and the tanks. The boiloff advantage is traded against the propellant losses due to the fluid transfer of propellant to the depot. Refer to Paragraph 3.0 of this appendix for a comparative analysis.

1. MODULAR MINI-DEPOT (MODULAR TANK TO DEPOT, FLUID OUT)

A. ROTATIONAL SETTLING - - - -  DEPOT EQUIP MOD + LOGISTIC/STORAGE TANK

B. LINEAR SETTLING - - - - - -  DEPOT EQUIP MOD + LOGISTIC/STORAGE TANK

2. PERMANENT TANKAGE MINI-DEPOT (FLUID IN, FLUID OUT)

A. ROTATIONAL SETTLING - - -  ONE TUG FILLING CAPACITY (LOG. TANK SIMILAR TO ABOVE)

B. LINEAR SETTLING - - - -  60' DEPOT INCLUDING TANKAGE (LOG. TANK SIMILAR TO ABOVE)

3. DIRECT TANK TO TUG TRANSFER (NO STORAGE)

A. LINEAR SETTLING - - - - -  TRANSFER CAPABILITY TANK

B. POSITIVE EXPULSION - - -  TRANSFER CAPABILITY EXPULSION TANK

4. TUG TO TUG; LINEAR TRANSFER (TWO TUG)

A. TRANSFER CAPABILITY TUGS - -  DEFINE IMPACT TO BASELINE (TUG)

B. TUGS WITH TRANSFER MODULE -  DEFINE IMPACT TO TUG & DEFINE TRANS MOD

FIGURE A9-1 OPERATIONAL CONCEPT MODULE IDENTIFICATION

The direct transfer options (3a and 3b) utilize a logistics tank that contains much of the transfer systems equipment, but would rely on the TUG for power and control during transfer. Direct transfer does not lend itself to rotational settling since there is no module to act as a boom for cg control. The positive expulsion tank would eliminate the need for settling acceleration. This advantage would be offset by the additional weight and larger propellant residual as discussed in Section 3.0 of this appendix.

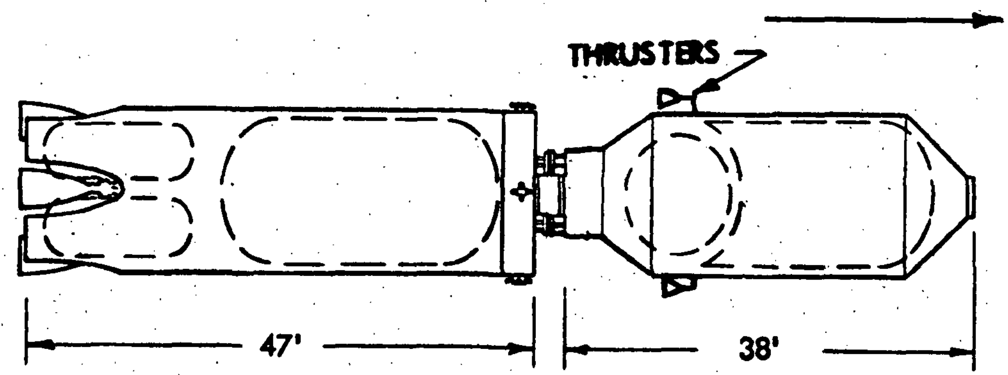
The "Tug-to-Tug" options (4a and 4b) operate by allowing one TUG to act as a temporary storage depot while the other performs a mission. Propellant would be brought in a logistic tank similar to the other options with fluid transfer to either TUG. The TUG would have some additional equipment added to permit transfer from Tug to Tug. Equipment would be kept to a minimum if the TUGs of option 4 are interfaced with a transfer capability tank of option 3. The transfer module defined for option 4b offers the alternative of having some of the transfer equipment in a separate module rather than as added weight on either the TUG or the logistic tank. This module could be controlled in orbit by the storage TUG.

In the course of evaluating these basic options, additional alternatives and/or combination have been recognized. Options 1a and 1b and 3a have received the most scrutiny, both in that they show the most promise and that the costing evaluation data from these can be extrapolated to provide the best overall comparisons.

2.2 TRANSFER CAPABILITY LOGISTIC TANK

The direct fluid transfer operation associated with option 7a is shown in Figure A9-2 with a transfer capability logistics tank docked to the TUG. The logistics tank module would be delivered by the Shuttle, attached to the receiver vehicle, separated from the Shuttle, and propellant transferred.

The transfer capability logistics tank shown in Figure A9-3 would be used for direct fluid transfer of propellant to the TUG or other user vehicles. Design criteria are basically the same as for other tanks, with the transfer systems added. Line interconnects engage the user receptacles and the necessary transfer compressors, lines, valves, actuation, flow metering and monitoring equipment are included with the tank. Thrusters for linear acceleration are included on the tank but are controlled by the user vehicle. This equipment has increased the weight of this module by over 400 pounds. For direct transfer, the tank must be controlled by the receiver vehicle during transfer and is dependent on the receiver vehicle for power supply, actuation commands, and data transmittal. The mode of propellant transfer is compatible only with linear acceleration for settling since a boom required for center of gravity (cg) control during rotational acceleration is not available.



TUG		TRANSFER CAPABILITY	LOGISTICS TANK
EMPTY	10,000 LB	EMPTY	4,800 LB
LH_2	11,400 LB	LH_2	9,300 LB
LOX	68,600 LB	LOX	50,900 LB

FIGURE A9-2 DIRECT TRANSFER, LINEAR ACCELERATION

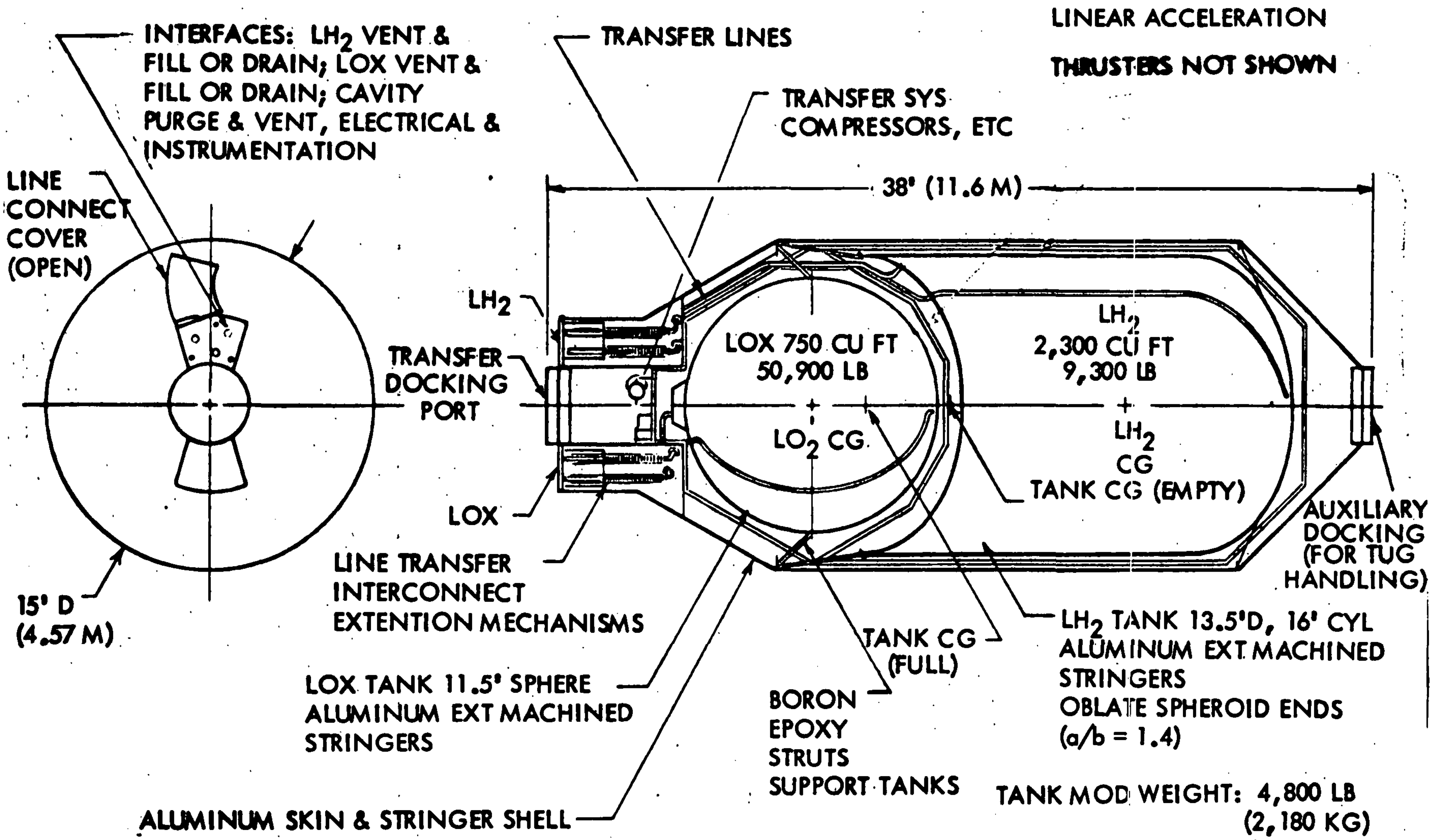


FIGURE A9-3 TRANSFER CAPABILITY LOGISTICS TANK

The tank module length has been kept at 38 feet to allow payload sharing considerations to apply. The length required for installation of the transfer line interconnect mechanism has been acquired by using an inverted bulkhead on the LH_2 tank to allow nesting of the LO_2 tank. The inverted LH_2 bulkhead may aid in propellant settling and reduce LH_2 residuals.

2.3 TRANSFER MODULE

To reduce the impact of adding transfer equipment to a space-based vehicle, an additional module was conceived for evaluation. Figure A9-4 defines the transfer module that could be used to transfer propellant from a tank to a space vehicle or from one space vehicle to another. The module contains only the transfer system and line interconnects. Power and orbital control would come from the delivery or receiving vehicle.

This module was considered in order to avoid the addition of transfer equipment to the receiver vehicle. This equipment, if on the TUG, would just be scar weight during the mission and thus reduce TUG performance. Since option 4 uses one TUG to act as a storage tank, and remain in orbit while the other goes on a mission, this module could remain attached and controlled by the storage TUG. It would require few launches during the program to provide such a module; and, due to its small size, could share with other cargo the costs of those launches.

2.4 MINI-DEPOT CONCEPTS

The modular mini-depot concepts 1a and 1b, shown in Figure A9-5, consist of an equipment module and a tank module and are so named because of the modular transfer (exchange of a full tank for previously emptied one) of propellant to the depot. Thus, the tank is designated a logistics/storage tank and is designed both for carrying propellant (in the orbiter cargo bay) to orbit and for short-term storage of propellant in orbit. The equipment module contains all the equipment required to support depot operations so that the tank is relieved of such systems and can be optimized for delivery of the maximum net quantity of propellant. The equipment module is launched by the Shuttle and remains in orbit. It has full capability of functioning unassisted in orbit and of accomplishing all propellant transfer operations.

Operations for the full Shuttle utilization mode can be illustrated starting at the time prior to the return of the TUG from a placement mission: the depot is alone in orbit, the orbiter having previously returned to the ground. The TUG returns and docks with the depot to receive stationkeeping propellant. The remaining propellant from the logistics/storage tank can either be transferred to the TUG at that time, or transfer just the stationkeeping propellant. In the latter case, the remaining propellant would go

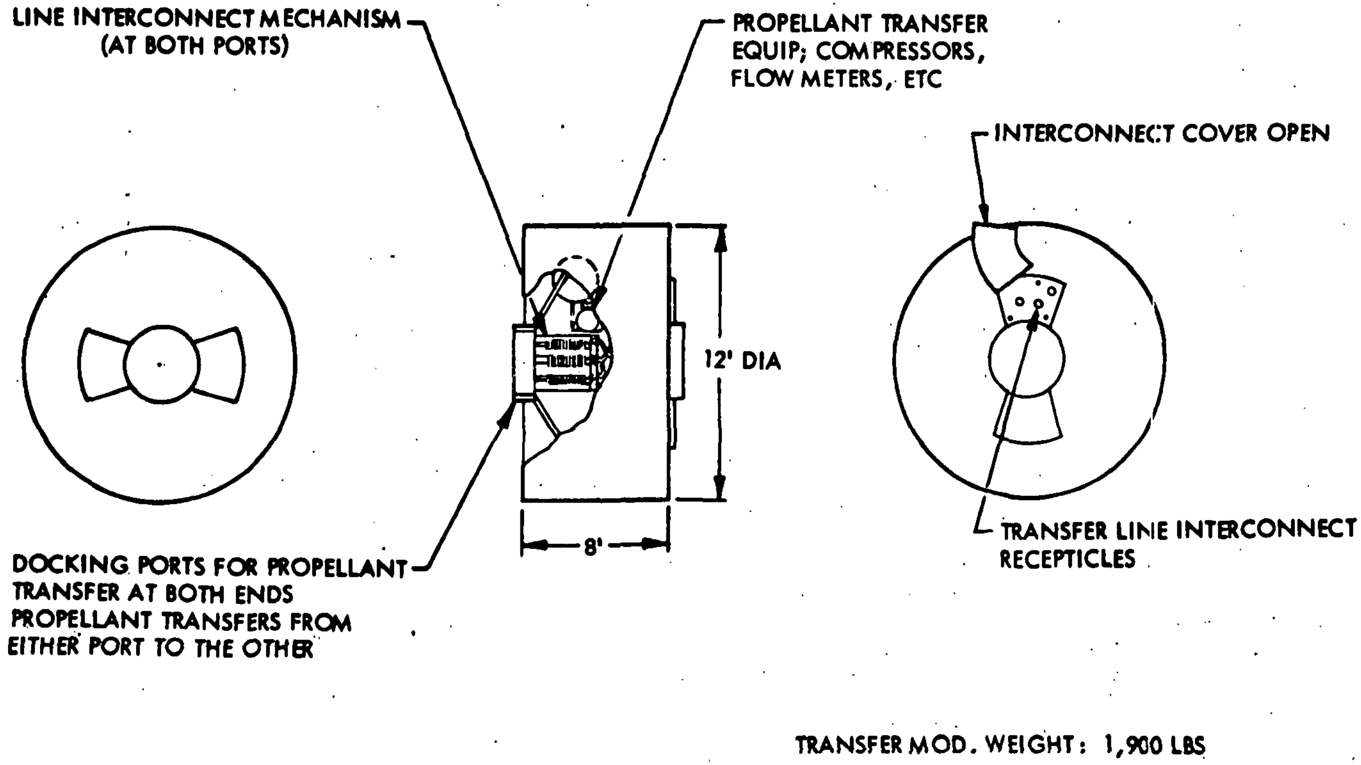
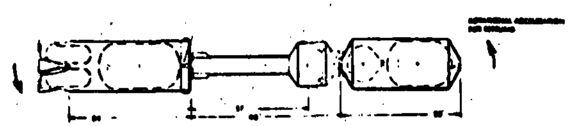
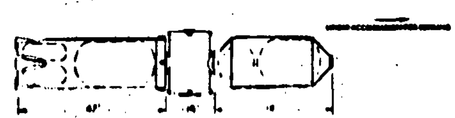


FIGURE A9-4 TRANSFER MODULE

1A MODULAR, ROTATIONAL MINI-DEPOT



1B MODULAR, LINEAR MINI-DEPOT

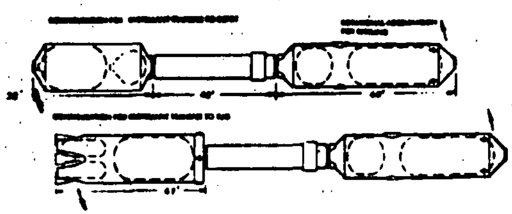


ELEMENTS: EQUIP MODULE & LOGISTICS/STORAGE TANK
FEATURES: ROTATIONAL SETTLING REDUCES PROP USE
EQUIP MOD REQS SEPARATE LAUNCH

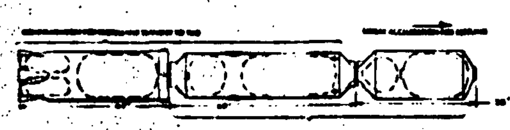
OPERATION: (BOTH) ORBITER DELIVERS PROPELLANT TANK
TANK BECOMES PART OF DEPOT UNTIL EMPTY
EQUIP MOD PROVIDES TRANS & SETTLING CAPABILITY
PROP TRANS FLUID TO TUG FROM L/S TANK
ORBITER RETURNS TANK WHEN EMPTY (NEXT TRIP)

ELEMENTS: EQUIP MODULE & LOGISTICS/STORAGE TANK
FEATURES: EQUIP MOD & TANK CAN SHARE LAUNCH
SHORTER TRANS LINES; LESS BOILOFF

2A PERMANENT TANKAGE, ROTATIONAL MINI-DEPOT



2B PERMANENT TANKAGE, LINEAR MINI-DEPOT



ELEMENTS: DEPOT MODULE, BOOM & LOGISTICS TANK
FEATURES: ROTATIONAL SETTLING REDUCES PROP USE
PERMANENT TANK REDUCES BOILOFF
BOOM REQS SEPARATE LAUNCH

OPERATION: (BOTH) ORBITER DELIVERS PROPELLANT TANK
TANK IS ATTACHED (TUG DETACHED) & PROP IS TRANSFERED (FLUID) TO DEPOT
ORBITER RETURNS EMPTY TANK TO GROUND
TUG DOCKS RECEIVES PROP (FLUID TRANS) FROM DEPOT TANKS
DEPOT PROVIDES TRANS & SETTLING CAPABILITY

ELEMENTS: DEPOT MODULE & LOGISTICS TANK
FEATURES: SINGLE LAUNCH FOR DEPOT
SHORTER TRANS LINES; LESS BOILOFF

FIGURE A9-5 MINI-DEPOT CONCEPTS COMPARISON to the TUG just prior to the next Shuttle arrival. That decision would depend on the quantity of propellant remaining in the storage tank, the length of time until the next mission, and the differences of boil-off rates between the storage tank and the TUG as traded against the additional propellant losses of the second transfer. Either way, the logistics/storage tank would normally be empty when the next Shuttle flight arrived at the depot.

Upon arrival, the orbiter could remove the empty tank from the equipment module, temporarily park it on the depot auxiliary port and attach the new propellant tank module to the depot. The depot would then transfer additional propellant to the TUG; to its optimum mission loading. The orbiter would remain in orbit (detached from the depot) and could stow the empty propellant tank module and check or prepare the scientific payload for the mission. The fueled TUG would rendezvous with the orbiter, receive the payload and go on its mission. The orbiter could then return to the ground. The depot (equipment module and logistics/storage tank with remaining propellant) would remain in orbit waiting for the return of the TUG and repeat of the operations cycle.

Should the full shuttle utilization approach lead to the accumulation of propellants in orbit in excess of the approximately 60,000 pounds capacity of the logistics/storage tank, a second tank can remain with the depot in orbit. Whenever the TUG's requirement for mission propellant does not empty the previously delivered propellant tank, that tank would remain in orbit along with the newly delivered tank and the orbiter would return empty. Because of the difficulties of cg control with two tanks in various states of loading for the rotational settling mode and the additional mass to be accelerated in the linear settling mode, the preliminary definition of the modules does not include the capability to transfer propellant from both tanks in the same transfer operation. It is assumed that when two logistics/storage tanks are needed for mini-depot capacity, transfer will take place only from one tank and (at least in the rotational settling mode case) with the other tank detached. The orbiter could maintain control of the detached tank.

The permanent tankage mini-depot concepts provide the same functions as the modular depots. The concepts are offered for comparison because of the potential economy of reduced boiloff. If the propellant logistics program calls for orbital propellant storage, the reduction in boiloff of just one pound per hour, day in and day out for 12 years, at a cost of approximately $100.00 per pound, can be a worthwhile potential savings. The permanent tankage depot can be optimized for low boiloff. Since the storage tanks will be launched only once for their lifetime, the need to compromise thermal protection to reduce weight is less stringent than in a logistic tank where extra weight reduces the quantity of propellants delivered each launch. In addition, the permament tankage depot module can be launched with empty tanks so that the tank supports can be minimized to reduce heat transfer paths.

Much of the foregoing rationale on depot operations also applies to the permanent tankage depot. An additional transfer operation (propellant settling and fluid transfer) must take place to get propellant from the logistic tank to the depot storage tanks. This would be accomplished at the time of propellant delivery by the orbiter with both the orbiter and the TUG detached. Normal operations could be based on having sufficient propellant previously on-board the depot to give the TUG its optimum mission loading just before arrival of the orbiter. The orbiter could then attach the logistics tank to the depot and while the depot was receiving propellant, the shuttle could attach the payload to the previously fueled TUG. The TUG could go on its mission and after propellant transfer to the depot is completed, the orbiter could return to the ground with the empty logistics tank.

The modular, rotational mini-depot is shown in Figure A9-6 in its operational configuration with the TUG attached. The mini-depot consists of an equipment module and a logistics/storage tank module. The equipment module contains depot operation and transfer systems, including power and attitude control, and remains in orbit independently. For rotational propellant settling, the combined TUG and depot cg, which is the center of rotation, will shift as the mass of propellant is transferred from the storage tank to the TUG. For proper liquid interface control, the center of rotation must not fall within any tank involved in the transfer.

The equipment module acts as a counterweighted boom to limit cg excursions. Propellant is delivered in the logistic storage tank which docks to the equipment module, then becomes part of the mini-depot until it is depleted and replaced with another tank from the next logistics flight. More than one tank can be left at the depot so that orbital storage capacity is not limited (and can therefore take advantage of each opportunity to launch propellant). However, with the present concept definition, propellant can be transferred from only one storage tank (the one adjacent to the equipment module) at a time. Due to the cg excursion considerations of the rotational propellant settling mode (as outlined on Figure A9-6) a second tank may or may not be able to remain attached to the configuration during transfer.

Figure A9-7 gives the conceptual definition of the equipment module for the rotational, modular mini-depot. This module contains all the equipment required for orbital propellant transfer and to support independent operation in orbit. If fuel cell and attitude control consumables are replenished by the frequent visits of the logistics tanks, the configuration of the module is determined (for the rotational propellant settling mode) by the necessity to control the location of the combined mini-depot

FOR ROTATIONAL PROPELLANT SETTLING THE COMBINED USER & DEPOT CG, WHICH IS THE CENTER OF ROTATION, WILL SHIFT AS THE MASS OF PROPELLANT IS TRANSFERED FROM THE STORAGE TANK TO THE USER. FOR PROPER LIQUID INTERFACE CONTROL THE CENTER OF ROTATION MUST NOT FALL WITHIN ANY TANK INVOLVED IN THE TRANSFER. THE EQUIPMENT MODULE ACTS AS A COUNTER WEIGHTED BOOM TO LIMIT CG EXCURSION

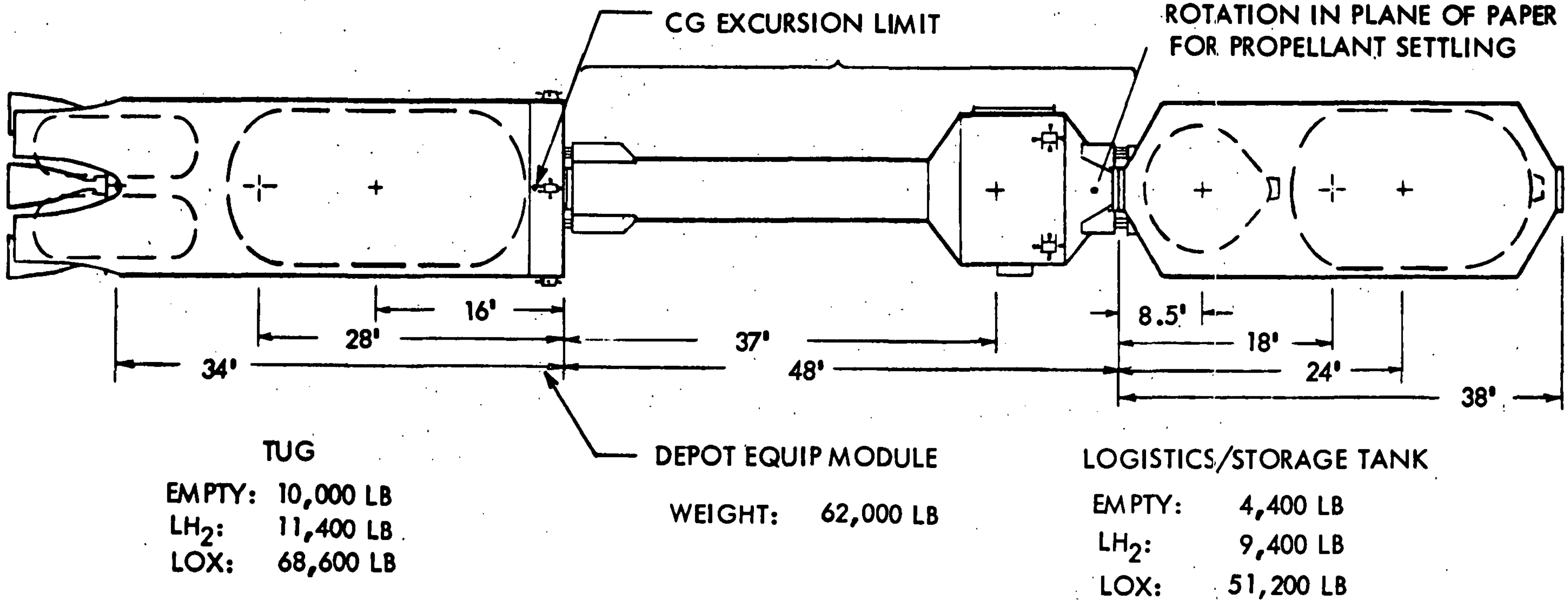
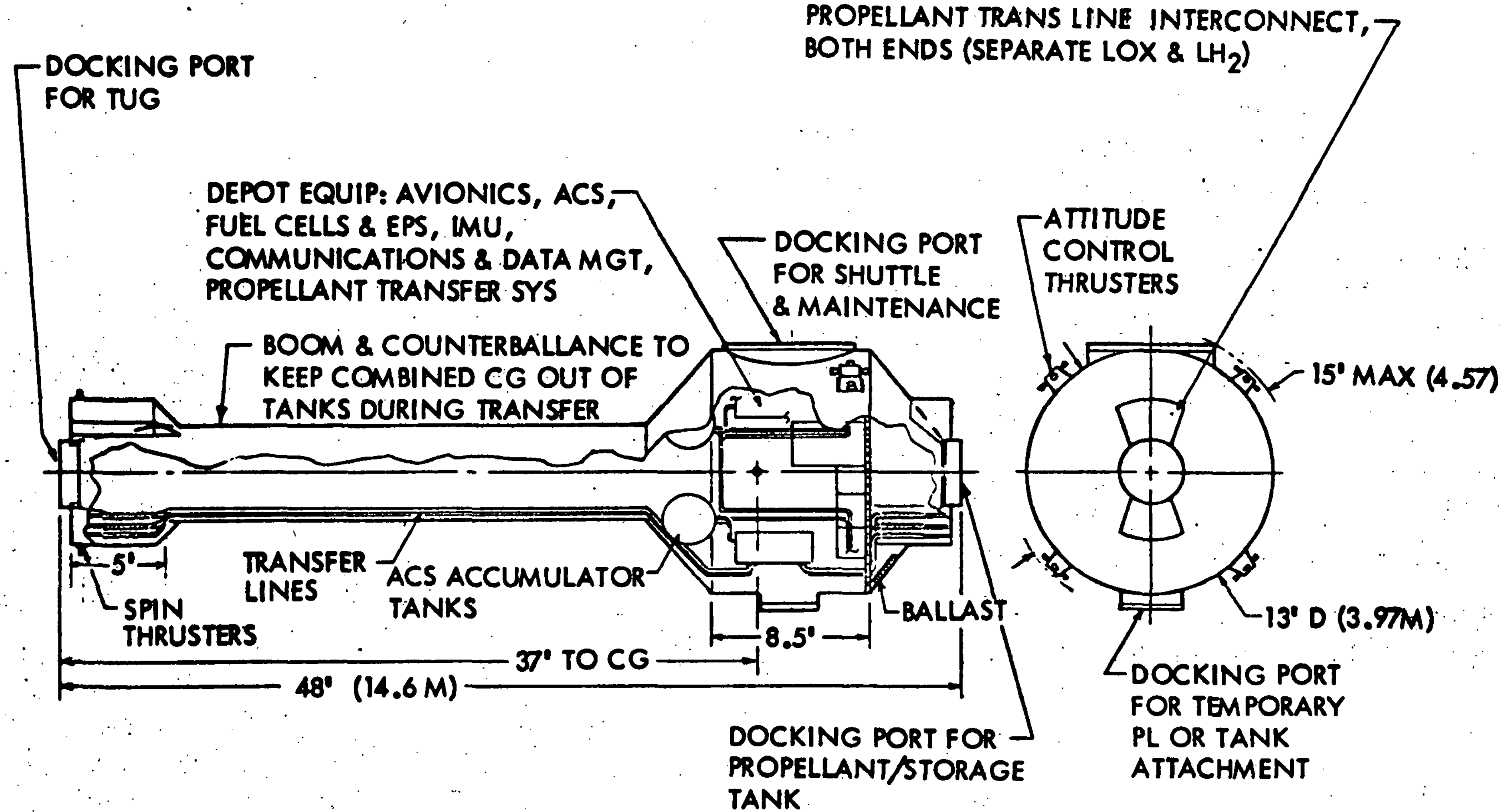


FIGURE A9-6 MODULAR, ROTATIONAL MINI-DEPOT



- MADEWITH BOILERPLATE TYPE STRUCTURE TO BUILD WEIGHT TO 62,000 LB (28,100 KG) FOR TRANSFER CG CONTROL

FIGURE A9-7 ROTATIONAL, MODULAR MINI-DEPOT EQUIPMENT MODULE and user cg to prevent rotation about a point that falls within the tanks involved in the transfer. The length, weight, and location of the equipment module cg could be altered somewhat; but the combination chosen gives the required cg control and is compatible with shuttle launch. The high weight of the module will allow use of inexpensive boilerplate type structure. The equipment module will be deployed by the shuttle orbiter and remain in orbit for its nominal six-year life. Minimum on-board maintenance will be provided with return to the ground for refurbishment for any major unscheduled required maintenance.

The logistics/storage tank module to be used with the rotational mini-depot, shown in Figure A9-8, is typical of the tank modules required by the other mini-depot options (1b, 2a, and 2b) of Figure A9-5. Its function is to bring propellant to the depot, to remain attached to the equipment module until it is empty, then to be returned in the shuttle orbiter for recycle. Major design criteria are low residuals, low boiloff and low inert weight. The tank module has no transfer equipment or orbital capability and must be dependent on and under (attached) control of the orbiter, TUG or equipment module at all times. The LO_2 and LH_2 tanks are sized for maximum utilization of the orbiter cargo capability. The indicated tank weight includes allowances for cargo bay installation of umbilical and re-entry pressurization systems which are not physically a part of the tank module.

Preliminary definition of the tank has included orbiter interface consideration of cg location, cargo umbilicals and payload sharing. For the linear settling (modular mini-depot) option, the logistics/storage tank required is considered identical to this for evaluation purposes. The only discernible difference being that the tanks and lines would be oriented for propellant settling toward the propellant transfer docking port. The logistics tank modules for the permanent tankage depots are also considered similar enough to this one for present evaluation. In more detailed definition phases, differences due to less emphasis on storage requirements (since the tank does not remain in orbit for an appreciable length of time) might reduce insulation or systems and allow a very slight increase in propellant capacity.

The modular, linear mini-depot, Figure A9-9 is shown in its operational configuration with the TUG attached. The operational concept is much, the same as for the rotational mini-depot, with an equipment module providing transfer and orbit-keeping capability and docked logistics tank providing propellant storage and being replaced when empty (modular propellant transfer to the depot). Linear acceleration for propellant settling eliminates the cg excursion problem and allows for a lighter, more compact equipment module. The module length chosen provides clearance for docking to the end ports (as would be used for temporary

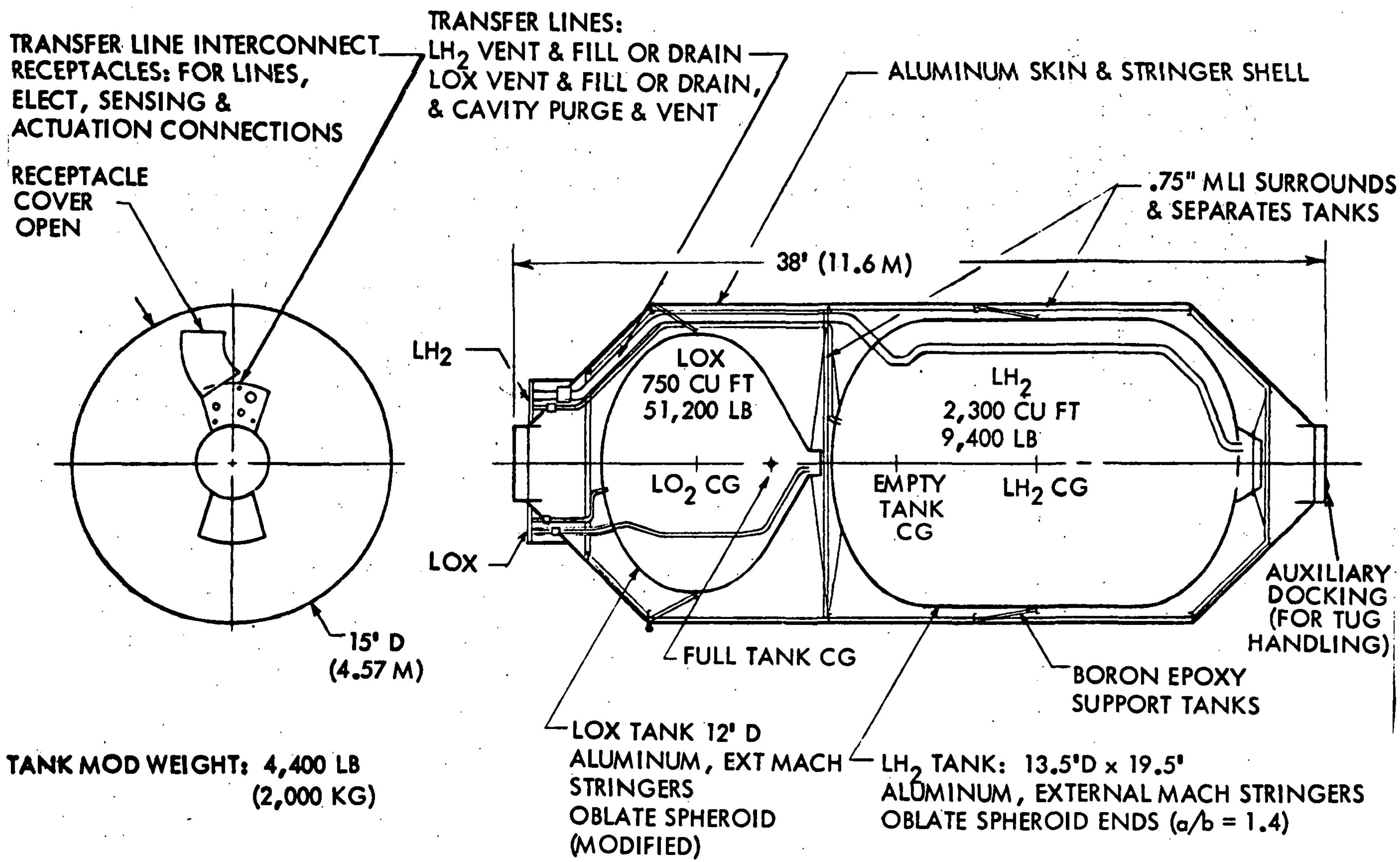


FIGURE A9-8 ROTATIONAL LOGISTICS/STORAGE TANK

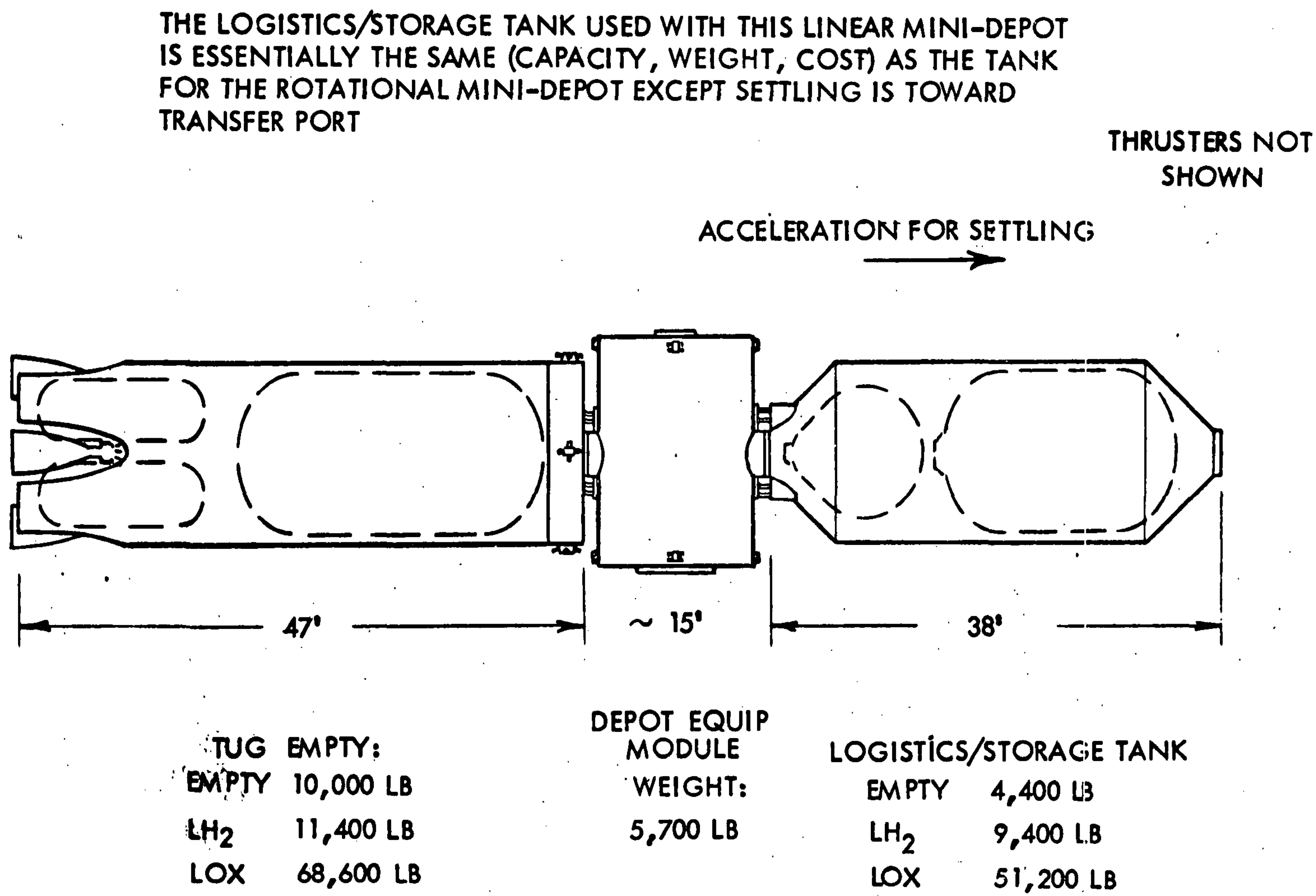


FIGURE A9-9 MODULAR, LINEAR MINI-DEPOT

placement during exchange of logistics tanks or for maintenance) and gives a satisfactory location for attitude control and settling thrusters. Also, the length is compatible with launch of both the logistics/storage tank and the equipment module in a single shuttle flight.

The logistics/storage tank for this mini-depot is quite similar to the rotational tank module. Structure, equipment, function, size, weight and cost are considered the same, though the transfer lines and tank ends will be reversed for settling of propellants toward the propellant transfer docking port.

Figure A9-10 defines the equipment module for the modular, linear mini-depot. It has essentially the same complement of equipment and functions as the rotational equipment module and much of the same design rationale applies. Systems allow the module to function independently in orbit with all monitoring, communication, rendezvous and docking, and attitude control provisions in addition to the checkout and propellant transfer systems compatible with the TUG. Fuel cells provide the power source and accumulation tanks, filled during transfer, hold the propellant for fuel cells and attitude control. The side docking ports have identical line interfaces and are interchangeable for use with the logistics/storage tank or the TUG. The volume of the module far exceeds what is required by the equipment. This will allow use of many "shelf" components and simplify fabrication and maintenance tasks. The module is launched with a six-year life expectancy with provisions to be returned to the ground for interim maintenance. The boom/counterweight approach is not required; and to reduce linear acceleration propellants expenditure, the structure would be of lightweight spacecraft design.

Figure A9-11 defines the permanent tankage, linear mini-depot. In addition to the propellant transfer equipment and orbital capability systems, this 60-foot module contains integral propellant storage tanks. The designs of a logistics/storage tank for the modular mini-depots must balance the amount of insulation installed to reduce boiloff against the corresponding additional weight that reduces the quantity of propellant that can be delivered per shuttle flight. The permanent tankage depot module requires launch (or return for maintenance and relaunch) only a few times throughout the program and due to its size, fills the volume of the cargo bay before approaching the shuttle cargo weight capability. Therefore, the weight of additional thermal protection systems is no penalty. In addition, the module can be launched with storage tanks empty which requires less tank support material. This reduces the heat transfer paths to the tanks and again minimizes boiloff.

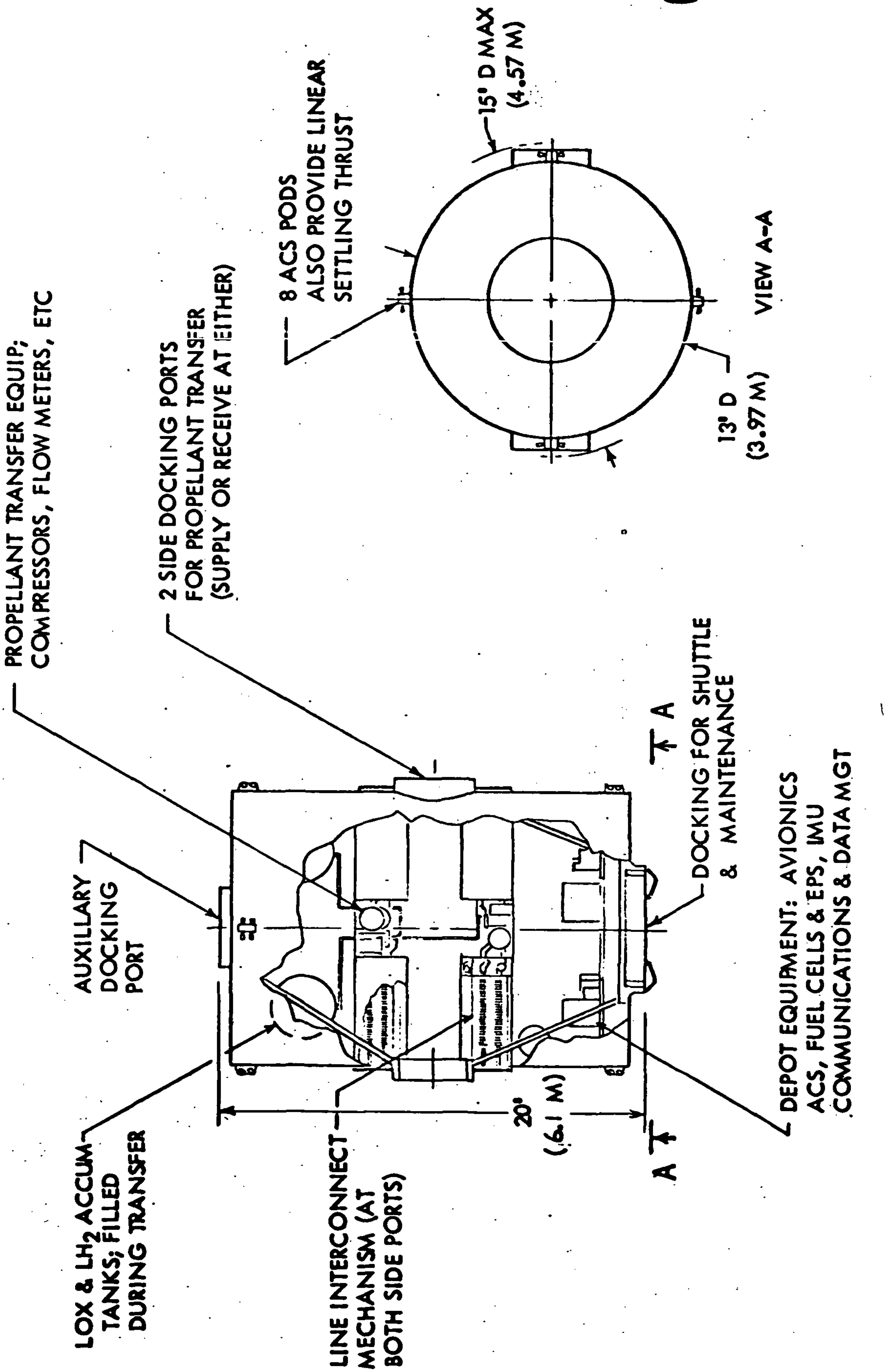


FIGURE A9-10 LINEAR MINI-DEPOT EQUIPMENT MODULE

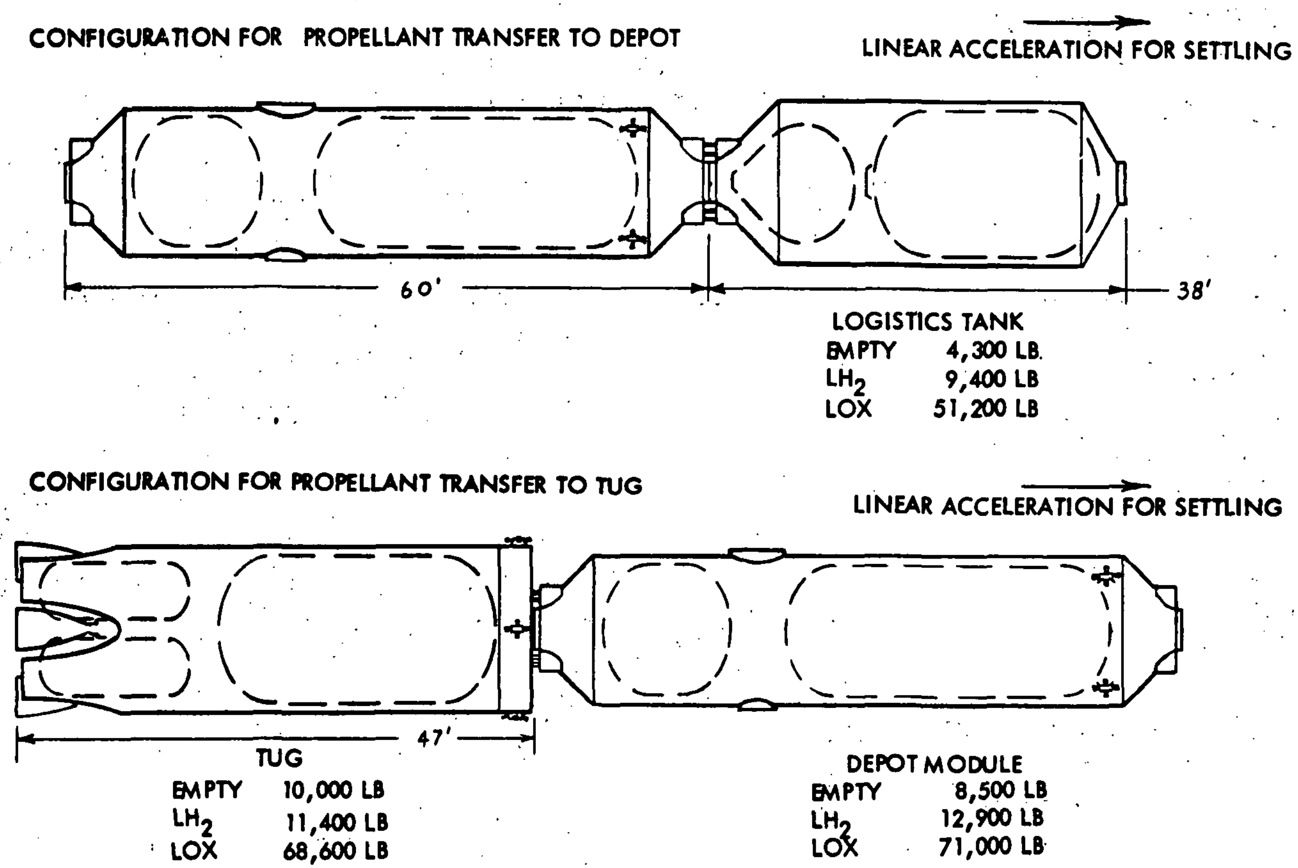


FIGURE A9-11 PERMANENT TANKAGE, LINEAR MINI-DEPOT

Figure A9-12 shows the operational configurations of the permanent tankage, linear mini-depot. Propellant transfer could take place with both the TUG and the logistics tank attached to the depot. Further study may indicate that this could be the desirable mode and that depot systems be included to allow simultaneous transfer from the logistics tank to the depot and from depot to TUG (or direct from tank through depot lines to the TUG). The figure, however, illustrates the configurations for separate transfer, in keeping with the foregoing operations rationale.

Figure A9-13 shows the operational configurations for the permanent tankage, rotational mini-depot. A preliminary definition of the depot module is not included, it being felt that extrapolation of the data from the other permanent tankage depot and the modular rotational depot would provide sufficient comparison data on this mini-depot candidate. The figures show a depot module similar to the linear depot module and a boom which is required for cg control. The depot module is slightly shorter and does not have propellant transfer capability at the outboard end docking port. The boom is a counter weighted, boilerplate type structure containing only the transfer lines (plus valves, insulation, etc.) and the line interconnect mechanism. Operation would be with separate transfer to the depot and to the TUG as shown by the two configurations. With both tank and TUG attached, the boom (as shown) would not prevent the combined cg from falling within the depot tanks during rotation and transfer.

2.5 LARGE ORBITAL PROPELLANT DEPOT

Candidate configurations of a large Orbital Propellant Depot (OPD) to support the TUG and CPS or TUG and RNS have been studied. Major considerations used in arriving at preliminary configurations include propellant storage capacity requirements, thermal control advantages, compatibility with propellant transfer subsystems and interface requirements. Table A9-1 is a listing of baseline system requirements that were considered in the development of the following OPD concepts: The concepts satisfy the needs of known potential space-based elements and are categorized into RNS or CPS supportive and non-modular or modular.

Reusable Nuclear Shuttle Supportive Depot

The depot configuration to support the RNS is illustrated in Figure A9-14 with details shown in Figure A9-15. Hydrogen tankage is provided by two LH_2 tanks derived from the use of Saturn S-II type structural elements. The tanks and the docking section are combined to form a single module 33 feet in diameter by 146 feet long. Located at both ends of the LH_2 module are attitude control system (ACS) thruster assemblies.

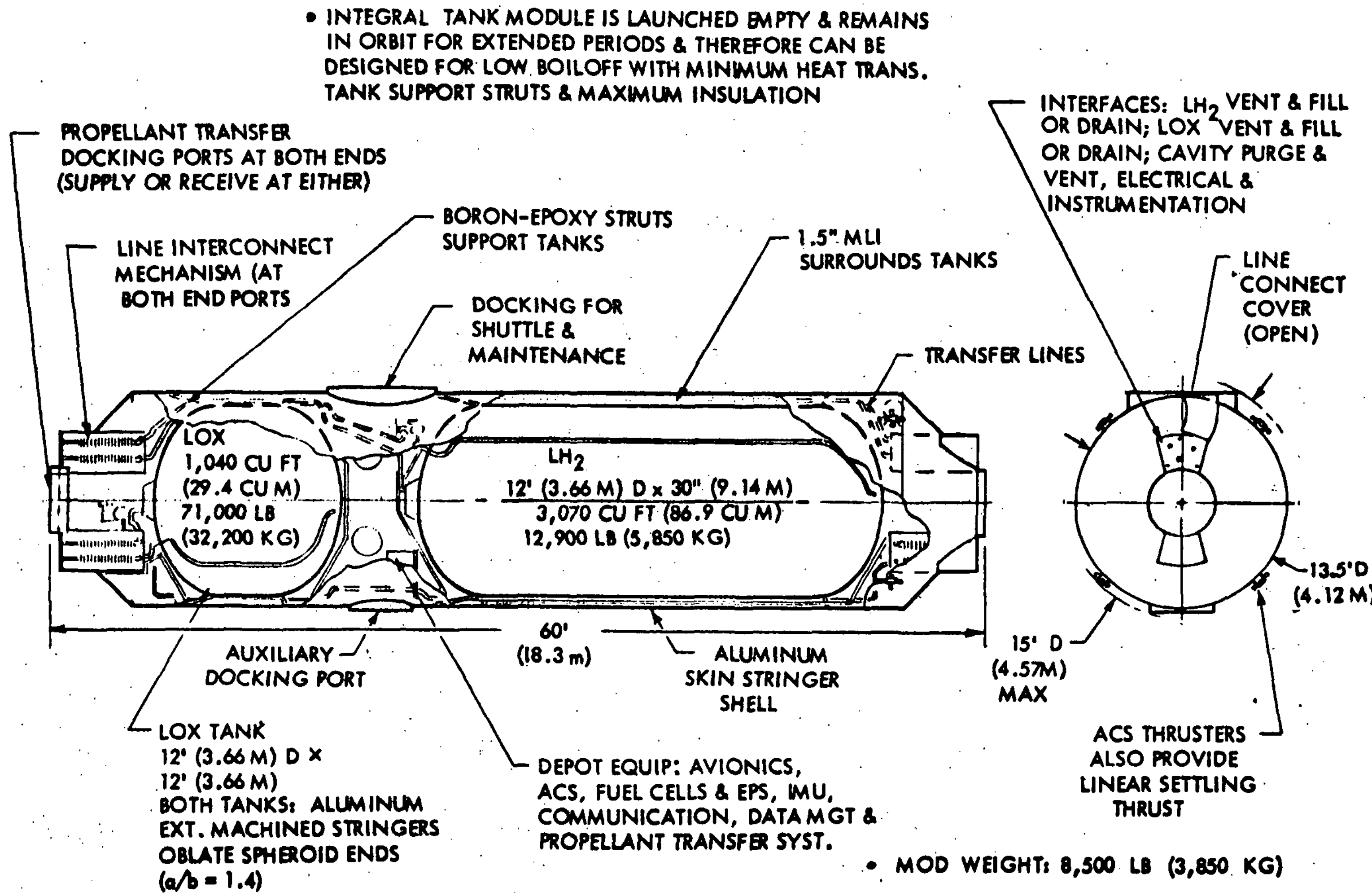


FIGURE A9-12 PERMANENT TANKAGE, LINEAR MINI-DEPOT MODULE

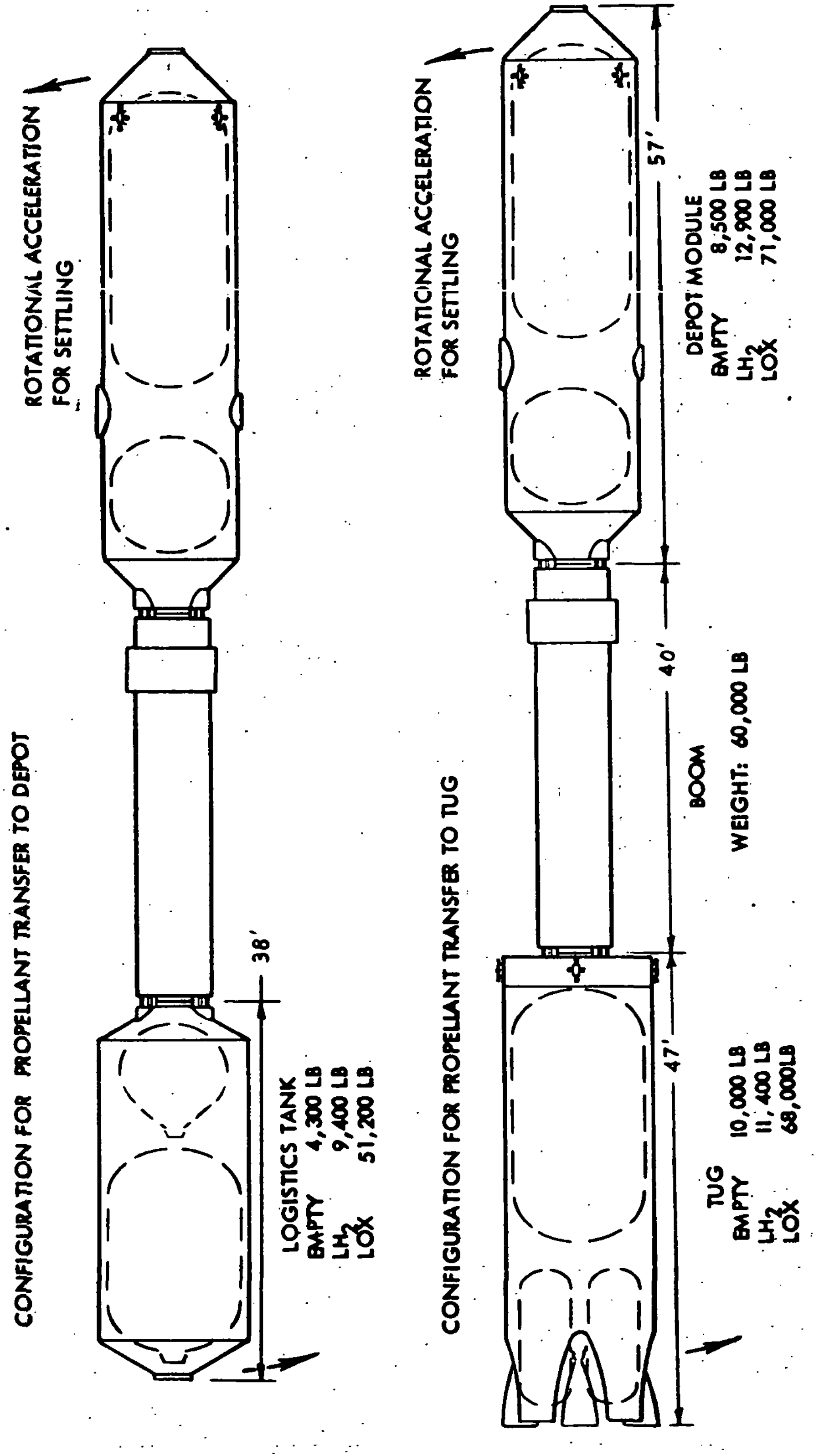


FIGURE A9-13 PERMANENT TANKAGE, ROTATIONAL MINI-DEPOT

TABLE A9-1 BASELINE OPD SYSTEM REQUIREMENTS

Design Element	Requirement
OPD configuration	Spoke type
Propellant logistics	EOS propellant P/L tank
Propellant transfer system	
Thermodynamic control	Vapor return
Pressurization	Gas generator heat exchanger
Expulsion	Pump
Liquid location	Rotational acceleration
Propellant gauging and instrumentation	Gravity-dependent mass sensors, temperatures, pressures, flowrates
Thermal control system	
Insulation	HPI and heat blocks
Vent system	Zero-g thermodynamic and regenerative heat removal
Destratification	Circulation pumps and spray nozzles
Orientation	Free drift
Propellant state	Saturated liquid
Rendezvous	Passive
Docking	
Propellant with crew provisions	Two ports (minimum)
Crew	One port
Alignment	Mechanical Indexing and latching
Crew provisions	Maintenance crew module
Attitude control systems	Low pressure, propellant tank bleedoffs
Operations	
Orbital assembly	SS or tug
Activation maintenance	
Ancillary modules	SS P/L replacement
Components	Modular (shirtsleeve)

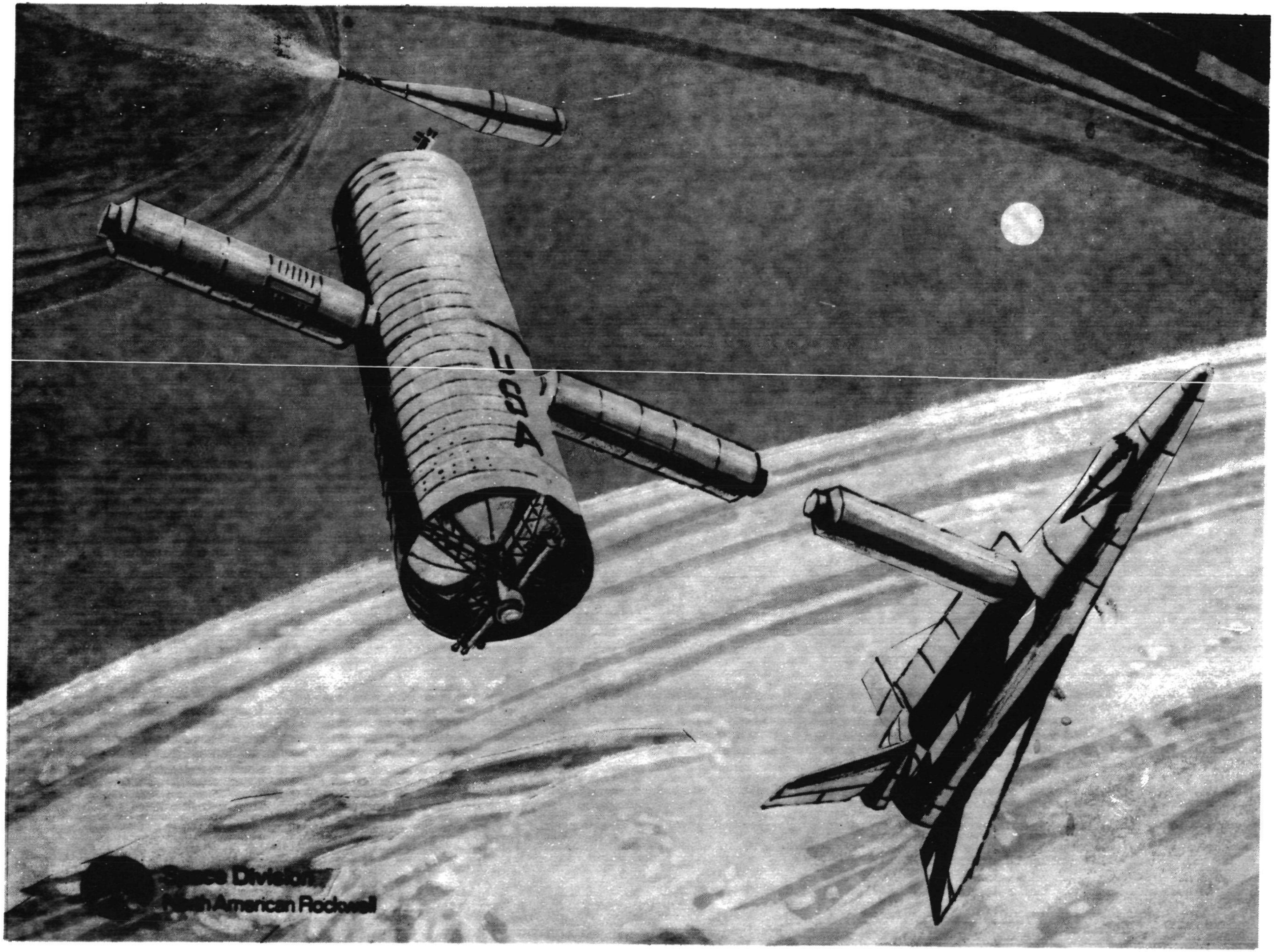


FIGURE A9-14 RNS SUPPORTIVE, NON-MODULAR ORBITAL PROPELLANT DEPOT

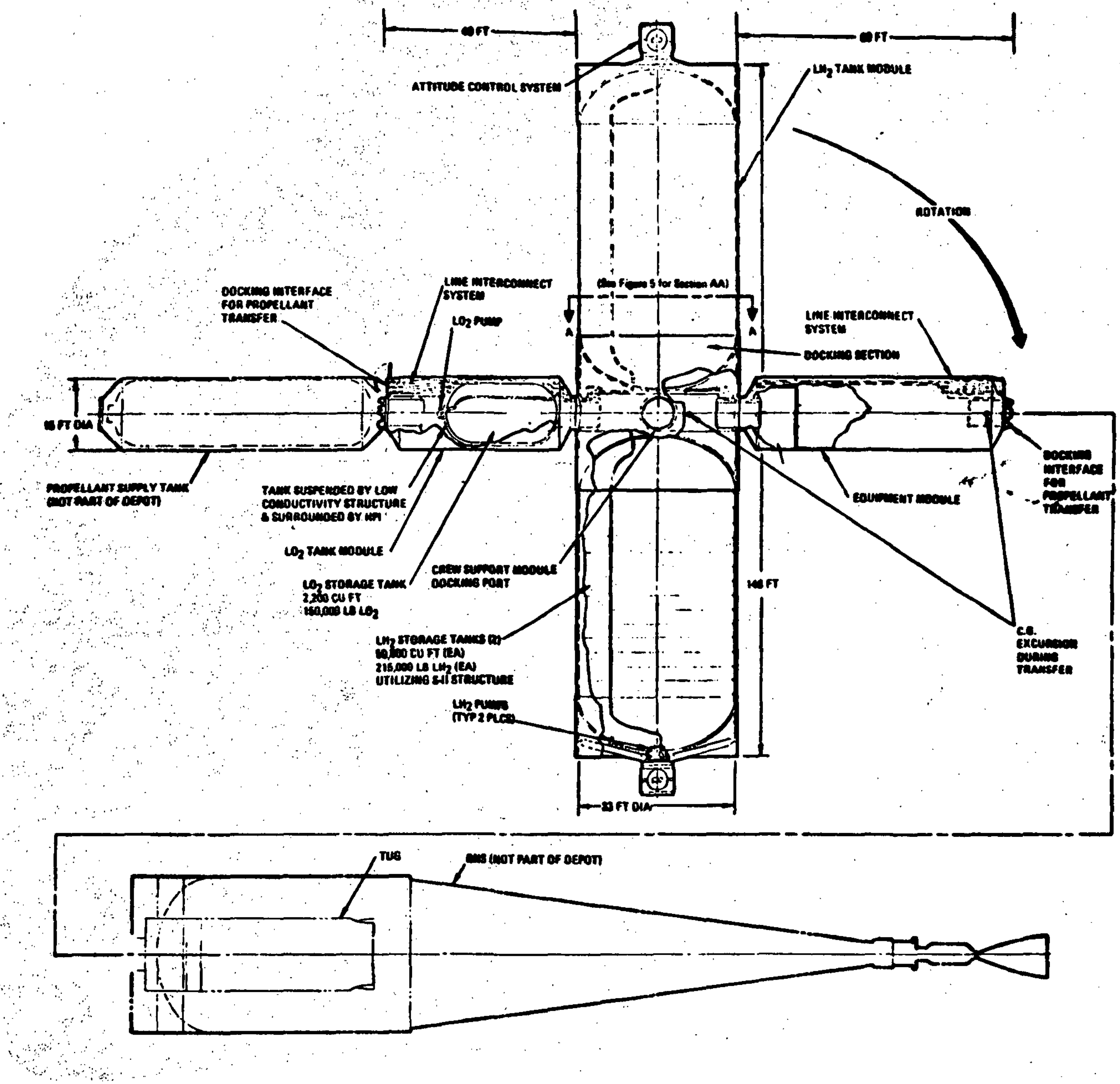


FIGURE A9-15 RNS SUPPORTIVE ORBITAL PROPELLANT DEPOT

The docking section has four docking ports. Two are depot assembly attach ports, the third is for crew module docking, and the fourth is an auxiliary docking port. The docking section contains some propellant transfer lines and other equipment. An airlock access is provided for maintenance of the transfer lines. Pumps at the outboard ends of the LH_2 tanks and a 4-inch transfer line inside the tank bring propellants to the docking section at the center of the module where they branch and are routed to each of the propellant transfer interfaces. The capacity of each LH_2 tank is 50,000 cubic feet.

The required oxygen is stored in a 15-foot diameter by 40-foot long LO_2 tank module connected to the LH_2 module at one of the assembly ports as shown in Figure A9-15. The tank module is an integral part of the depot but can be removed and returned to earth in a shuttle cargo bay, if required, for major maintenance or modification. The LO_2 tank proper is 12 feet in diameter by 23 feet long and has a capacity of 2200 cubic feet. It is an aluminum tank suspended within the external structure of the module to minimize heat transfer. A docking interface consisting of a seven-foot ring/cone docking fixture and a line interconnect fixture are located at the outboard end of the LO_2 module. Both LH_2 and LO_2 can be transferred at this interface, the primary docking port for the earth to earth-orbit propellant logistics tank for resupply of the depot.

A 15-foot by 60-foot equipment module is connected to the other assembly port. It is an integral part of the depot but can be returned to earth if required for major maintenance or modifications. The equipment module contains most of the subsystems required for depot operation. The inboard end of the module contains a crew station which includes equipment for on-board control, monitoring, and checkout of the depot by the maintenance crew. There is access from the inboard end of the crew compartment to the passageway in the docking section. A docking interface consisting of a seven-foot diameter ring/cone docking fixture and line interconnect fixture on the outboard end of the equipment module provide for vehicle docking and propellant transfer primarily to the user vehicles. An airlock is provided for crew ingress and egress.

A crew module may be docked at one of the two docking ports on the 33-foot docking section during manned occupancy of the depot. Alternatively, the crew module can dock at other ports on the depot for inspection or maintenance at those locations. The module will provide the crew with living quarters and life support requirements during its depot stay and will share a common generic design with other elements such as the TUG, RNS, or CPS.

It is necessary to properly settle propellants within the tanks to facilitate their transfer to the using vehicle. This is done by rotating the depot to generate an artificial gravity field during transfer. Liquid gas interface control allows pumping of the liquid to the receiver tank and separate return of the ullage gas from the receiver tank. Rotation will take place about the combined center of gravity of the depot and the user. Initially, the cg is close to the center of the depot. As the mass of propellants is transferred to the user vehicle, the cg and therefore the center of rotation will shift toward the vehicle being serviced. The cg must remain on the depot side and outside of the tanks directly involved in the transfer to ensure proper settling during transfer. The symmetrical arrangement and simultaneous transfer from the opposed LH_2 tanks will maintain the cg location along the centerline of the smaller depot modules. The equipment module acts as a boom to extend the user tank beyond the reach of the cg excursion. The location of the LO_2 tank opposite the user docking port acts as a counterbalance to reduce cg travel.

Because of these center-of-gravity considerations, the RNS and TUG will normally receive propellants at the equipment module docking port. As a contingency, the TUG can refuel at the opposite port. The depot can be resupplied at either port enabling the shuttle to alternately dock the full logistics tank to one end and retrieve the empty tank (previously delivered) from the other. The two transfer interfaces also allow simultaneous transfer.

Propellant transfer equipment will include pumps and pressurization (gas generator, heat exchanger) systems.

Thrust for propellant settling spinup and despin is provided by the attitude control system (ACS). The outboard location of the thrusters on the LH_2 module provides maximum moment arm. The ACS uses the low-pressure LO_2 and LH_2 from the depot tanks for propellants. Other major ACS components are part of the ACS modules attached to the ends of the LH_2 module by use of standard docking hardware. The entire module can be replaced and returned in the space shuttle to earth for refurbishment if required.

Propellant losses due to boiloff are minimized by the use of high performance insulation (HPI) and low thermal conductivity structural materials at discrete locations. The LH_2 module has high-performance insulation external to the cylindrical portion of the module and at the tank ends. Heat conduction through the structure to the tank walls is controlled by the use of titanium for the skirt and docking section structures. The LO_2 tank is suspended within the module shell by low thermal conductivity struts or spokes and is surrounded by HPI. Additional insulation and thermal control will be provided as required for maintenance crew occupancy of portions of the modules and for subsystems equipment.

Chemical Propulsive Stage Supportive Depot

The depot configuration to support the CPS is illustrated in Figure A9-16 and shown in more detail in Figure A9-17. The depot is comprised of four separately launched modules that are joined in orbit. The two larger modules are 33 feet in diameter and provide LH_2 and LO_2 storage. The dual symmetrical tank arrangement of the depot allows for cg balance during propellant transfer.

As shown in Figure A9-17, the tank modules are joined at the 33-foot diameter docking section. The docking section has two assembly ports, one for joining the equipment module and one for joining the propellant transfer module: It also has two docking ports, one for the crew module, and one which is an auxiliary port. The outboard ends of the equipment and propellant transfer modules provide for docking of user and supplier vehicles.

The principal docking port for users is at the outboard end of the equipment module. To simplify shuttle supply tank handling, propellants can be supplied to the depot at either outboard docking port. Rotation provides artificial gravity for propellant settling during transfer. Sequential transfer of first LH_2 and then LO_2 to the CPS is required to keep the combined cg from falling within the LH_2 tank during LO_2 transfer. The equipment module and the crew module are similar to those of the RNS supportive OPD configuration. A propellant transfer module opposite the equipment module provides clearance for docking and houses the line interconnect system.

Thermal control considerations are similar to the RNS supportive depot. Tank modules will have external HPI and meteoroid protection attached before launch and the tanks will be isolated by low conductivity skirts. The tank ends also will have external HPI and, with other heat paths minimized, the thrust structure and engines will not require complete insulation.

Attitude control jet packages are the same as on the RNS depot. They will be attached to the aft docking ring of the tank modules during orbital assembly of the depot.

Modular Depot

A modular configuration for the RNS depot is illustrated in Figure A9-18. The configuration, shown in more detail in Figure A9-19 consists of 16 LH_2 modules, an LO_2 module, an equipment module, a crew support module, an intersection module, and 6 boom segments assembled in the geometric arrangement shown.

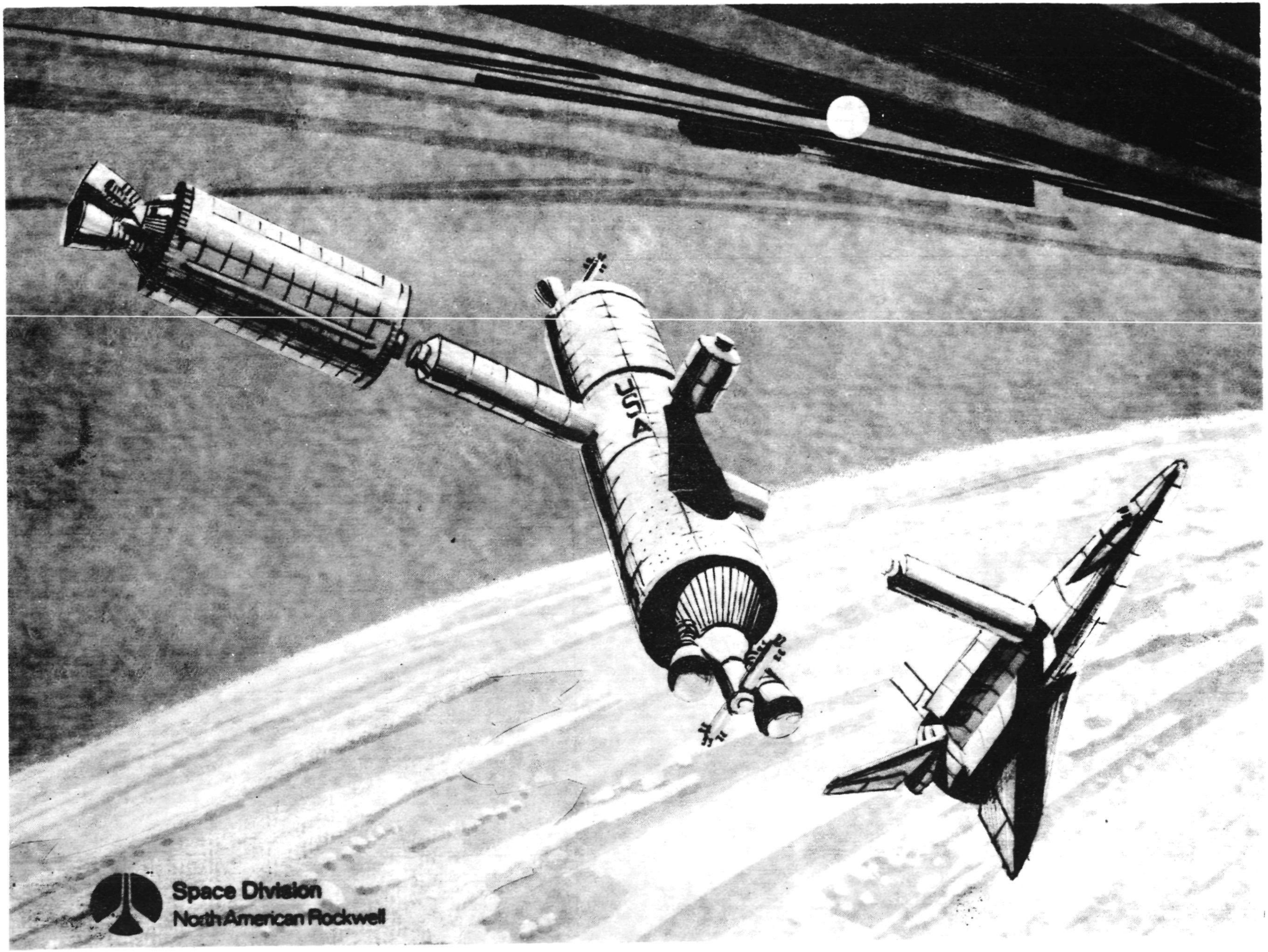


FIGURE A9-16. CPS SUPPORTIVE, NON-MODULAR ORBITAL PROPELLANT DEPOT

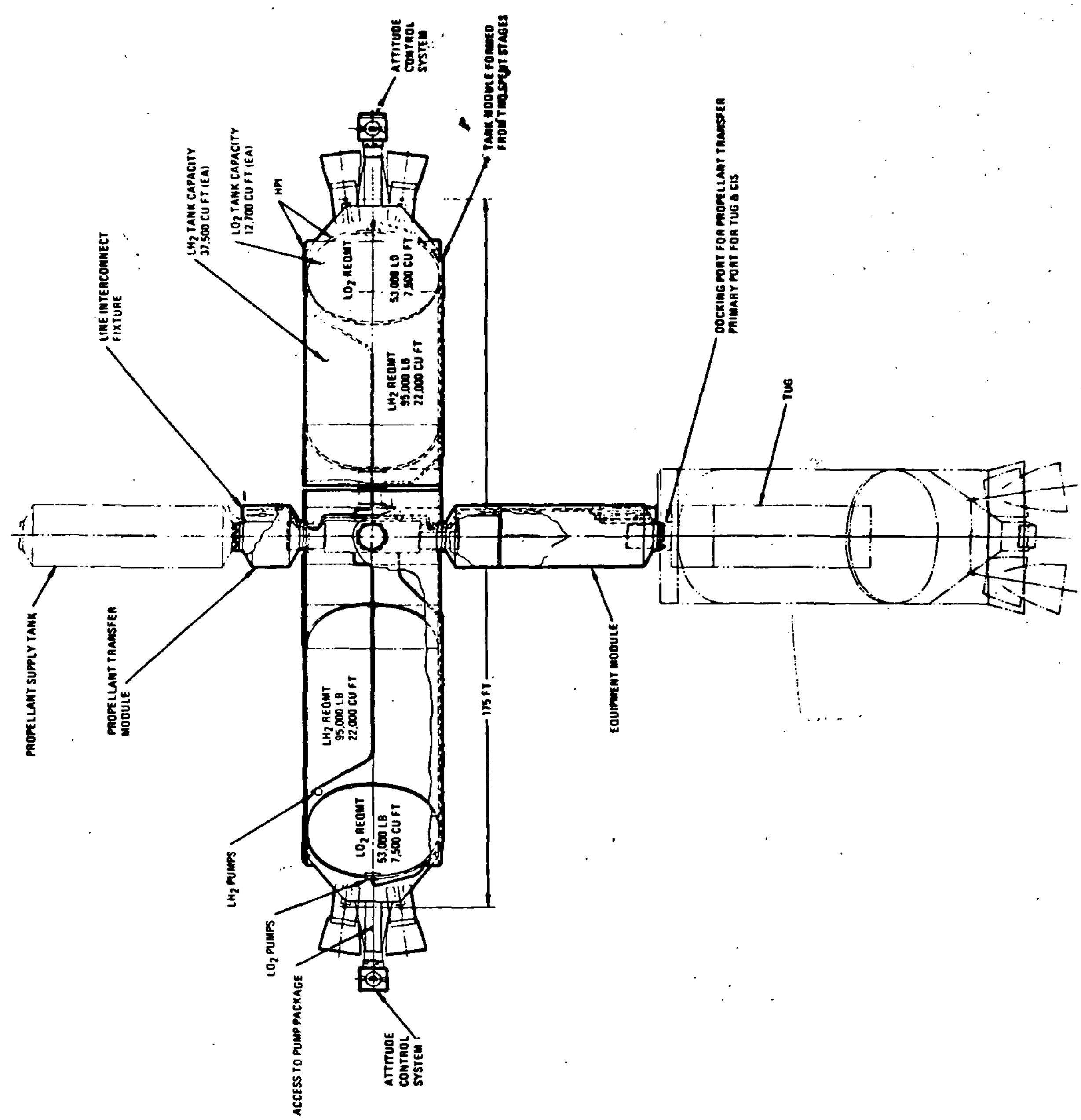


FIGURE A9-17 CPS SUPPORTIVE ORBITAL PROPELLANT DEPOT

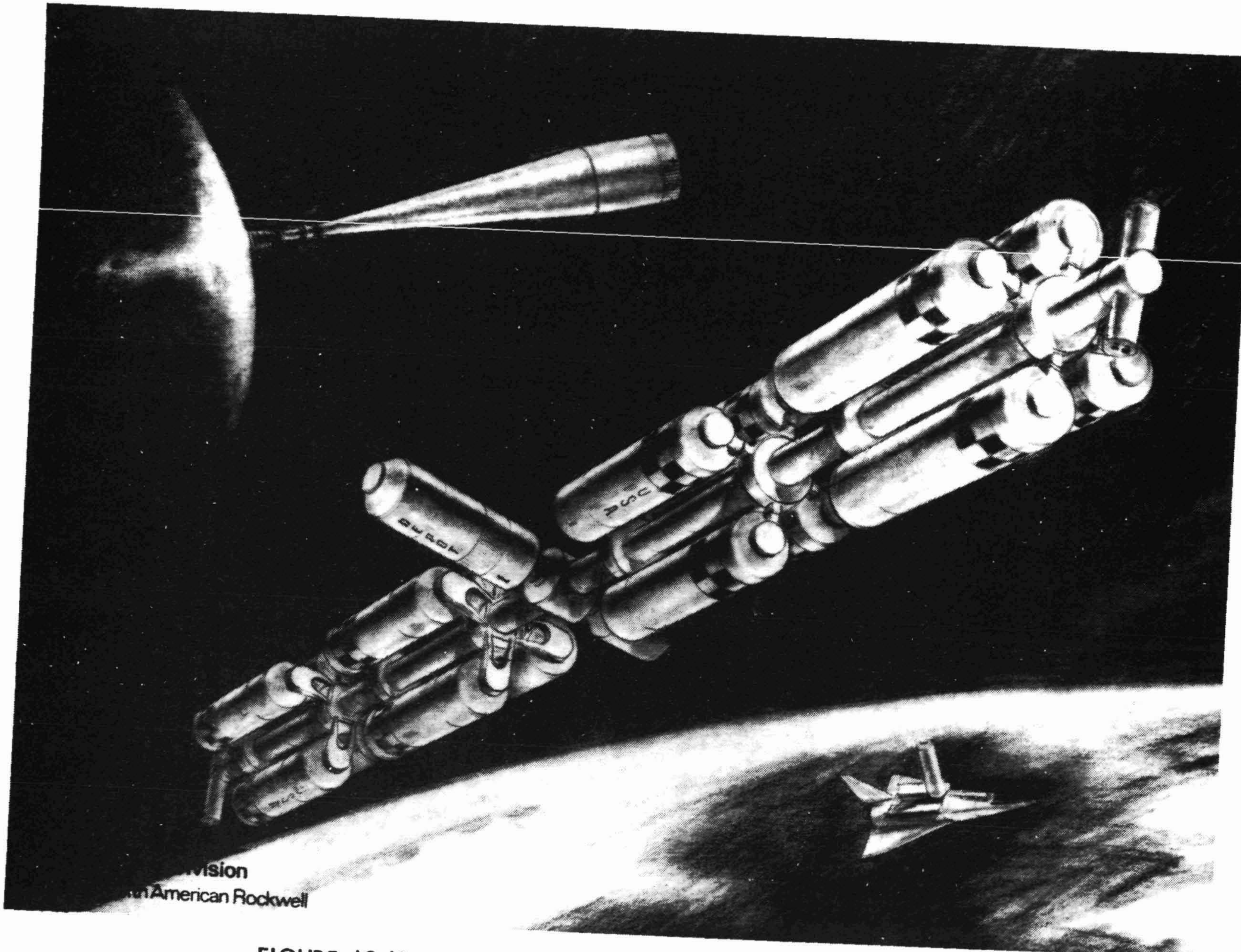


FIGURE A9-18 RNS SUPPORTIVE, MODULAR ORBITAL PROPELLANT DEPOT

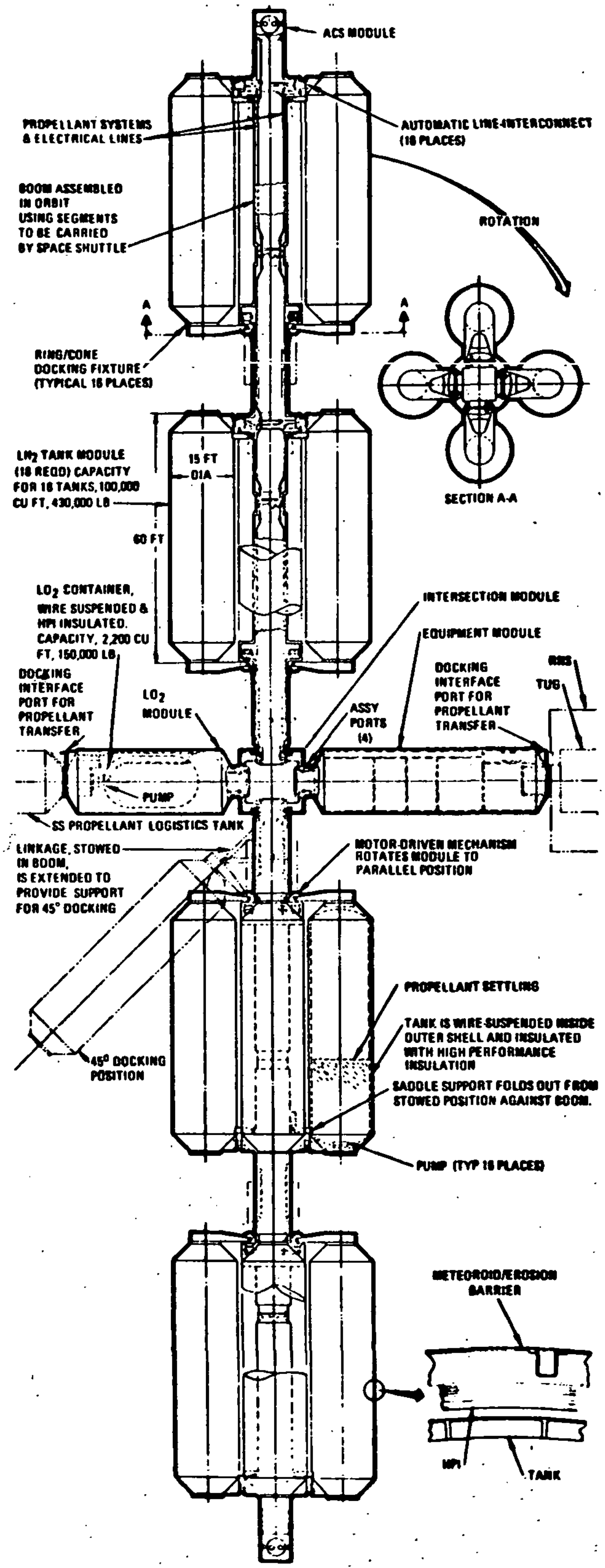


FIGURE A9-19 RNS SUPPORTIVE, MODULAR ORBITAL PROPELLANT DEPOT SCHMETIC

The intersection module is 15 feet in diameter and has four assembly ports and one docking port. It is the hub of the depot with three connected boom segments attached to each side of the module at opposite assembly ports and the LO_2 module and equipment module attached at the two end assembly ports, and at right angles to the boom. The remaining docking ports can receive the crew module.

The boom consists of six 60-foot segments assembled in three sections end to end on each side of the intersection module. Boom segments provide for attachment of tank modules by 7-foot diameter ring/cone docking fixtures, attached to movable arms. LH_2 tank modules are docked either 45 degrees or perpendicular to the boom, as is desirable operationally and rotated by electric motors to a position parallel to the boom.

The 16 LH_2 modules are each 15 feet in diameter and 60 feet long and have a total capacity of 100,000 cubic feet. The design incorporates a tank-within-a-tank concept, low heat-leak suspension, and high-performance insulation. The 16 modules are attached in four clusters of four to the boom.

The LO_2 module is 15 feet in diameter and 40 feet in length and has a capacity of 150,000 pounds of LO_2. It is attached to one end of the intersection module with its centerline at a right angle to the boom and in line with the equipment module on the opposite end of the intersection module. It has a docking port on the outboard end for receiving propellants from the shuttle orbiter propellant logistics tank.

The equipment module is 15 feet in diameter and 60 feet long. It is longer than required for the equipment in order to confine the center-of-gravity excursion during fluid transfer to an area outside the user tank. It has a docking port at the outboard end to which the using vehicles dock to take on propellants.

The LO_2 module, equipment module, and crew module are identical to those described for the RNS depot and are subject to the same maintenance provisions. All of these modules and the boom segments are capable of return to earth in the shuttle for replacement or refurbishment.

Propellant lines are connected to a manifold inside the boom by a line interconnect fixture during orbital assembly. The fixture is oriented perpendicular to the boom in a location that will match the line receiver fittings near the tank end, opposite the docked end. Together, the tank modules and boom segments contain all necessary plumbing, valves, and pumps. Electrical connections between modules are made by the interconnect mating operation.

The depot tanks are a minimum boiloff design. To achieve this, it is necessary to minimize structural support (thus reducing heat transfer paths) below that required to withstand space shuttle launch and orbiter abort loads with full tanks. Therefore, the depot tanks are deployed empty and become fixed depot equipment. The propellant is transferred into the depot and from the depot to the user vehicles by fluid flow.

Passageways through all boom segments can be entered from the crew module via the intersection module. Sufficient clearance is provided to permit access to all boom compartments for maintenance and checkout. Pressurization and environmental control of the boom will be limited to critical compartments.

The attitude control system package required for maneuvering and propellant settling by rotation will be attached to the outboard boom segments and will utilize propellants from the supply tanks attached to that segment. Operating procedures will ensure that LH_2 is available at all times in the outboard tanks to support maneuver and attitude control.

Since all of the modules that make up this concept are within the 15-foot diameter by 60-foot long dimensional limitations of the space shuttle cargo bay, the shuttle will be the only means required to transport the depot modules to orbit. It may be possible to control assembly in orbit from the space shuttle orbiters, but space tug assistance is required as a baseline.

The configuration for support of the RNS also is adaptable for CPS support as shown in Figure A9-20 by arranging six LH_2 tank modules symmetrically about the centerline through the user vehicle. These will be attached to booms in the same manner as is done in the RNS version in a single cluster of three on each side of the centerline. The LO_2 tanks will be attached in tandem to the central docking hub opposite the equipment module. This will influence the center of gravity in that direction and avoid shifting it into the user tank during propellant transfer. All other considerations for the CPS version remain the same as for the RNS version.

2.6 PROPELLANT GAUGING

For fluid transfer regardless of the particular transfer concept, some type of propellant gauging is required for status monitoring and control of the fluid transfer operation.

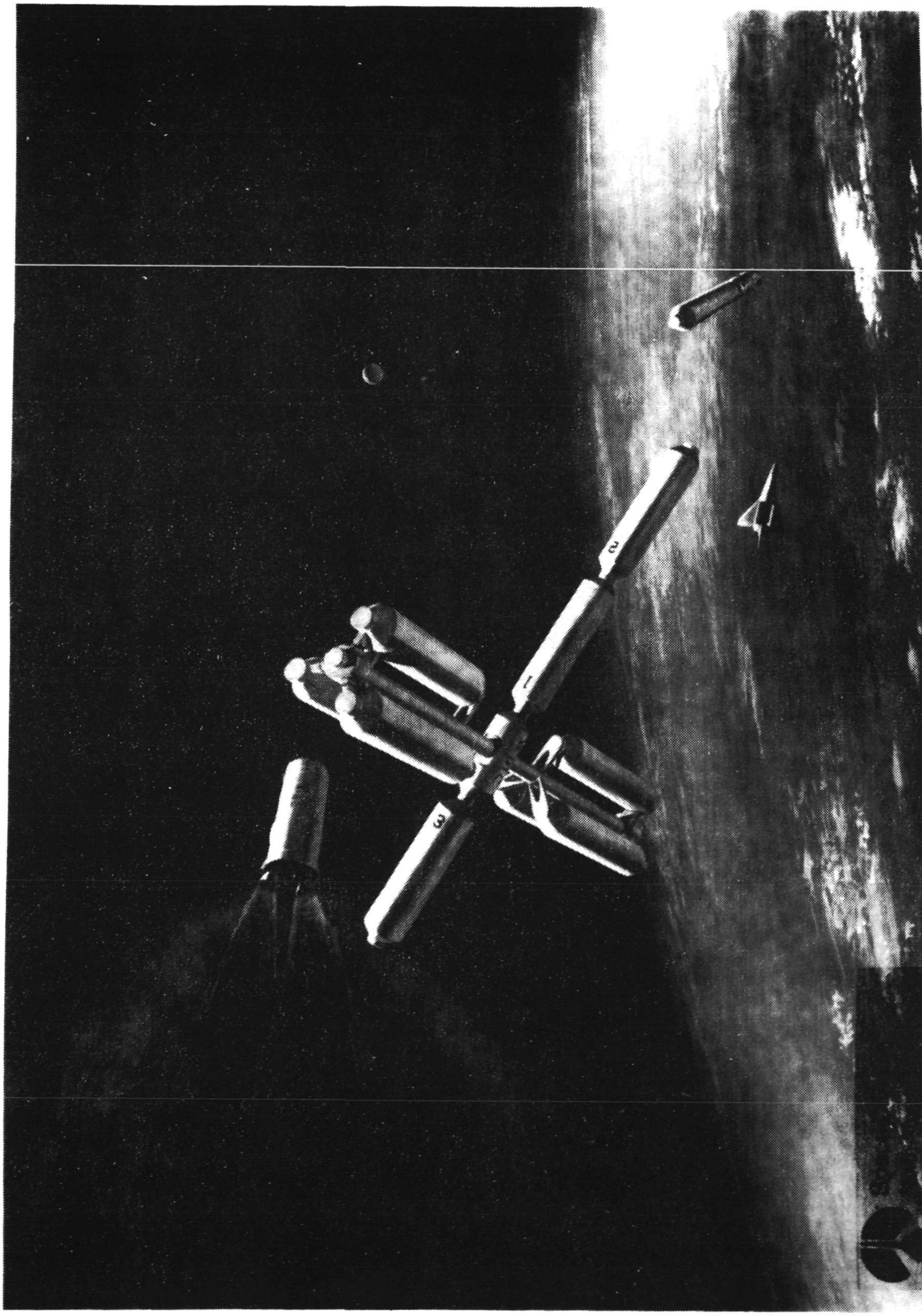


FIGURE A9-20 CPS SUPPORTIVE, MODULAR ORBITAL PROPELLANT DEPOT

Propellant gauging, under positive gravity conditions, is generally simplified due to the presence of an acceleration vector that allows prediction of the propellant location and shape within the confines of the tank. Point sensors of the ultrasonic, optical, or heated wire type are all capable of accurate propellant level (volume) measurements. Propellant volume measurement devices of the echo-ranging and capacitance type which provide continuous measurements are also adaptable. These systems, however, are useful for propellant quantity indication only when a reference vehicle acceleration is coincident with the vehicles longitudinal axis allowing prediction of the propellant locations. Under zero g conditions, propellants are randomly oriented within a tank and cannot be measured by normal methods. However, only general quantity gauging may be required. Infrasonic, radio frequency, or radiation systems may satisfy the zero g requirements but further development is required.

For the orbital propellant depot, a propellant gauging system was developed to satisfy the particular needs of fluid transfer at the transfer acceleration level. This system is conceptual and has been defined as a baseline only. The OPD propellant gauging system description and requirement are as follows:

Accuracy	± 1 percent of full tank
Power	100 watts maximum at 28 vdc input
Response	10 seconds maximum for full-scale change
Output	0-to 5 vdc for full-scale change
Stability	750 hours minimum between service and calibration
Linearity of electronics	± 0.25 percent of full scale
Warmup	20 minutes maximum
Checkout	Capability to simulate various propellant tank modes for automatic checkout purposes

The system selection analysis indicated that a combination of the segmented capacitance mass sensor system and a backup discrete point sensor system best satisfied the requirements of the OPD propellant gauging system for the non-modular RNS and CPS baseline configurations. The segmented capacitance mass sensor system provides a self-correcting feature that overcomes the undesirable effects of capillary and meniscus characteristics of the low gravity environment of the OPD. The discrete level sensor system is not affected by the low-gravity environment of the OPD and each system is capable of cross-checking the other system.

The segmented capacitance sensor system consists of a series of independent variable capacitance sensors that extend over the length of the propellant tank and one electronics assembly for each sensor. It is expected that four capacitance sensors per tank will comprise the capacitance sensor system. A schematic diagram for one unit of the segmented capacitance sensor system is shown in Figure A9-21.

Each capacitance sensor is made up of two concentric cylinders that form the plates of a capacitor. As the propellant rises in the tank, the propellant between the capacitor plates increases the dielectric value between the plates and results in increased capacitance developed by the sensor. The electronics assembly associated with each sensor converts the capacitance variations to an analog voltage that is indicative of the level of propellant within the sensor.

During tank fill at the low-gravity conditions of the OPD, the propellant within each sensor will rise above the propellant level of the tank due to capillary effects. Because the top of each lower sensor is at a higher elevation than the bottom of the next upper sensor, the capillary rise is revealed at the time the next upper sensor detects propellant. Once the capillary rise is known for the respective segment, it is continuously subtracted from the sensor indication and the sum of the corrected values for all the sensors represents the propellant level within the respective OPD tanks.

During tank depletion at the low-gravity conditions of the OPD, liquid can be retained within the capacitance sensors due to meniscus effects. By utilizing area heating on the outside surface of the capacitance sensor, the retained liquid is evaporated and the respective capacitance sensor will properly gauge the rate of tank depletion.

Both capillary action and meniscus effects can be reduced by increasing the plate separation of the capacitance probes. Adding vertical slits up the side of the probes also will be beneficial. Both of these solutions require additional study to determine the details of the probe structure and possible effects on signal strength.

The discrete level sensor system consists of a series of independent variable resistance sensors that extend over the length of the propellant tank and one electronic controller for each sensor. It is anticipated that 20 discrete sensors per tank will comprise the discrete level sensor system. A schematic diagram for one unit of the discrete level sensor system is shown in Figure A9-22.

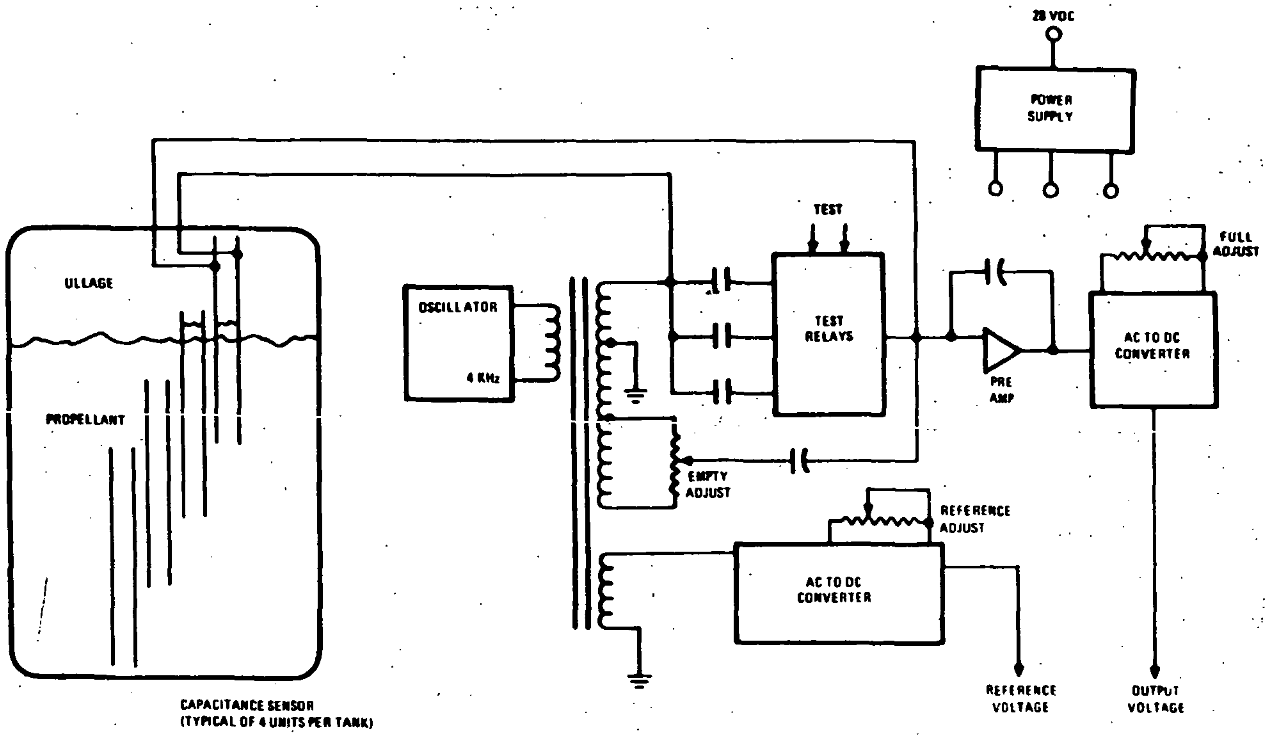


FIGURE A9-21 SEGMENTED CAPACITANCE SENSOR PROPELLANT GAUGING SYSTEM

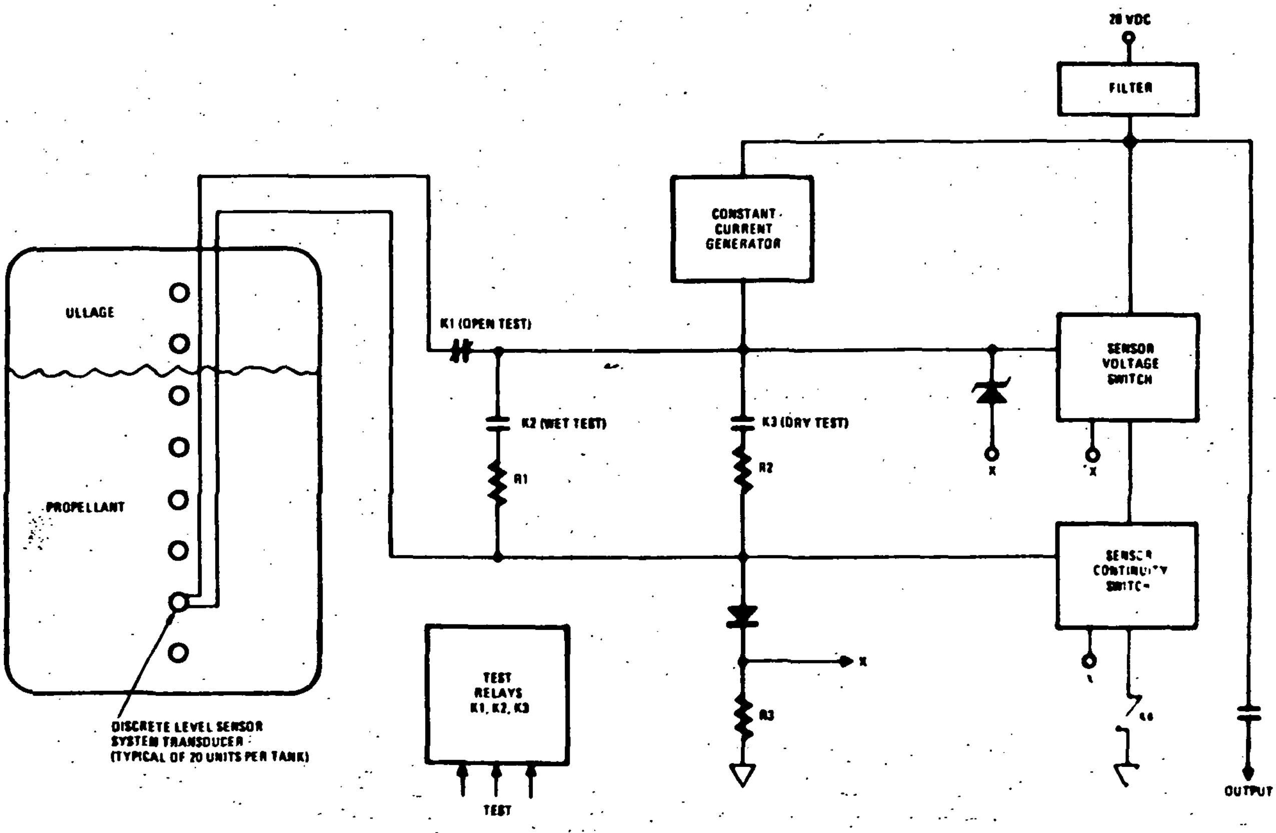


FIGURE A9-22 DISCRETE LEVEL SENSOR PROPELLANT GAUGING SYSTEM

Each discrete sensor is made up of a coil of wire that conducts constant current and undergoes a sharp resistance change downward when transferred from cryogenic vapor to cryogenic liquid. The electronics assembly associated with each sensor converts the sensor change to an output voltage change that is indicative of whether the sensor is "wet" or "dry".

The discrete level sensors are mounted in the tank in such a manner as to overcome the effects of capillary action due to the low-gravity environment of the OPD during tank fill. The liquid retained in each discrete sensor during tank depletion by the meniscus effects at low-gravity conditions of the OPD is readily evaporated by the heating effect of the constant current technique employed by the discrete level sensor system.

While both the capacitance probe and point sensor systems will be affected by propellant sloshing, it is hoped that proper tank design including baffles and other devices will greatly reduce the slosh effect.

To complement the segmented capacitance probe and discrete point sensor gauging systems, flow metering devices will be placed in all OPD fill and drain lines to determine the amount of propellant being transferred into or out of the OPD.

3.0 FLUID TRANSFER ANALYSIS

An analysis was conducted to define the functional requirements for in-orbit fluid transfer and to establish the characteristics and merits of the various techniques by which the transfer may be accomplished. The results of the analysis have been used for selecting the technically preferred techniques for in-orbit fluid transfer operations.

The potential fluid transfer logistic interfaces considered are depicted on Figure A9-23. The two major transfer options considered for transfer of propellant between the various space elements are: (1) modular transfer (the transfer of packaged or contained propellant as a unit), and (2) fluid transfer. Modular transfer results in minimum propellant loss and is the first preference if and when the receiver element is suited to this transfer technique. The remainder of this section deals exclusively with the fluid transfer mode of propellant transfer.

In-orbit fluid transfer can be divided into four major subsystems. Each subsystem must provide the necessary control to accomplish the following functions:

A. <u>Liquid/Vapor Interface Control</u>

Provide control to insure acceptable supplier tank outflow, liquid phase or acceptable quality through the transfer lines, and acceptable receiver inflow conditions.

B. <u>Receiver Tank Thermodynamic Control</u>

Provide control to insure acceptable inflow characteristics, prevent unnecessary overboard venting of liquid or vapor, and maintain or establish receiver propellant thermodynamic conditions which fulfill the receiver vehicle's propulsion system or outflow requirements.

C. <u>Expulsion</u>

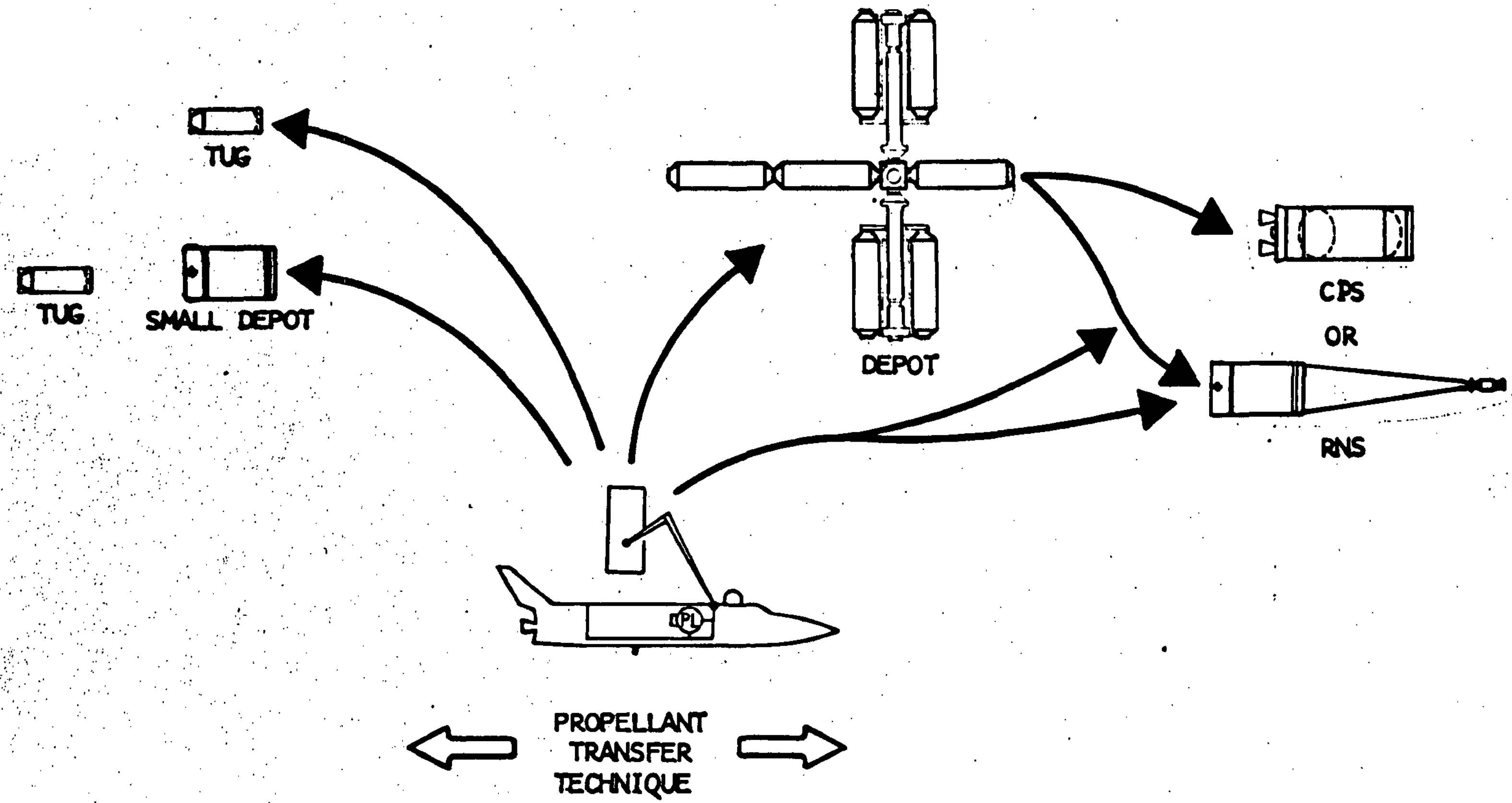
Provide the energy and/or means of expelling the propellant from the supplier into the receiver.

D. <u>NPSP Control</u>

Vapor pressure control must be provided to establish subcooled or acceptable quality propellants to fulfill the requirements as established by the total transfer system.

3.1 LIQUID/VAPOR INTERFACE CONTROL

Probably the most critical and most difficult requirement to achieve in support of orbital propellant transfer is liquid/vapor interface control. Previous data have shown that the concepts shown in Figure A9-24 are most promising. The concepts employ either acceleration or surface tension for liquid/vapor interface control. Criteria used to select the technically preferred technique included:



MODULAR

- NO PROPELLANT LOSS
- DEPENDENT ON RECEIVER CONFIGURATION

FLUID

- SIGNIFICANT PROPELLANT LOSS
- COMPATIBLE WITH USER VEHICLE CONFIGURATIONS

FIGURE A9-23 IN-ORBIT PROPELLANT TRANSFER INTERFACES

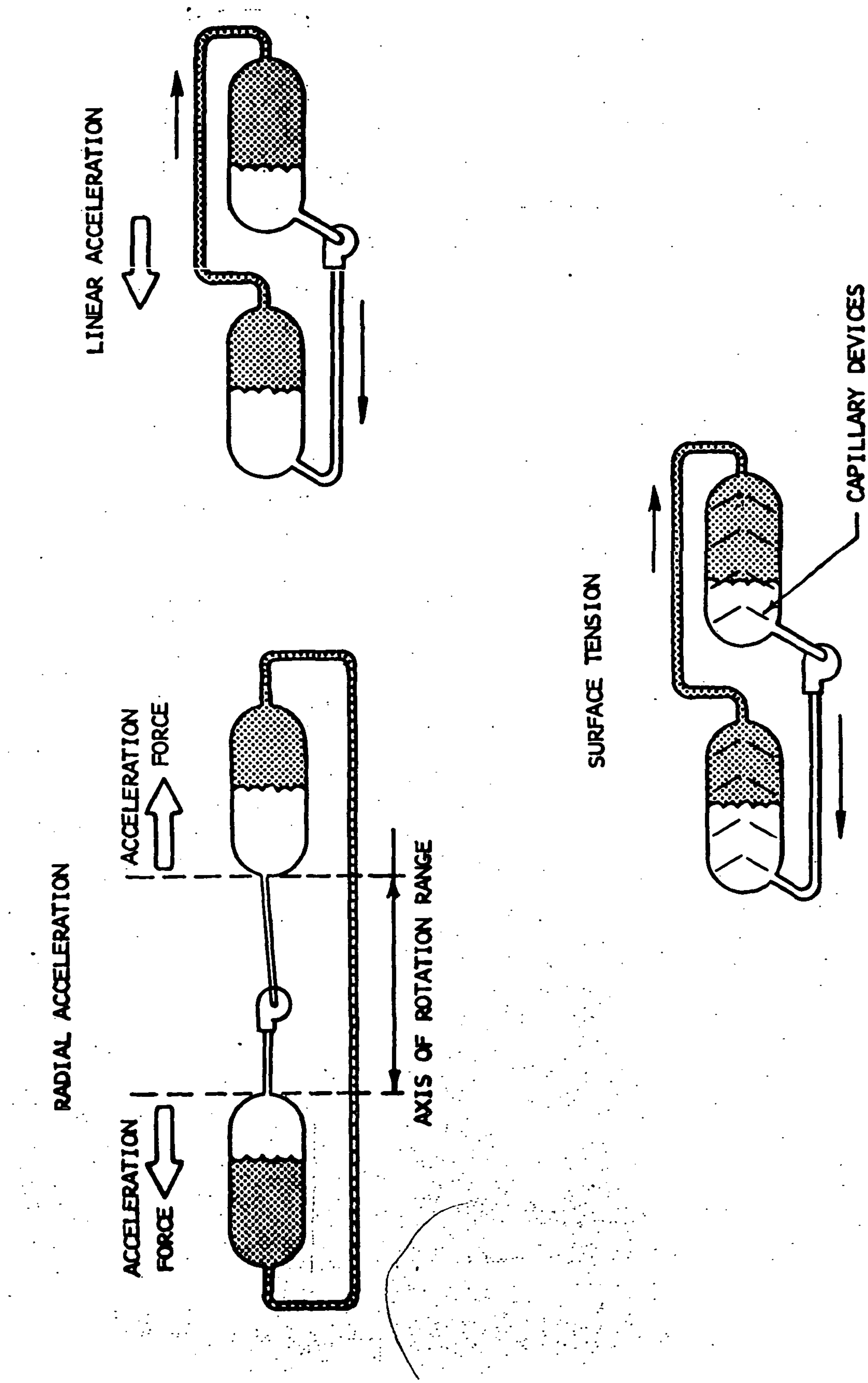


FIGURE A9-24 LIQUID/VAPOR INTERFACE CONTROL CONCEPTS

Technical feasibility
Propellant transfer losses
Compatibility with user and logistic vehicle system
Development risk
Safety

Propellant transfer losses include such items as jet propellant, logistic tank residuals, pressurization, pumping power, transfer line heat leak, transfer line residuals and tank and line chilldown.

Figure A9-25 presents the data used to relate the logistic tank propellant residuals with the jet propellant (the propellant consumed to provide constant linear acceleration of 10^{-4} G).

The minimum loss transfer time for an acceleration factor of 10^{-5} G was determined to be approximately 80 hours. In order to reduce this time to a more practical value, the factors were determined for an acceleration of 10^{-4}; the total loss curve for the 10^{-5} G condition is shown for reference. OPD draw down data were used to estimate the residual propellant characteristics. This evaluation indicates that the minimum propellant loss for the 10^{-4} G condition is approximately 4.5 percent of the transferred propellant with a transfer time of approximately 15 hours.

The linear acceleration technique requires constant thrust application for the full duration of the transfer with the transfer losses primarily a direct function of time and an inverse function of acceleration. Conversely, the rotational acceleration technique requires thrust for spin-up and spin-down only; the propellant transfer is accomplished with the system free spinning. Therefore, the propellant transfer losses are primarily a function of acceleration and are relatively insensitive to transfer time. Figure A9-26 presents the propellant loss characteristics of a logistic tank to TUG rotational acceleration system as a function of acceleration. A boom or mini-depot was used to prevent migration of the center of rotation beyond acceptable limits. A 15-hour transfer time was used to provide a meaningful comparison with the characteristics of the linear system. Seven and one-half and thirty-hour propellant loss points have been included to indicate the time sensitivity of the system. The minimum propellant loss for a 15-hour transfer is 0.8 percent of the transferred propellant at an acceleration of approximately 10^{-2} G.

Figure A9-27 shows a concept for liquid/vapor interface control by rotation about the longitudinal or minor axis of the system. For the vehicle configurations considered, this technique is unattractive due primarily to its inherent poor outflow characteristics resulting in high propellant residuals.

The orbital mechanics of a linear acceleration transfer technique were analyzed to establish viable modes of operation. Thrust vector orientation was analyzed for two modes, in-plane and cross-plane (relative to Shuttle orbital plane). Thrusting in both cases was at constant inertial vector. The line of sight separation distance between the thrusting TUG/logistic tank and the quiescent orbiter as a function of time was described for ten orbits (one 15-hour loading cycle). Also, the altitude above the earth was computed for

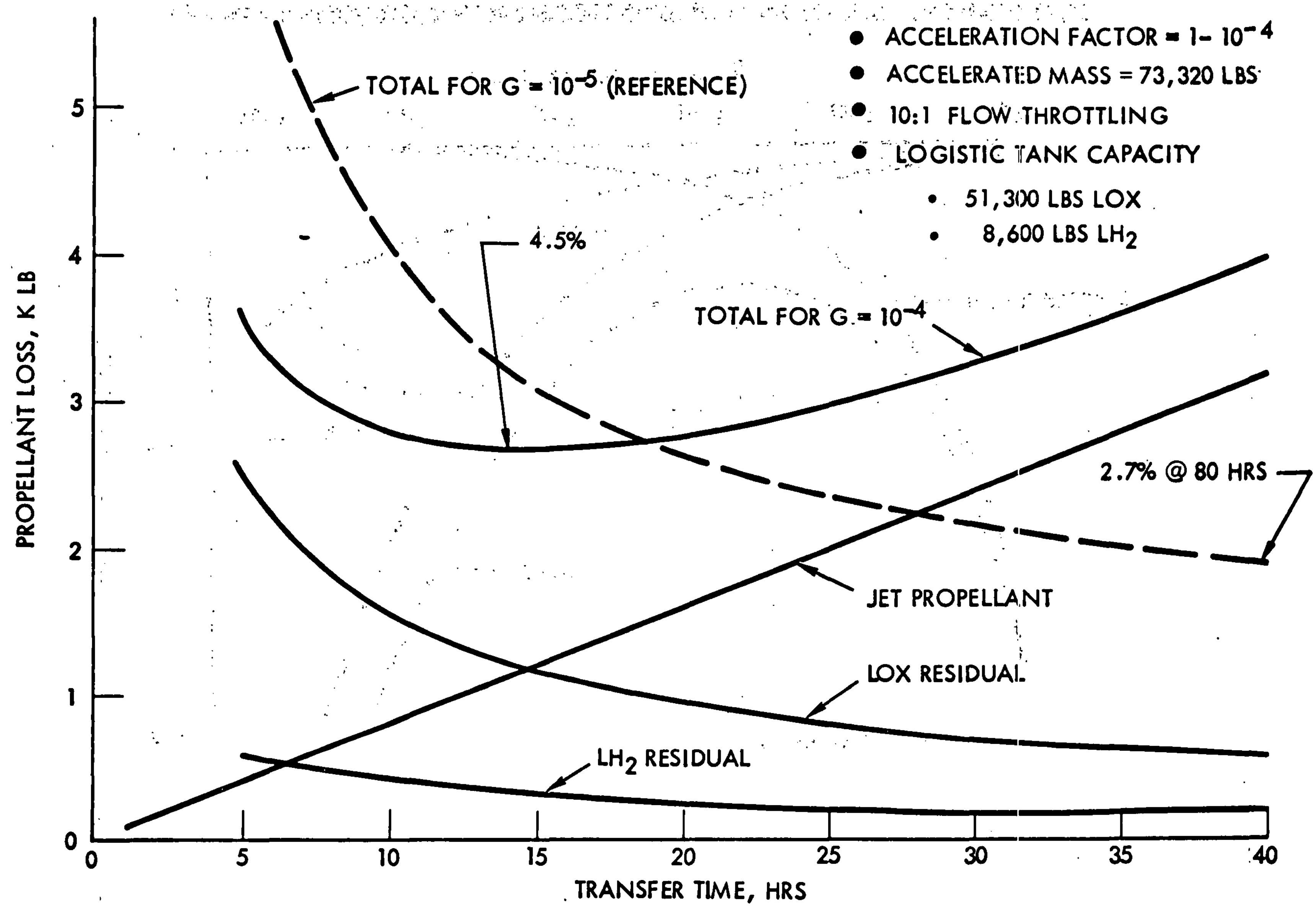


FIGURE A9-25 LINEAR TRANSFER LOGISTIC TANK TO TUG

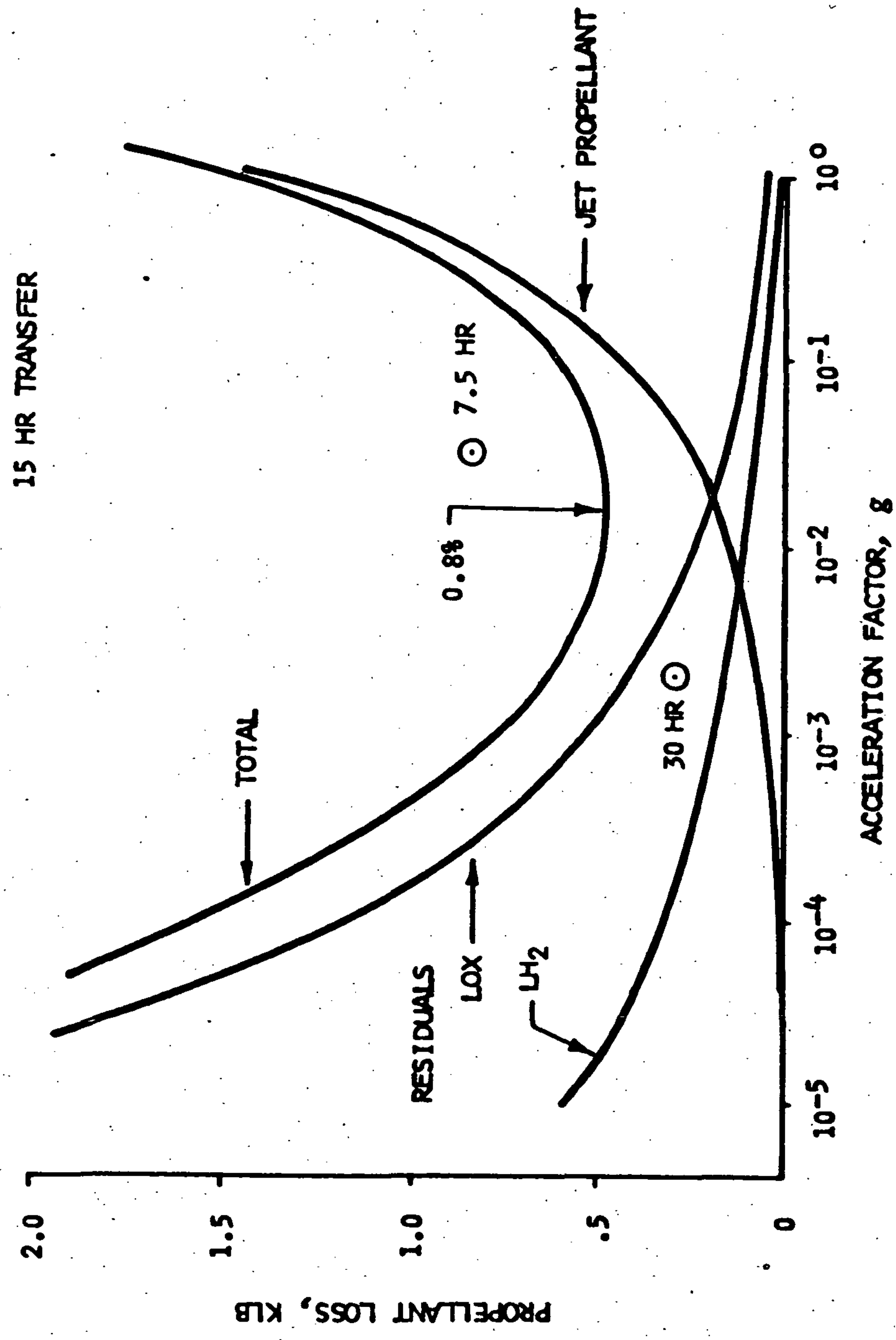


FIGURE A9-[illegible] ROTATIONAL TRANSFER LOGISTIC TANK TO TUG VIA MINI-DEPOT

ADVANTAGE

- BOOM NOT REQUIRED FOR CG CONTROL

DISADVANTAGES

- HIGH LIQUID RESIDUALS
- POOR LIQUID CONFIGURATION STABILITY
- POOR VEHICLE ROTATIONAL STABILITY
- CONFIGURATIONAL COMPLEXITY

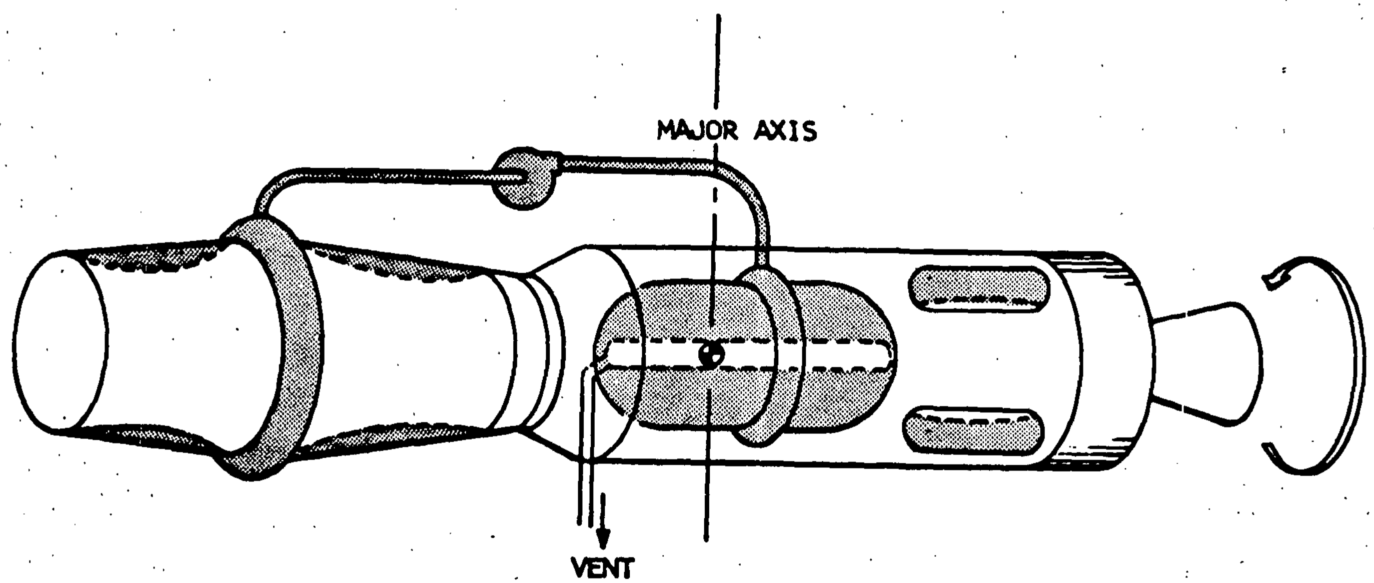


FIGURE A9-27 RADIAL ACCELERATION SPIN

the TUG/logistic tank during its thrusting cycle. These results are presented in Figure A9-28. The cross-plane thrusting produces a minor separation distance from the orbiter at completion of the propellant transfer, i.e., less than 1/2 nautical mile. The in-plane concept, assuming constant inertial attitude as shown, causes a divergence in separation distance which only partially recovers at the end of each orbit. Although not shown in Figure A9-28, the in-plane thrusting could be performed with the TUG maintaining an in-plane attitude maneuvering rate equal to 1/2 orbital rate, rather than constant inertial attitude. If this were done the thrusting inertial direction would change 180 degrees each orbit, and would almost completely compensate for the separation distance produced in the previous orbit.

A concept employing capillary systems for liquid/vapor interface control is presented in Figure A9-29. Characteristics and problems associated with the concept are also presented. Although this concept has some very attractive features such as it being a passive system and it develops no orbit disturbances, the development risk for cryogenic orbital transfer and user vehicle compatibility problems prevent the selection of this concept at this time.

A summary of the advantages and disadvantages of rotational acceleration, linear acceleration, and surface tension to provide liquid/vapor interface control follows:

TECHNIQUE	ADVANTAGES	DISADVANTAGES
Rotational acceleration	Low propellant loss	Configuration must be maintained between tanks
	Minimum orbital maneuvering (relative to Shuttle orbit)	
Linear acceleration	No configuration cg problem	Propellant loss greater than other methods
		Orbital maneuvering required
Surface tension	Passive system	Incompatible with some user configurations
	No orbital maneuvering required	Development risk

Linear acceleration is selected as a viable baseline concept for both direct fluid transfer and for transfer operations involving the mini-depot. The mini-depot would require an exceptionally long boom for c.g. control for fluid transfer to the CPS or RNS with radial acceleration. A capillary system, although not selected at this time because of development problems, could ultimately prove to be the most desirable method.

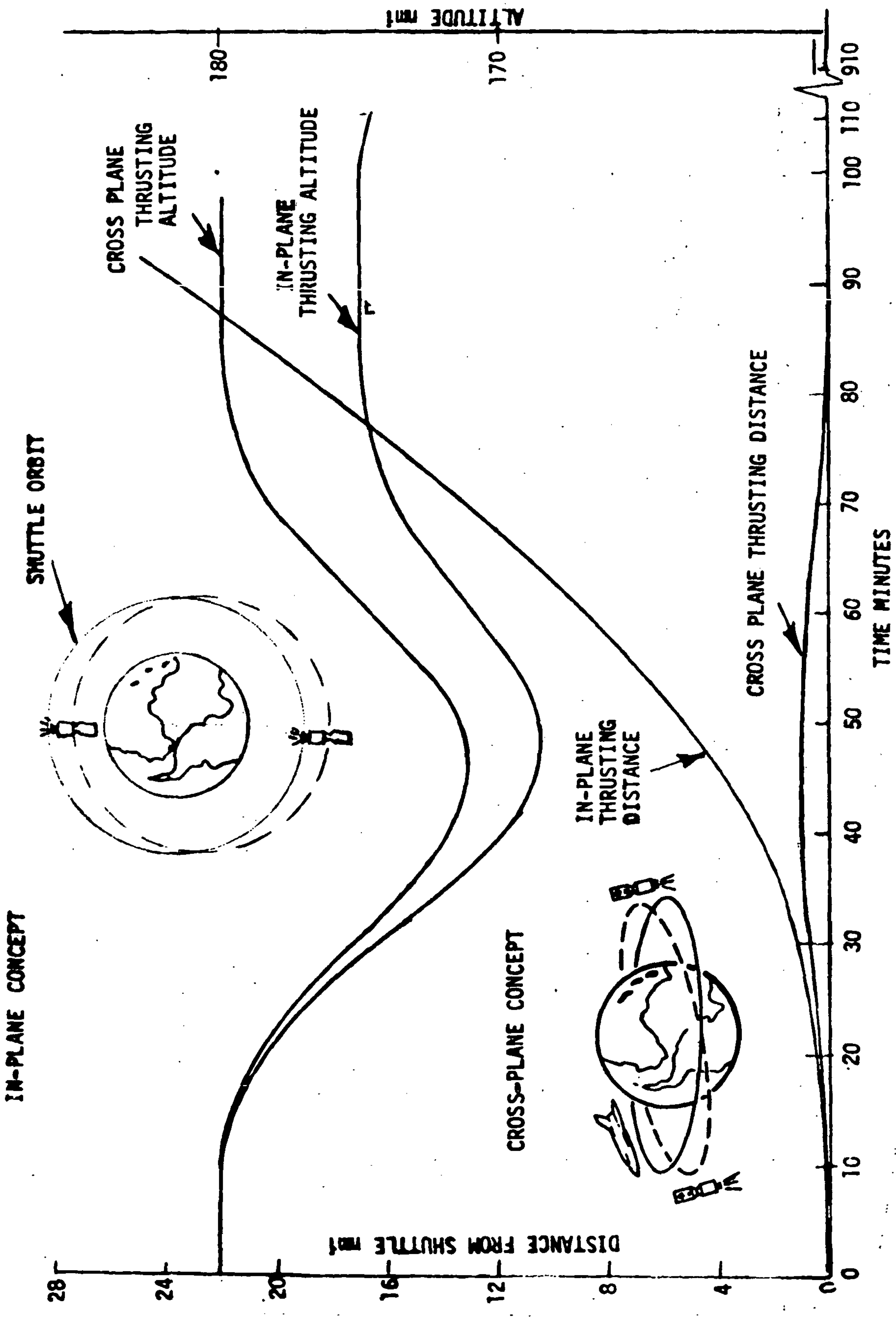


FIGURE A9-28 LINEAR ACCELERATION/PROPELLANT SETTLING ORBITAL MECHANICS

- CHARACTERISTICS
  - SURFACE TENSION FORCES
    - PREVENT GAS PASSAGE THROUGH SCREENS
    - CONDUCT LIQUID IN CHANNELS
  - PASSIVE SYSTEM
  - NO EXTERNAL FORCES REQUIRED
  - POTENTIAL FOR LOW RESIDUAL LIQUID
  - NO ORBIT DISTURBANCES
- PROBLEMS
  - SUBCOOLED LIQUID REQUIRED DURING STORAGE AND TRANSFER
  - RECEIVER TANK MUST HAVE SIMILAR DEVICES OR ZERO G VENT
  - POTENTIAL FOR GAS ENTRAPMENT DURING REFILL
  - HIGH DEVELOPMENT RISK FOR CRYOGENIC APPLICATION
  - LOW SLOSH DAMPING FORCES IN RECEIVER

SPACE SHUTTLE PHASE B

LOX FEED TANK

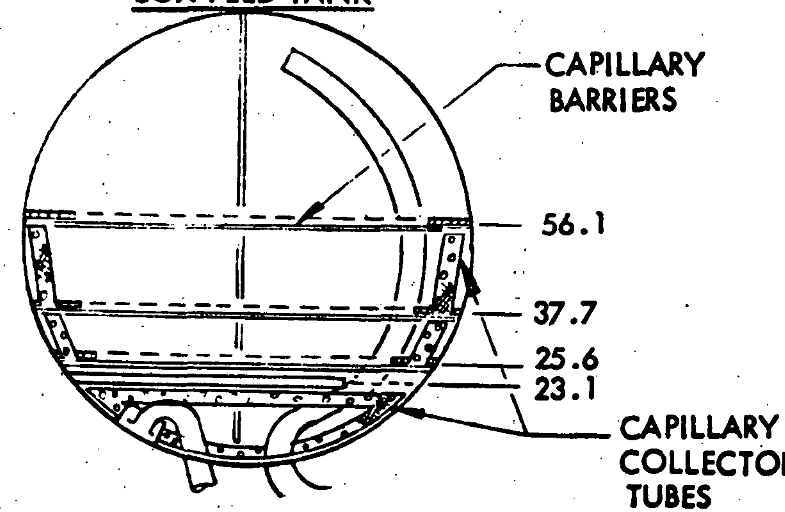


FIGURE A9-29 LIQUID/VAPOR INTERFACE CONTROL CAPILLARY SYSTEMS

Feasibility of Linear and Rotational Acceleration

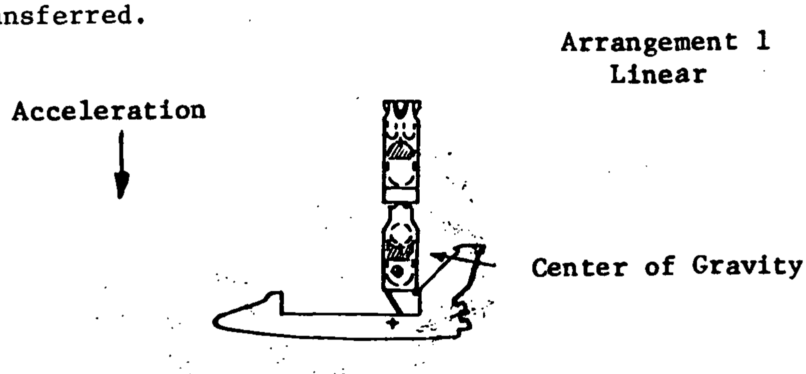
An analysis was conducted concerning the deployment of the logistic propellant tank relative to the orbiter cargo-bay and the receiver element for linear and rotational acceleration. Four potential arrangements were considered for linear acceleration, and five potential arrangements were considered for rotational acceleration. Acceleration thrust was provided by the orbiter for both the linear and rotational cases.

Criteria for an acceptable deployment included the location of propellant during transfer relative to its position on the pad during launch, the center of gravity location, the capability of visually monitoring the docking and transfer operations and the capability of the manipulators to deploy the tanks in an acceptable docking position.

Linear Acceleration Arrangements

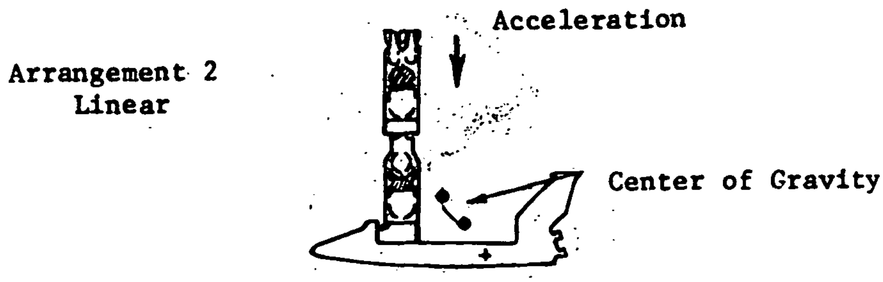
Arrangement 1 extends the logistic tank perpendicular to the orbiter center line about a point in the aft of the cargo bay.

Direction of thrust settles propellant to opposite ends of tanks as when loaded on pad. Orbiter, logistic tank and TUG center of gravities are nearly in line allowing thrust to be continuously applied in one direction as propellant is transferred.



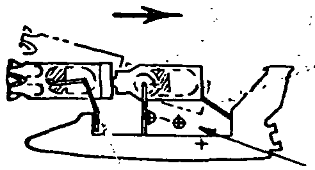
Arrangement 2 extends the logistic tank perpendicular to the orbiter center line about a point in the forward end of the cargo bay.

Direction of thrust settles propellant to same end of tanks as when loaded on pad. Orbiter, logistics tank and TUG's center of gravities are not in line and thruster would have to be balanced to react through combined cg. Balance would have to change as the cg shifted during transfer.



Arrangement 3 extended the logistics tank outward from the cargo bay but parallel to the orbiter centerline. The tank attaches to TUG as shown.

Direction of thrust settles propellant to opposite ends of tanks as when loaded on pad. Deployment is complex and requires support fixtures and detachment from cargo bay. Center of gravities of orbiter, logistic tank and TUG are not in line and thrust resultant would have to follow cg shift during transfer.



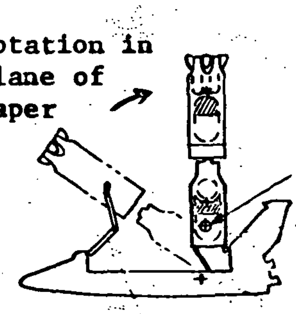
Arrangement 4 is shown in Figure A9-30. The logistic tank is extended at an angle from the cargo bay so as to keep center of gravities of orbiter, logistics tank and TUG in line. Propellant settles to same ends of tanks as on pad during acceleration. This arrangement appears to be the most desirable for linear acceleration with orbiter attached.

Linear acceleration with the logistics tank detached from the orbiter was also considered in the analysis. The logistics tank in these cases would provide the settling thrust. Typical configurations are shown on Figures A9-31, A9-32, and A9-33, for TUG, CPS, and RNS, respectively.

Rotational Acceleration Arrangements

The following rotational arrangements were also analyzed. Arrangement 1 extends the logistic tank perpendicular to the orbiter center line about a point in the aft section of the cargo bay as shown.

With this arrangement, the combined center of gravity (center of rotation) falls inside the tank making transfer impractical.



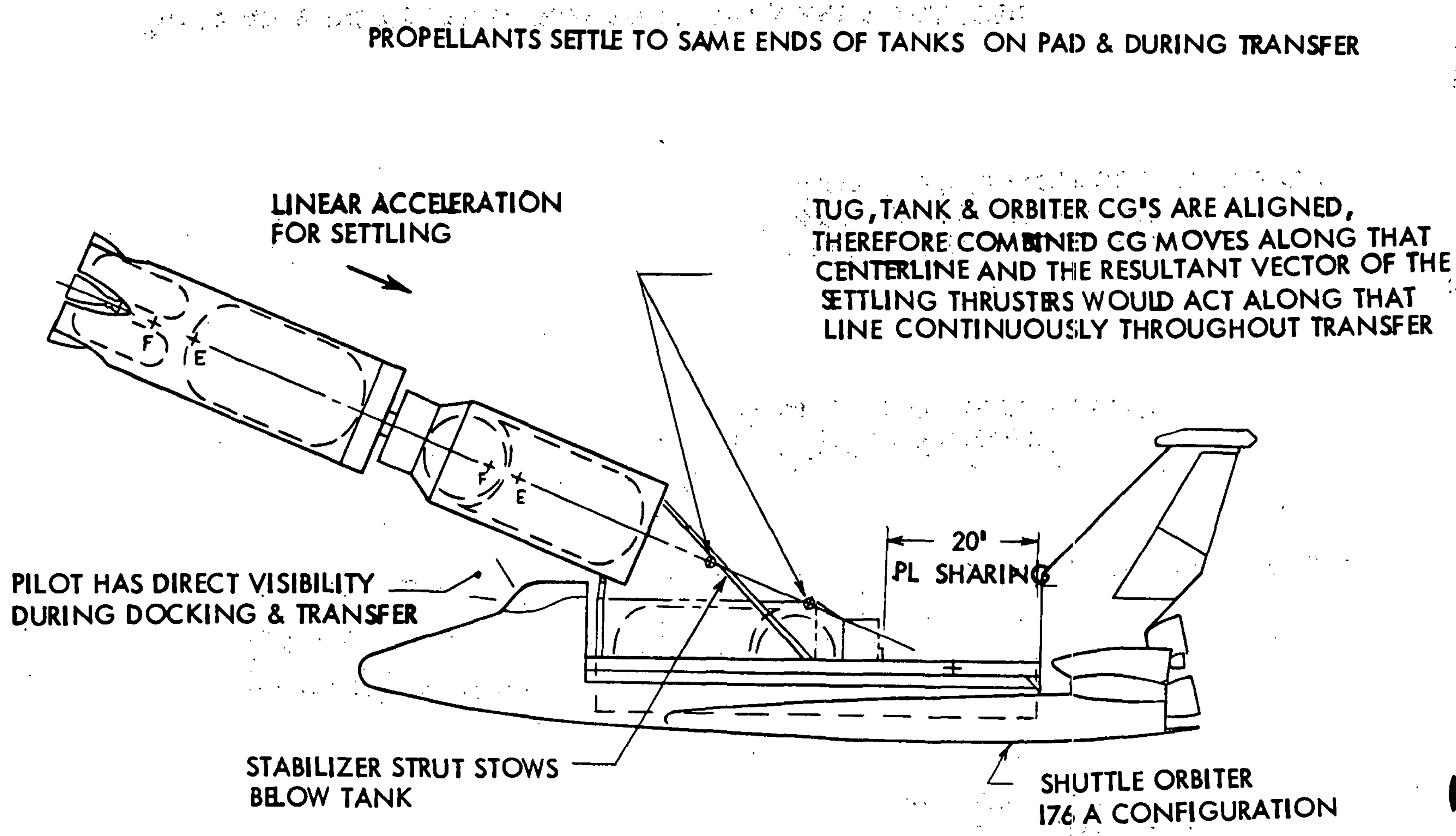


FIGURE A9-30 LINEAR TRANSFER WITH ORBITER ATTACHED

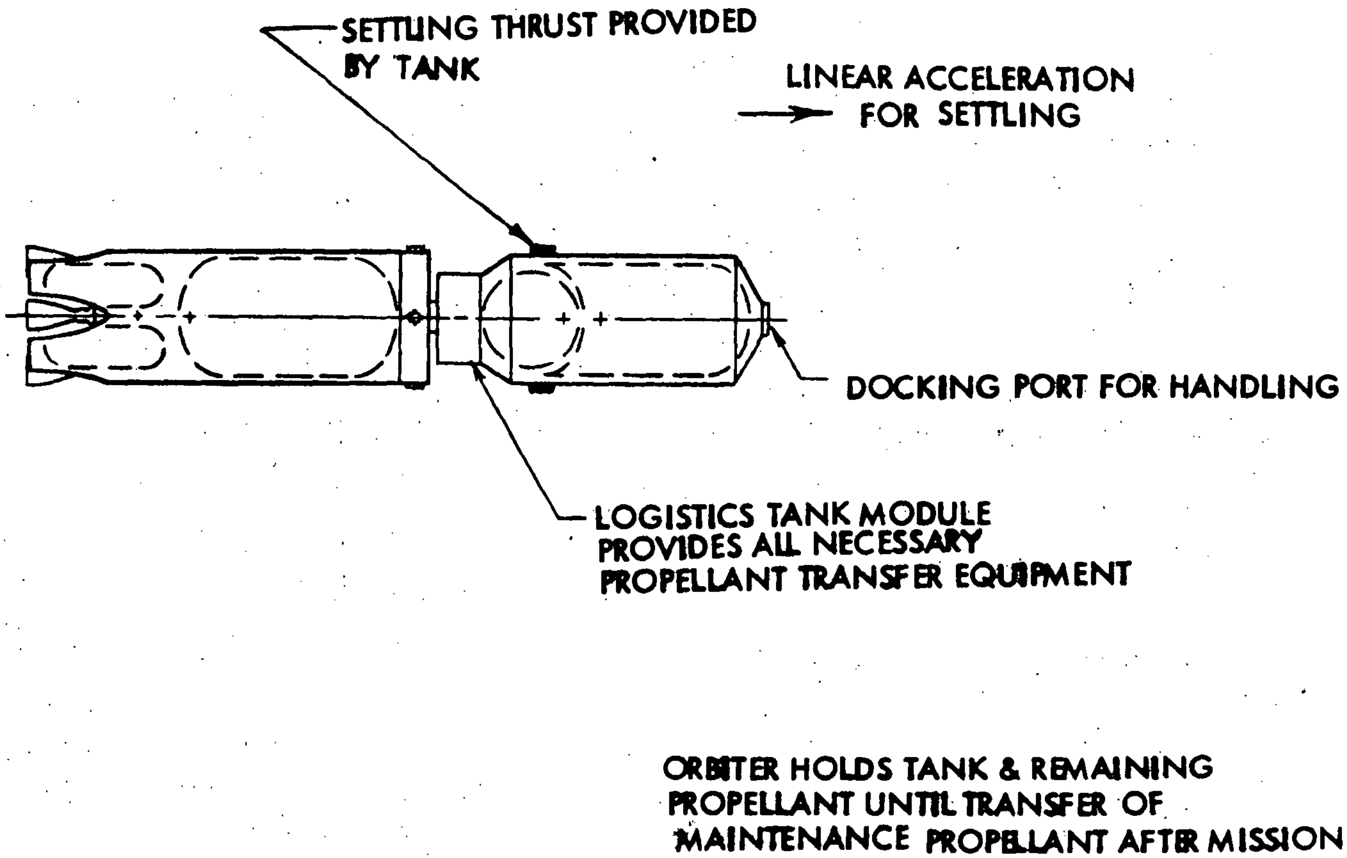


FIGURE A9-31 DIRECT LINEAR TRANSFER TO TUG (ORBITER DETACHED)

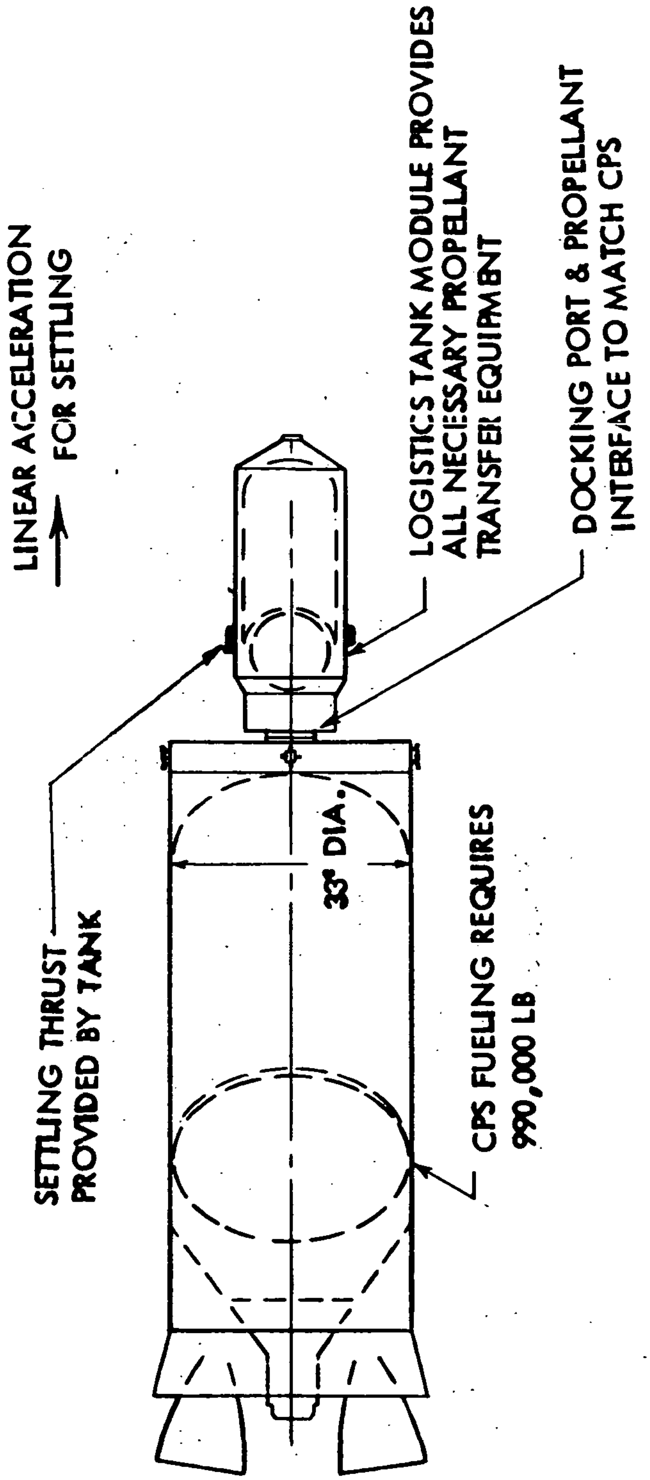


FIGURE A9-32 DIRECT LINEAR TRANSFER TO CPS

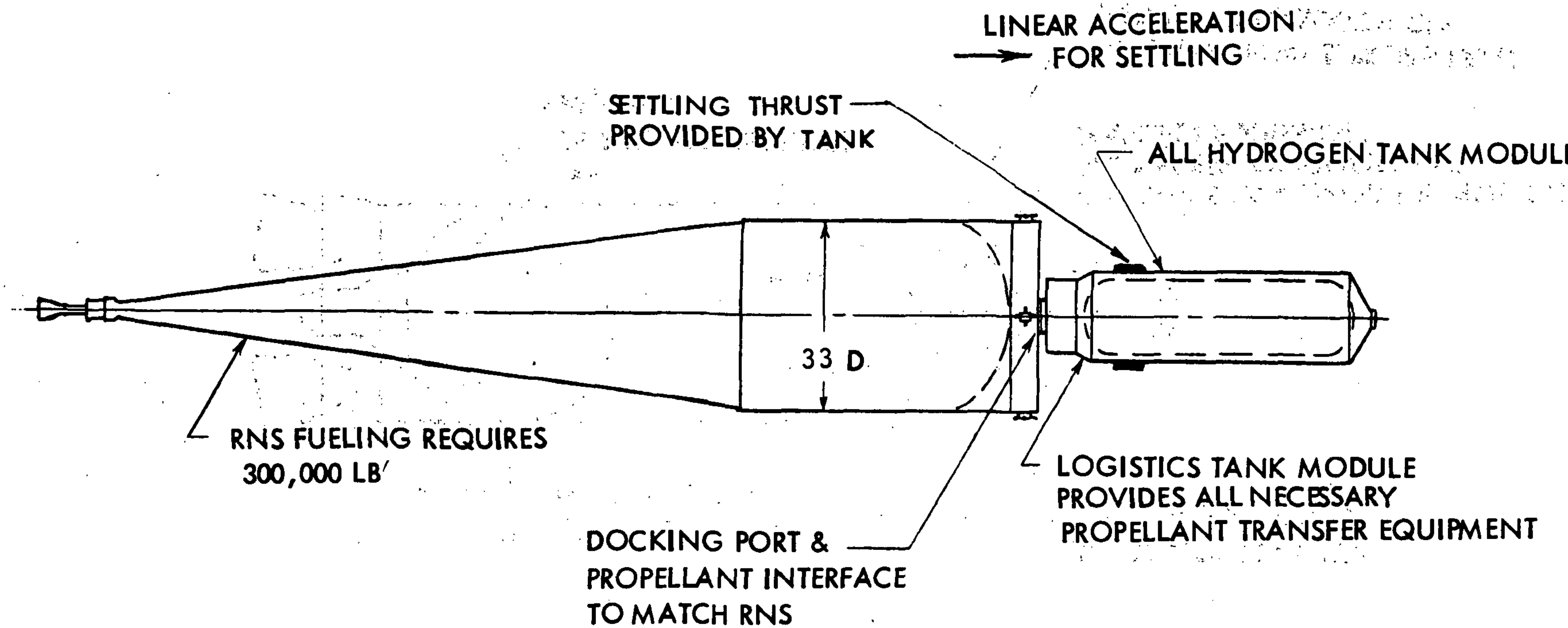
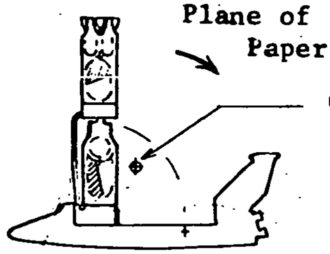


FIGURE A9-33 DIRECT LINEAR TRANSFER TO RNS

Arrangement 2 extends the logistic tank perpendicular to the orbiter center line about a point in the forward section of the cargo bay, as shown.

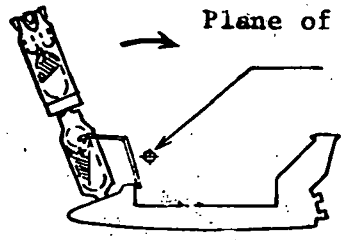
With this arrangement, the combined cg (center of rotation) is outside the tanks; however, propellants are settled at the tank sides not towards the tank ends. This arrangement is also impractical.

Arrangement 2
Rotational



Arrangement 3 extended the logistic tank forward of the cargo bay as shown below. Propellant would also settle at side of tank and transfer would not be practical.

Arrangement 3
Rotational



Arrangement 4 is shown in Figure A9-34. This arrangement appears feasible because the cg remains aft of tanks during transfer. However, the cg does more during transfer causing some eccentricity in rotation. Arrangement 5 rotational transfer with logistics tank in orbit cargo bay is shown in Figure A9-35. This arrangement also appears feasible because the cg remains out of the tanks during transfer. Again the cg moves during transfer and a higher rotation rate is required because of the close proximity of the cg to the tanks. An additional problem is the location of the vent and gaging systems hardware.

As a result of the analysis, arrangement 4 using linear (Figure A9-30) acceleration, and deployed from orbiter, direct transfer using linear acceleration detached from the orbiter (Figure A9-31), arrangement 4 (Figure A9-34), using rotational acceleration and deployed from orbiter, and arrangement 5 (Figure A9-35) using rotational acceleration with tank nestled in orbiter were selected as the most viable concepts for liquid/vapor interface control. Further analysis using propellant losses, operational complexity, safety, and dynamic stability resulted in the potential selection of rotational arrangement 5 as the most viable method of liquid/vapor interface control for

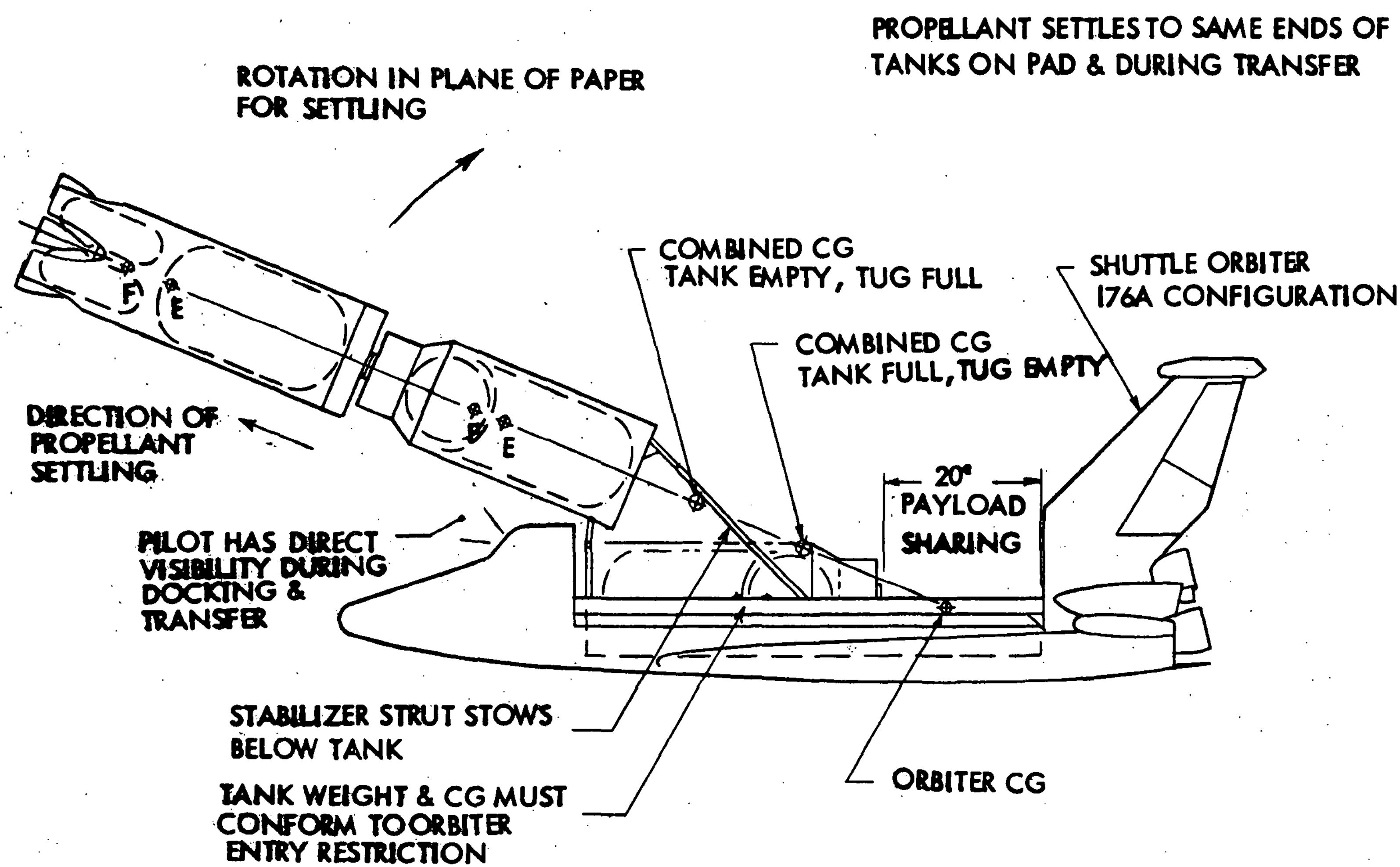


FIGURE A9-34 ROTATIONAL TRANSFER WITH ORBITER ATTACHED

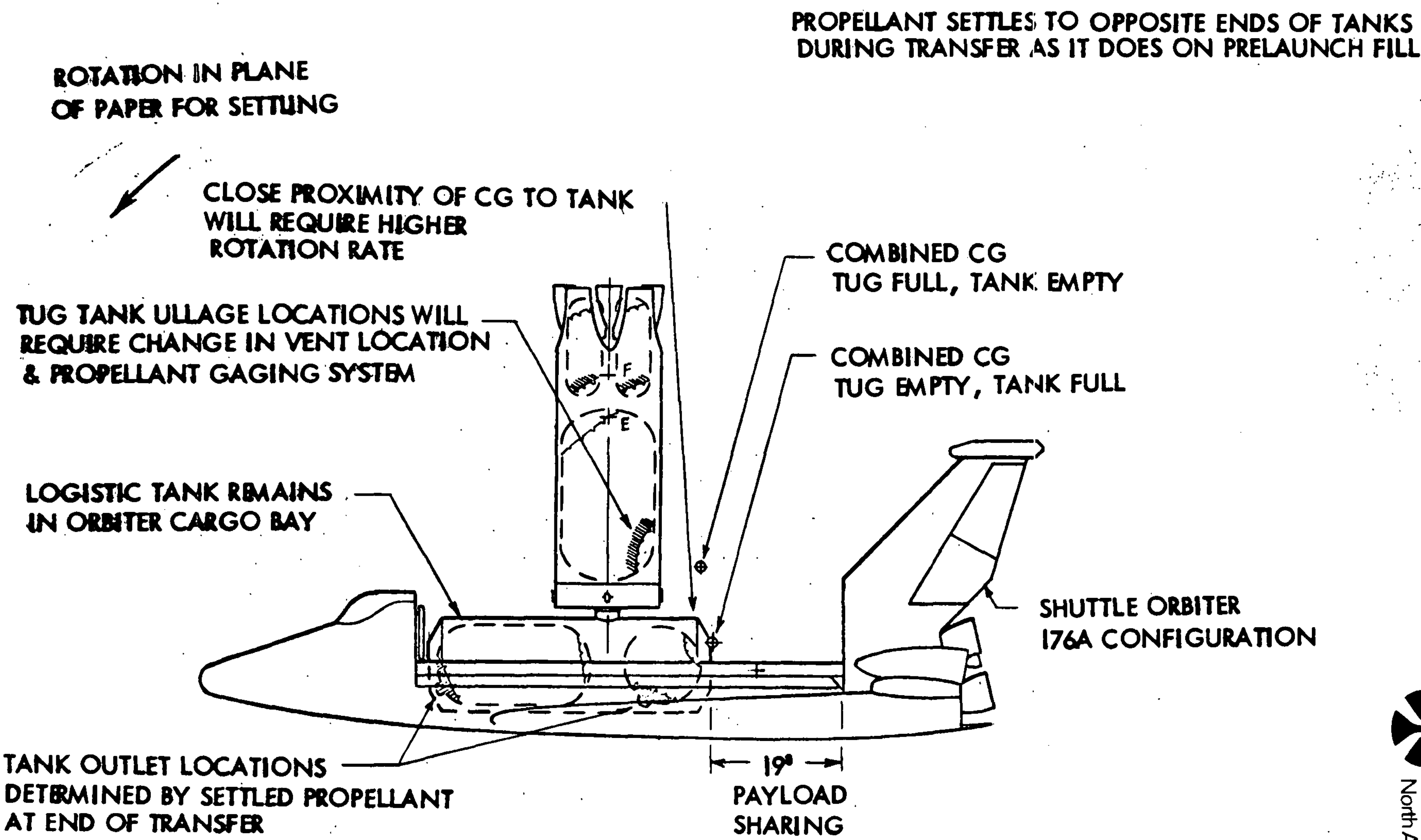


FIGURE A9-35 ROTATIONAL TRANSFER WITH LOGISTIC TANK IN CARGO BAY propellant transfer to the TUG. No single factor was identified as a strong driver in the selection. When adding other considerations such as commonality relative to CPS or RNS propellant resupply, the selection changes to direct transfer using linear acceleration detached from the orbiter. This method is compatible with TUG, CPS, and RNS and provides for transfer to take place remotely from the orbiter allowing it to be free for other duties during the transfer period. Therefore, direct transfer as shown in Figures A9-31, A9-32, and A9-33 for TUG, CPS and RNS, respectively, has been selected for the baseline liquid/vapor interface control technique.

3.2 RECEIVER TANK THERMODYNAMIC CONTROL

The concepts for providing the required thermodynamic control involve either (1) connecting the ullage of the supplier and receiver vehicle tanks, (2) providing overboard venting of the receiver tanks prior to transfer, or (3) providing overboard venting of the receiver tank during transfer. These concepts are shown schematically on Figure A9-36. The connected ullage concept results in minimum propellant losses but requires liquid/vapor interface control in the receiver as well as supplier and requires additional line interfaces. A technique which vents the receiver tank to space prior to transfer eliminates the need for liquid/vapor interface control in the receiver since the actual transfer is to the voided receiver tank with no additional venting required.

This concept would involve the total loss of the pre-transfer residual propellants in the receiver and would also require very rigorous thermodynamic control.

Figure A9-37 shows the characteristics of the prior-to-transfer vent concept. Ideal mixing of the fluid to achieve necessary mass transfer to result in a homogeneous temperature fluid in the receiver was assumed and therefore represents idealized conditions. In practice, one might expect the full tank ullage pressure to be higher than shown which might require thermodynamic venting and the associated propellant losses to restore the propellant to the desired thermodynamic balance.

Figure A9-38 shows the propellant losses of the vent during transfer concept for RNS and TUG. The losses represent the gas that must be vented from the receiver to restore stabilized conditions at the conclusion of the transfer. Complete liquid/vapor separation with no two phase venting is assumed.

Advantages and disadvantages of the three concepts are summarized below with the connected ullage selected as the baseline subsystem because of low propellant losses.

TECHNIQUE	ADVANTAGES	DISADVANTAGES
Connected ullage	min. propellant vent loss	Liquid/vapor interface control required for receiver
	Provides source tank liquid displacement	Additional line interfaces
	Liquid/gas interface control not critical	

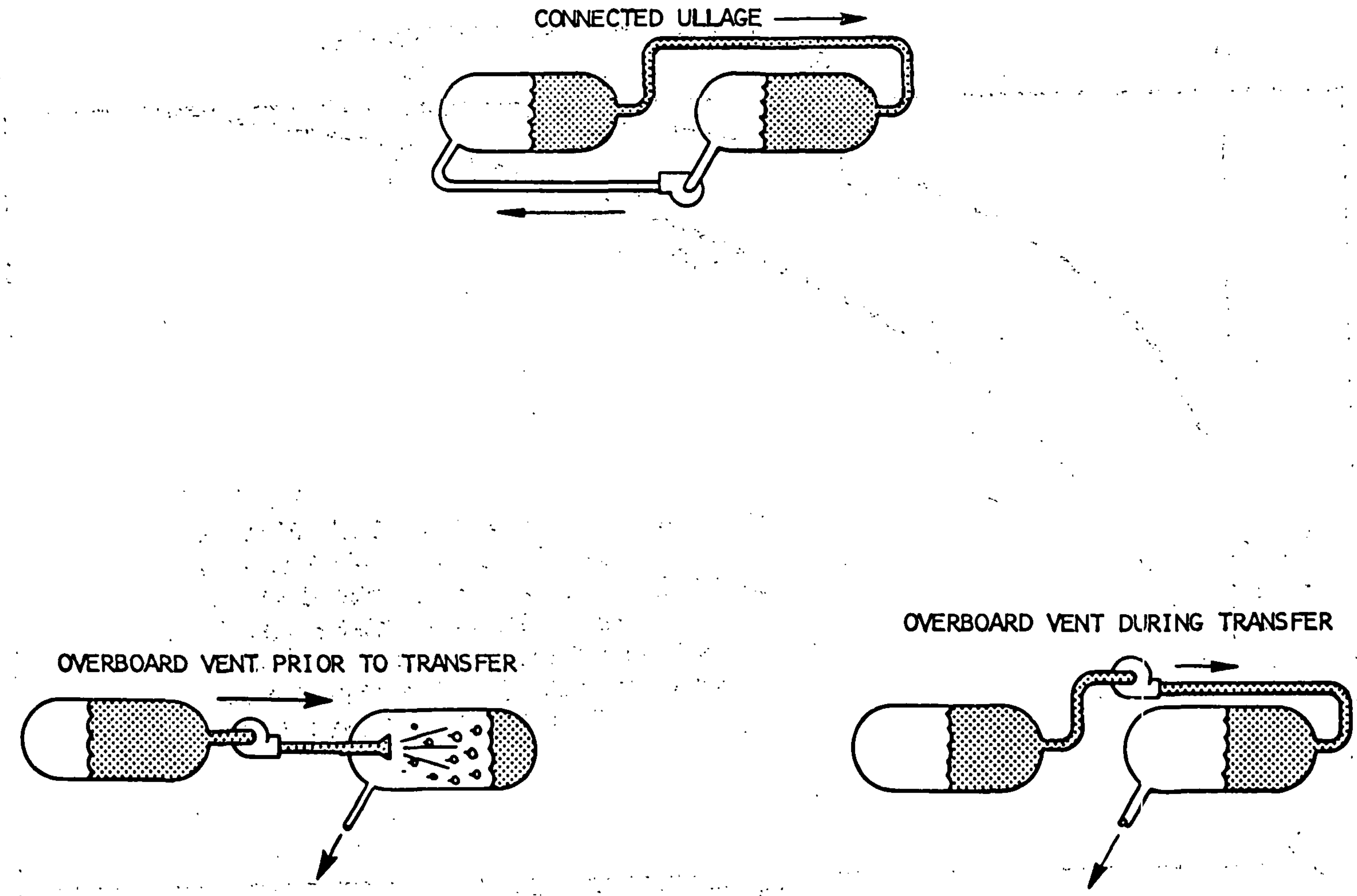


FIGURE A9-36 RECEIVER TANK THERMODYNAMIC CONTROL CONCEPTS

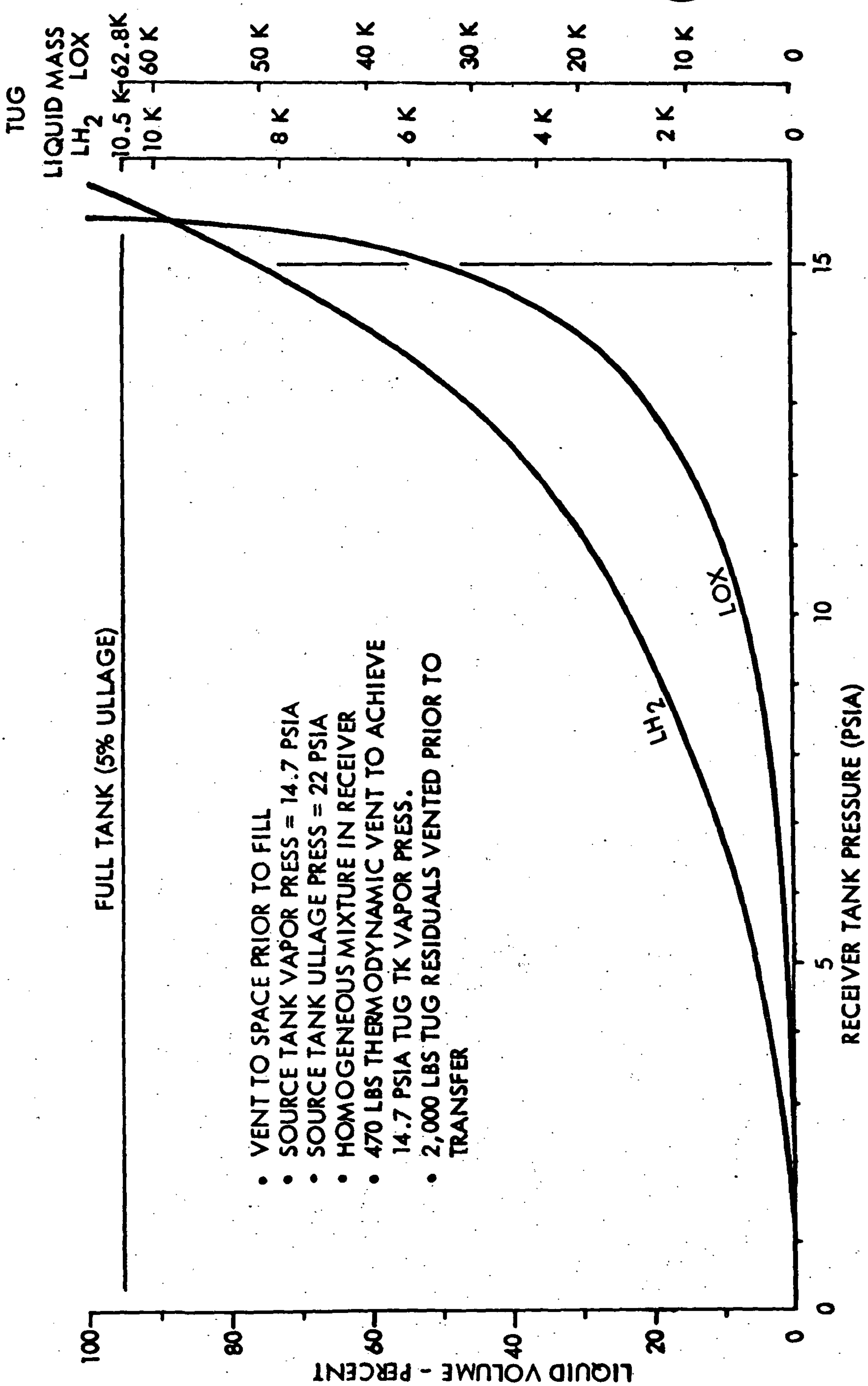


FIGURE A9-37 OVERBOARD VENT PRIOR TO TRANSFER

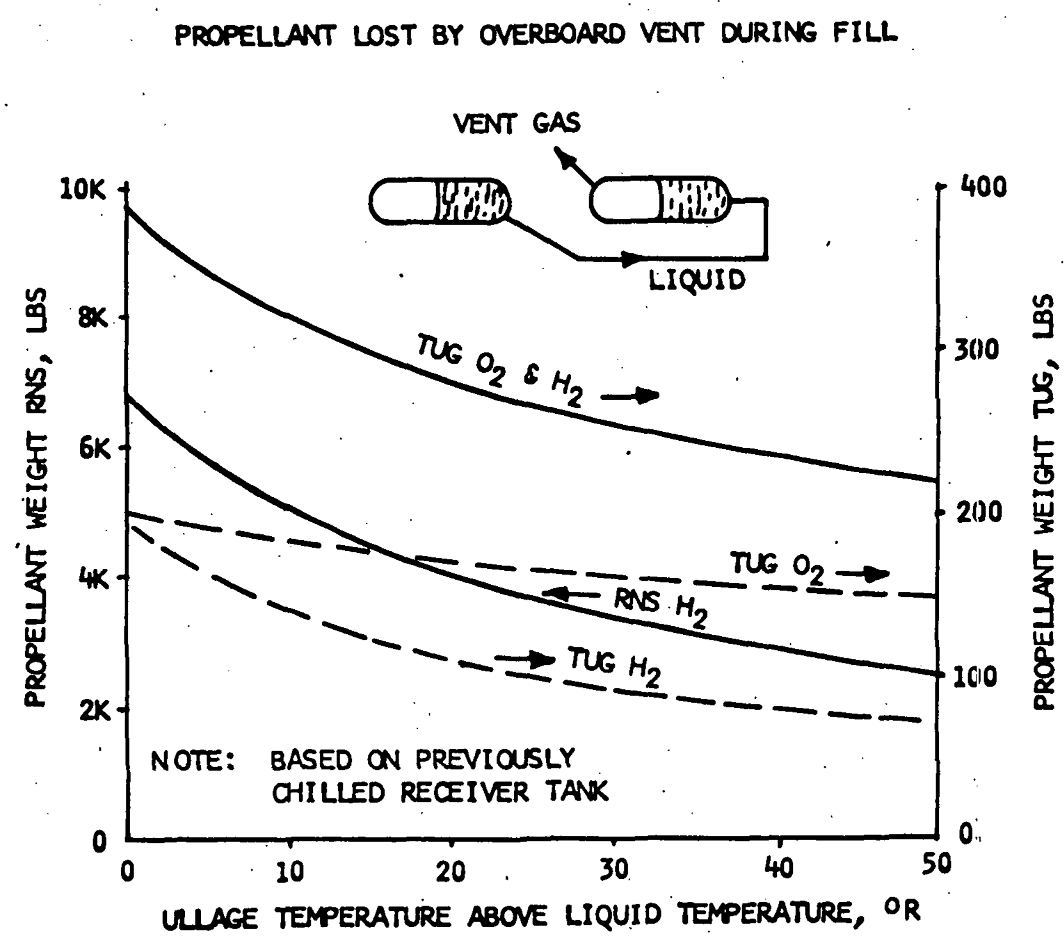


FIGURE A9-38 OVERBOARD VENT DURING TRANSFER

TECHNIQUE	ADVANTAGES	DISADVANTAGES
Overboard vent prior to transfer	No receiver tank liquid/vapor interface control required	Loss of user vehicle initial propellant residuals
		Good fluid mixing required
Overboard vent during transfer	No gas return line required	Source tank gas supply required for liquid displacement
		Liquid/vapor interface control critical

3.3 PROPELLANT EXPULSION

The most promising concepts for expulsion are (1) displacement of the fluid by liquid pumping, (2) displacement of the fluid by gas pumping, (3) positive displacement of the fluid by mechanical devices, (4) displacement of the fluid by pressurization using a liquid to gas conversion device, and (5) displacement of the fluid by pressurization using a stored gas. These concepts are shown schematically in Figure A9-39.

Figure A9-40 shows the parametric hydrodynamic characteristics for a 15-hour LH_2 logistic tank to TUG liquid pump transfer concept. The significant penalty factors, pumping power, line residual, line weight and line boiloff, are presented as a function of transfer line diameter. As shown by the Figure, the penalty is only 0.02 percent for a line size of approximately 0.8 inch diameter. Attention is called to the pumping power curve which indicates that the pumping power loss is insignificant for larger size lines.

A weight penalty comparison of oxygen pressurization, helium pressurization, and pump expulsion for a 15-hour logistic tank to TUG transfer is presented in Figure A9-41. The data indicate that the pump concept for LO_2 transfer has the lowest weight penalty and oxygen pressurization transfer has the highest penalty. Although not presented here, similar data for the LH_2 system show the pump with the lowest and the helium pressurization with the highest penalty.

Some of the more salient features of a positive displacement bellows concept is presented in Figure A9-42. The most attractive features of a positive displacement concept are no liquid/vapor interface control is required and it has the potential of reducing the liquid residuals to a minimum. However, there are a number of fabrication and operational characteristics which prevent the selection of this concept at this time. Some of the principal disadvantages are: large sizes are difficult to manufacture (equipment and technique not available for sizes over 40" in diameter), hardware weight is high, and as listed on Figure A9-42, an unusually high number of compatibility and operational problems are anticipated.

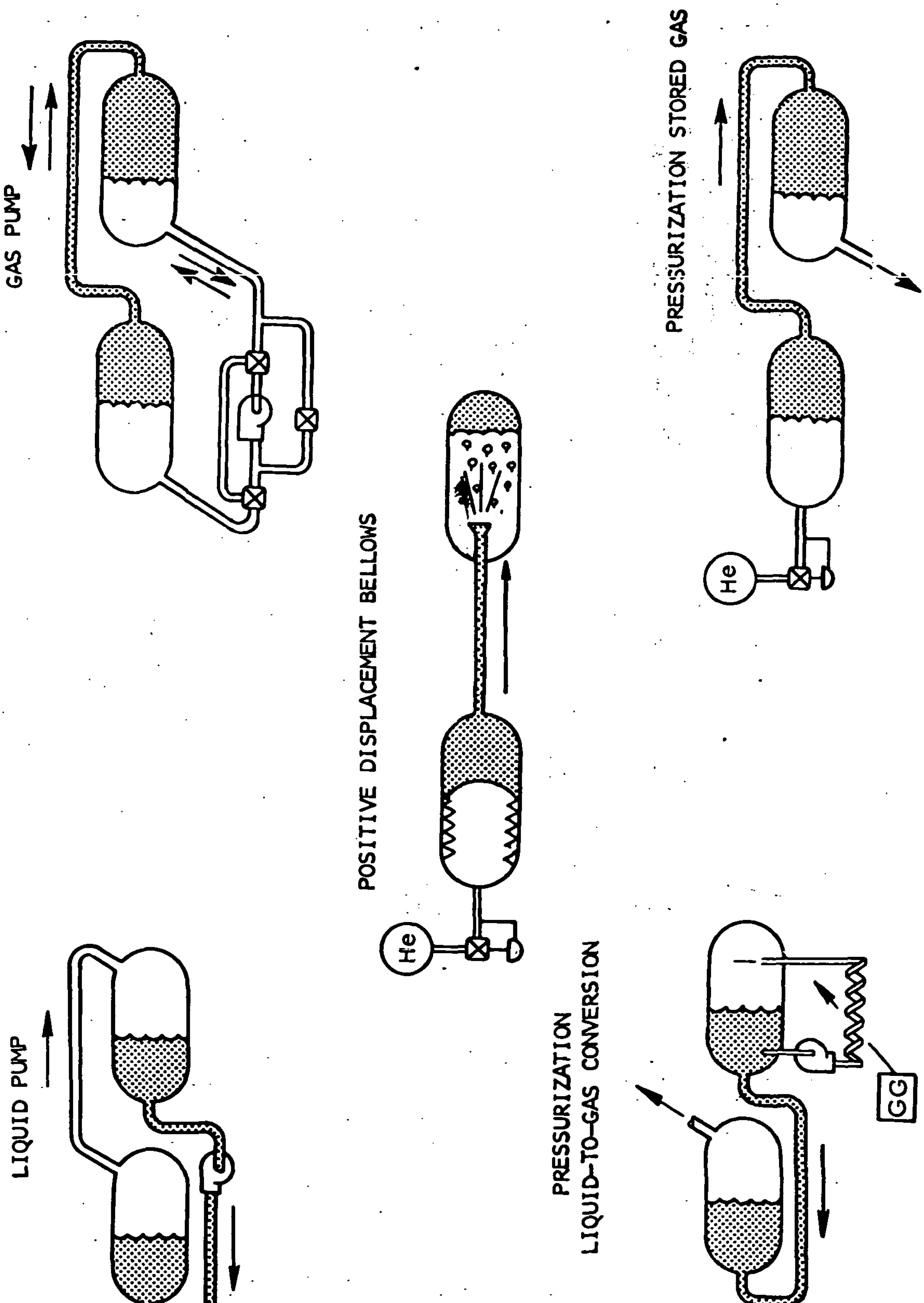


FIGURE A9-39 LIQUID EXPULSION CONCEPTS

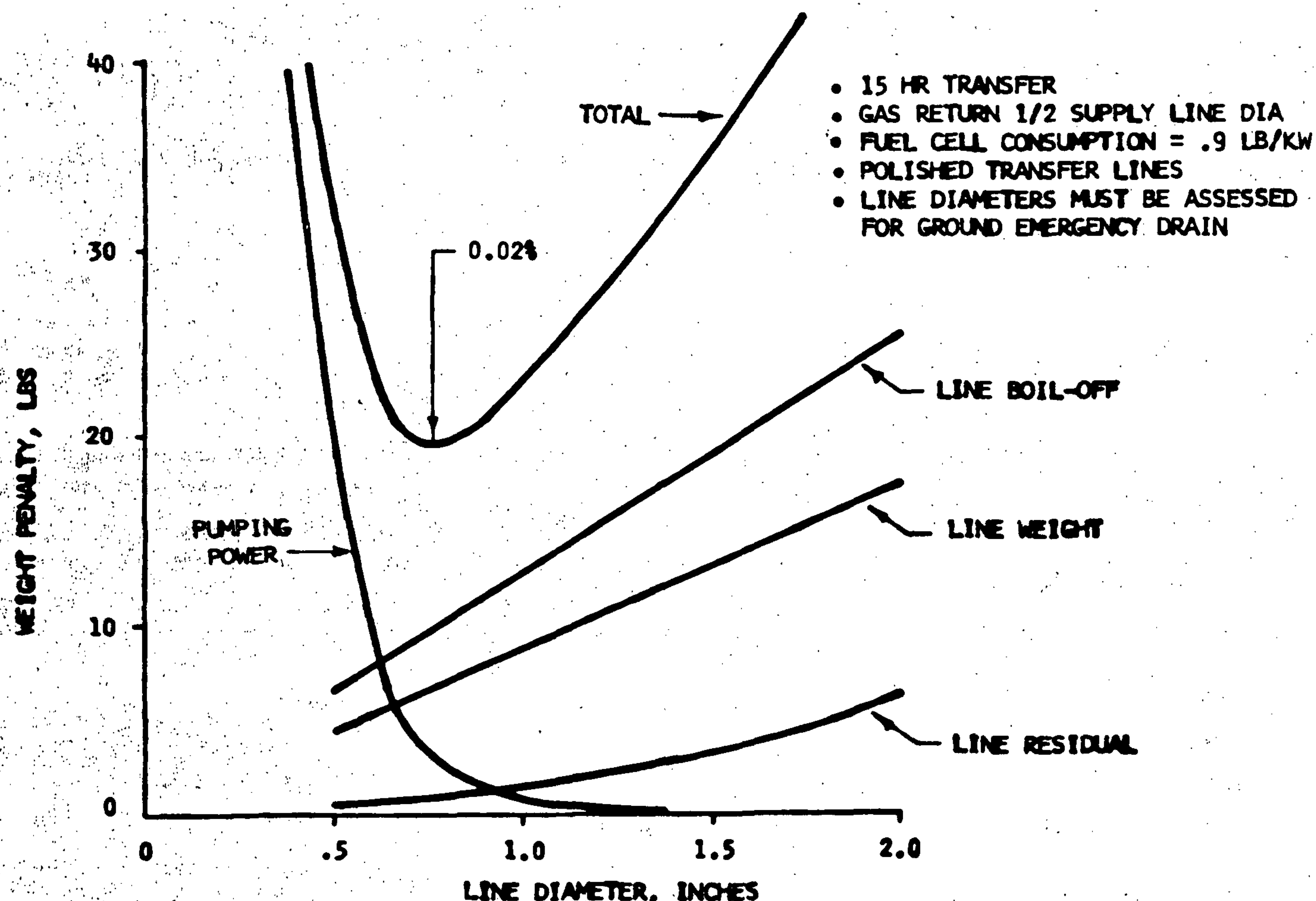


FIGURE A9-40 LOGISTIC TANK TO TUG LH_2 TRANSFER LINE DIAMETER

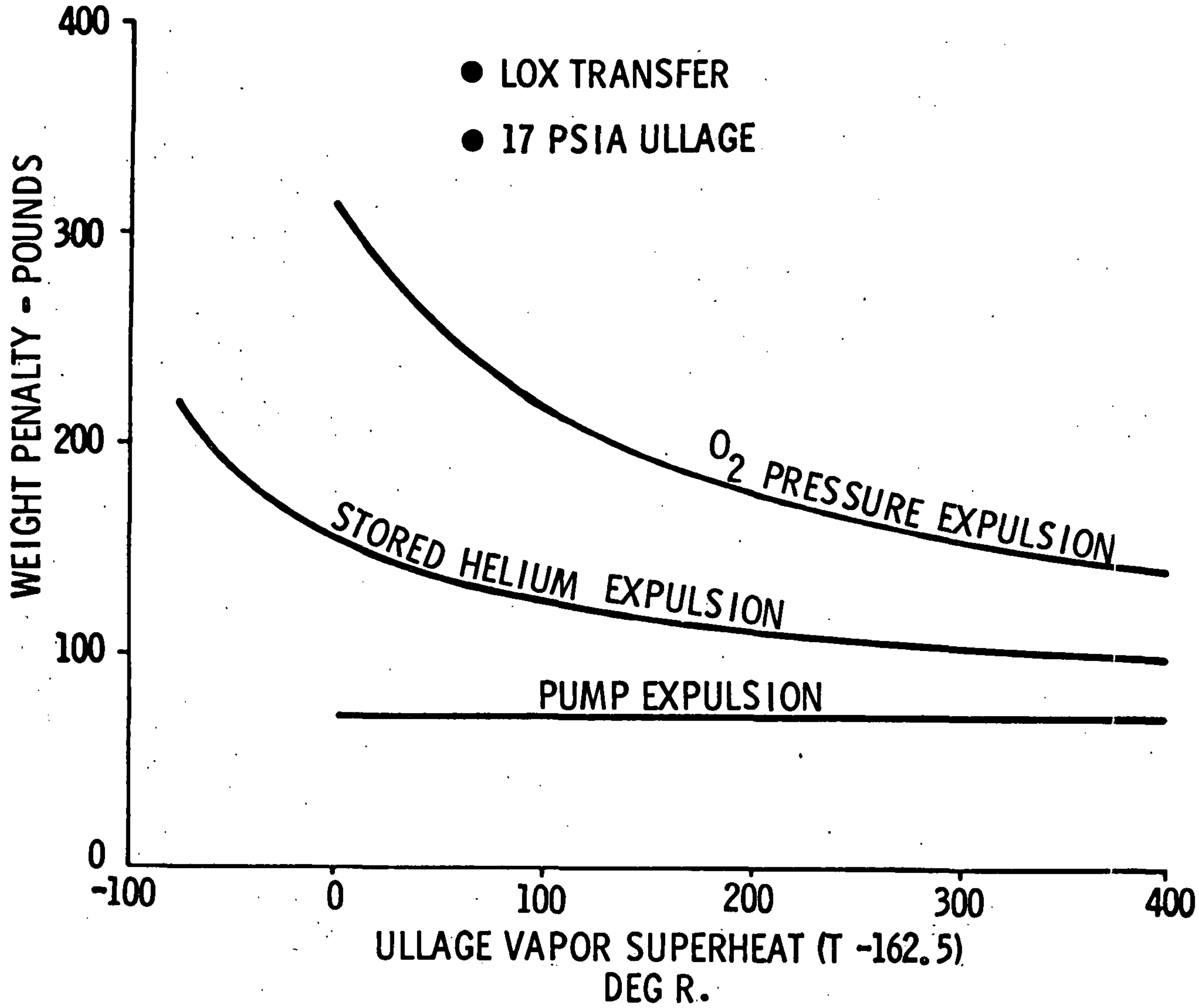
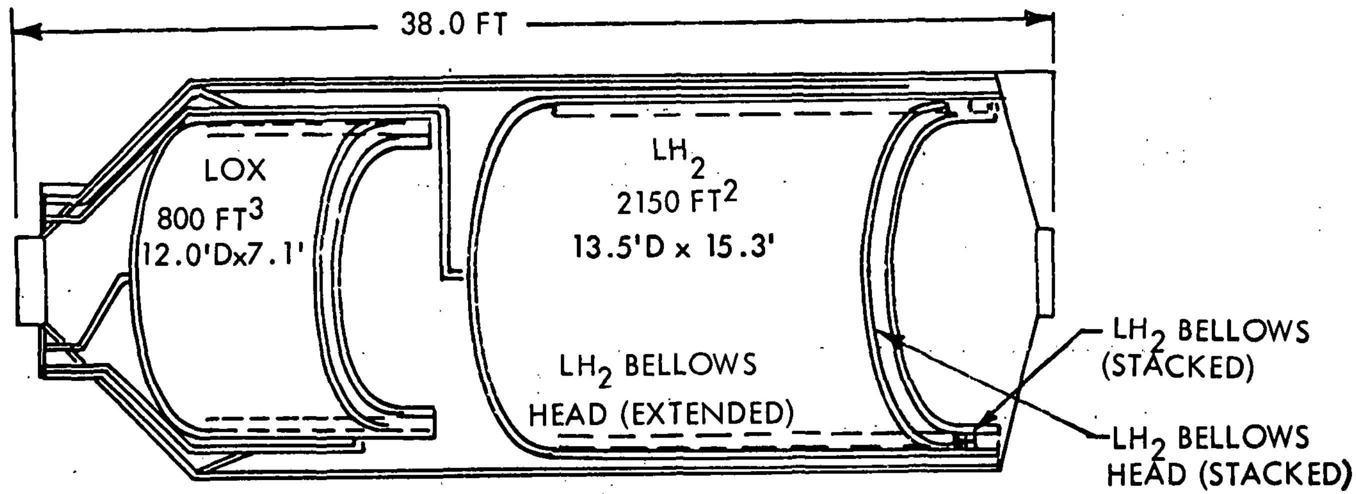


FIGURE A9-41 COMPARISON OF EXPULSION CONCEPTS (LOGISTIC TANK TO TUG TRANSFER)



ADVANTAGES

1. NO PROPELLANT SETTLING REQUIRED
2. SHUTTLE CAN REMAIN ATTACHED DURING TRANSFER OPERATIONS
3. RESIDUALS MAY BE LOWER THAN FOR OTHER CONCEPTS
4. NO TRANSFER PUMPS REQUIRED
5. NO GAS RETURN LINE REQUIRED
6. SMALL VOLUME TO INERT OR PURGE

DISADVANTAGES

1. PROPER VENTING OF BELLOWS DIFFICULT
2. ΔP CRITICAL TO PREVENT BUCKLING
3. SUPPORT OF EXPANDED BELLOWS FOR ACCELERATION LOADING DIFFICULT
4. REQUIRES ACTUATION SYSTEM
5. PURGING OR INERTING OF CONVOLUTIONS DIFFICULT
6. TWO PHASE FLOW WHEN USED WITH VENT PRIOR TO FILL
7. INCOMPATIBLE WITH "TOPPING-OFF"
8. INCOMPATIBLE WITH MOST STATE-OF-THE-ART LIQUID GAUGING SYSTEMS
9. FLOW DIRECTION NOT REVERSIBLE

FIGURE A9-42 POSITIVE DISPLACEMENT EXPULSION BELLOWS CONCEPT

The concept with a pump mounted in the vapor return system was selected as baseline for this subsystem. The performance of the gas pump concept and the liquid pump concept is essentially the same; however, the configurational design, flexibility and the serviceability of the gas pump concept appears superior.

A summary of advantages and disadvantages of the various concepts follows:

TECHNIQUE	ADVANTAGES	DISADVANTAGES
Liquid pump	Low-weight penalty	In-tank pump
		Receiver pump required for transfer reversal
		Maintenance - poor accessibility
Gas pump	Safety - transfer reversible	Higher speed
	Low-weight penalty	Shorter life
	Maintenance - external pump	
Positive displacement	No propellant setting required	Incompatible with user configuration
Liquid/gas conversion pressurization	Lighter than stored gas for LO_2	Weight penalty for gas generation
		Incompatible with connected ullage
Stored gas pressurization	Tank purging initiated if required	Weight penalty for gas supply
	Lighter than liquid/gas conversion for LH_2	Incompatible with connected ullage

3.4 NET POSITIVE SUCTION PRESSURE (NPSP) CONTROL

Propellant transfer accomplished by pumping fluid between the tanks will require control of the net positive suction pressure. The ullage pressure must be maintained above the vapor pressure with sufficient margin to prevent boiling and two phase flow in the transfer lines and receiver tank inlet. Alternately, the complete system could be designed for mixed phase flow. Figure A9-43 illustrates these two concepts. Mixed phase conditions in the transfer lines and tanks could be expected with the self-pressurization concept. An active pressurization system can be designed to provide the margin required to insure liquid phase transfer.

Data generated during Study 8, Cryogenic Acquisition and Transfer, as part of the Saturn S-II advanced technology studies (Reference SD71-768), shows that bubble collapse times resulting from a sudden increase in ullage pressure can be relatively long even though the fluid is instantly subcooled by the

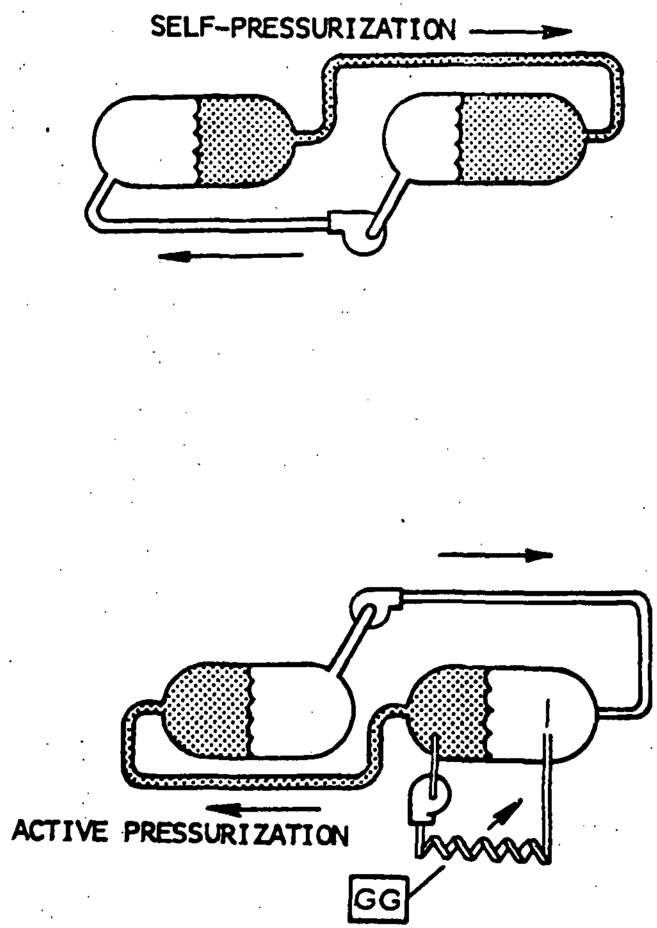


FIGURE A9-43. NPSP CONTROL CONCEPTS

pressure level change. Data are shown in Figure A9-44. These data assume that the bubbles are collapsed by convection heat transfer in the liquid and mass transfer between the liquid and the bubble. An analysis of this type will tend to produce conservative data or maximum collapse times with factors such as propellant agitation and the existence of a firm liquid/vapor interface reducing the collapse time. Although collapse times in an operating system would tend to be shorter than those shown, it is concluded that it is undesirable for the bulk propellant to become saturated during transfer operations such as may be the case if a self-pressurization concept were to be employed.

Figure A9-45 shows the weight penalty including system hardware for stored helium and liquid to gas active NPSP control systems for both LH_2 and LO_2. The weight of the propellant required to provide a 2 psi NPSP for the logistic tank, TUG, and CPS is shown on Figure A9-46.

The advantages and disadvantages of the NPSP control concepts discussed are presented below:

TECHNIQUE	ADVANTAGES	DISADVANTAGES
Self-pressurization	Passive system	Boiling (two-phase flow)
	No propellant losses	Gas entrapment in user capillary devices
		Poor performance predictability
Active pressurization	All liquid transfer flow	Significant propellant losses
	Good performance predictability	Active system required

It was concluded that the development risk for a self-pressurization was unacceptable at this time; therefore, active pressurization was selected as baseline.

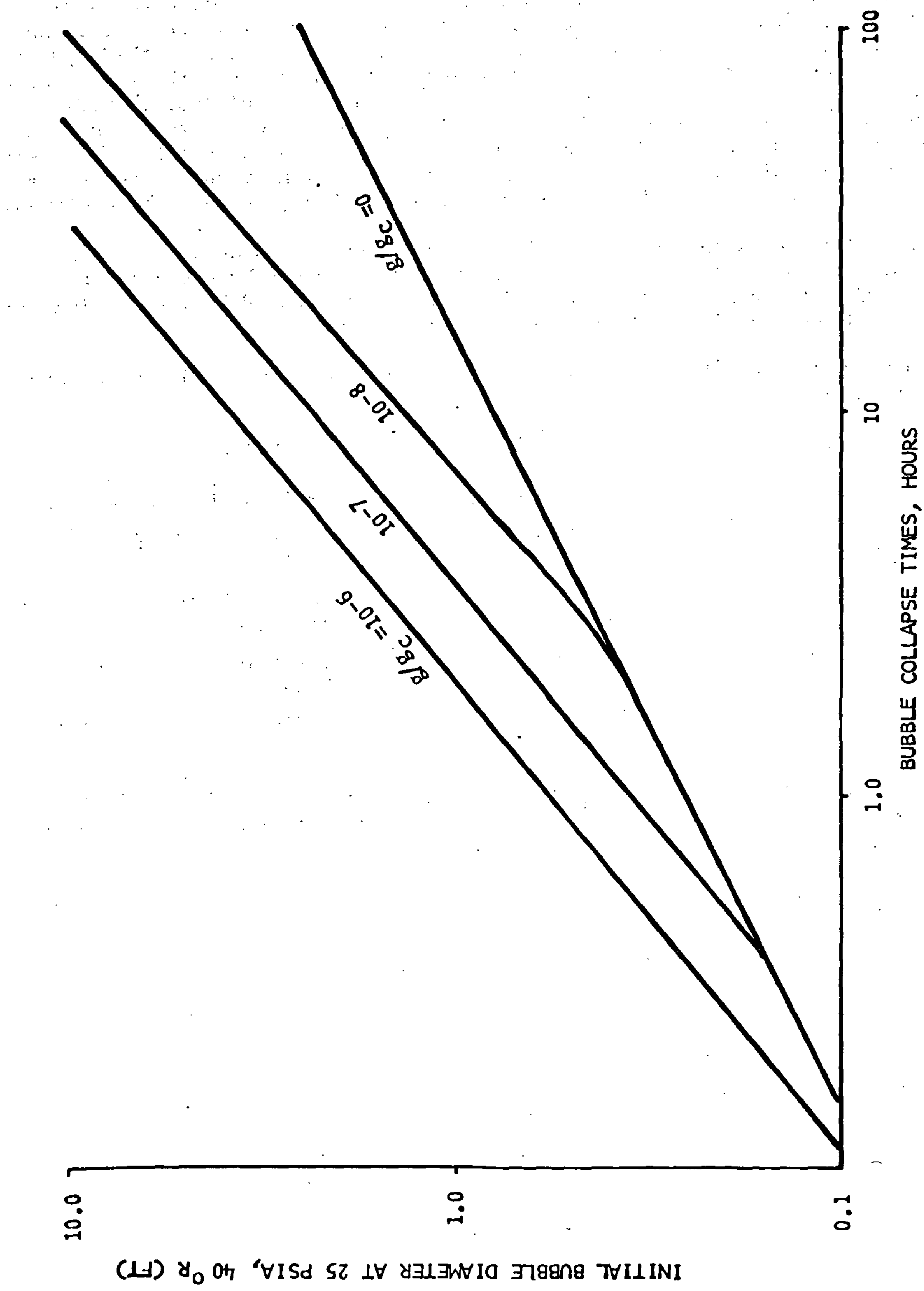


FIGURE A9-44 BUBBLE COLLAPSE TIMES FOR HYDROGEN VAPOR IN LIQUID SYSTEM SUDDENLY PRESSURIZED FROM 15 TO 25 PSIA

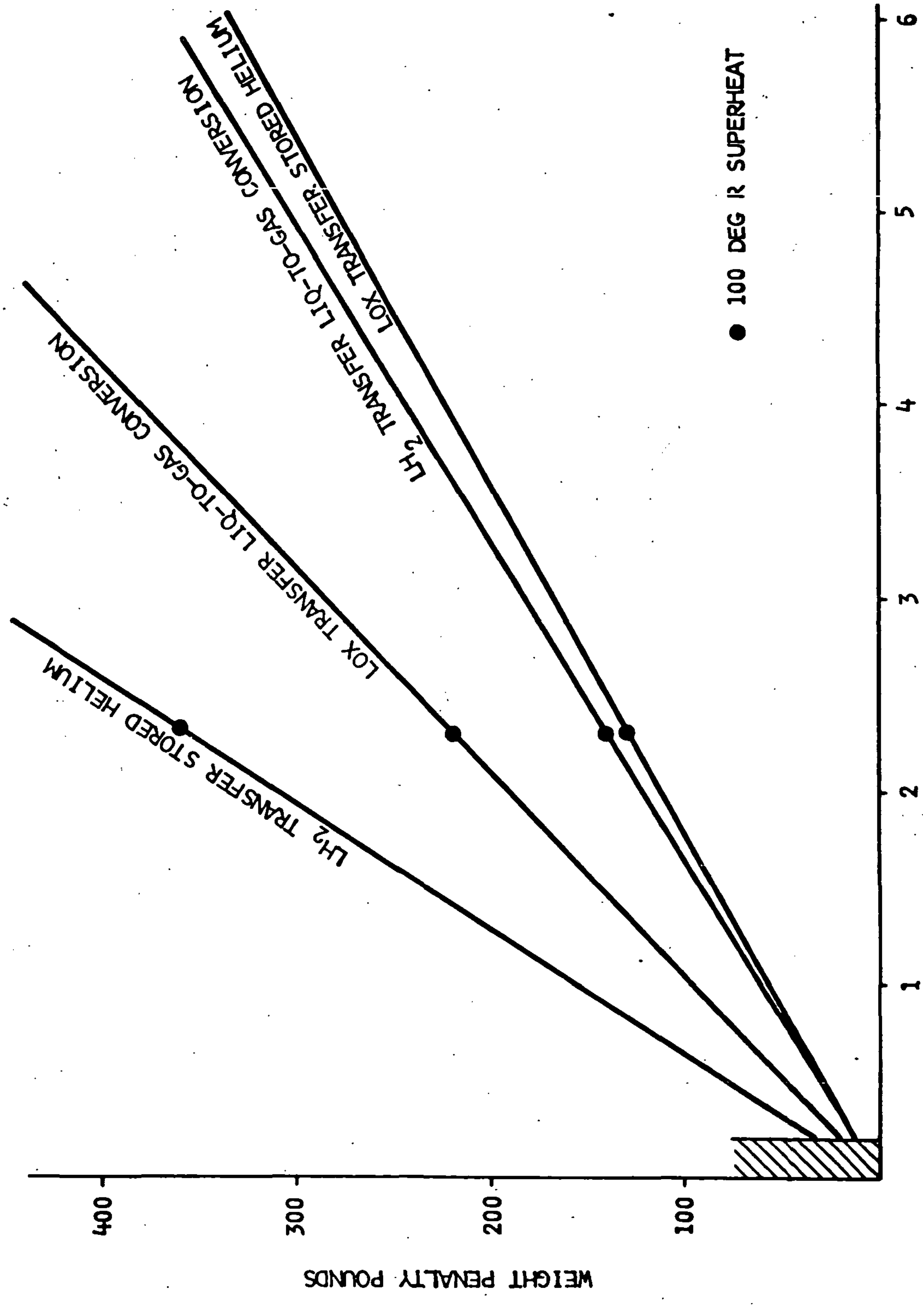


FIGURE A9-45 LOGISTIC TANK TO TUG TRANSFER NPSP CONTROL

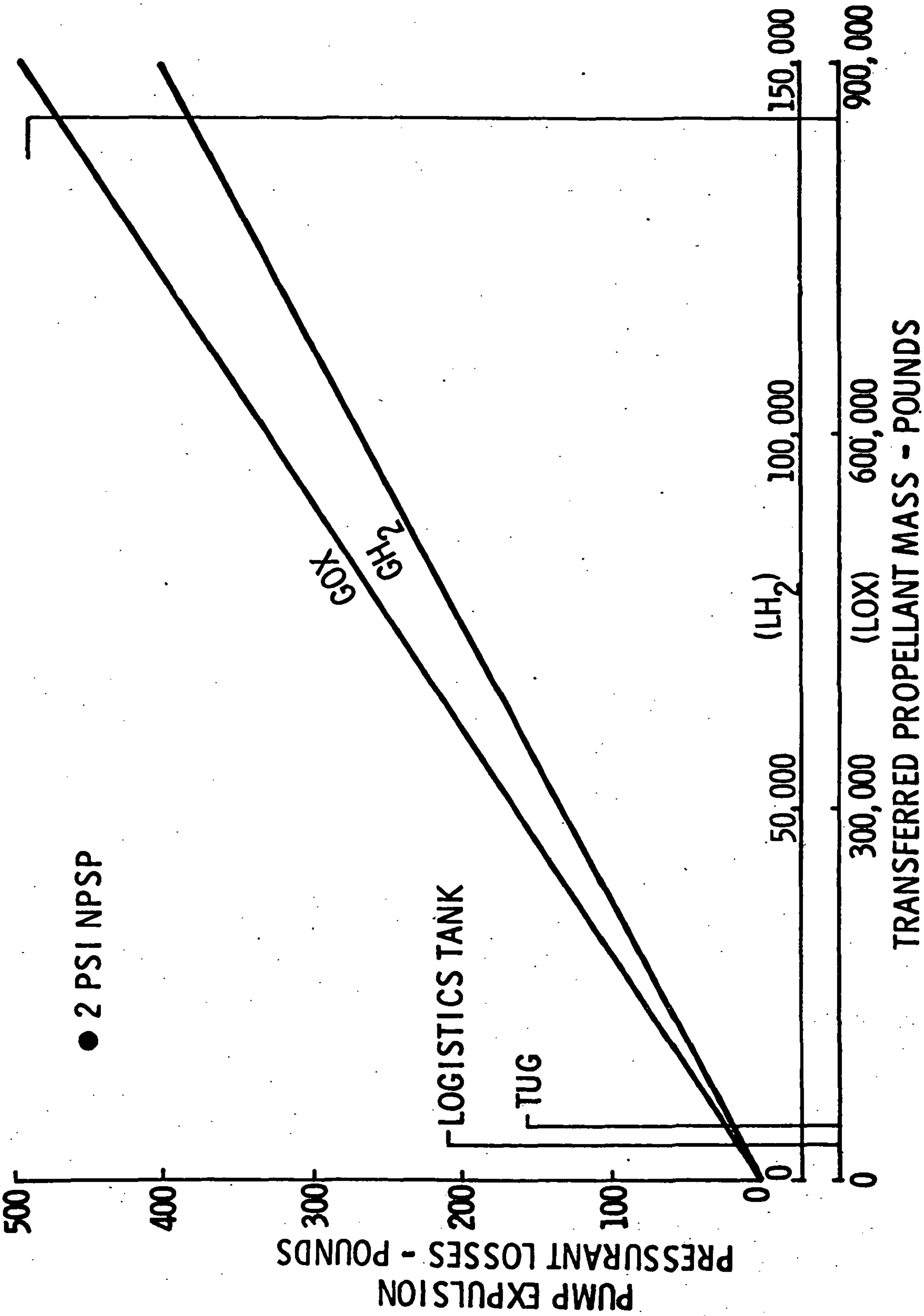


FIGURE A9-46 PRESSURANT LOSSES FOR NPSP CONTROL

4.0 LOGISTICS TANK TO VEHICLE PHYSICAL INTERFACES

4.1 FLUID TRANSFER INTERFACES

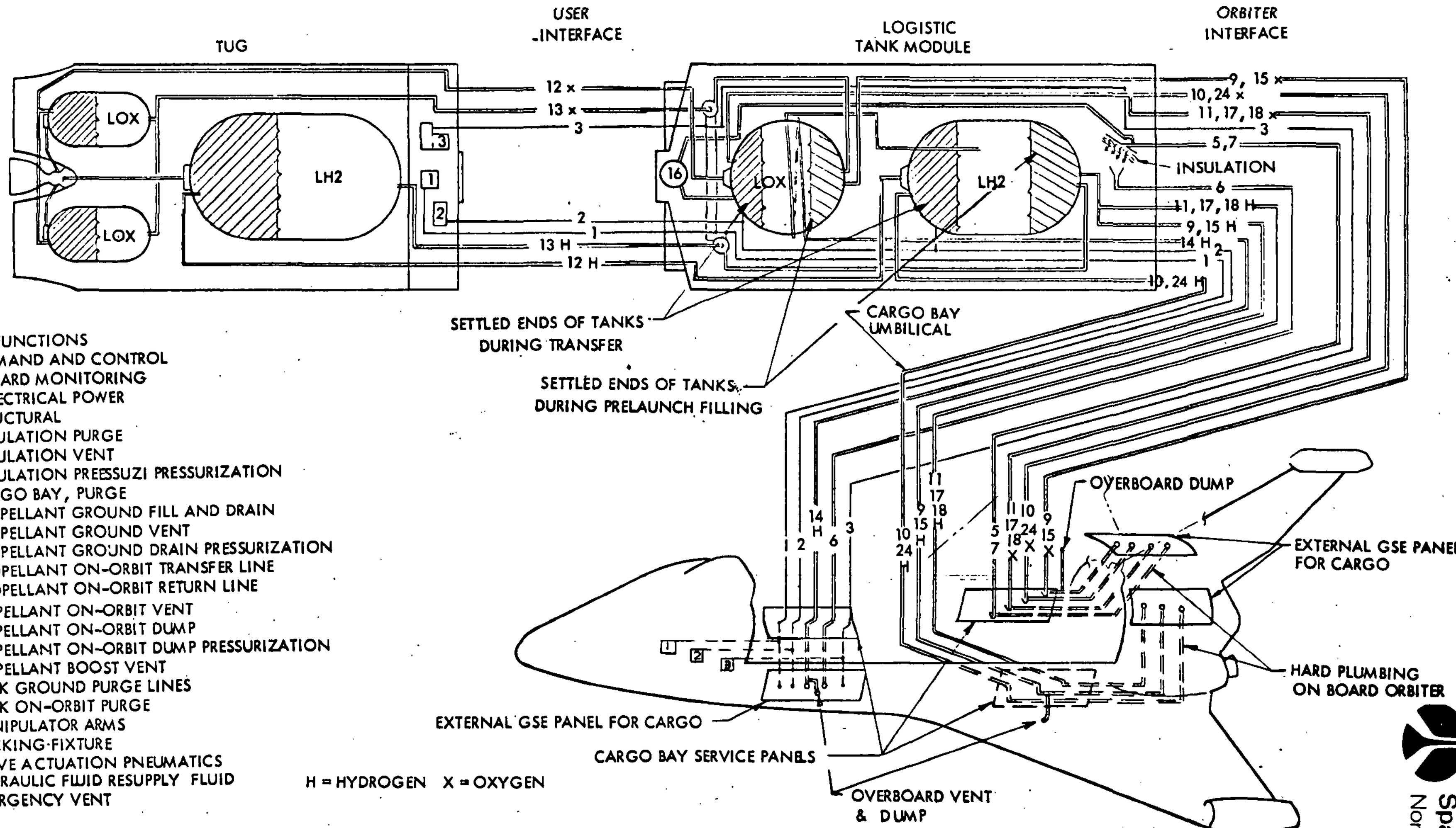
An analysis was conducted to identify the potential interfaces associated with an on-orbit fluid transfer involving a Shuttle orbiter, logistics tank, and a user vehicle. The analysis identified twenty-four potential interface functions. These functions were evaluated and a schematic was prepared showing the interface functions between the orbiter and GSE, logistics tank and user vehicle. The schematic is presented on Figure A9-48 along with the list of functions.

The schematic shows the fluid and electrical interfaces with the user vehicle and the orbiter cargo bay. Structural, docking, and handling interfaces (functions 4, 20, & 21) are not illustrated. The orbiter interfaces can be traced through the on-board plumbing to the GSE external service panels. The locations of service panels, line runs, dumps, and vents are not necessarily intended to reflect actual location requirements. It is anticipated that a cargo bay umbilical will be installed as a kit for logistics missions and will join the cargo bay service panels (wherever located) with the tank interfaces.

The user (or propellant transfer) interface is shown separate from the cargo bay interface to help clarify the schematic. This would be the actual case in some configurations of logistic tank modules. (TUG supportive tank modules will not likely be the same as a tank module for CPS or RNS. The schematic is of a "typical tank". Depending on combined configuration, location in orbiter and deployment of the tank, it may be preferable to combine the user interface and the cargo bay umbilical interface. This would combine the transfer supply and return lines (12 & 13) with the corresponding hydrogen or oxygen (H or X) ground fill and vent lines (9 & 10), respectively. This type of combining is shown on the tank schematic Figure A9-49.

Line routings in the tank module reflect a situation where the propellants would be settled toward one end of the module during ground (prelaunch) filling and settled toward the opposite end during artificial g settling for on-orbit transfer. This would be a worst case condition. If possible, (based on tank module cg location in the orbiter, transfer mode configuration and tank deployment), the tank module would be oriented so that propellants settle to the same ends of the tanks during prelaunch filling and during transfer.

No fluid lines are shown for valve actuation pneumatics or hydraulic fluid resupply (functions 22 & 23). It is anticipated that user vehicles will require resupply of these fluids as well as the primary propellants, and they were included on the list for that reason. If it should be determined that pneumatic valve actuation is required by the propellant transfer system, it would be combined with the on-orbit dump pressurization system (16) and the function to fill the helium bottle (22) would be added to the existing insulation purge and pressurization line (5 & 7). It is not currently anticipated that a propellant logistics module would be used in the resupply of such fluids as would be required to support manned operations.



FUNCTIONS

1. COMMAND AND CONTROL
2. HAZARD MONITORING
3. ELECTRICAL POWER
4. STRUCTURAL
5. INSULATION PURGE
6. INSULATION VENT
7. INSULATION PREESSUZI PRESSURIZATION
8. CARGO BAY, PURGE
9. PROPELLANT GROUND FILL AND DRAIN
10. PROPELLANT GROUND VENT
11. PROPELLANT GROUND DRAIN PRESSURIZATION
12. PROPELLANT ON-ORBIT TRANSFER LINE
13. PROPELLANT ON-ORBIT RETURN LINE
14. PROPELLANT ON-ORBIT VENT
15. PROPELLANT ON-ORBIT DUMP
16. PROPELLANT ON-ORBIT DUMP PRESSURIZATION
17. PROPELLANT BOOST VENT
18. TANK GROUND PURGE LINES
19. TANK ON-ORBIT PURGE
20. MANIPULATOR ARMS
21. DOCKING FIXTURE
22. VALVE ACTUATION PNEUMATICS
23. HYDRAULIC FLUID RESUPPLY FLUID
24. EMERGENCY VENT

H = HYDROGEN X = OXYGEN

FIGURE A9-48 ORBITER-LOGISTIC TANK-USER INTERFACE SCHEMATIC

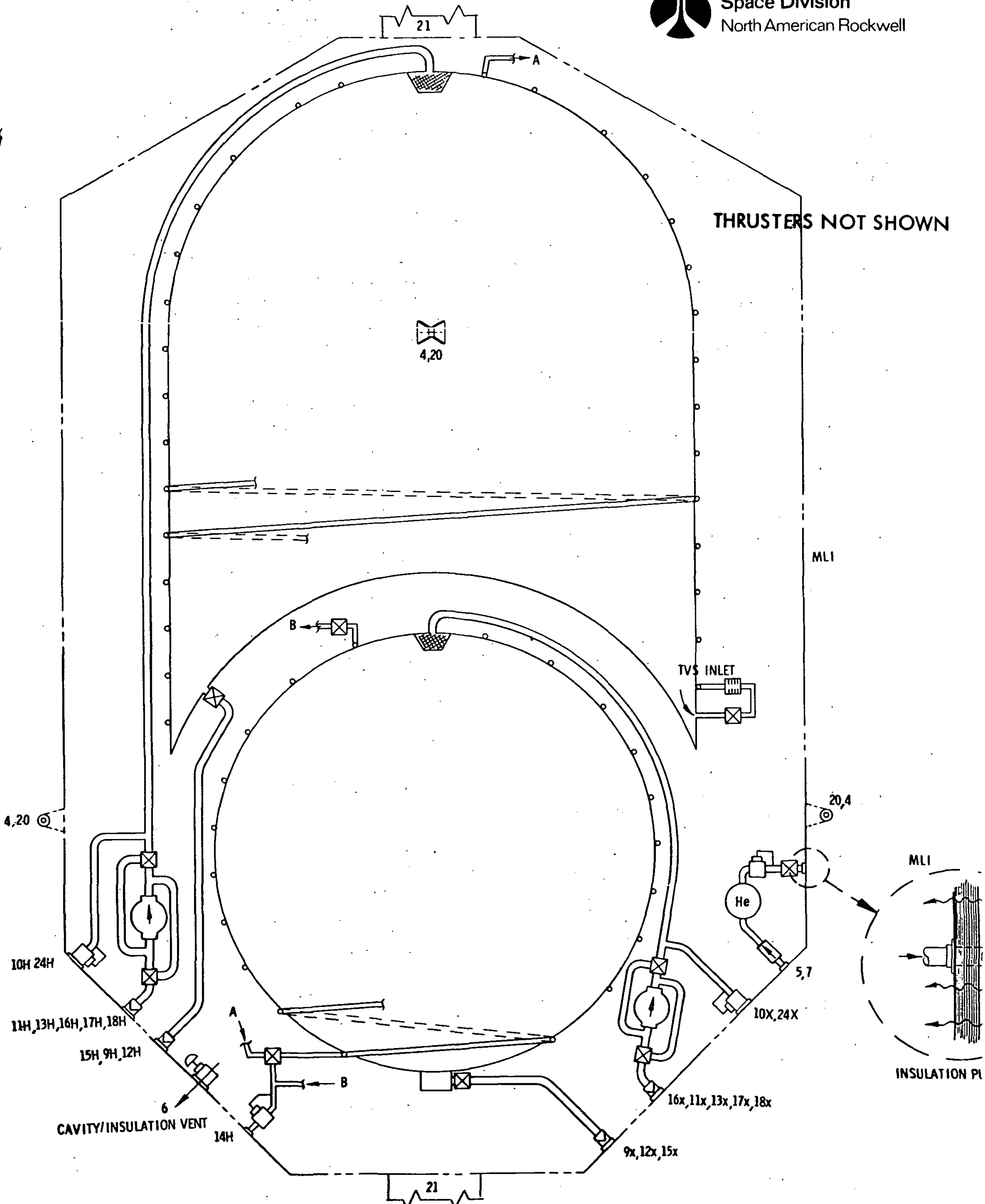


FIGURE A9-49 LOGISTIC TANK INTERFACE SCHEMATIC

The hydrogen on-orbit dump (15H) is shown combined with the LH_2 fill and drain line (9H), but it may be determined that hydrogen will not be dumped in an abort condition. In that case, the hydrogen overboard dump shown on the orbiter would not be required.

As can be seen on Figure A9-48 interface functions 1, 2, and 3 are required across the user interface and orbiter interface. Each of these functions are required during ground operations as well as orbit propellant transfer operations. The function will be electrical and interface hardware will be electrical connectors. Separate interfaces for electrical power, hazard monitoring and command and control will be necessary for safety purposes. Automatic connection will be required. A self-aligning electrical connector concept is shown in Figure A9-50.

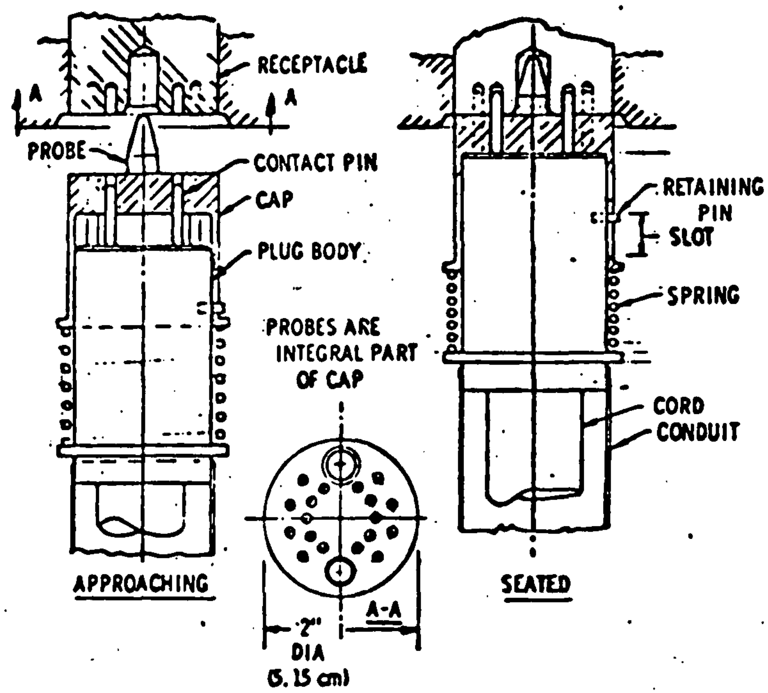
In addition to functional interfaces 1, 2, and 3 functional interfaces 12 and 13 for LO_2 (X) and LH_2 (H) are shown across the user interface on Figure A9-48. Figure A9-49 shows that only one interface connection is possibly required for each of the LH_2 or LO_2 functions 9, 12, and 15 and only one interface connection is possibly required for LH_2 or LO_2 functions 11, 13, 16, 17, and 18. These functions may require individual interface connection for ground operation between the orbiter and GSE, but not for orbiter to tank interface. Further study would be necessary to establish firm requirements. Also, safety or operational requirements may add additional interface connections to the logistic tank.

To accomplish the transfer operation, a docking interface function is necessary to attach the supplier to the receiver vehicle. Docking usually involves an active fixture on one vehicle and a passive ring on the other. The active fixture provides the attenuators, pulldown capability and final latching capability.

Figures A9-51 and A9-52 shows a design concept for a docking fixture and a line interconnect fixture. The line interconnect fixture provides dedicated and isolated propellant and non-propellant lines required to satisfy the applicable functions. Indexing probes are included for alignment and mating with bellows used for line extension to meet the stationary fixture of the receiver element. The triangular rack is integral with the structure with indexing probes acting as corner posts and truss work as side panels. Space inside the rack is compartmentized using insulated panels to separate LH_2 and LO_2 lines and non-propellant lines. Interlock is verified by signal reception.

Multiple docking ports are shown on Figure A9-49 because it is expected that transfer may require sequential transfer of the propellant tanks but only one fluid transfer interface is expected to be required.

Figures A9-53 and A9-54 give information on the cargo bay interface showing the orbiter attach points and retention system. These figures apply to any cargo, such as the logistics tanks and depot modules. The attach point locations and directions of load carrying capability are for the baseline 161 C orbiter but would be subject to change were further work done on that orbiter or for the later drop tank orbiter configuration.



- OPERATION:
  - THICK BASE OF CAP PROTECTS PINS BEFORE ENGAGEMENT
  - CAP HELD IN EXTENDED POSITION BY SPRING
  - PROBES ON CAP PRODUCE RADIAL & ANGULAR ALIGNMENT OF PINS
  - PROBES ON CARRIER PRODUCE PRIMARY ALIGNMENT
  - PINS FORCED THRU HOLES IN CAP AND ENGAGE RECEPTACLE
  - CARRIER CONTAINS LOCK
- PROJECTED TOLERANCE CAPABILITIES
  - MISALIGNMENT - 1/8 INCH
  - ANGULARITY:
    ROLL - 5°
    PITCH - 5°

FIGURE A9-50 SELF-ALIGNING ELECTRICAL CONNECTOR

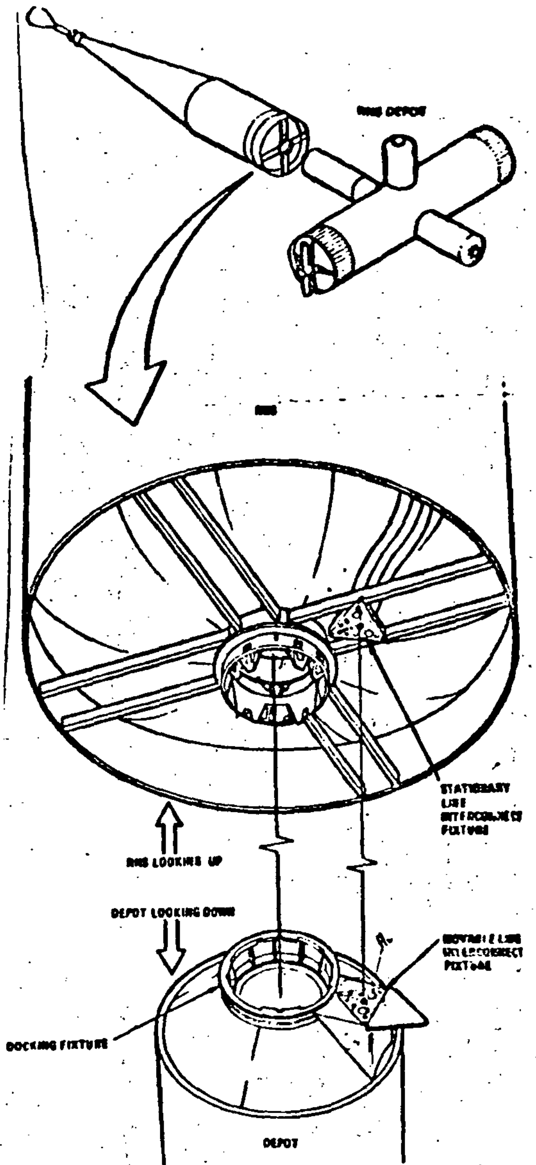


FIGURE A9-51 DOCKING AND PROPELLANT TRANSFER LINE MATING CONCEPT

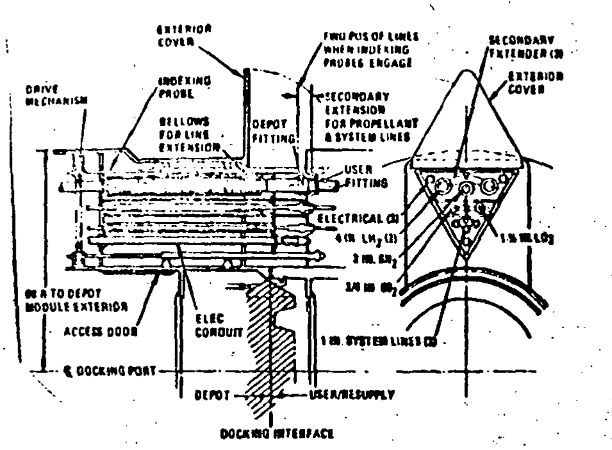


FIGURE A9-52 LINE INTERCONNECT FIXTURE

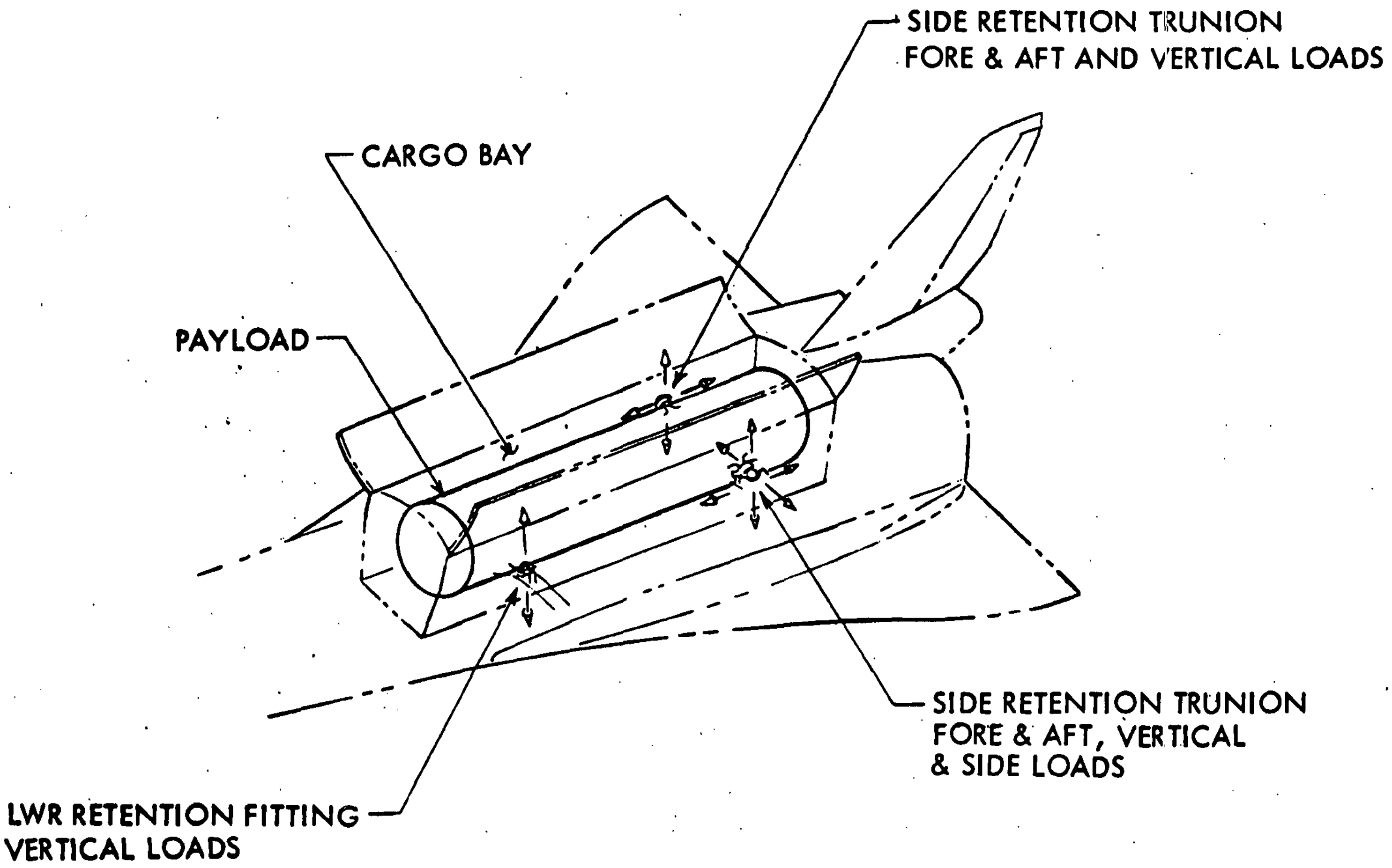


FIGURE A9-53 ORBITER PAYLOAD RETENTION SYSTEM

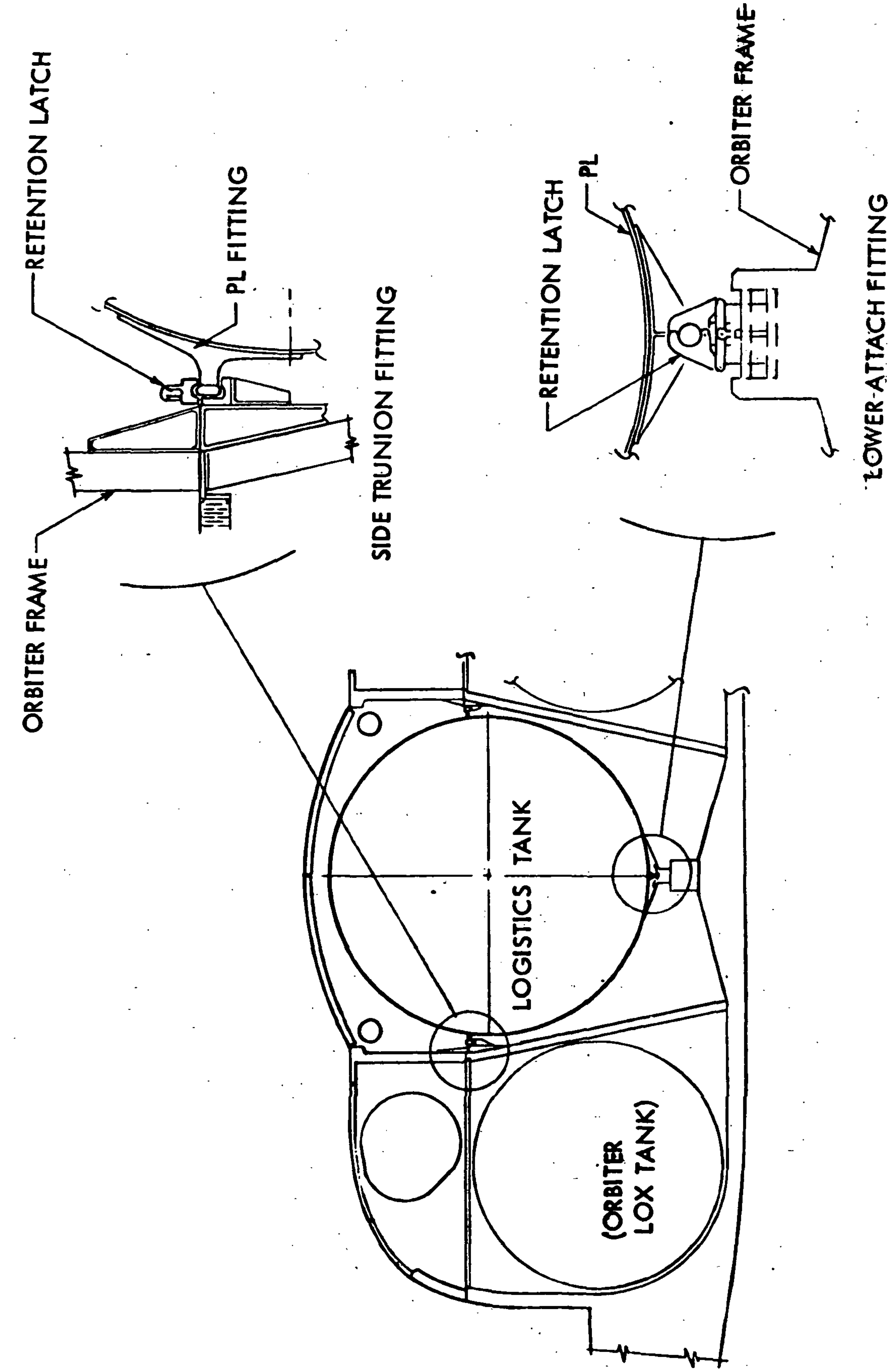


FIGURE A9-54 LOGISTIC TANK & SHUTTLE CARGO BAY INTERFACE ATTACHMENT

Payload handling in orbit, for removal from and return of the logistics tanks to the orbiter cargo bay, is assumed to be accomplished by the orbiter manipulator arms. Orbiter has design requirements for perfecting such a cargo handling system and present definition tank design will rely on those provisions with appropriate manipulator attach fittings included (similar to the attach fittings) on the exterior of the tank shell.

Figure A9-55 illustrates the logistics tank module in the orbiter cargo bay. Its location is based on taking the radial load of the base retention fitting at the major structural ring of the tank. This is compatible with c.g. location and leaves room for umbilical connection at the forward end of the tank but slightly compromises the space remaining for payload sharing. LO_2 and LH_2 settle to the same end of the tanks during ground filling as they will during propellant settling mode during orbital transfer. This eliminates duplication of plumbing and valving for proper fill and vent locations and of quantity gaging and monitoring systems.

Figure A9-56 shows the transfer capability tank module in the orbiter cargo bay. The interface considerations are the same. Maintaining ground fill settling in the same direction as orbital settling has been slightly compromised utilizing the transfer interconnects with the cargo bay umbilical. Payload sharing is much the same. In general, the cargo bay interfaces can be considered similar enough for all the logistics tank configurations so as not to be a determining factor in operation concept tradeoffs.

4.2 MODULAR TRANSFER INTERFACES

The modular transfer concept provides for delivery of full propellant tanks to the user vehicle, and substituting the full tanks for previously emptied tanks. The interfaces for modular transfer tanks (i.e., plug-in tank sets) appear to be similar to those required between integral propellant tanks and engines on existing or proposed space vehicles. Tank isolation valves would be required to maintain the propellants in the tank during tank delivery. Thermal and structural connections of the tanks to the space vehicle engine and payload would be required. Subsystem interfaces, such as dictated by the vehicle configuration, such as electrical, purging etc., would also be required. A docking mechanism and fluid transfer interconnect similar to that shown for fluid transfer in Figure A9-51 and A9-52 could be used.

Several problems associated with this concept of propellant transfer can be seen. One problem that makes the concept questionable is the weight penalty that must be considered when the space vehicle tank is also used to bring propellant from earth surface to orbit. This extra weight is then part of the space vehicle and the mission must provide sufficient propellant to overcome the extra weight. Studies have shown that it takes approximately eight pounds of propellant to support one pound of extra weight for a geosynchronous equatorial placement mission for the TUG. Additional weight would be required to provide adequate attachment fixtures for the propellant and pressurization system lines and structure.

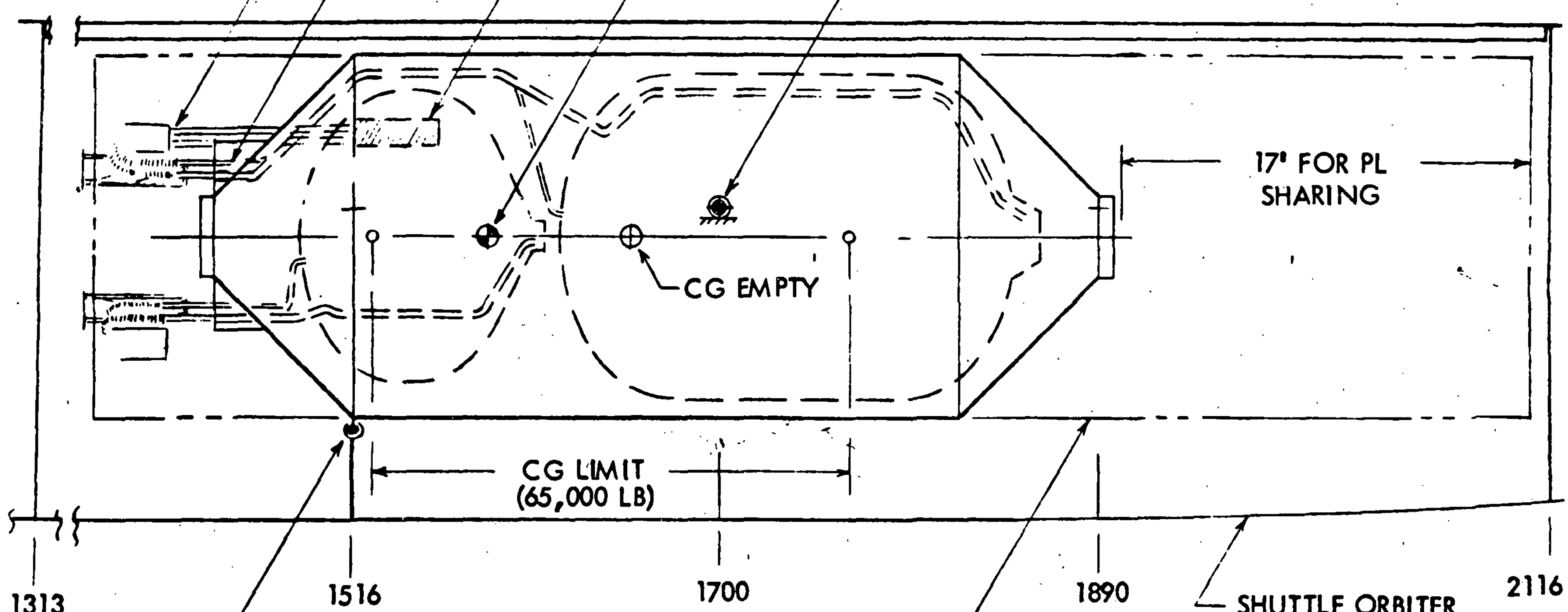


FIGURE A9-55 LOGISTIC TANK & SHUTTLE CARGO BAY INTERFACE

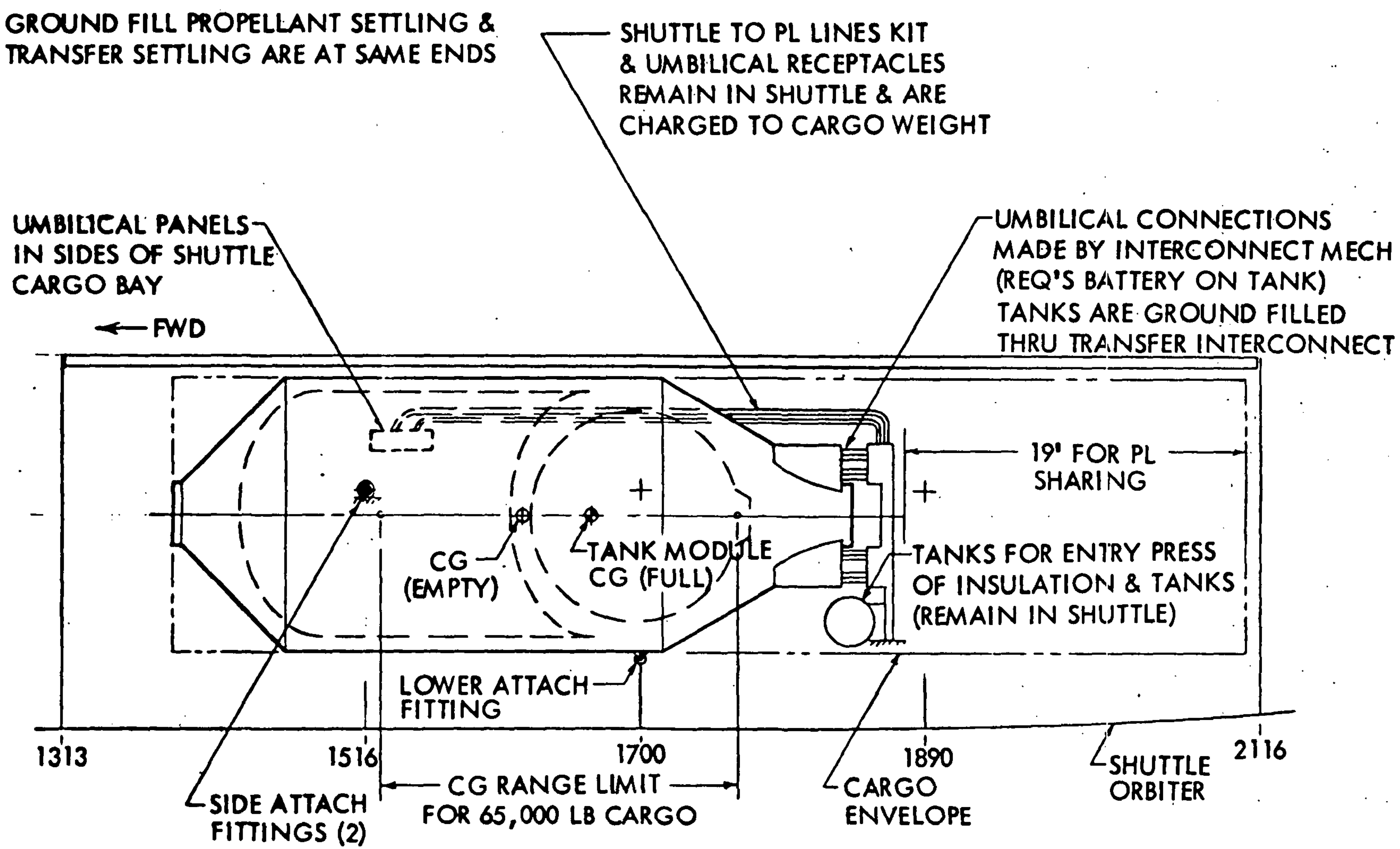


FIGURE A9-56 TRANSFER CAPABILITY TANK & SHUTTLE CARGO BAY INTERFACE

Another penalty is due to ascent and space storage boiloff. Boiloff losses could be large enough to require propellant topping, which requires fluid transfer. Thus, fluid transfer systems and interfaces are required, and tank module sets have the disadvantage of both systems.

In summary, modular transfer appears to be feasible. However, considerable study is required before detailed interface definition could be developed.

5.0 NUMBER OF FLIGHTS FOR PROPELLANT RESUPPLY

Each logistic option considered above involves several steps in the delivery of propellants to the user vehicle. Each step has some propellant loss chargeable to the transfer process. These losses are categorized and loss percentages based on quantity of propellant transferred have been developed. These values are based on data obtained from other studies (DS-451 and ISPLS). The loss values used are identified in Table A9-2.

In addition to the loss data, a typical propellant delivery time line was developed for supply of the TUG, CPS, and RNS to allow calculation relative to losses for each supply mission. An assumption of seven days was made relative to the time between propellant delivery flights. This was based on the availability of more than one shuttle for propellant resupply and the shuttle having a 21-day or less turn-around capability. A summary of the time line (or sequence of events) is presented in Table A9-3.

Using the above developed data, the payload capability of the shuttle orbiter and the propellant loads required by the user vehicle, the quantity of propellant supply flights for each user can be determined. The shuttle payload capability is shown in Figure A9-57. It is assumed that the orbital maneuvering system abort propellants of the shuttle orbiter can be used for mission completion since this would support an efficient operation. Also, it is assumed that the shuttle orbiter can deliver the propellant payload to the user vehicles parking orbit. Propellant quantities required by the user vehicles as identified in Appendix C are:

TUG	78,946 pounds
CPS	1,080,000 pounds
RNS	300,000 pounds

Table A9-4 identifies the number of flights required to support the tug, CPS, or RNS for the four propellant logistic options chosen for analyses, as shown in Figure A9-58. These logistic concepts are defined in Section 3, Part 4 of Vol. II. A sample calculation in determining the number of flights follows:

Using TUG for Concept 11 as an example, it can be seen that boil-off losses are 1.9 pounds per hour (Ref. Table A9-2) for a period of 7 1/2 hours (Reference Table A9-3). This establishes 14 pounds loss for boil-off in the shuttle orbiter during delivery.

$$1.9 \text{ pound/hr} \times 7\ 1/2 \text{ hr.} = 14 \text{ pounds} \qquad (1)$$

Loss for fluid transfer from the logistic tank to the mini-depot is determined by identifying the propellant loss percentage 2.1% (Reference Table A9-2) of the tank capability (60,000 pounds).

$$2.1 \times 60{,}000 = 1260 \text{ pounds} \qquad (2)$$

TABLE A9-2 PROPELLANT TRANSFER LOSSES

	TUG	CPS	RNS
Boil-off in Shuttle during delivery or in modular tank.	1.9 lb/hr.	1.9 lb/hr.	1.9 lb/hr.
Losses for fluid transfer from delivery or storage element to storage or user element.*,**	2.1%	5.7%	3.1%
Boil-off losses in storage or user during refill waiting period of 7 days.	0.5%	0.5%	0.5%

NOTE: * Loss percentages are based on an average 60,000 pound delivery of LH_2 and LO_2 for TUG and CPS and 34,000 pound delivery of LH_2 only for RNS. Delivery to TUG and CPS is weight limited; delivery to RNS is volume limited.

**Losses include propellant required for settling thrust, attitude control system operation, task and line residuals, pumping power, line chilldown, heat leak, and NPSP control.

TABLE A9-3 TYPICAL PROPELLANT DELIVERY TIME LINE

	SEQUENCE OF EVENTS	DURATION
1.	Shuttle liftoff with supply tank	
2.	Orbiter maneuvers to orbit and docks supply tank with space based user element.	5 hours 30 min.
3.	Supply tank and space based user pre-transfer hookup and checkout	1 hour, 45 min.
4.	Transfer propellant or modular tank	10 hours
5.	Post transfer checkout and securing	1 hour, 30 min.
6.	Space based user separates from supply tank or extra module.	
7.	Space based user waits for additional resupply if required.	7 days

TABLE A9-4 PROPELLANT RESUPPLY FLIGHTS REQUIRED

LOGISTIC CONCEPT (SEE FIGURE A9-58)	USER VEHICLE	BOILOFF IN SHUTTLE DURING DELIVERY OR IN MODULAR TANK	LOSS FOR FLUID TRANSFER FROM DELIVERY OR STORAGE ELEMENT TO STORAGE OR USER ELEMENT		BOILOFF LOSSES IN STORAGE OR USER DURING REFILL PERIOD	TOTAL LOSSES PER FLIGHT	NO. OF FLIGHTS REQUIRED
			TO STORAGE ELEMENT	TO USER ELEMENT			
		LB	LB	LB	LB	LB	
1	TUG	14	19	1260	300	1593	1.36
	CPS	14	19	3420	300	3753	19.20
	RNS	14	19	1054	170	1257	9.16
7	TUG	14	19	-	300	333	1.32
	CPS	14	19	-	300	333	18.10
	RNS	14	19	-	170	303	8.90
8	TUG	14	-	1260	300	1574	1.35
	CPS	14	-	3420	300	3734	19.19
	RNS	14	-	1054	170	1238	9.16
11	TUG	14	1260	1260	300	2834	1.38
	CPS	14	3420	3420	300	7154	20.44
	RNS	14	1054	1054	170	2292	9.46

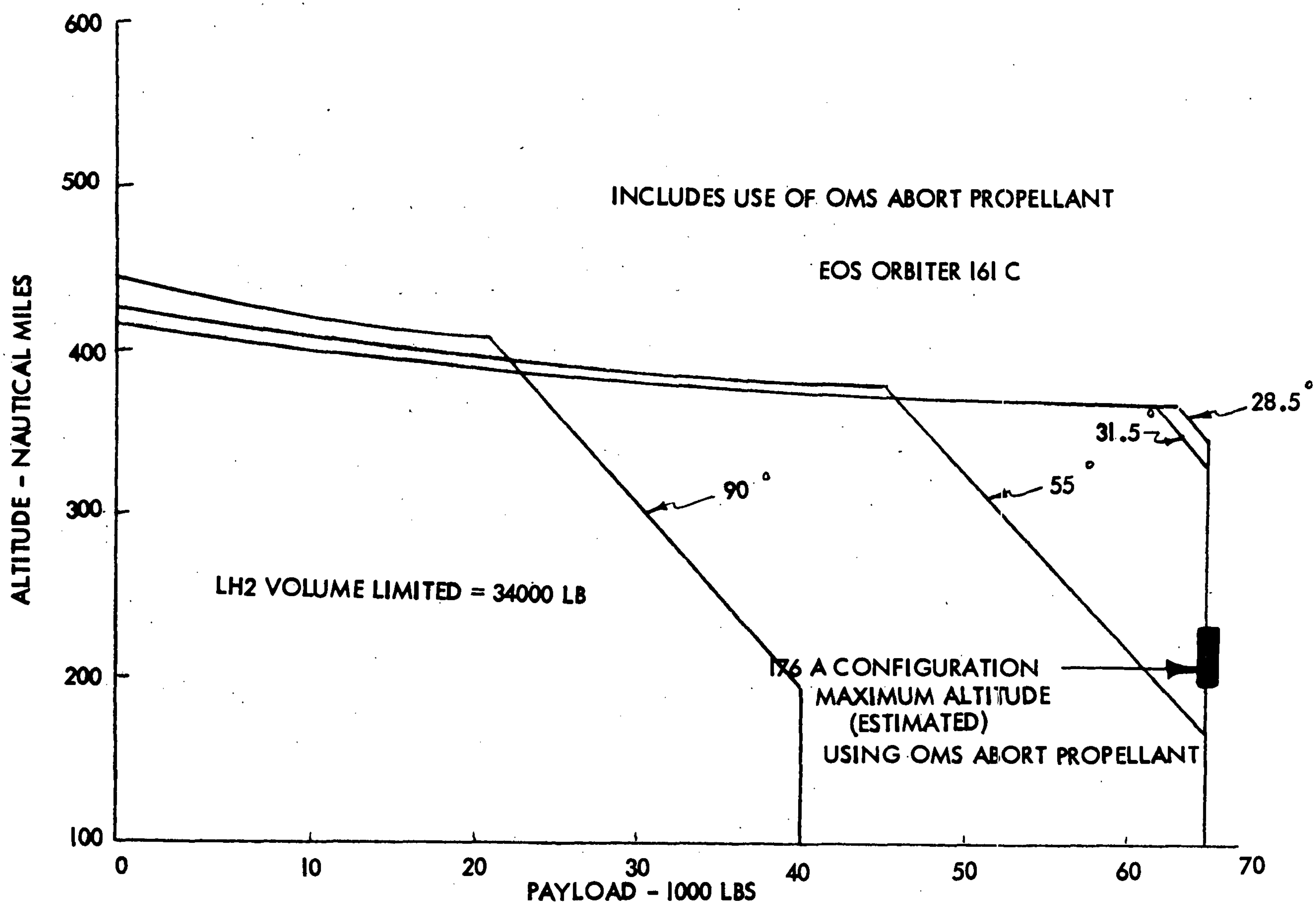


FIGURE A9-57 SHUTTLE PAYLOAD CAPABILITY

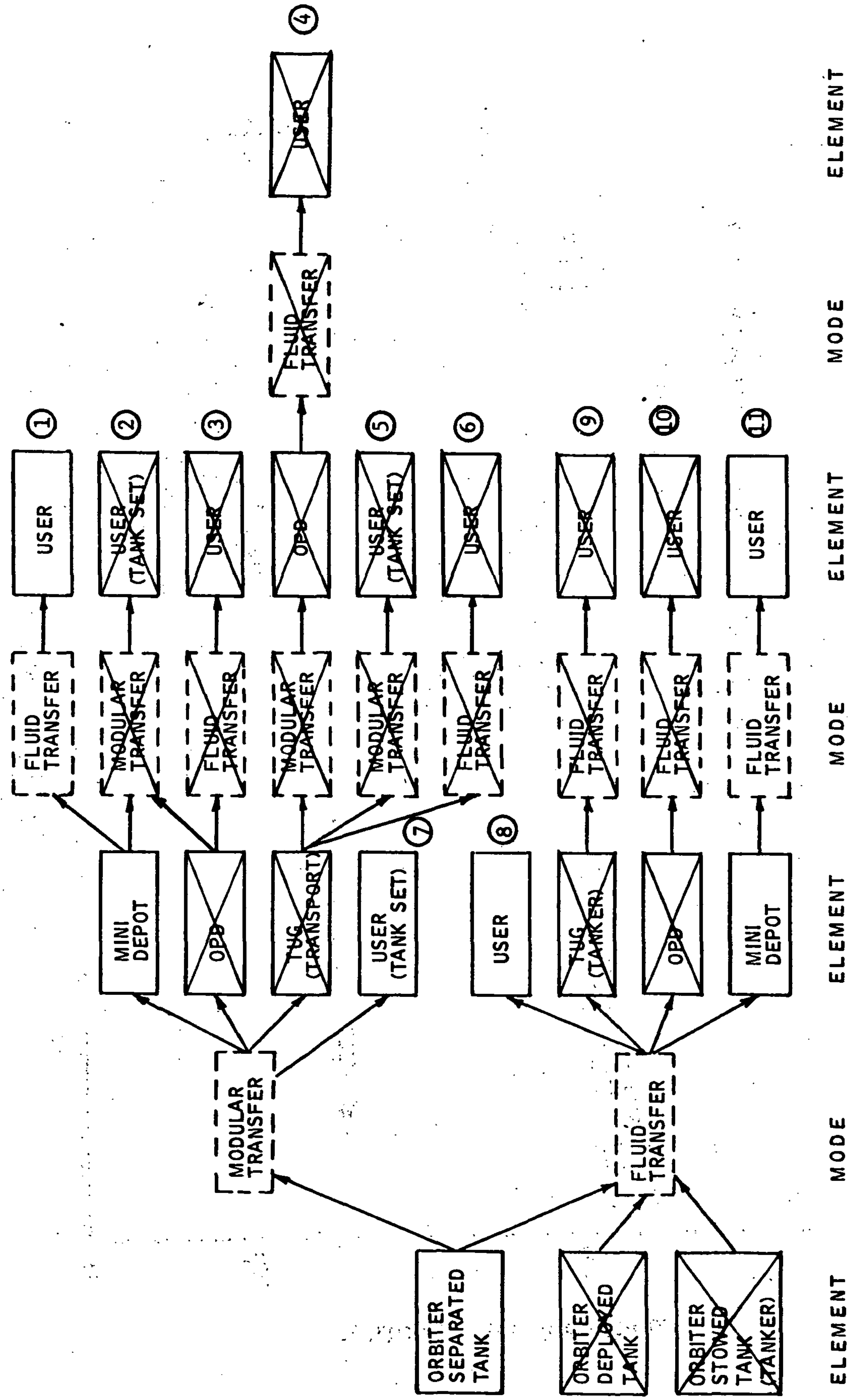


FIGURE A9-58 PROPELLANT TRANSFER LOGISTICS OPTIONS

An identical loss is determined for the transfer loss from the mini-depot to the user.

$$0.021 \times 60,000 = 1260 \text{ pounds} \quad (3)$$

Storage in the user for seven days involves a loss of .5 percent (Reference Table A9-2) for boiloff.

$$0.005 \times 60,000 = 300 \text{ pounds} \quad (4)$$

Totaling (1) + (2) + (3) + (4) = 2834 pounds as the total loss per shuttle trip which provides 57,166 pounds of usable propellant per shuttle delivery to the user.

$$60,000 - 2834 = 57,166 \text{ pounds} \quad (5)$$

With the TUG requiring 78,946 pounds of propellant, it can be seen that 1.38 shuttle loads are required to fill the TUG.

$$78,946 \div 57,166 = 1.38 \text{ trips} \quad (6)$$

The losses of the modular tank were assumed to be at the rate of 1.9 pounds per hour with a transfer time including checkout of 10 hours. This resulted in losses of 19 pounds for a modular transfer.