

It is with great pleasure that we present this revised and expanded edition of the Inertial Upper Stage (IUS) User's Guide. This document is intended to introduce both commercial and government users to the IUS system. It presents essential data for preliminary mission planning and analyses, spacecraft conceptual design and spacecraft/launch vehicle compatibility assessment.

The IUS is built under Air Force contract as an upper stage for both the Space Shuttle and the Titan 34D. In either case, the IUS mission is the same: the placement of spacecraft into higher energy Earth orbits.

Since the last user's guide was published, the IUS program has moved from development to production and operation. Boeing's contracts with the Air Force now include 14 Inertial Upper Stages.

The first operational flight of the Inertial Upper Stage successfully delivered the DSCS II and DSCS III satellites to geosynchronous orbit using a Titan 34D boost vehicle. Precise accomplishment of the mission objectives on October 30, 1982 culminated a demanding 6-year design/development program.

The second IUS mission was launched on April 4, 1983. While this mission was not as successful as the first, the IUS provided 96% of the total transfer energy required to place TDRS-A in its desired geosynchronous orbit. Since that time, Boeing has dedicated itself to reestablishing the flawless performance record set by the first mission by improving design margins, introducing design enhancements and by reinspection/retest.

The IUS and its derivatives offer exciting possibilities to a wide spectrum of spacecraft users. Design activities are now underway for four derivative vehicles: a performance improvement option, a reduced cost option, an integrated propulsion module, and a perigee stage. In addition, the IUS has structurally demonstrated a heavy payload carrying capacity.

The Boeing Aerospace Company is committed to the spacecraft community. We stand ready to work with users in the creation or expansion of their space roles and missions. The IUS is a proven asset which can be used effectively to meet unique and diversified demands.



D. O. Love  
IUS Program Manager



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## CHAPTER I - INTRODUCTION

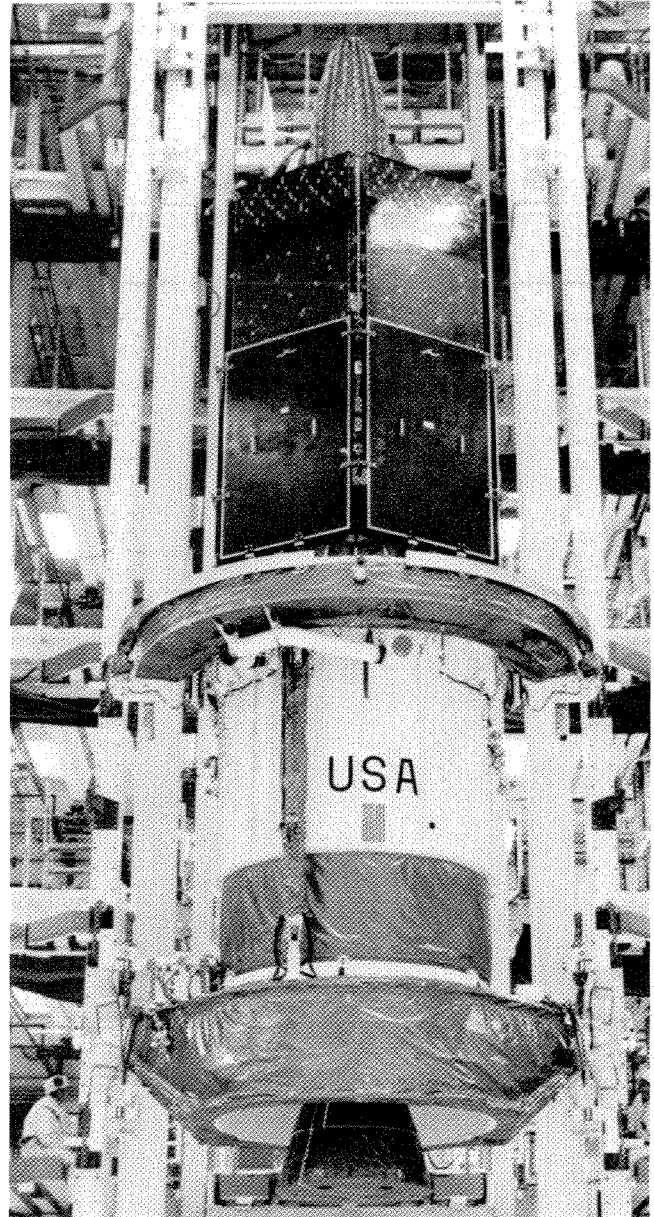
The Inertial Upper Stage (IUS) is a new-generation high altitude booster which is used in conjunction with either the Shuttle or Titan launch vehicle. It is built by the Boeing Aerospace Company and is managed by the Air Force Systems Command Space Division. Developed for both the Air Force and the National Aeronautics and Space Administration (NASA), the IUS has become an integral part of the nation's Space Transportation System (STS).

As an innovation in space technology, the IUS has scored several significant firsts. These are:

- o The *FIRST* completely redundant avionics system ever developed for an unmanned space system. It has the capability to correct failures within milliseconds.
- o The *FIRST* application of a 65,000 word solid state computer in a boost/space environment.
- o The *FIRST* strap-down redundant inertial measurement unit in space.
- o The *FIRST* upper stage designed for ultra-high reliability (0.98).
- o The *FIRST* upper stage offering extensive spacecraft services, e.g., electrical power, telemetry, and command.
- o The *FIRST* upper stage guidance system designed to fly from either a Shuttle or Titan launch vehicle.
- o The *FIRST* guidance system capable of retargeting in order to accommodate different Shuttle deployment opportunities.
- o The *FIRST* flight propulsion system using an extendible exit cone.

### USER BENEFITS

The IUS offers a full spectrum of benefits to the user.



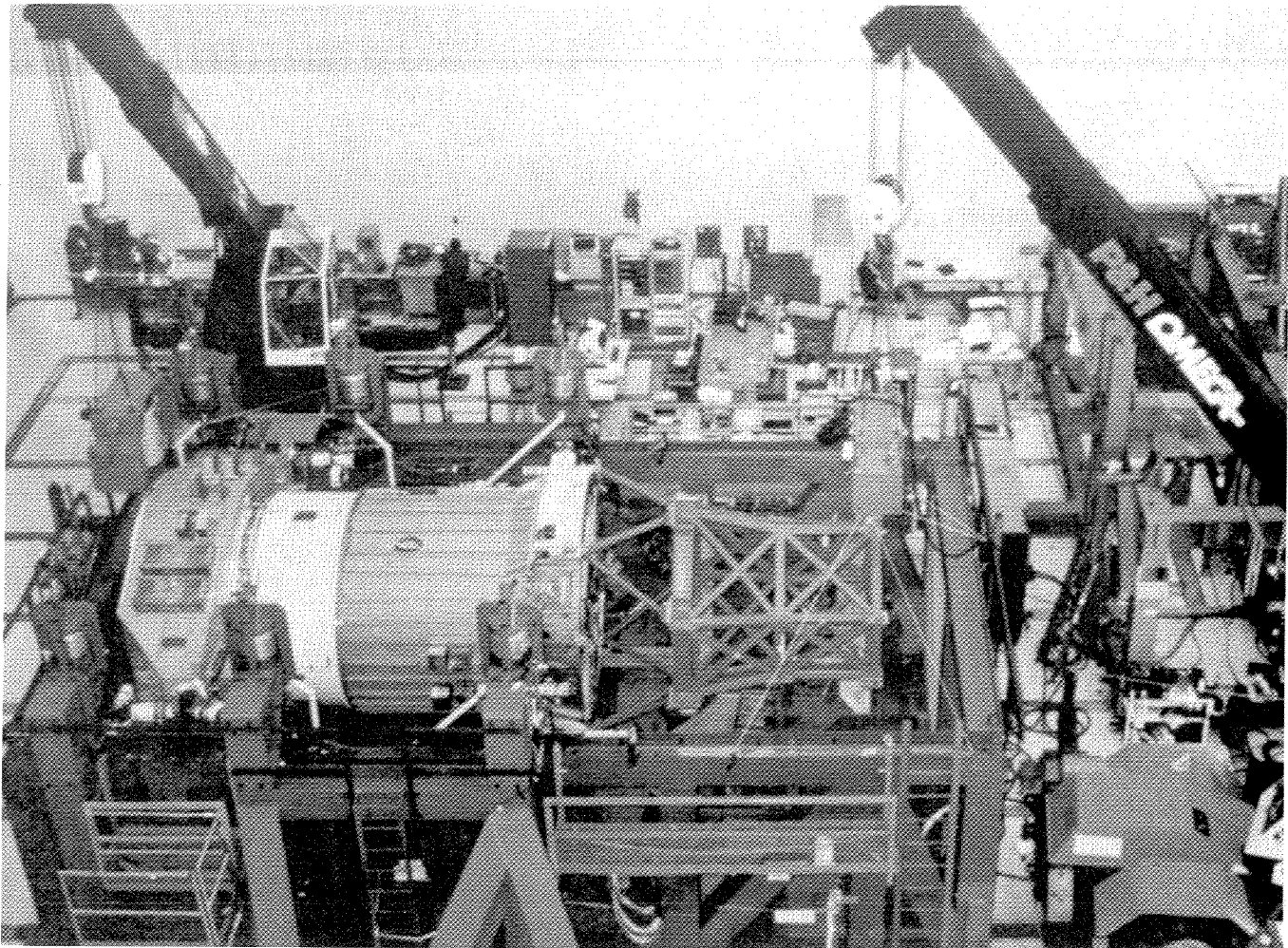
*IUS-1 and TDRS-A in Vertical Processing Facility at the Eastern Launch Site*

- o The user can select his choice of launch systems: the Shuttle or the Titan T34D, now available for commercial use.
- o The user can choose from a wide range of earth orbits and even has the option of a planetary trajectory. (See Performance Capability, Chapters 3 and 4)

- o The spacecraft's on-orbit useful life is extended because the IUS is capable of precisely placing the spacecraft in orbit. This allows the spacecraft to keep more of its propellant. (See Performance Capability, Guidance Accuracy, Chapter 4)
- o The IUS offers more services to the spacecraft than ever before. (See Spacecraft Services, Chapters 3 and 4).
- o A load alleviation system has been incorporated into the design of the Shuttle-qualified airborne support equipment to minimize impacts on the spacecraft design. (See IUS/Shuttle/Spacecraft Physical Interfaces, Chapter 2 and also Appendix A)
- o Spacecraft mission success is virtually assured by IUS redundancy in all mission-critical functions. (See Avionics, Chapter 2)
- o Risks to the launch are minimized because of the safe IUS propulsion system. (See Propulsion, Chapter 2)

## IUS DEVELOPMENT

Originally designed as a temporary stand-in for a reusable space tug, the IUS vehicle was first named the Interim Upper Stage. The Interim Upper Stage concept was first devised by Vice President Agnew's Space Task Group in 1969. At that time, the IUS concept was merely a modification of an



*STS Modal Survey Test on IUS Structural Test Vehicle*

existing expendable upper stage which could be readily adapted to the Shuttle's orbiter. The advanced technology Space Tug was to be developed by 1984.

When it became apparent that the Space Tug would not be developed in the foreseeable future, the "interim" upper stage began its gradual evolution to the "inertial" upper stage ("inertial" refers to the guidance technique).

After two and a half years of design competition, Boeing was selected in August 1976 to begin preliminary design of the Inertial Upper Stage. During that time, the vehicle design had changed from a simple growth version of the Boeing Burner II upper stage with virtually no new technology to an all-new upper stage. The primary objective of the new design was to provide an upper stage with higher reliability and with greater payload capability than the existing upper stages then used on expendable launch vehicles.

The IUS two-stage configuration is a complete upper stage system that eliminates the need for an apogee kick motor in the spacecraft. Boeing chose solid propellant motors over liquid propellant engines because of their relative simplicity, high reliability, low cost and safety. A modular approach, another fundamental design feature, was selected so that the basic hardware could be more efficiently used.

In 1981, the IUS successfully completed its development and qualification program. During the system level qualification testing (conducted to the very severe environmental limits prescribed by MIL-STD-1540A), no part failures occurred. This is due to the stringent parts procurement requirements specified in SAMSO-STD-73-2C which were subsequently implemented on the IUS program and to the vigorous component testing performed prior to system level testing.

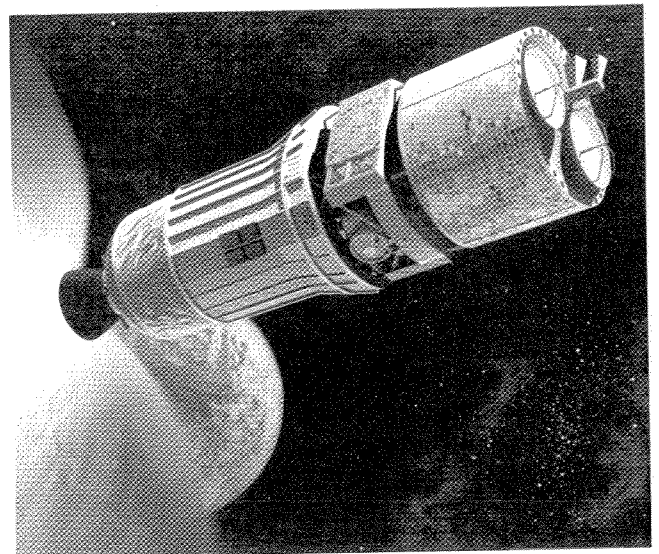
The qualification test vehicle was shipped to the Eastern Launch Site in December of 1981. Renamed Pathfinder-D, it has been used to verify IUS installation and checkout in the Shuttle Processing Integration Facility (SPIF). Similarly, Pathfinder-C has validated the IUS groundflow for the Titan launch.

## First Launch

Liftoff for the first IUS mission which involved both a new launch vehicle, the Titan T34D, and a new spacecraft, DSCS III, took place October 30, 1982.

This first mission placed two Defense Satellite Communication System spacecraft (DSCS II and DSCS III) into near-synchronous equatorial orbit following insertion into a 80x95 nautical mile park orbit by a Titan T34D boost vehicle.

All mission events occurred within the expected time intervals and in the correct sequence. All boost and separation events were normal. Nominal IUS/T34D separation was accomplished 8 minutes 14.1 seconds after launch. SRM-1 ignition commenced near the first ascending node as planned; the SRM-1 burn duration was 164 seconds. The IUS on-board gamma-guidance system calculated the resultant velocity, detected a short-fall, and extended the follow-on RCS-1 burn to achieve the target velocity. During the next 5 hours the IUS performed thermal rolls and roll reversals at equal intervals as scheduled. Separation of the IUS first stage occurred nominally 2 minutes and 20 seconds before SRM-2 ignition. SRM-2 ignition occurred within one minute of predicted ignition time, and SRM-2 burn duration was 81 seconds. The follow-on RCS-2 trim burn



*Artist's Conception of IUS-2 and its DSCS Payload*

placed the IUS/DSCS II/DSCS III in the planned final orbit ready to orient for DSCS II release. After the IUS maneuvered to the release attitude, the DSCS II separated as planned at an altitude of 22,780 n.m. and 2.493 degrees inclination. The target orbit for DSCS II was 22,793 n.m. altitude with 2.484 degrees inclination. The IUS then reoriented, increased its velocity by 5 feet per second, oriented to the DSCS III release attitude and separated DSCS III as planned at 22,804 n.m. altitude and 2.502 degrees inclination. The target orbit for DSCS III was 22,817 n.m. altitude with 2.483 degrees inclination.

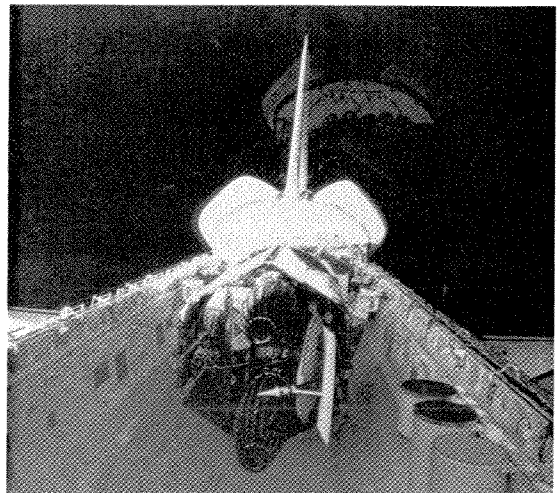
The final orbit parameters indicated a near "bull's-eye." The DSCS II/III spacecraft deployment attitude was termed "perfect." Boeing received the maximum incentive award available for the success of the DSCS II/III mission. For more data on the accuracy of this mission, see Chapter 4, Table 4-3.

### Second Launch

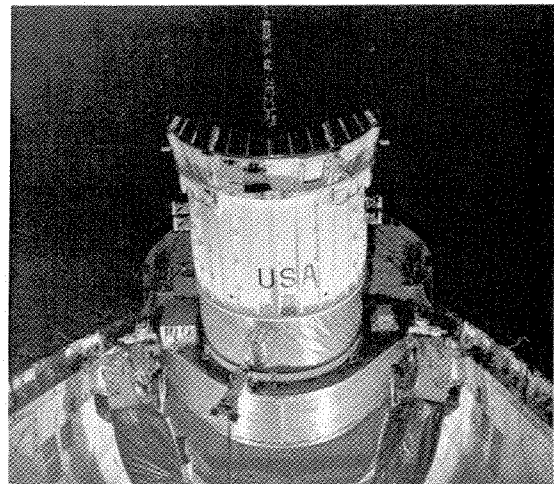
A second IUS launch occurred on April 4, 1983 from the Space Shuttle *Challenger* (STS-6). This milestone launch marked four firsts: it was the *first* launch of a Tracking and Data Relay Satellite (TDRS-A) in the satellite system of the same name; it marked the *first* flight of the orbiter *Challenger* (OV-099); it was the *first* use of the IUS-unique airborne support equipment; and it was the *first* time the IUS had been flown aboard the Space Shuttle. It is well to note at this time that these "firsts" are vital to the long term viability of the space transportation system.

Although the second mission was not the success that the first mission was, it did prove that the design decisions discussed in this Guide are correct.

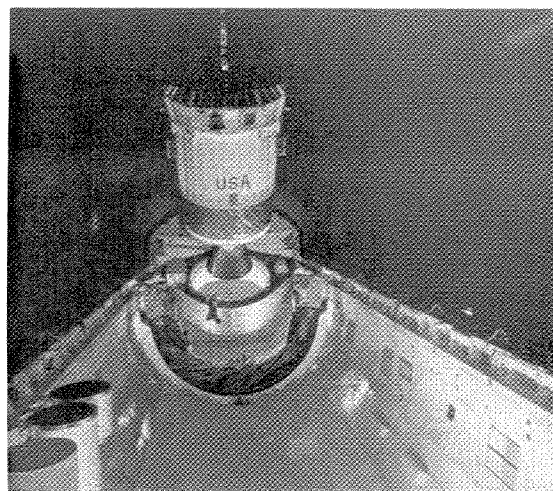
The deployment from the orbiter was clean and smooth. Following deployment, the IUS completed a thermal roll to control TDRS temperature, and then oriented itself to fire its first stage rocket to inject the combined payload into transfer orbit. During the next six hours, the IUS completed additional thermal rolls and other maneuvers including five reorientations to direct TDRS telemetry



*IUS and TDRS in Shuttle Payload Bay  
Prior to Launch*



*IUS in Deployment Position in ASE*



*IUS Released from ASE*

*Deployment Sequence of IUS-1 from the  
Challenger's Cargo Bay*



antennas toward Earth before approaching the geosynchronous orbit altitude. All of this occurred as programmed in the data load with the IUS-unique software.

At 83 seconds into the 107-second burn of the second stage solid rocket motor, the IUS lost its control stability. At the 83rd second mark, the IUS had provided 96% of the total transfer energy required to place TDRS in its desired geosynchronous orbit.

The telemetry data, combined with laboratory simulations, has led to the conclusion that loss of IUS control was caused by an uncommanded change in the position of the SRM-2 nozzle which forced the motor thrust vector to be pointed away from the vehicle's center-of-gravity. This movement could not be corrected by the thrust vector control actuators which control the nozzle's position.

The cause identified for the nozzle movement was the collapse of the Techroll seal (TRS). The Techroll seal is a flexible rubber and fabric toroid filled with silicone oil and located in the motor nozzle. Its action allows the nozzle to rotate.

With the collapse of the TRS, the nozzle assembly rapidly moved aft. The thrust vector control actuators could not extend fast enough to compensate for the nozzle's aft motion, and as a consequence, the pitch actuator rotated about its attachment and was driven into a bracket mounted on the motor dome. This induced a bending moment in the actuator which froze its shaft. As a result, the nozzle was offset approximately 1.4 degrees from alignment with the vehicle's center-of-gravity.

The resultant uncontrolled tumble of the IUS and TDRS caused the telemetry antenna to sweep across the earth once every two seconds, rather than point constantly at the ground stations. Consequently the ground stations were unable to display telemetry data to IUS flight controllers. After motor burnout, the centrifugal forces from the tumble caused the IUS accelerometers to react as if the motor were still burning. This prevented the IUS from performing the

scheduled separation of the IUS and the TDRS.

In order to separate the TDRS satellite, IUS flight controllers generated and transmitted the required separation commands. Within seconds of issuing this real-time command, the IUS released the TDRS spacecraft in an elliptical orbit lower than the desired 22,236-mile circular orbit.

After separation, TDRS circularized its orbit by using its own propulsive capability. A series of 39 TDRS thruster firings moved the spacecraft to its desired geosynchronous orbit of 22,236 statute miles. Its remaining propellant supply is approximately twice the amount needed for the 10-year long TDRS mission. In fact, TDRS is already operational and was used to support the STS-8 mission. TDRS will also support the STS-9 spacelab mission after it is moved to its permanent location at 41°W longitude.

Efforts to identify the specific cause of the loss of control continues. Motor firings are planned to revalidate the thermal models and to test new design enhancements. All SRM-2 motor nozzles are being rebuilt to incorporate the design enhancements and will undergo an intensified inspection process.

All other motor hardware is undergoing reinspection as a further precaution. Launch of the next IUS is anticipated in 1984.

Boeing is dedicated to correcting the problem encountered on the IUS-1 mission, so that future IUS flights will provide flawless performance as an integral part of the United States Space Transportation System.

### **Current Activities**

Besides the activities that are now underway to resolve the anomaly for a 1984 launch, spacecraft integration activities are currently being conducted for various spacecraft. These are in various developmental stages from supporting preliminary design to the generation of final mission data loads. Preliminary evaluations have also been conducted for other prospective users.

Preliminary designs have been accomplished for twin-stage and three-stage configurations. Studies of on-orbit storage and on-orbit assembly have also been conducted. These designs and study results are available for mission-unique requirements.

Preliminary design activities are now underway for four derivative vehicles. The first is a performance improvement option; the second is a reduced cost option which is achieved by deleting some redundant elements while still retaining a high-reliability system; the third is a three-axis-controlled perigee stage design using the large IUS motor and the fourth is a two-stage, integrated propulsion module. Additionally the IUS has been structurally tested and has demonstrated heavy payload carrying capability. (See Chapter 6, IUS Derivatives)

IUS and its derivatives are planned to provide an upper stage capability for the Air Force, NASA, and commercial users for the 1990s.

### **IUS PROGRAM ORGANIZATION**

The United States Government IUS management team includes both Air Force and NASA

personnel. IUS project offices are located at the Air Force Systems Command's Space Division and NASA's Marshall Space Flight Center (MSFC). Contracting responsibility rests with the Space Division.

Space Division's Inertial Upper Stage Program Office, Los Angeles is the program office responsible for the development, procurement, operation, and spacecraft integration of the IUS.

Initial spacecraft program requests for IUS integration planning should be submitted to Space Division, Inertial Upper Stage Program Office, or NASA's Marshall Space Flight Center. Integration activities should begin approximately four years before launch.

The Boeing Aerospace Company is the prime contractor for design, integration, checkout and launch of the IUS. Much of this work is done at the Kent Space Center, south of Seattle, Washington. The IUS program/Boeing Aerospace Company/Air Force customer relationship is represented in Figure 1-1. Customer relationships for other government agencies and commercial users are shown in Figure 1-2.



*IUS Vehicle and Airborne Support  
Equipment Assemblies in Kent Facility  
Manufacturing Area*

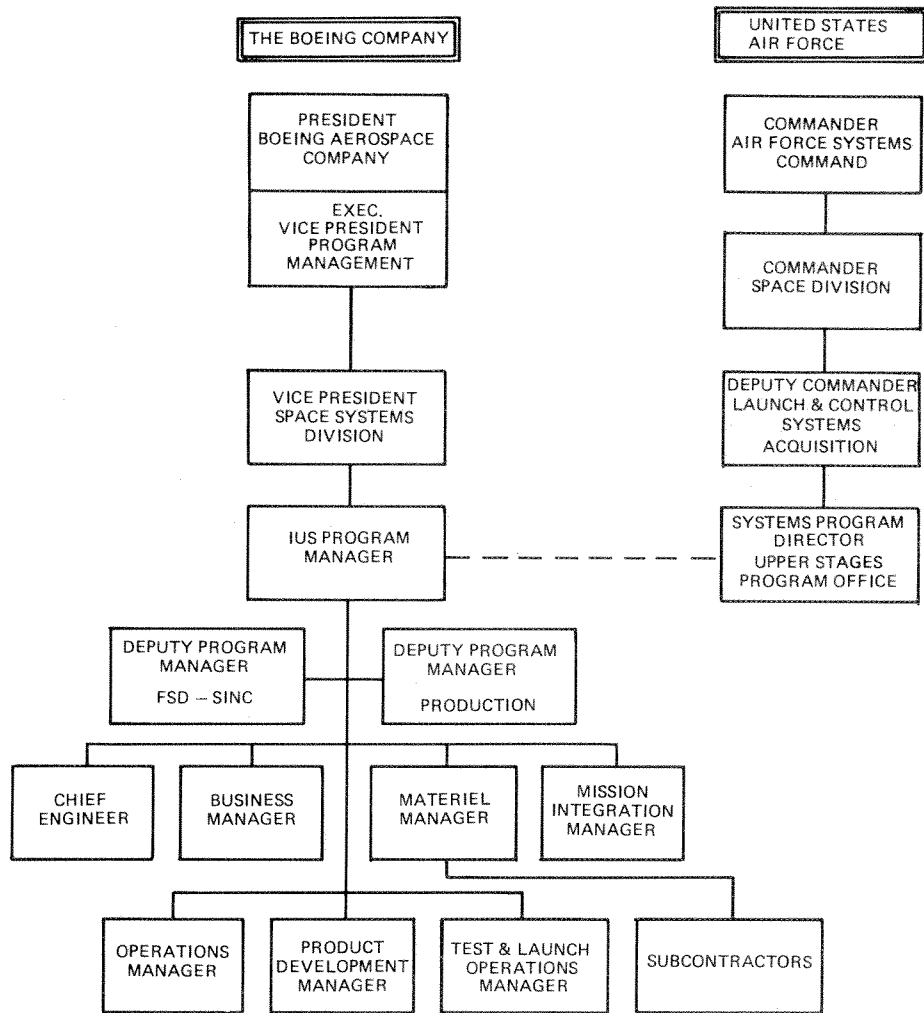


Figure 1-1: IUS Program, Boeing Aerospace Company, and USAF Customer Relationship

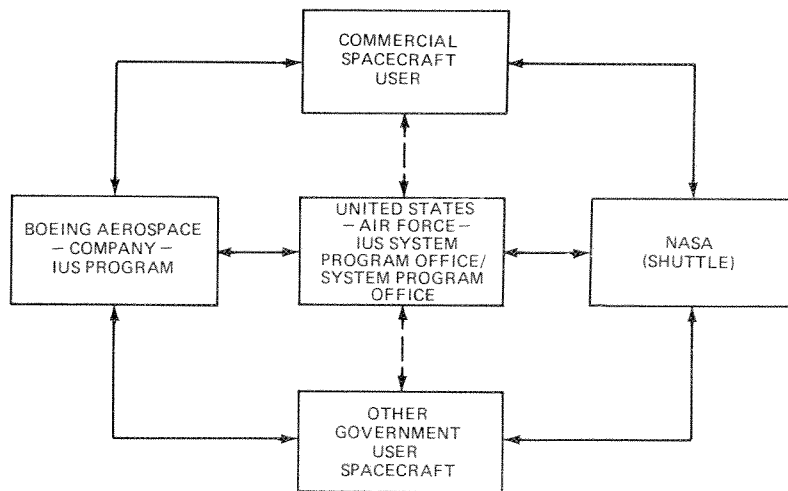
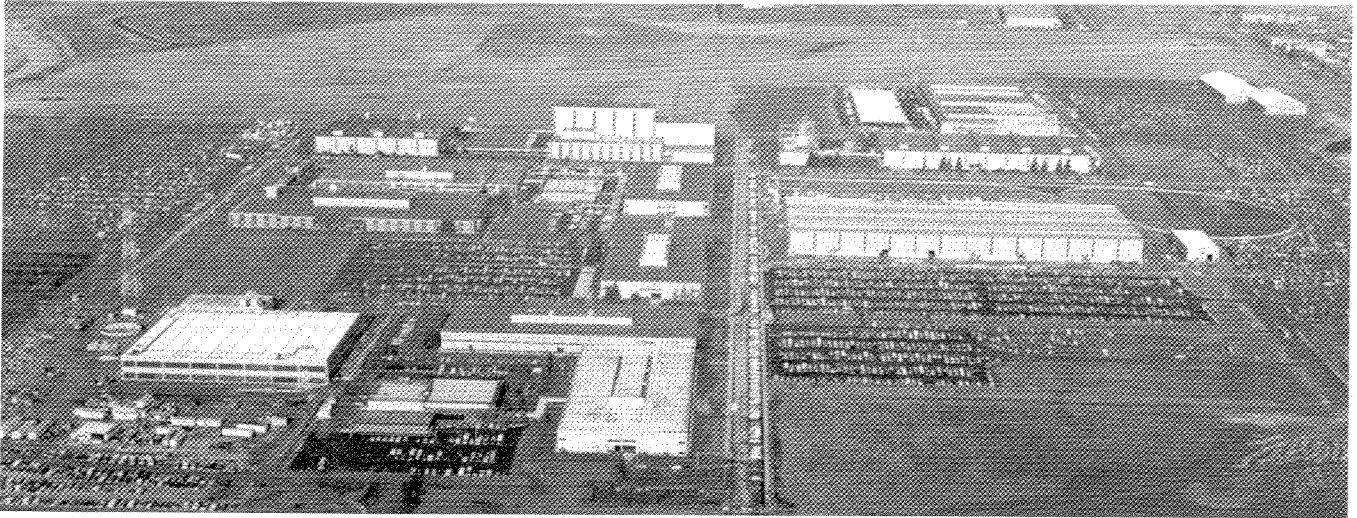


Figure 1-2: IUS Program, Boeing Aerospace Company, and All Customer Relationships



*Aerial View of the Boeing Aerospace Company,  
Kent Facility, Kent, Washington*

Subcontractors on the IUS Program are Chemical Systems Division of United Technologies (solid rocket motors and thrust vector control servos), Hamilton Standard (reaction control engine module and inertial measurement unit), DELCO (computer), Ball Aerospace (star scanner), and TRW (space ground link transponder, converter regulator

unit, and communications interface unit).

Within a year, the IUS Program will complete its full-scale engineering development phase which produced nine flight vehicles. The program will then proceed into a full production and operations phase.



## CHAPTER II-IUS SYSTEM SUMMARY

The IUS is a two-stage, solid-rocket-propelled, three-axis-controlled, inertially navigated upper stage that can deliver a spacecraft of approximately 5,000 pounds (2,268 kg) from the Shuttle parking orbit to a geosynchronous orbit. The generic IUS vehicle can be configured to be used on the Shuttle or the Titan 34D boost vehicles. The basic generic vehicle includes: stage structure; solid rocket motors and the reaction control subsystem for propulsion; avionics for telemetry, tracking and command, guidance, navigation and control, data management, and thrust vector control; electrical power sources and electrical cabling; destruct hard-

ware; and airborne software. The IUS vehicle also provides the interface between the spacecraft and the launch vehicle.

The generic vehicle as shown in Figure 2-1 consists of the major components listed in Table 2-1. This table and Figure 2-2 show the differences between the two generic configurations.

Spacecraft Services, Chapters III and IV, discuss production options which are available to revise the IUS's generic configuration and/or capabilities to meet mission-specific requirements.

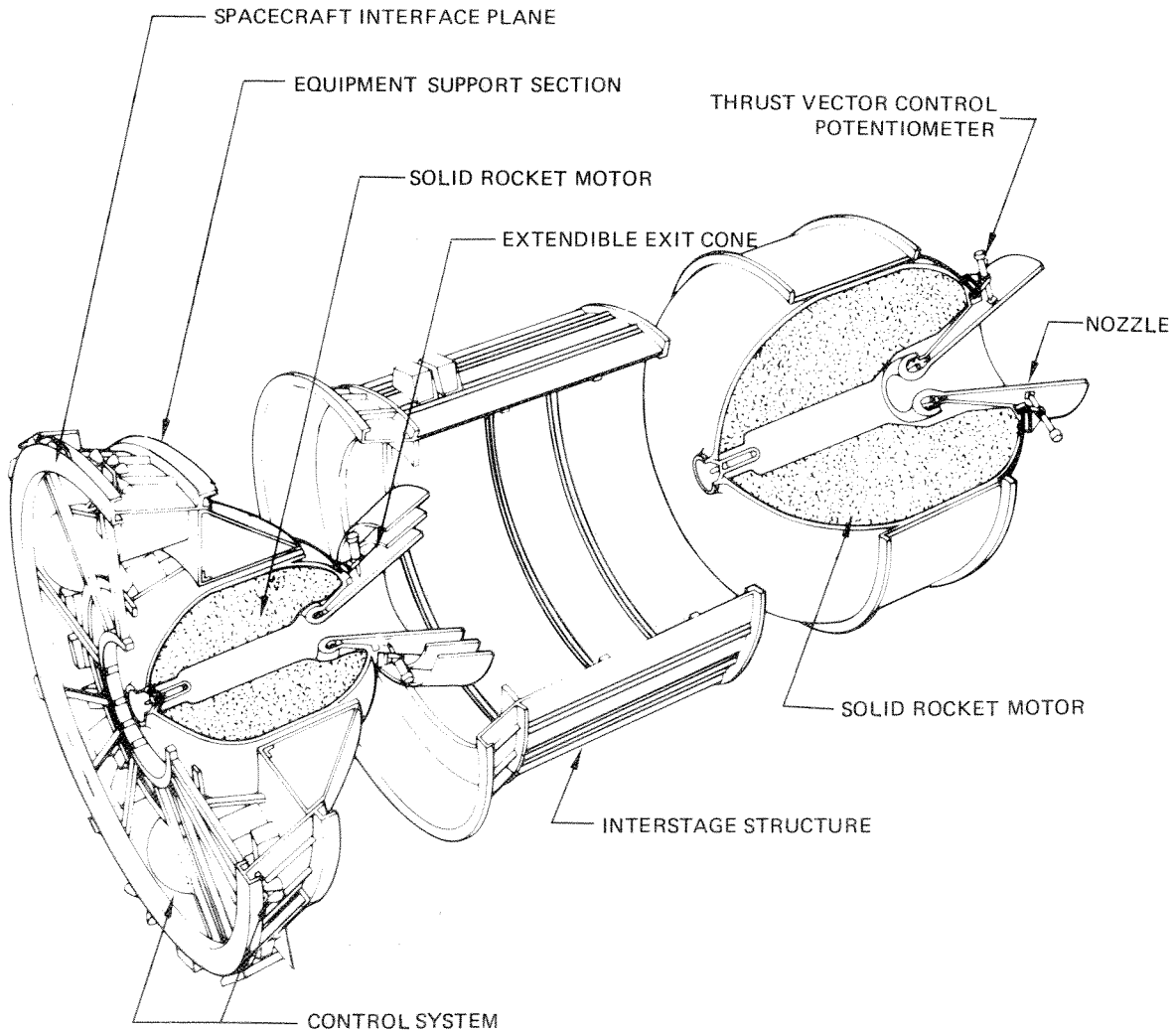
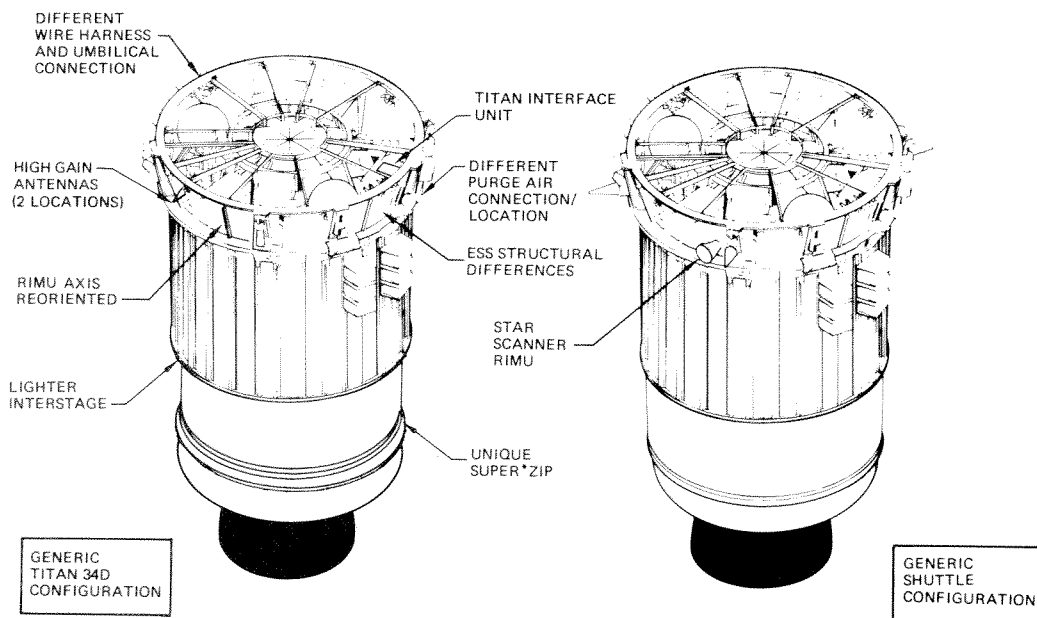


Figure 2-1: Inertial Upper Stage

Table 2-1: Generic IUS Vehicle: Major Components List

Shuttle Qty	T34D Qty	Nomenclature	Shuttle Qty	T34D Qty	Nomenclature
1	1	Solid Rocket Motor - 1 (Large Motor)	0	1	Two-Stage T34D IUS Vehicle Structural Subsystem
1	1	Solid Rocket Motor - 2 (Small Motor)	1	0	Two-Stage STS/IUS Vehicle Structural Subsystem
4	6	Ignition Safe and Arm Device	1 set	1 set	Separation Devices
1 set	1 set	Thrust Vector Control Servo System	0	1 set	Separation Devices
1 set	1 set	RCS Rocket Engine Module	2	2	Reaction Control System Propellant Tank
1	1	SGLS Transponder, S-Band	1	1	Reaction Control Subsystem Manifold
1	1	20-watt Power Amplifier	0	1 set	IUS Destruct Device
1	1	Signal Interface Unit	2	0	Antenna, S-Band, Omni
8	4	Avionics Battery	1	2	Antenna, S-Band, Medium Gain
5	4	Utility Battery	1 set	1 set	Cabling
2	2	Power Distribution Unit	0	2	Destruct Battery
1	1	Pyro Switching Unit	1 set	1 set	RF System Components
1	0	Power Transfer Unit	1	1	Operational Flight Software
2	2	Computer Unit	1 set	1 set	Thermal Control Subsystem
1	1	Redundant Inertial Measurement Unit	0	1 set	Environmental Measurement Unit
2	2	Signal Conditioner Unit	1	0	Encryptor
1	1	SCU A Code Plug	1	0	Decryptor
1	1	SCU B Code Plug			
1	0	Star Scanner			
0	2	Titan Interface Unit			



- Requires flight termination system
- No spacecraft batteries or power transfer unit

Figure 2-2: Generic IUS Differences: Shuttle Configuration vs. T34D Configuration

## IUS/SHUTTLE/SPACECRAFT PHYSICAL INTERFACES

Significant IUS/Shuttle/spacecraft physical interfaces are the geometric envelope for the IUS/spacecraft combination, the envelope's center-of-gravity location, and the spacecraft's mechanical and electrical connections to the IUS.

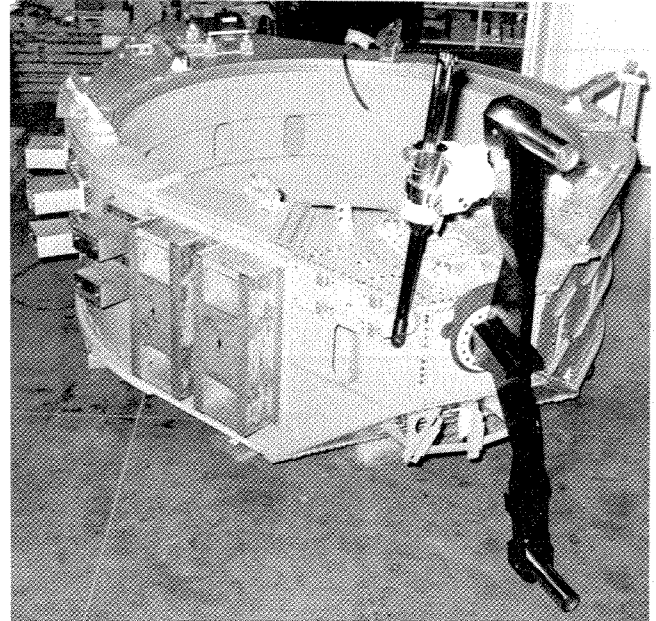
### IUS/Shuttle Interfaces

The Shuttle installation is unique in that special airborne support equipment (ASE) is used to support the IUS in and deploy the IUS from the Shuttle cargo bay.

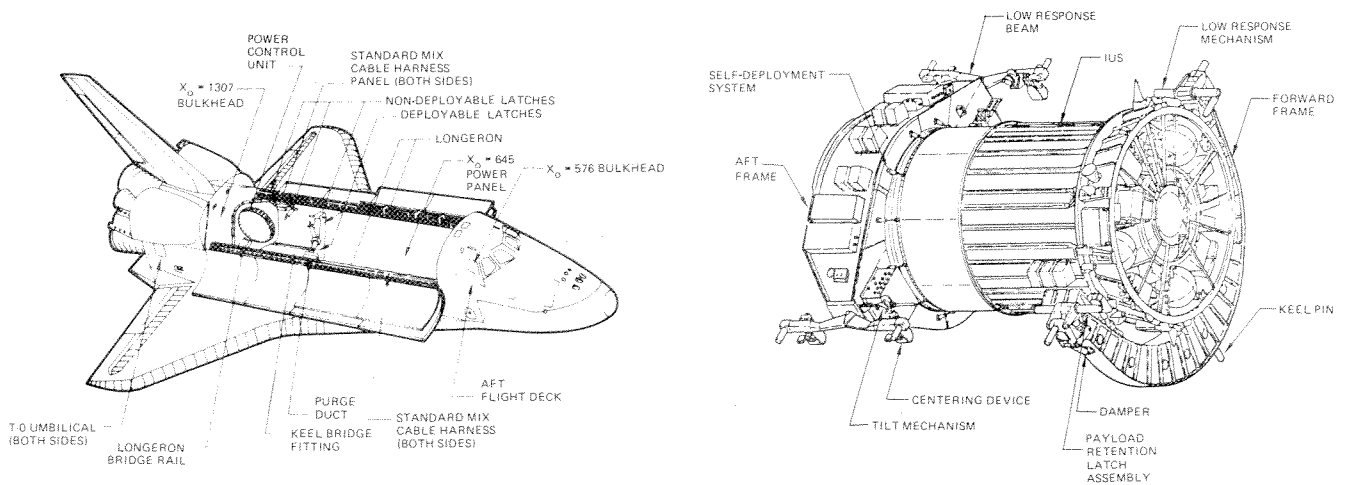
Figure 2-3 shows the ASE and its physical interfaces and attach points in the orbiter. The structural interfaces in the payload bay consist of standard orbiter nondeployable attach fittings on each longeron which mate with the ASE trunnions and the active attach fitting at the keel which mates with the ASE keel trunnion. Ducting from the orbiter purge spigot interfaces with the IUS at the forward ASE.

The IUS ASE consists of the forward and aft frames and incorporates a low response spreader beam and a torsion-bar mechanism that reduce the normal spacecraft dynamic loads by one-third at the center-of-gravity. Greater reductions are realized for the

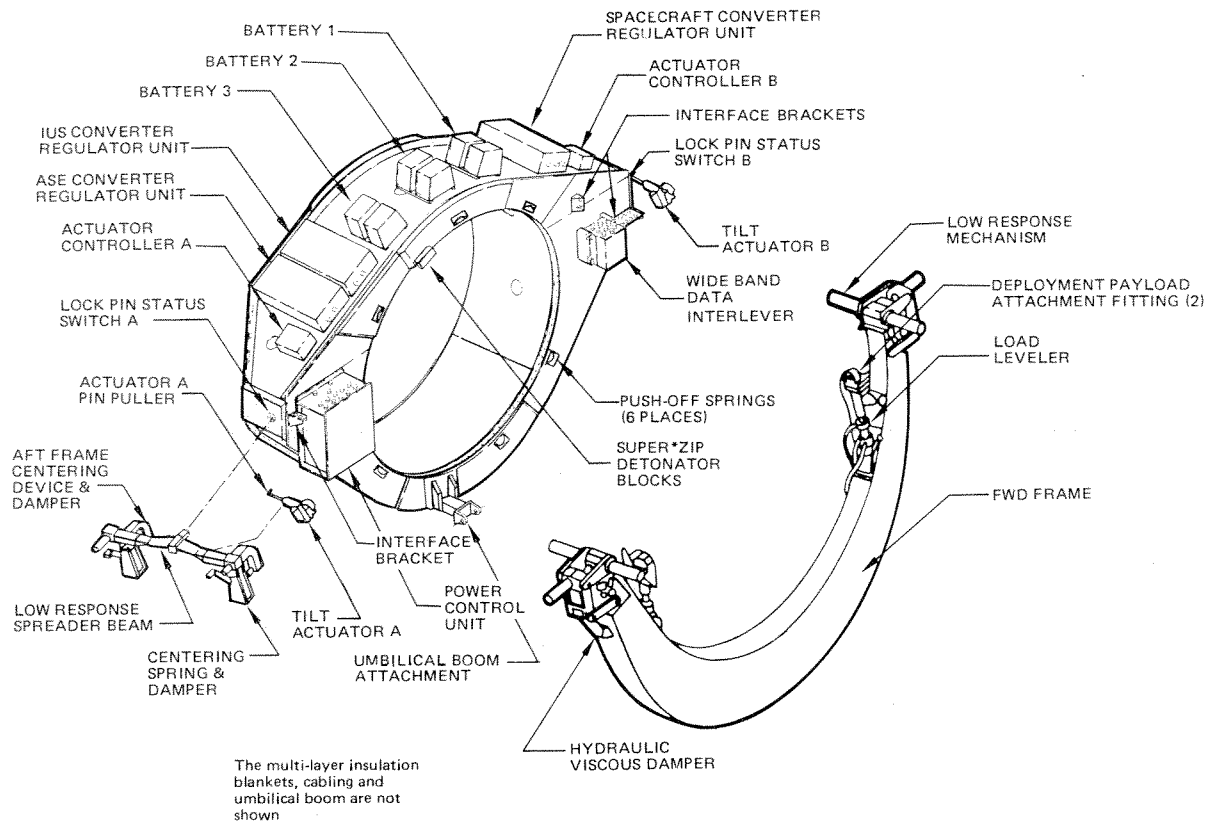
appendage responses. In addition, the forward ASE frame includes a hydraulic load leveler system to balance loading at the forward trunnion fittings. The aft frame interfaces with the IUS solid rocket motor through the Super\*Zip separation system. After firing the Super\*Zip, the IUS vehicle is ejected from the aft frame by six separation springs. (See Appendix A, ASE Structural and Mechanical Design Data).



*Low Response Spreader Beam and Aft Tilt Frame Actuator Attached to the Aft ASE*



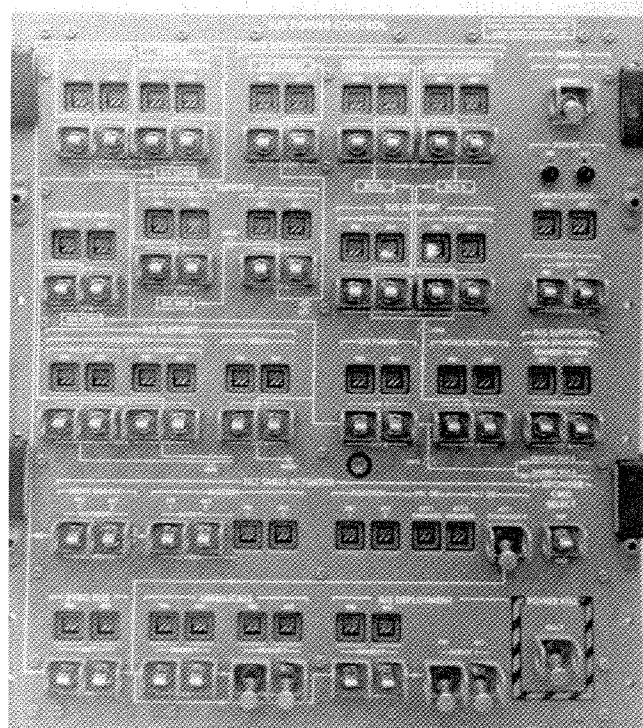
*Figure 2-3: IUS/Orbiter Physical Interfaces and Airborne Support Equipment*



**Figure 2-4: Airborne Support Equipment Component Locations**

The ASE includes equipment (Figure 2-4) which provides and/or distributes and controls electrical power to the IUS/spacecraft, and provides and/or distributes and controls communication paths between the IUS and/or spacecraft and the orbiter. Communications and power control equipment are mounted on the aft flight deck at the payload station and are operated in flight by the payload/mission specialist. Electrical power and signal interfaces to the orbiter are located at the IUS equipment connectors. Cabling to orbiter equipment and between separated IUS elements (IUS or payload bay to aft flight deck) are provided by the orbiter.

The ASE data subsystem provides for the transfer of data and commands between the IUS/spacecraft and the appropriate orbiter interfaces. Telemetry data includes spacecraft data received over dedicated circuits via the vehicle and spacecraft telemetry streams. An interleaved stream is provided to the orbiter for transmission to the ground or for transfer to ground support equipment.



**Airborne Support Equipment Power Panel**

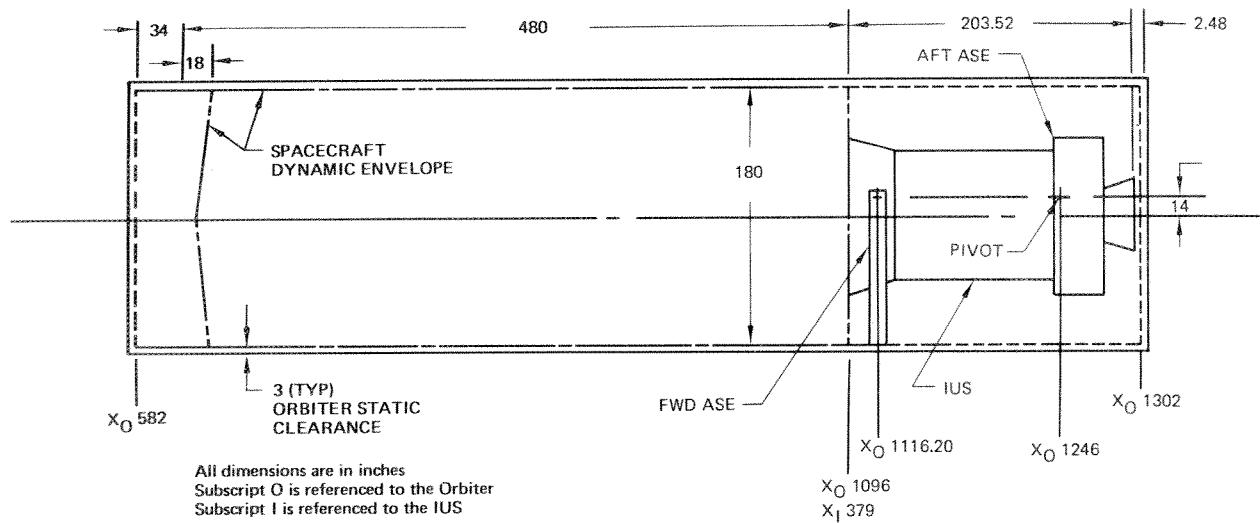


Figure 2-5: Spacecraft Volumetric Envelope

### Allowable Spacecraft Envelope

Figure 2-5 represents the allowable spacecraft envelope in the Shuttle when the IUS is installed in its ASE. The IUS/spacecraft thermal and dynamic envelope is a cone-ended cylinder 15-feet in diameter and 686-inches long on the centerline. The forward end of the envelope is a cone (nominal 78.69° half angle). Its apex is forward, sloping back 18 inches at the 15-foot diameter. The envelope contains all thermal, dynamic and static deflections and clearances for the IUS/spacecraft/ASE combination (except for the orbiter attachment fittings) from the time of installation in, to deployment from the orbiter.

### IUS Structural Capability

The IUS is capable of supporting various spacecraft weights at specific distances above the spacecraft interface plane. Figure 2-6 shows the allowable center-of-gravity locations as a function of spacecraft weight based upon test data and the extrapolation of parametric loads analysis.

### Spacecraft Center-of-Gravity Envelope

The allowable spacecraft center-of-gravity envelope is based on the assumption that the IUS and spacecraft center-of-gravity lateral offsets are generally in the same direction.

The spacecraft allowance can be increased in the specific interface control document (ICD) if the lateral center-of-gravity offsets for the IUS vehicle and the spacecraft are opposed.

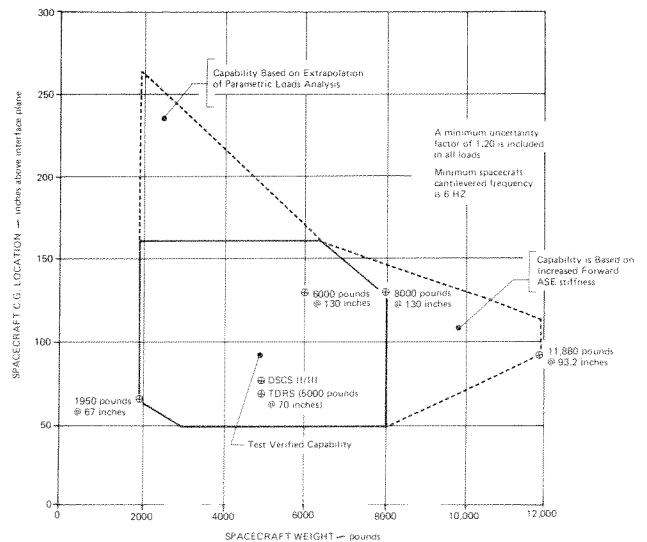
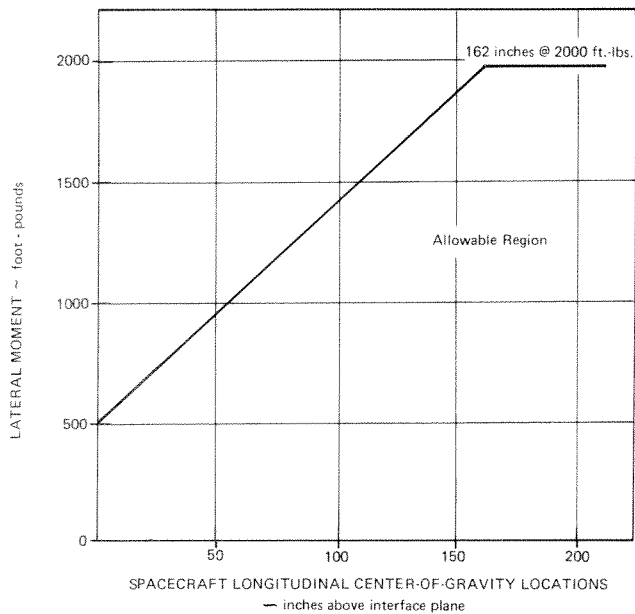


Figure 2-6: IUS Structural Capability - Shuttle Configuration

The spacecraft lateral center-of-gravity offset, combined with a longitudinal acceleration of 1.0g, shall not produce a moment greater than that shown in Figure 2-7 about any axis through the IUS geometric centerline in the plane of the IUS/spacecraft interface. In no case, however, shall the spacecraft lateral center-of-gravity offset exceed eight inches.



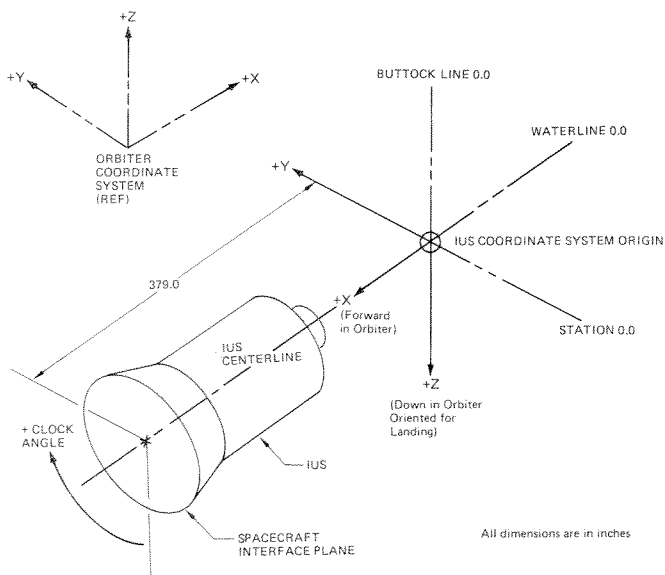
**Figure 2-7: Spacecraft Center-of-Gravity Envelope**

### Coordinate Systems and Clocking

The relationship between the IUS and ASE is shown in Figure 2-8. The spacecraft coordinate system is defined in each specific ICD. The clocking of the spacecraft attachment points is shown in Figure 2-9.

### Mechanical Attachment

Physical attachment of the spacecraft to the IUS is provided by eight attachment points.



**Figure 2-8: IUS Vehicle/ASE Coordinate System**

This configuration was originally selected to correspond to the Titan/Transtage spacecraft attachment concept. It provides substantial load-carrying capability while minimizing thermal conduction across the interface. Details of the fitting at each attach location are illustrated in Figure 2-9.

The spacecraft separation mechanism is provided by the spacecraft. The separation plane will typically be forward of the IUS/-spacecraft attachment interface, and the spacecraft adapter will remain with the IUS after separation occurs.

The IUS structure is capable of transmitting all internal loads generated by a cantilevered spacecraft during orbiter operations and IUS free flight. The IUS is designed to accommodate the limit loads imposed by the spacecraft weights, center-of-gravity locations and load factors shown in Table 2-2.

This capability depends upon a cylindrical spacecraft adapter that distributes loads in a manner that makes all eight IUS/spacecraft attachment points fully effective. Interface capability will be reduced by use of conical-type adapters that induce radial loads into the IUS/spacecraft interface ring or by adapters that will not effectively attach to all eight interface attachment points. The Boeing Aerospace Company should be contacted, when these conditions occur.

### Electrical Connectors

The IUS provides for several interface connectors which accommodate the transmission of power and data services. The physical location of these interface connectors is shown in Figure 2-10. Access to these can either be provided on the spacecraft side of the interface plane, or through the access door in the IUS equipment bay (Figure 2-10).

Figure 2-11 illustrates the IUS system functional interfaces available to the spacecraft for distribution of the electrical power, ordnance power, command discrettes, telemetry data, spacecraft dedicated circuits, and radio frequency signals.

All dimensions are in inches.

A master gage will be used to establish the interface hole locations (GFE)

The spacecraft to IUS fasteners are provided by the spacecraft contractor.

The shear pins are provided by the IUS contractor.

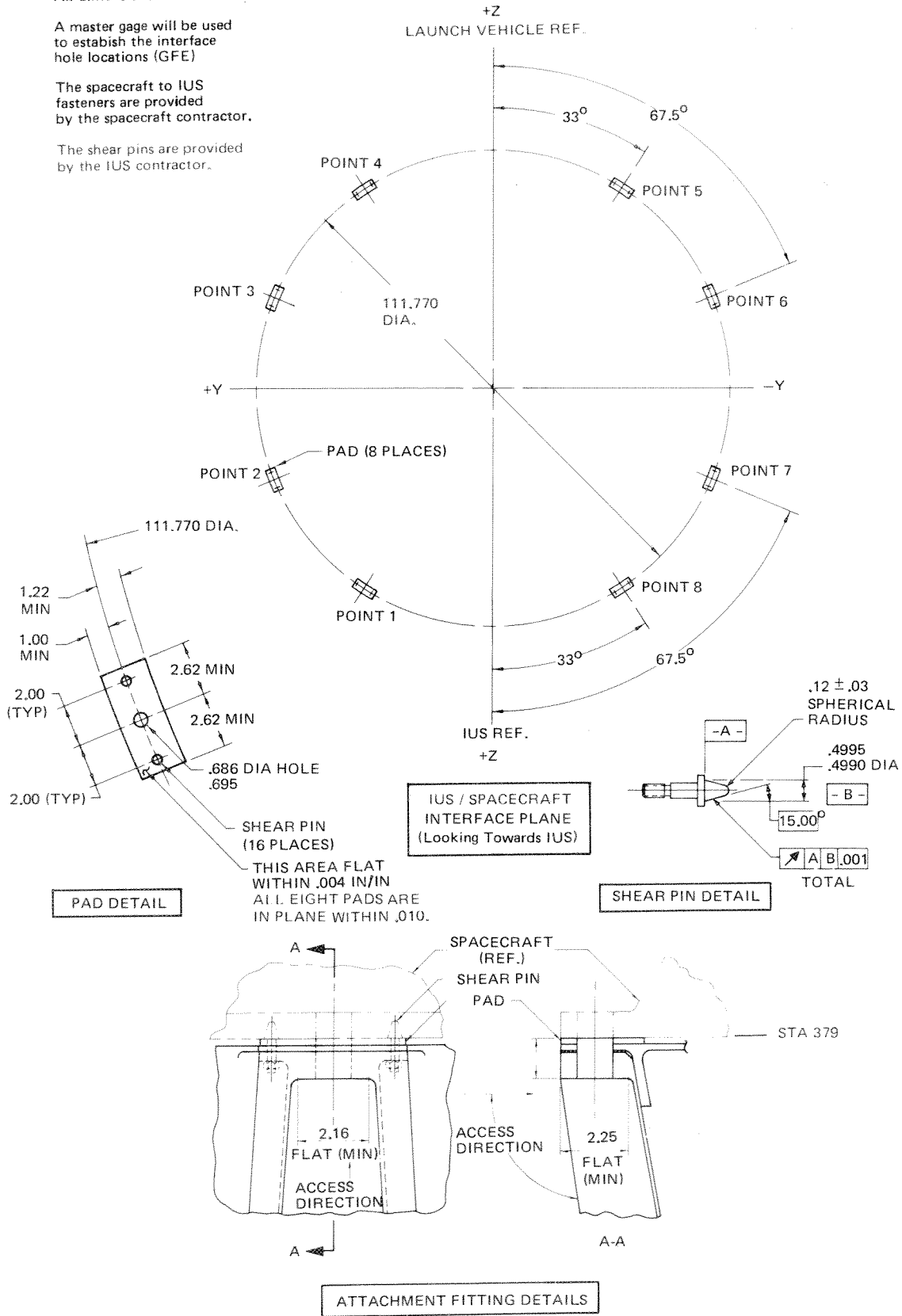




Figure 2-9: IUS/Spacecraft Mechanical Attachment

Table 2-2: Design Limit Load Factors for the Spacecraft

EVENT	AXIS 	STEADY STATE	SPACECRAFT WEIGHT & CENTER-OF-GRAVITY			
			1950 lb. @ 67"	6000 lb. @ 130" 5000 lb. @ 162"	5000 lb. @ 70"	
			SPACECRAFT DYNAMIC LOAD FACTORS (G)			
LIFTOFF	X	-1.5	± 3.5	± 3.5	± 3.5	
	Y	-	± 2.4	± 1.2	± 2.4	
	Z	-	± 3.4	± 2.0	± 3.0	
LANDING	PITCH MAX	X	-	± 1.0	± 1.0	
		Y	-	± 1.0	± 1.0	
		Z	+1.0	± 4.0	± 2.0	± 3.5
	YAW MAX	X	-	± 0.9	± 0.9	± 0.9
		Y	-	± 2.0	± 2.0	± 2.0
		Z	+1.0	± 2.0	± 1.4	± 1.5

 Orbiter coordinate system

### Thermal Considerations

The IUS/spacecraft interface is designed to be relatively adiabatic. Physical contact occurs at the eight attachment points and at the 11 electrical interface connectors (See Figures 2-9 and 2-10). Appropriate contact conductance is applied at the specific contact locations. The IUS structural temperatures at these locations can vary between

120°F and -50°F when spacecraft heating effects are neglected. The 120°F temperature extreme is a combination of sun, earth and albedo environments on the IUS avionics bay. The 120° temperature occurs when the sun is in the normal-to-hot location with high heat dissipating from the IUS equipment in the vicinity. The -50°F temperature extreme represents a lack of external environments without any heat dissipating from

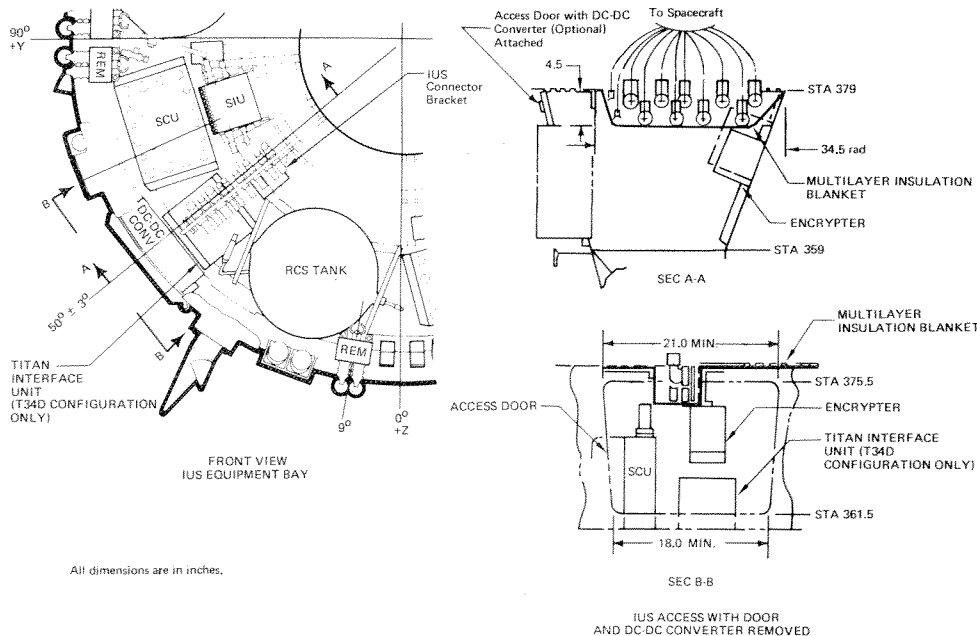


Figure 2-10: IUS/Spacecraft Electrical Connector Locations



the IUS equipment in the attach point vicinity. Specific temperature histories are mission-dependent and are provided in specific ICD's. However, the spacecraft adapter temperature at the attachment locations will not exceed 100°F when IUS heating effects are neglected.

The IUS/spacecraft interface has a multi-layer insulation blanket. The blanket consists of an outer layer of aluminized Beta cloth and three layers of aluminized Kapton which are separated by polyester net spacers, one layer of polyester filter cloth and one inner cover of aluminized Kapton.

Thermal blankets are grounded to the IUS vehicle structure, and they are vented into the IUS cavity. The gases within the IUS cavity are then vented into the orbiter payload bay. The only gas flow between the

spacecraft and the IUS is through the electrical connectors.

Radiation is the heat transfer mode across the area enclosed by the 111.77-inch diameter bolt circle. On the spacecraft side, the radiation properties are: for the hemispherical emittance ( $\epsilon_{ir}$ ),  $0.85 \pm 0.05$ ; for the solar absorptance coefficient ( $\alpha_s$ ),  $0.40 \pm 0.05$ ; for the solar reflectance, they are diffuse. The insulation blanket should be treated as two radiation shields. The effective emittance ( $\epsilon^*$ ) between the shields is approximately 0.02. The IUS side temperatures are 105°F for the hot case and 35°F for the cold case. These thermal interface requirements are depicted on Figure 2-12.

A thermal analysis is required for each spacecraft/IUS interface so that the verification analysis can be supported, and accept-

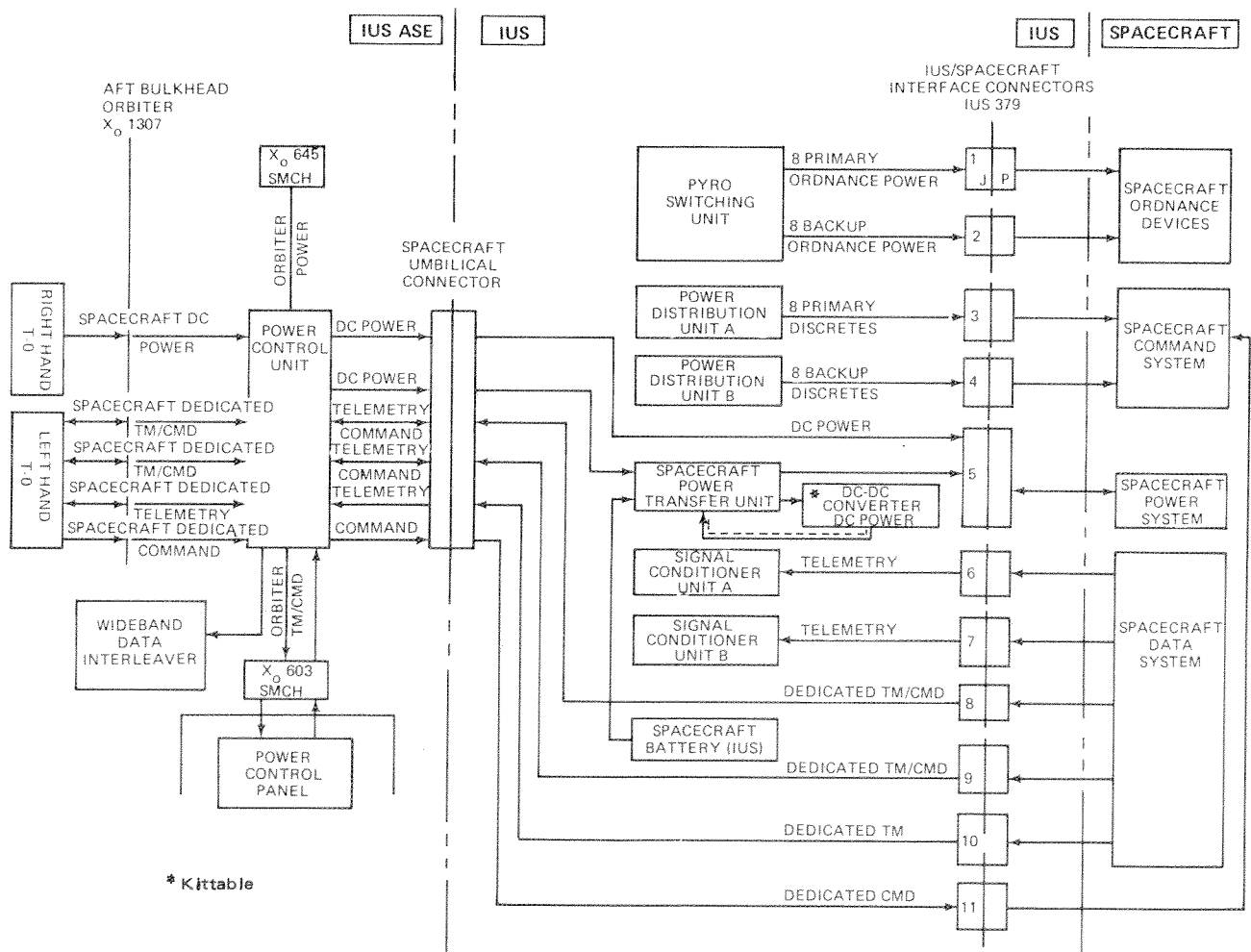


Figure 2-11: IUS/Spacecraft Electrical Interfaces

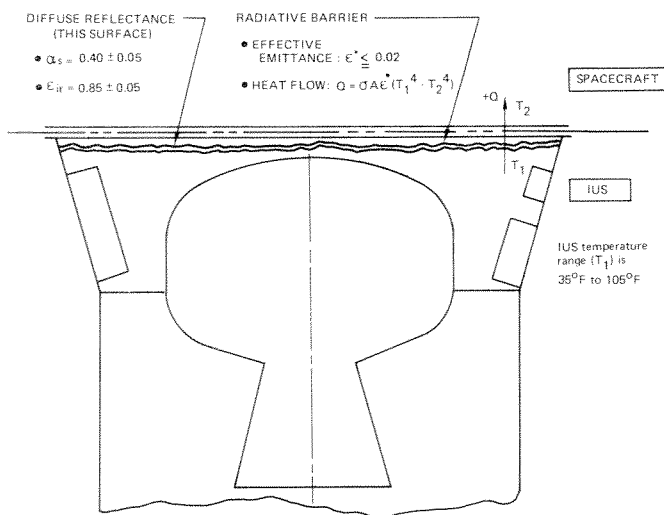


Figure 2-12: IUS/Spacecraft Thermal Interfaces

able temperature conditions on each side of the interface can be ensured. The spacecraft models will be identified in the specific ICD. The IUS thermal analytical models shall be as authorized by the IUS System Program Office (SPO). The current model is described in Boeing Document D290-10953-1 "IUS/TDRS-A STS-6 First Flight Thermal Model Report".

NASA's Johnson Space Center authorizes any orbiter thermal math models. The current models are described in Document ES3-77-3 "390 Node Atmospheric Orbiter Midsection/Payload Bay Thermal Math Model Description" and ES3-76-1, Rev D, "Orbiter Midsection/Payload Bay Thermal Math Model Description (Model No. SSX001D)".

## IUS/T34D/SPACECRAFT PHYSICAL INTERFACES

The significant IUS/T34D/spacecraft physical interfaces are the geometric envelope for the IUS/spacecraft combination, the envelope's center-of-gravity location, and the spacecraft's mechanical and electrical connections to the IUS.

### Allowable Spacecraft Envelope

Figure 2-13 illustrates the envelope which is available to the spacecraft in the Titan 34D payload fairing. This envelope contains all thermal, dynamic and static deflections and

clearances for the IUS/spacecraft combination.

### IUS Structural Capability

The IUS is capable of supporting various spacecraft weights at specific distances above the spacecraft interface plane. Figure 2-14 shows the allowable center-of-gravity locations as a function of spacecraft weight based upon tests.

### Spacecraft Center-of-Gravity Envelope

Same as Shuttle configuration.

### Coordinate Systems and Clocking

The relationship of coordinates between the IUS and T34D is shown in Figure 2-15. The spacecraft coordinate system is defined in each specific ICD. The clocking of the spacecraft attachment points is shown in Figure 2-9.

### Mechanical Attachment

Same as Shuttle configuration.

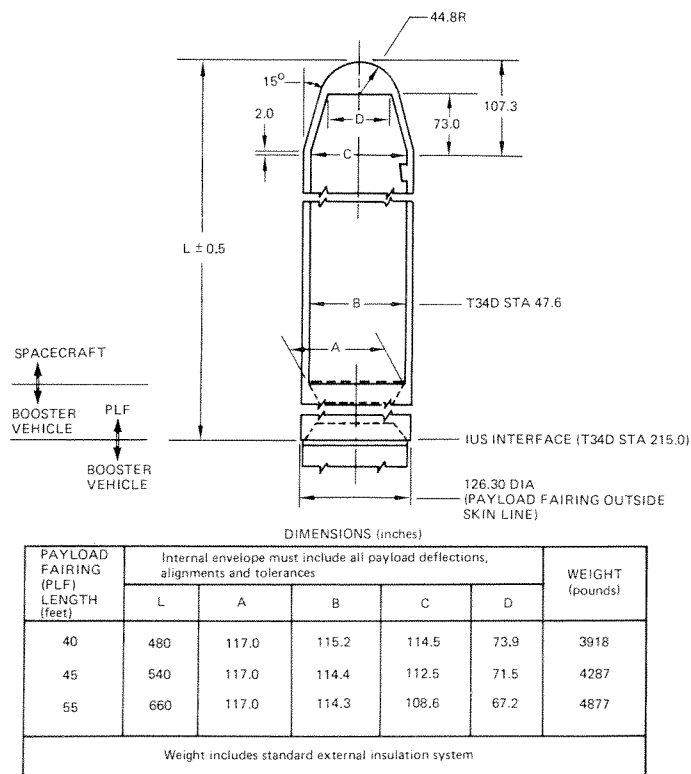


Figure 2-13: Spacecraft Volumetric Envelope

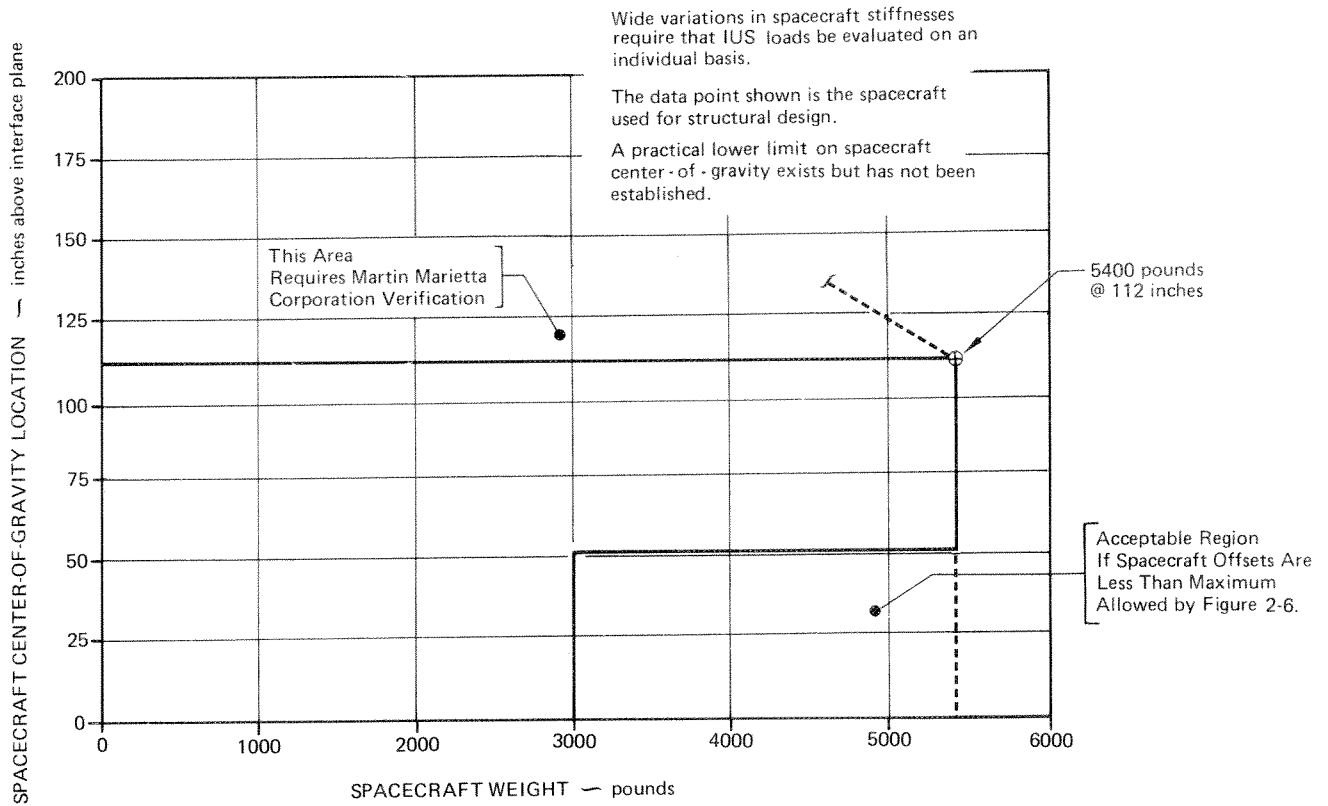


Figure 2-14: IUS Structural Capability - T34D Configuration

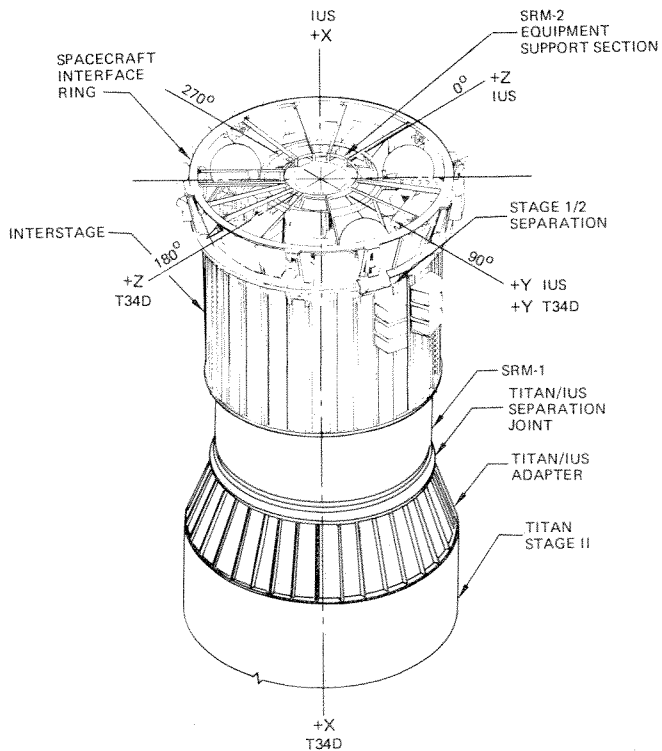


Figure 2-15: T34D;IUS Vehicle Coordinate Systems

### Electrical Connectors

The IUS provides for several interface connectors which accommodate the transmission of power and data services. The physical location of these interface connectors is shown in Figure 2-10. Access to these connectors can either be provided on the spacecraft side of the interface plane, or through the access door in the IUS equipment bay (see Figure 2-10).

Figure 2-16 illustrates the IUS system functional interfaces available to the spacecraft for distribution of the electrical power, ordnance power, command discretes, telemetry data, spacecraft-dedicated circuits, and radio frequency signals.

### Thermal Considerations

Same as Shuttle configuration.

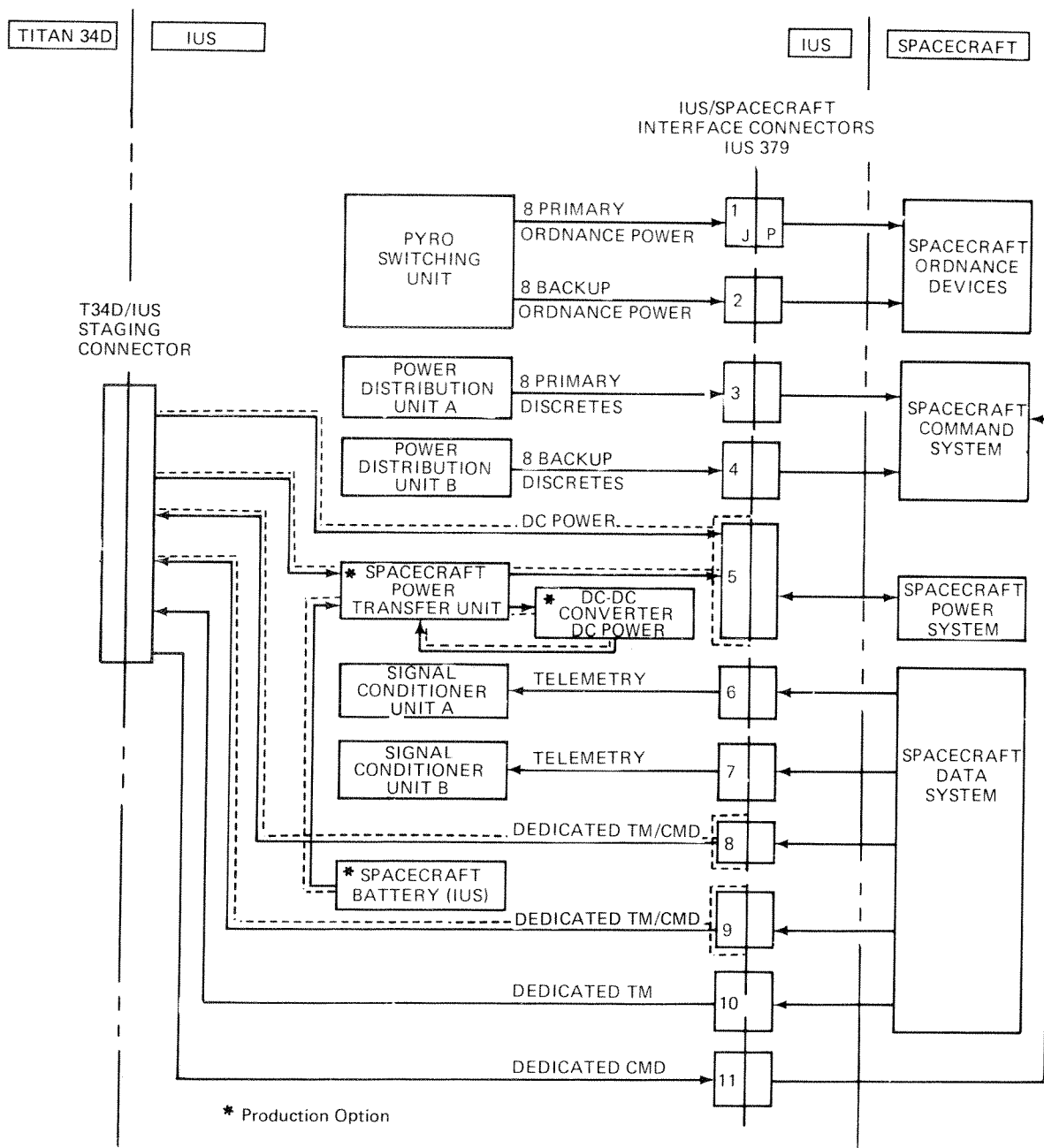


Figure 2-16: IUS/Spacecraft Electrical Interfaces

### STRUCTURES, MECHANISMS AND THERMAL CONTROL

The structure supports the spacecraft, all the equipment and solid rocket motors in the IUS and provides the mechanisms for stage operation. The major structural assemblies of the two-stage IUS are the equipment support section (ESS), and the interstage structure.

The ESS is the second stage structure. It houses the small motor, most of the avionics

and provides the spacecraft interface. It is a semi-monocoque aluminum structure with eight longerons to provide the primary load paths at the spacecraft attachment fittings. The top of the ESS contains an interface mounting ring and electrical interface connector segment for the mating and integration of the spacecraft.

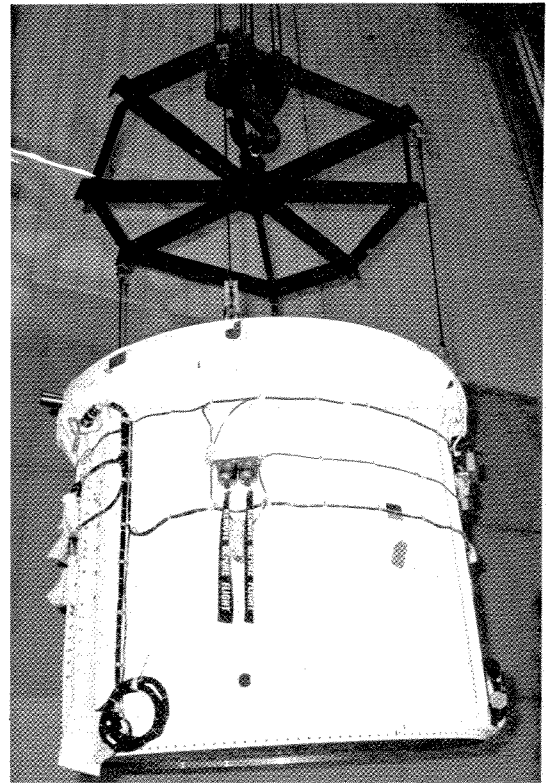
The interstage is an aluminum skin stringer conical/cylindrical structure between the large motor and the ESS and is jettisoned

with the first stage. The IUS stage 1-to-stage 2 separation system uses redundant low shock, ordnance-initiated separation nuts to minimize the shock environment on the spacecraft.

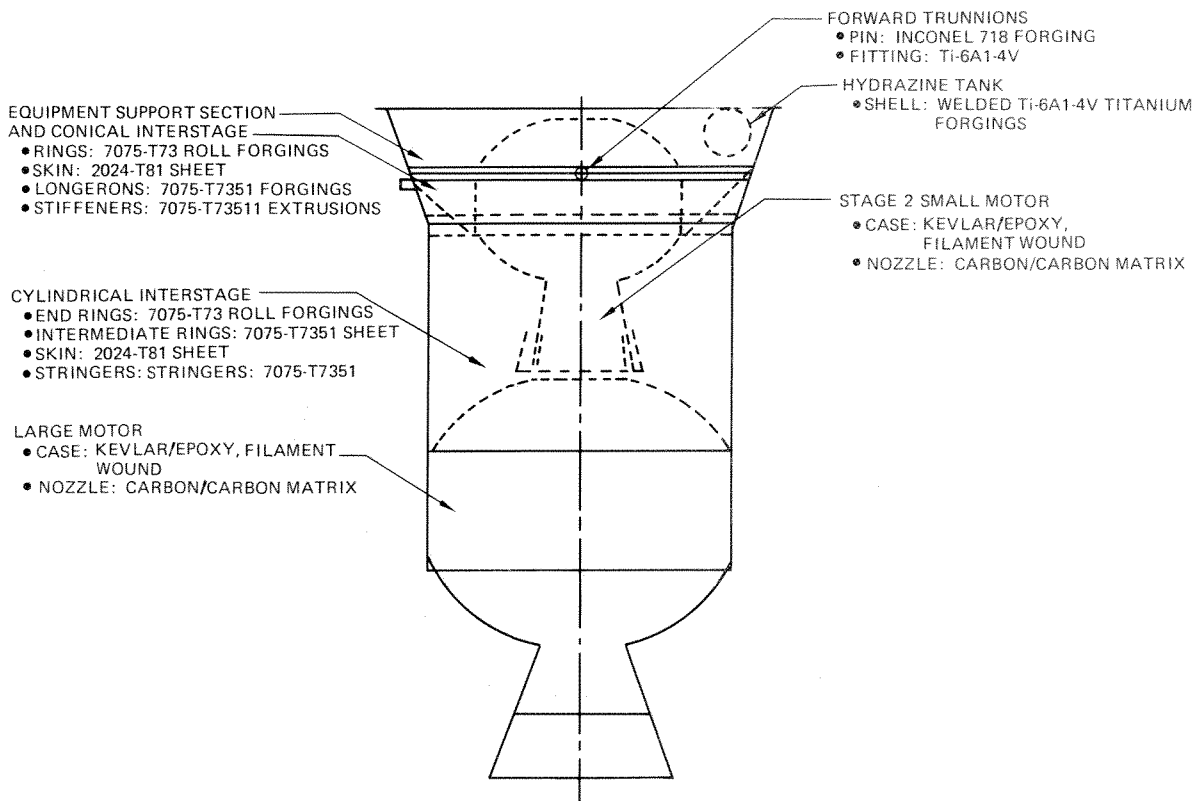
The IUS structural materials are shown in Figure 2-17.

### Equipment Arrangement

The equipment support section houses the majority of the IUS avionics and control subsystems. Exceptions in the Shuttle version include the batteries (six for stage 1 and spacecraft power), two safe and arm firing devices, three separation switches which are mounted on the interstage structure, and two thrust vector control actuators and potentiometers which are mounted on each motor. All line replaceable units (LRU) mounted in the ESS can be removed and replaced via access doors even with a mated spacecraft. This includes all equipment except the RCS tankage. The RCS tanks, prior to spacecraft mating, are filled and monitored for leaks to eliminate any later need for access.



*IUS Interstage*



*Figure 2-17: IUS Materials*

Arrangement of the ESS equipment is shown in Figure 2-18.

### Thermal Control

During the prelaunch period, conditioned air at 45°F to 80°F is required for cooling temperature-critical components and for general flushing of the equipment support section. An IUS purge duct provides for this air distribution and interfaces with either the orbiter or ground servicing facilities for pre-launch checkout. A thermal blanket prevents the conditioned air from crossing the spacecraft interface. Overboard venting is through the IUS structure.

Other than the spacecraft interface blanket, system level thermal control is provided by a coating of white thermal paint on the vehicle's outer surfaces, primer coating on the interior surfaces, and mounting provisions for the individual components. Additional thermal control for individual components is

achieved by insulation, coatings or heaters (Figure 2-19).

### PROPULSION

The IUS incorporates two propulsion systems: the solid rocket motors for main propulsion and a redundant reaction control subsystem for vernier burns and for attitude control during the coast phases of the mission.

#### Solid Rocket Motors

The solid rocket motors (SRM) are designed to use common approaches between the large and small motors to improve reliability and reduce cost. The IUS two-stage vehicle uses a large solid motor (SRM-1) with approximately 21,000 pounds of propellant (Figure 2-20) and a small solid motor (SRM-2) with approximately 6,000 pounds of propellant (Figure 2-21). These figures show the key design features such as Kevlar filament wound cases and three-dimensional

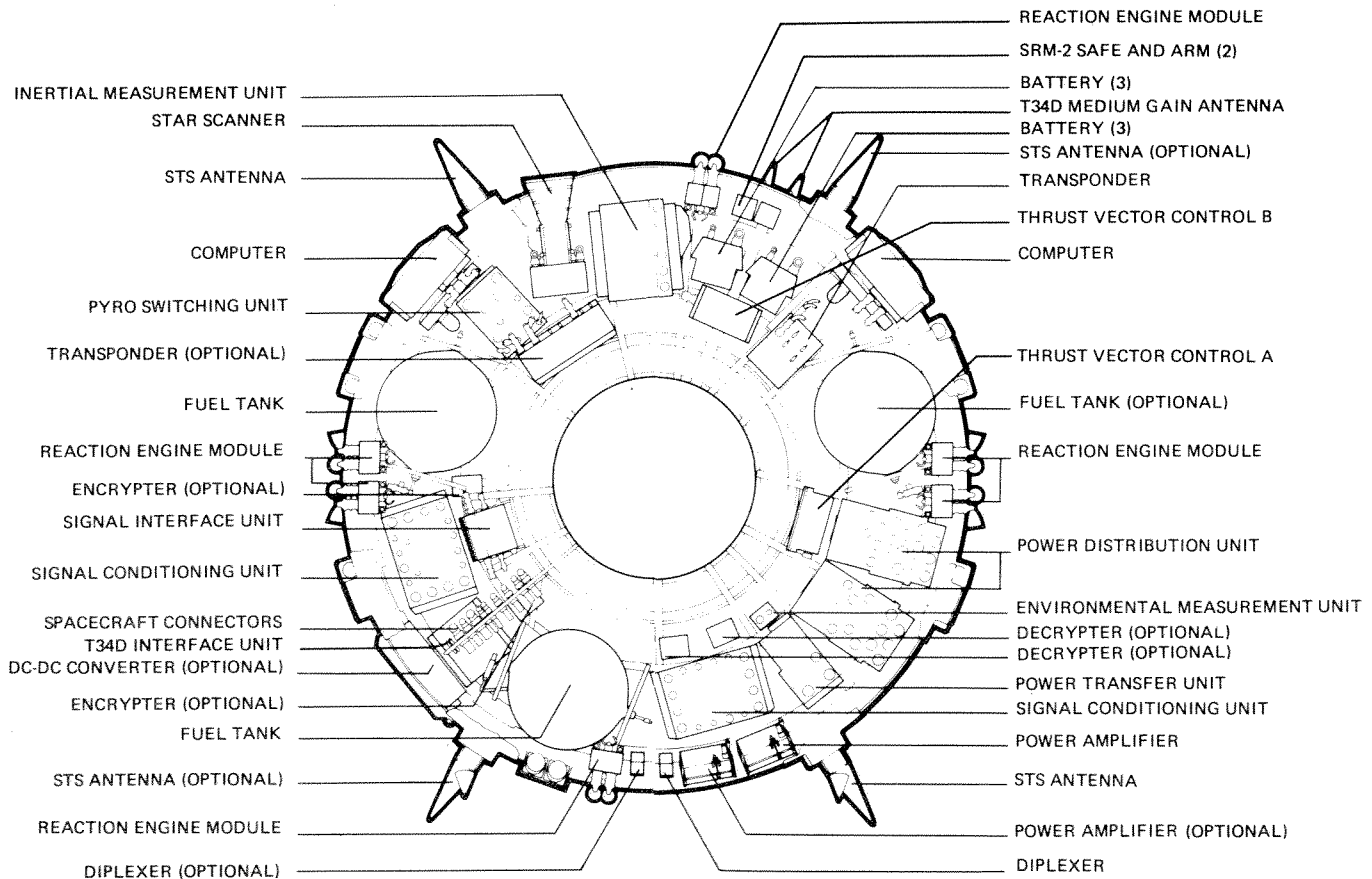
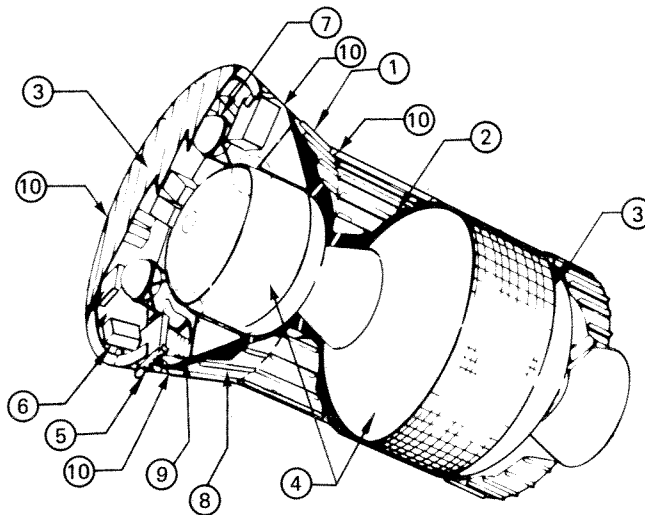


Figure 2-18: Equipment Arrangement



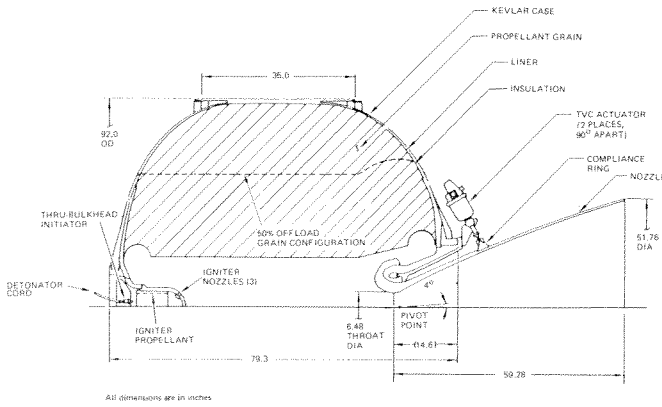
1. OUTER SURFACES – All surfaces with potential sun exposure are painted with white thermal control paint.
2. INNER SURFACES – All inner surfaces are painted with primer.
3. OUTER INSULATION BLANKETS – Five layers of aluminized Kapton, Dacron net spacers and an outer layer of beta cloth with aluminized backing.  
Used on:  
SRM-1 Aft Dome  
SRM-2 Aft Dome  
Spacecraft/IUS Interface  
External Batteries  
Star Scanner  
ESS Closures
4. INNER INSULATION BLANKETS – Five layers of aluminized Kapton with Dacron net spacers.  
Used on:  
SRM-1 Forward Dome and Cylinder  
SRM-1 Nozzle  
SRM-2 Forward Dome and Cylinder  
Avionics Equipment  
Safe & Arm Devices
5. HEATERS – Electric patch heaters  
Used on:  
Rocket Engine Modules  
Propellant Tanks  
Propellant Lines  
Batteries  
RIMU  
Power Amplifier Frames
6. FLEXIBLE OPTICAL SECONDARY REFLECTORS – Outside - mounted silver-backed Teflon  
Used on:  
20-watt Amplifiers  
ESS Exterior adjacent to RIMU
7. INSULATION WRAP – Single layer of aluminized Kapton  
Used on:  
Propellant Tanks  
Propellant Lines
8. PURGE SYSTEM – Ducting used for routing ground cooling air
9. SHUNTS – Layered aluminum strips which provide a low resistance path to dissipate LRU-generated heat. Used on:  
Power Amplifier  
Computers  
Wide Band Data Interleaver  
REM Plumes
10. PLUME DEFLECTORS – Nextel fabric and Rene steel mounted externally on the ESS and interstage to provide shielding from REM plumes.

**Figure 2-19: Thermal Control Subsystem Elements**

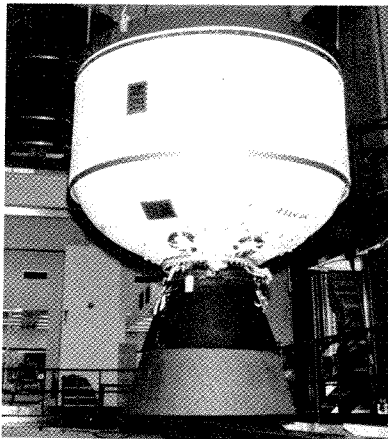
carbon/carbon integral throats and two-dimensional carbon/carbon exit cone nozzles. These motors also have independently redundant motor ignition systems. Movable nozzles are used for thrust vector control and are driven by redundant electromechanical

actuators to provide up to 4° of nozzle movement on the large motor and 7° on the small motor.

Variations in user mission requirements are met by tailored propellant offloading. Both



**Figure 2-20: First Stage Solid Rocket Motor**



**SRM-1 (Large Motor)**

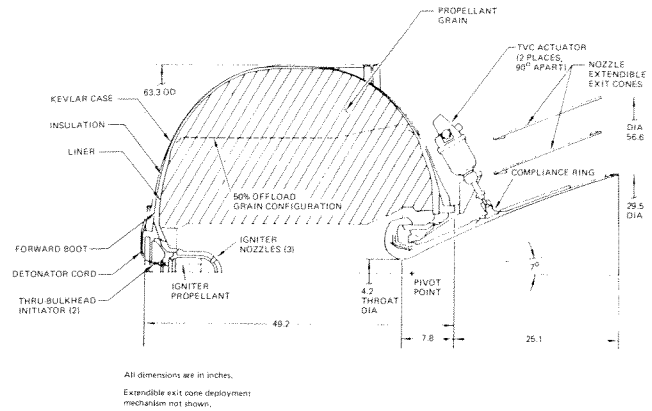
motors have a tubular-grain design that allows offloading flexibility. The motors are cast with a mandrel and then machined to nominal full propellant load. Offloading is accomplished by continuing to machine to the nominal motor burnback pattern. This approach does not alter the thrust profile.

The motor igniter design will accommodate offloads up to 50%. The cylindrical sides of the SRM-1 motor case carry the IUS vehicle structural loads. Integral skirts are provided at the forward and aft ends of the motor for bolt attachment to the IUS vehicle's structural sections.

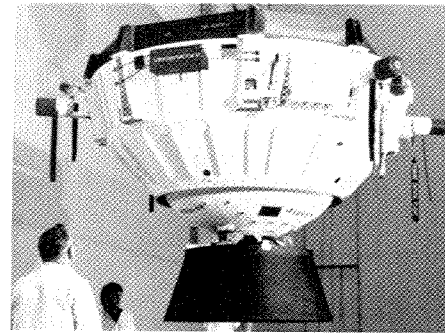
SRM-2 can be flown with or without its extendible exit cone (EEC). The EEC provides an increase of 14.4 seconds in the delivered specific impulse. Spacecraft missions which are not performance critical may elect to save the cost and fly without the

EEC. A trade must be made, however, with Shuttle usage charges. If the combined payload is weight driven, the EEC will permit more IUS propellant to be offloaded.

The design characteristics of the motors are summarized in Table 2-3.



**Figure 2-21: Second Stage Solid Rocket Motor**



**SRM-2 (Small Motor with Equipment Support Section)**

**Table 2-3: Motor Design Characteristics**

	SRM-1	SRM-2
Nominal Propellant Load (lb)	21,404	6,061
Minimum Propellant Load (lb)	10,700	3,000
Burntime (sec) 0% Offload (sec)	152.02	103.35
Average Thrust (lb)	41,611	17,629
Delivered Specific Impulse (sec)	295.5	289.1 (without EEC) 303.5 (with EEC)



## Reaction Control Subsystem

The IUS reaction control subsystem (RCS), Figure 2-22, uses hydrazine ( $N_2H_4$ ) in a monopropellant positive expulsion system that maneuvers and controls the vehicle attitude during the IUS coast periods. Thrust from rocket engine modules (REM) provides roll control during SRM burns, stability during coast periods, velocity impulses for accurate orbit injection, and collision avoidance maneuvers after spacecraft separation. The system is also used to provide the velocities for spacing between multiple spacecraft deployments.

Six two-nozzle REM's are used. Series-redundant valves for each nozzle provide redundancy in the fail-open or fail-closed valve failure modes. The placement of the twelve nozzles on the IUS provides two-nozzle redundancy for each of the six steering functions. To avoid spacecraft contamination, the IUS has no forward-facing thrusters.

The generic IUS contains two RCS tanks, each with a capacity of 122.5 pounds of hydrazine. A production option is available to add a third tank, if required. A tank can also be deleted, if desired. The system is

serviced prior to spacecraft mating and is sealed. Propellant is isolated in the tanks by squib-operated valves which are activated approximately five seconds before the IUS separates from Titan or after the IUS is deployed from the Shuttle. The tank and manifold safety factors are such that no safety constraints are imposed on operations in the vicinity of the serviced tanks.

## AVIONICS

The IUS avionics subsystem includes the electronic and electrical hardware used to perform all of the computation, signal conditioning, data processing, and software formatting which is associated with navigation, guidance, control, data management and redundancy management. The avionics subsystem provides communication between the IUS and the orbiter or the ground stations. It also provides for electrical power distribution.

The IUS has the most reliable avionics subsystem ever developed for unmanned space applications. This unparalleled reliability is obtained by a combination of stringent parts requirements through testing, and complete redundancy of all mission-critical command and control equipment.

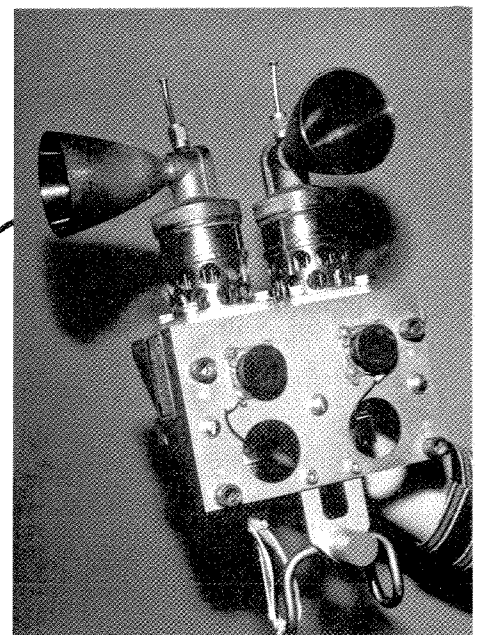
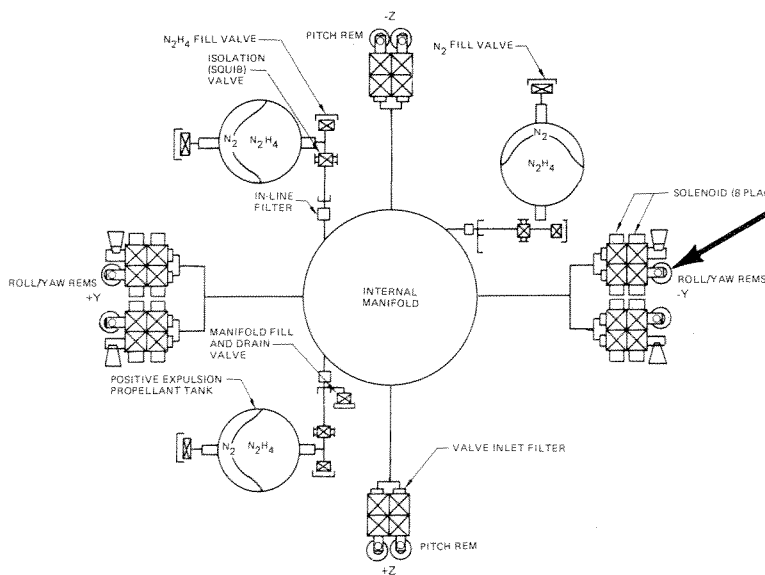


Figure 2-22: Reaction Control Subsystem

Reaction Control Subsystem:  
Rocket Engine Module Assembly

The redundancy architecture is illustrated in Figure 2-23. The single-string functions in the active avionics and control subsystems are not critical to mission completion.

The avionics consists of the following five subsystems:

- o Telemetry, tracking and command (TT&C) subsystem
- o Guidance and navigation subsystem (GNS)
- o Data management subsystem (DMS)
- o Thrust vector control (TVC) subsystem
- o Electrical power subsystem (EPS)

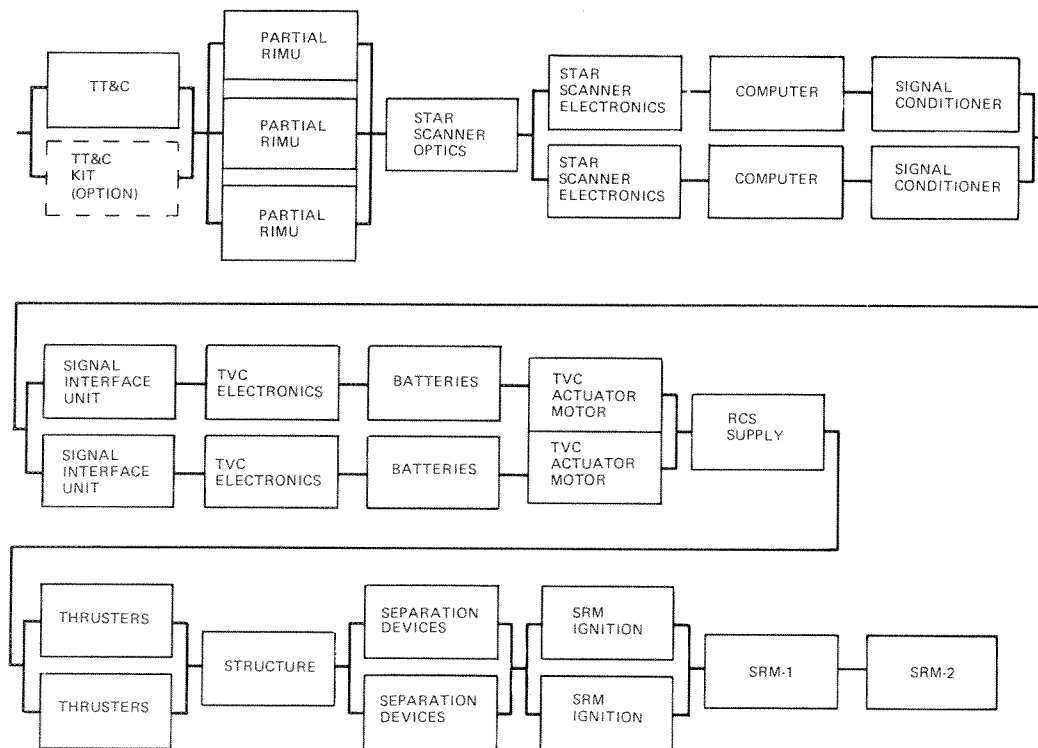
**Telemetry, Tracking and Command**

The baseline telemetry, tracking, and command subsystem is single string because the IUS performs an autonomous mission. A redundant set is optional to insure the command/communications link. The IUS has two telemetry transmission bit rates: 16 kbps and 64 kbps. The subsystem operates with ground stations during predeployment checkout and

during orbital flight. It also operates with the orbiter while the IUS is in the cargo bay and after IUS deployment at ranges of up to 20 nautical miles. The general characteristics for the 16-kbps format and the 64-kbps format are given in Table 2-4.

**Table 2-4: Telemetry Formats**

	<b>16 kbps</b>	<b>64 kbps</b>
Word Length	8 bits	8 bits
Main Frame Length	80 words	160 words
Main Frame Rate	25/sec	50/sec
Sub-Frame Length	25 main frames	50 main frames
Sub-Sub-Frame Length	8 sub-frames/ 200 main frames	4 sub-frames/ 200 main frames
Master Frame Rate (Equivalent to 1 Complete Sub-Sub-Frame)	1/8/sec	1/4/sec



**Figure 2-23: IUS Avionics System Redundancy Architecture**

At the 16-kbps rate, up to 1 kbps of data from each of the spacecraft (as many as 4) can be interleaved into the IUS telemetry format. At the 64-kbps data rate, up to 4 kbps from each of the spacecraft can be interleaved. The data will be included in the main frame words shown in Table 2-5.

**Table 2-5: Spacecraft Data Locations**

	Mainframe Words	
	16 kbps	64 kbps
Spacecraft 1	61-65	121-130
Spacecraft 2	66-70	131-140
Spacecraft 3	71-75	141-150
Spacecraft 4	76-80	151-160

At the spacecraft maximum data ranges of 1 kbps and 4 kbps, all mainframe words allotted for a particular spacecraft will contain updated data each mainframe. If the spacecraft is transmitting data at a data rate below the maximum rate allowed, the following applies: when sufficient data has been received from a spacecraft to fill all the mainframe words allotted to the spacecraft, the data will be downlinked in the next telemetry mainframe. A bit for each spacecraft in mainframe word 60 (16 kbps), or 120 (64 kbps) will be set to 1, noting that new spacecraft data is present in the mainframe words allocated to that particular spacecraft. The bit in word 60 (16 kbps) or 120 (64 kbps) will then be set to zero until the next new batch of spacecraft data is received.

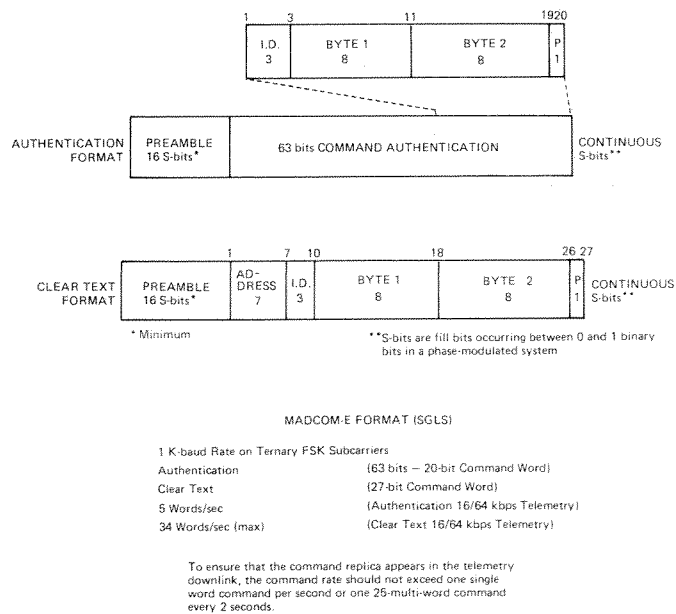
The lower bit rate is used during the coast phases after deployment from the orbiter. Telemetry transmission on the IUS radio frequency link begins after the vehicle is tilted for deployment from the orbiter. Before the vehicle is tilted, telemetry data is transmitted via the orbiter umbilical link.

The telemetry, tracking and command subsystem is capable of receiving update commands from ground control or the orbiter at a bit rate of two kbps. Upon receiving an uplink carrier, the receiver acquires and demodulates the uplink baseband. The basic command format, (compatible with the space

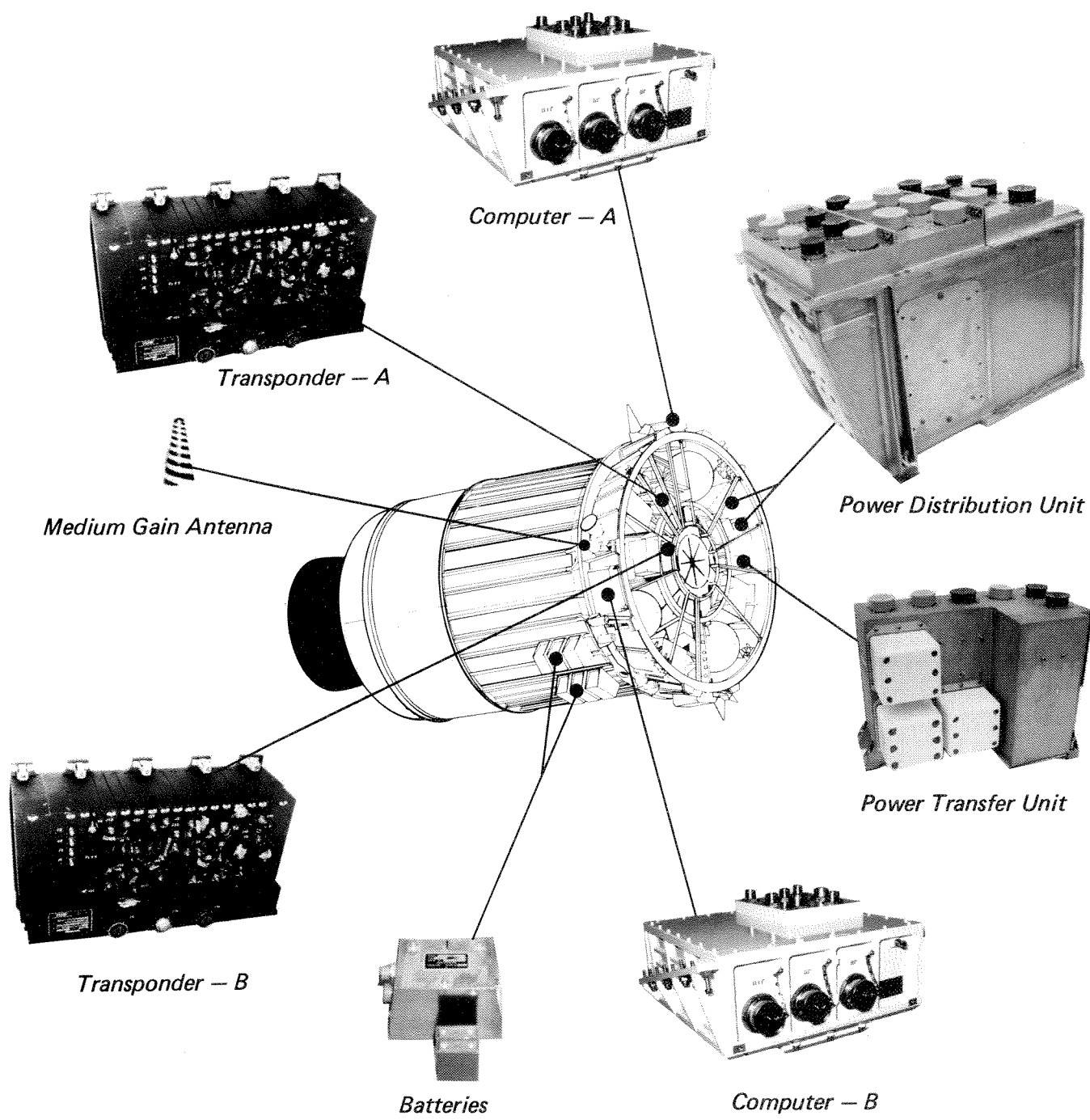
ground link subsystem (SGLS)) is shown in Figure 2-24.

The clear text format for the IUS (Figure 2-25) has three types of commands:

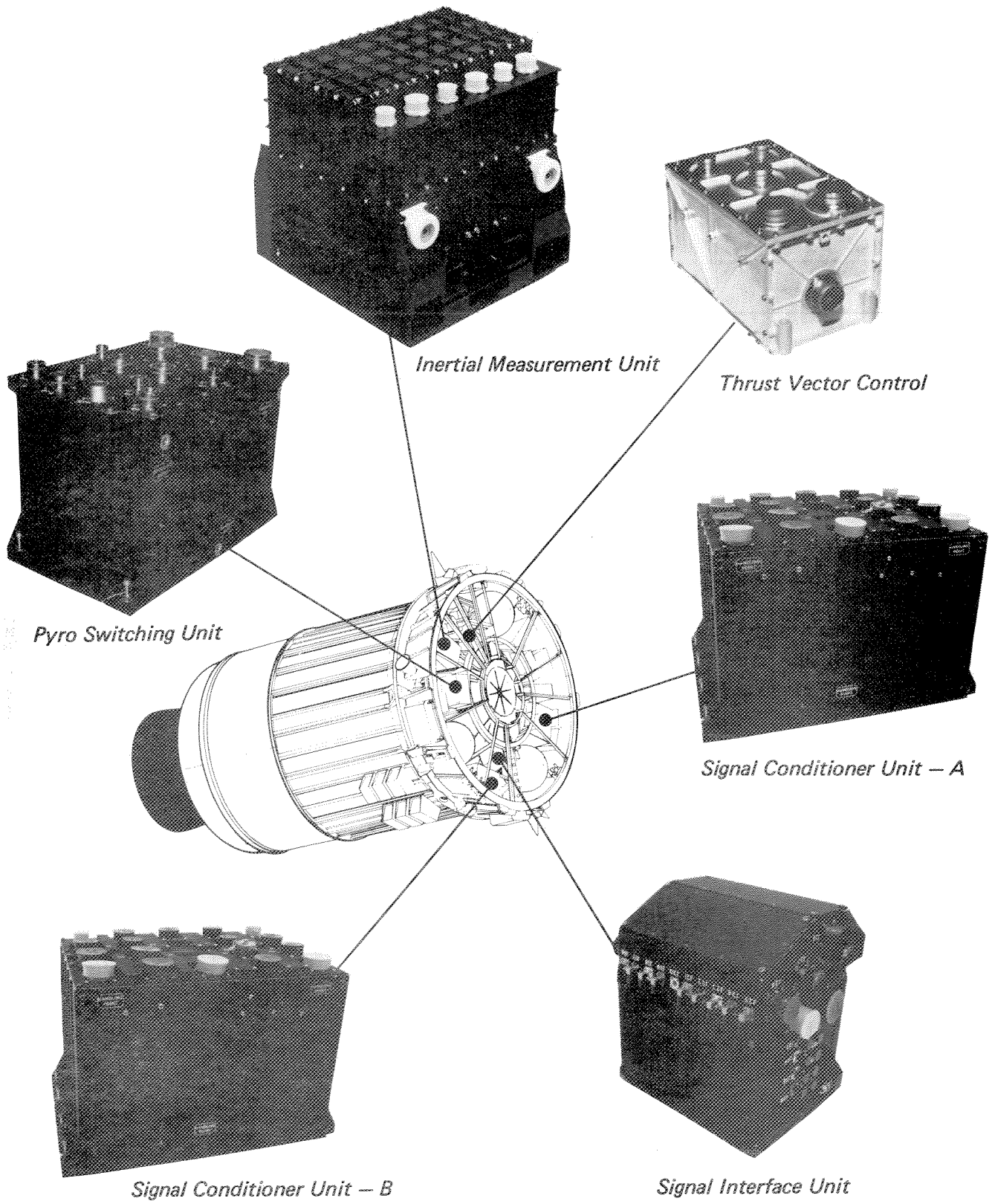
- (1) A two-stage command consists of one to twenty-four words with verification via telemetry. Ground verification of the command word replica is then followed by an execute command. (When more than one word is transmitted as a command, it is called a multi-word command.) The two-stage command has two basic parts: the command (stage 1) is followed after ground verification by EXECUTE (stage 2). EXECUTE is a command complement of the first word.
- (2) A one-stage command consists of the same command as (1) but is followed immediately by EXECUTE without ground verification. This is faster but the probability for error increases because EXECUTE is issued blindly. EXECUTE in this mode is a command complement of the first word.



**Figure 2-24: SGLS-Compatible Command Format**

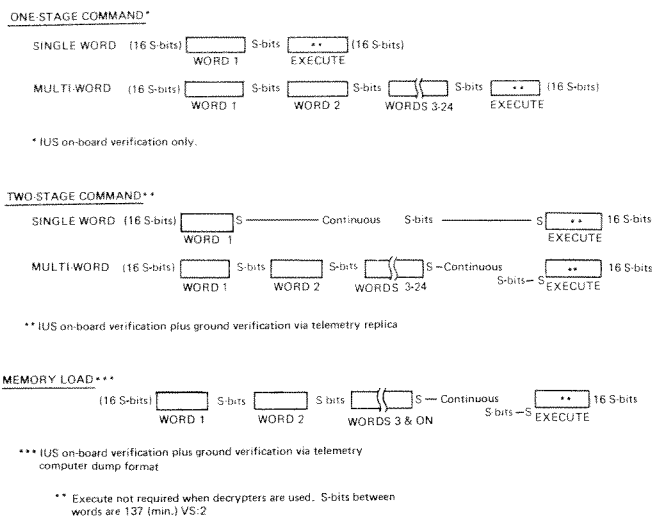


Typical Line Replaceable Units (LRU's)



Typical Line Replaceable Units (LRU's) (Continued)

- (3) A memory load consists of a continuous transmission of words with a beginning and ending memory address. Verification is done via telemetry with a change in telemetry format, i.e., a computer dump. This allows large amounts of data to be uplinked at maximum rates. EXECUTE may be transmitted after ground verification of the load is complete.



**Figure 2-25: Command Transmission and Verification Format for Clear Text**

## Guidance and Navigation

The stellar inertial guidance and navigation subsystem (GNS) provides measurements of angular rates, linear accelerations and other sensor data to the data management subsystem for appropriate processing by the computer software.

The GNS consists of a strapped-down redundant inertial measurement unit (RIMU) and a star scanner.

The RIMU consists of five Hamilton Standard RI-1010 rate-integrating gyros, five Singer Kearfott 2401 accelerometers, and associated electronics. The electronics provide conditioned power, digital control, thermal control, synchronization, and the necessary computer interfaces for the inertial sensors. As configured, the electronics provide three fully independent channels of data to the computers. Two channels each support two

sets of sensors, and the third channel supports one set. Data from all five gyro/accelerometer sets is simultaneously sent to both computers.

The RIMU is calibrated and aligned on the launch pad. The navigation function is started at liftoff, and data from the RIMU is integrated in the navigation software to determine the current state vector. Although RIMU drift errors accumulated prior to IUS deployment have proven to be negligible, it is possible to update the IUS attitude reference while in the orbiter. This requires the orbiter to perform a maneuver so that the IUS star scanner can scan two stars separated by 60 to 120 degrees.

If, for any reason, the computer is powered-down prior to deployment, the navigation function is reinstated by transferring orbiter position, velocity, and attitude data to the IUS vehicle. An attitude update is then performed as described above.

A star scanner attitude update can also be performed after deployment. For example, an update can be used shortly before SRM-2 ignition to correct any attitude drift accumulated during transfer orbit coast.

The IUS vehicle uses an explicit guidance algorithm (gamma guidance), to generate thrust steering commands, SRM ignition time, and RCS vernier thrust cutoff time. Prior to each SRM ignition and each RCS vernier burn, the vehicle is oriented to a thrust attitude based on nominal performance of the remaining propulsion stages.

During SRM burn, the current state vector determined from the navigation function is compared to the desired state vector and commanded attitude is adjusted to compensate for the buildup of position and velocity errors due to off-nominal SRM performance. RCS vernier thrust is used to compensate for velocity and position errors resulting from SRM impulse and cutoff time dispersions.

Attitude control, in response to guidance commands, is provided by thrust vector control (TVC) during powered flight and by reaction control thrusters during coast. Measured

attitude from the guidance and navigation subsystem is compared with guidance commands to generate error signals. During solid motor burn, these error signals drive the motor nozzle actuator electronics in the TVC subsystem. The resulting nozzle deflections produce the desired attitude control torques in pitch and yaw. The RCS roll-axis thrusters maintain roll control.

During coast flight, error signals are processed in the computer to generate RCS thruster commands which are used to maintain the vehicle attitude or to maneuver the vehicle. For attitude maneuvers, quarterturn rotations are used. Spacecraft requirements for thermal control, telemetry, etc., establish the attitudes/maneuvers during the coast periods.

### **Data Management**

The data management subsystem (DMS) executes the computation, data processing and signal conditioning associated with guidance, navigation and control; implements the safe and arm and firing of the solid rocket motor and electro-explosive devices; commands the decoding and telemetry formatting; performs the redundancy management; and issues the spacecraft discrettes.

The DMS consists of two operational channels with a computer, a signal conditioner unit (SCU) for each channel, and a signal interface unit (SIU) which is functionally divided into two electrically redundant channels.

The IUS avionics computers are modular, general purpose digital computers manufactured by DELCO electronics. The DELCO model designator is M362S. This computer uses the operational flight software (OFS) to perform in-flight calculations and to initiate vehicle thrust and attitude control functions which are necessary to guide the IUS payload through a predetermined flight path to a final orbit. A stored program including data, known as the on-board digital data load (ODDL), is loaded into the flight computer memory from magnetic tape through the memory load unit (MLU) during prelaunch operations.

The computer's high speed is achieved

through functional parallelism in the central processing unit (CPU). This parallelism allows independent functional units within the CPU to operate on a common register file. Up to five instructions, at different stages of execution, can be handled simultaneously. The input/output section consists of a dual-redundant serial data bus interface, including internally generated and external interrupts, and a direct memory access. The memory capacity is 65,536 (64K) 16-bit words. Five hundred and twelve words of memory may be directly addressed, and the remaining words may be accessed through relative, stack, and indirect addressing modes.

The signal conditioner unit provides the interface for the commands and measurements between the avionics computers and the pyros, power, RCS, TVC, TT&C, star scanner, and the spacecraft. The SCU consists of two channels of signal conditioning and distribution for the command and measurement functions. The two channels are designated channel A and channel B. Channel B is redundant to channel A for each measurement and command function.

With one exception, all of the electronics for command processing electronics are packaged in the SCU. The command output interface electronics, which switch either inductive loads or high currents, are packaged with the power switching and distribution electronics in the power distribution unit and the pyro switching unit.

The signal interface unit performs buffering, switching, formatting and filtering of TT&C interface signals. These command and telemetry interface functions are shown in Figure 2-26.

### **Thrust Vector Control**

The thrust vector control (TVC) subsystem consists of two controllers, four actuators, and four potentiometer units. The actuators and potentiometers are mounted in opposition to each other on each engine's aft dome and are attached to the engine's nozzle. The pitch and yaw actuators are clocked at 90° from each other.

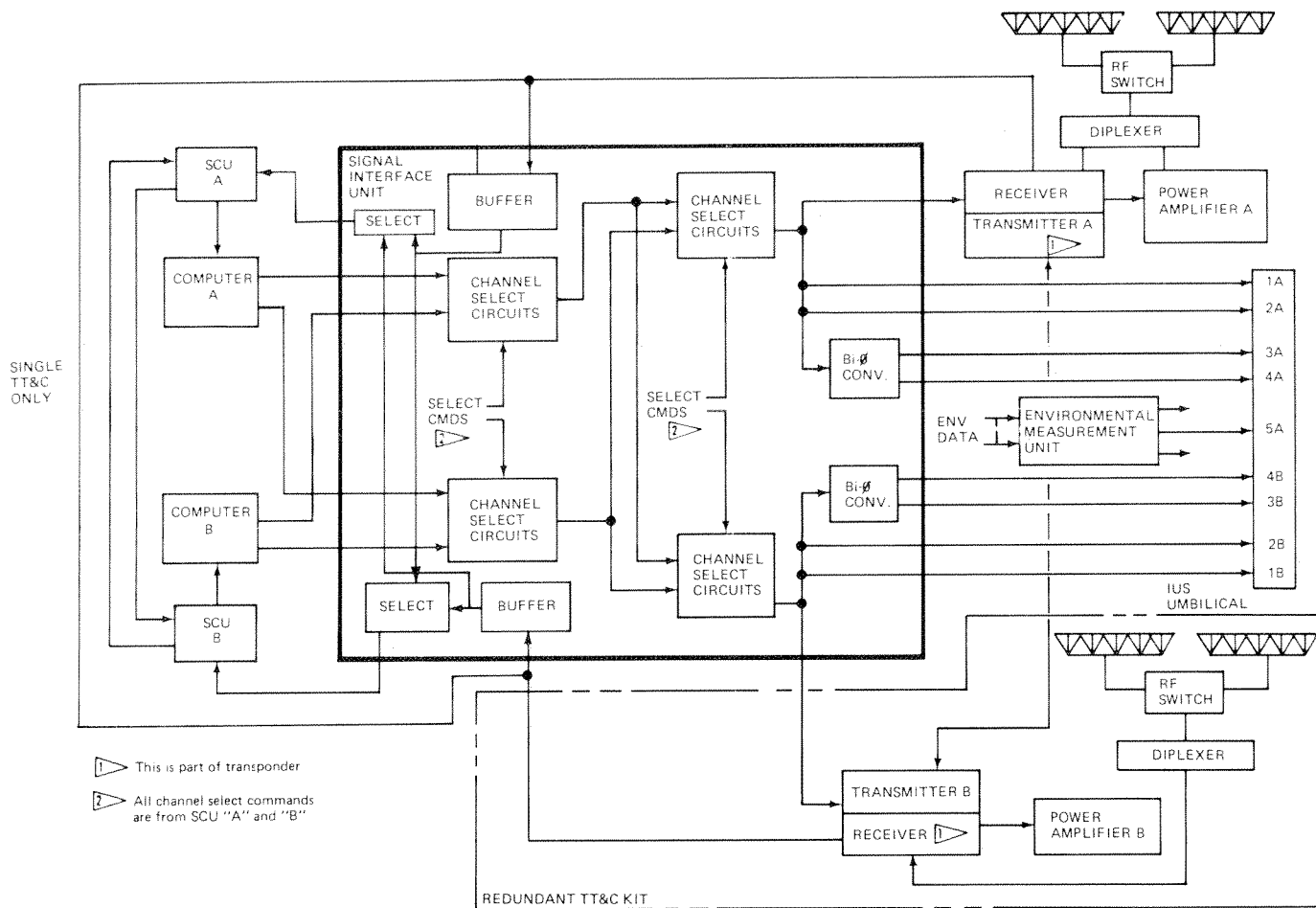
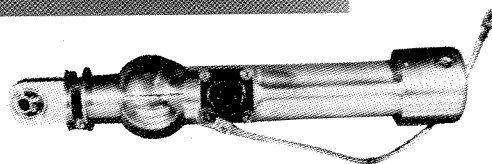
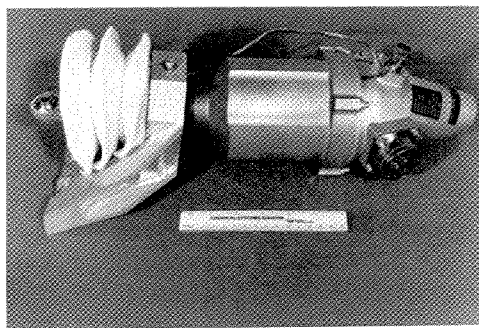


Figure 2-26: Signal Interface Unit Command and Telemetry Interfaces

The TVC subsystem provides the interface between the IUS guidance, navigation and control system and gimballed solid rocket motor nozzles so that powered-flight attitude control can be performed. Two complete electrically redundant channels are provided to minimize single-point failures.

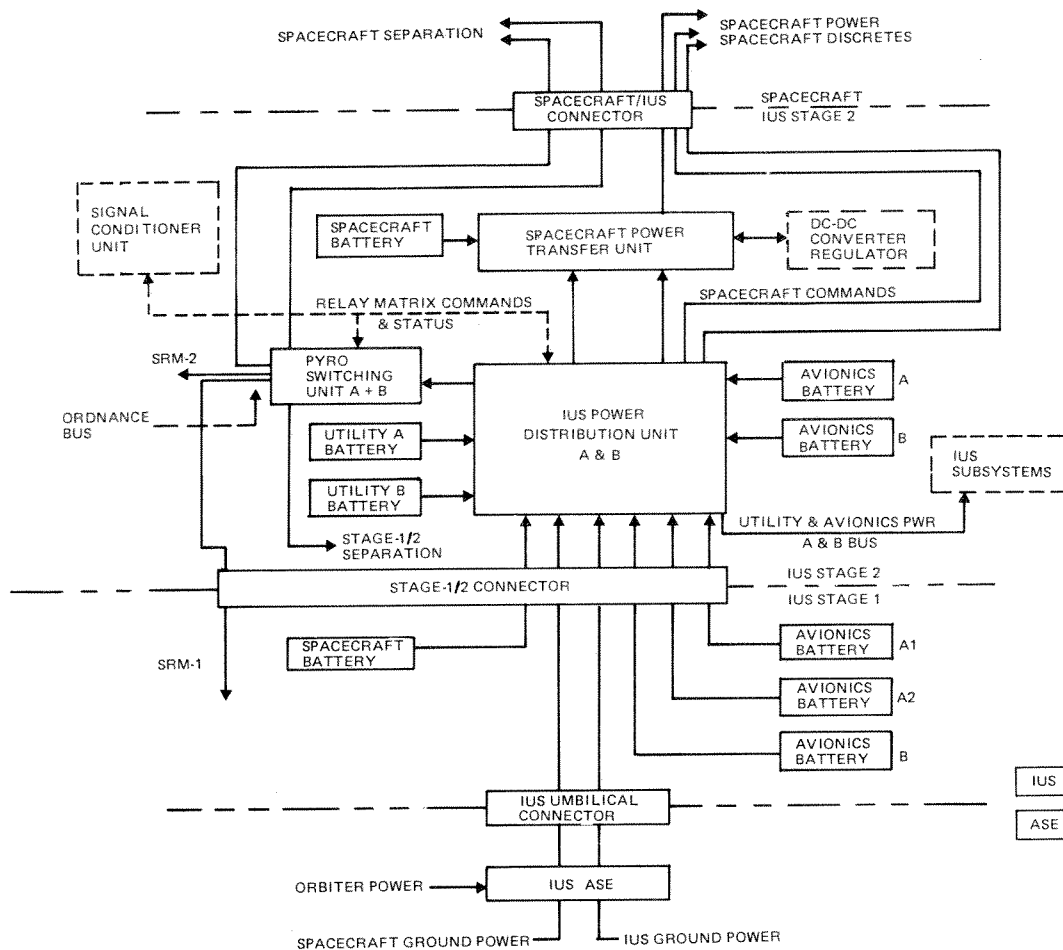
Power is supplied through the SCU to the thrust vector controller which controls the actuators. The controller receives analog pitch and yaw commands which are proportional to the desired nozzle angles and converts them to pulse-width, modulated voltages which power the actuator motors. The completely redundant motors drive a ball screw which extends or retracts the actuator to position the SRM nozzle. Potentiometers provide for servo loop closure and for nozzle position instrumentation. A staging command from the SCU allows switching of the

controller outputs from actuators on one stage to actuators on the second stage.



Thrust Vector Potentiometer and Actuator





**Figure 2-27: IUS Electrical Power Subsystem**

### Electrical Power

The IUS power subsystem provides power for the IUS vehicle equipment and the spacecraft. The power subsystem consists of the following major components: avionics batteries, spacecraft batteries, IUS power distribution units, power transfer unit, utility batteries, pyro switching unit, IUS wiring harness and umbilical, and staging connectors (Figure 2-27).

The power subsystem can use electrical power supplied via the ASE from ground support equipment or from the orbiter. If the spacecraft requires regulated power, provisions are made for installation of a dc-to-dc converter production option in the IUS vehicle. The power subsystem switches and distributes electrical power to the IUS vehicle and its spacecraft via dual buses. The dual buses ensure that no single-point power

system failure can disable both the channel A and channel B avionics. Four batteries (two 100 ampere-hour and one 140 ampere-hour for the avionics, and one 100 ampere-hour for the spacecraft) are carried in the first stage to provide power until stage 1 separates from stage 2. Five 13 ampere-hour batteries (two, avionics; two, utility; and one for the spacecraft) supply power throughout the second stage flight and after spacecraft release.

The separate utility battery source supplies power to the thrust vector control motors, RCS valves, motor-driven power transfer switches, and all ordnance devices. This increases the likelihood of freedom from conducted interference.

All the batteries are silver-zinc type and supply the power required at the equipment interfaces at a voltage of  $28 \pm 4$  volts.

Battery complements are selected to satisfy the particular mission needs as dictated by the mission timelines and spacecraft power requirements.

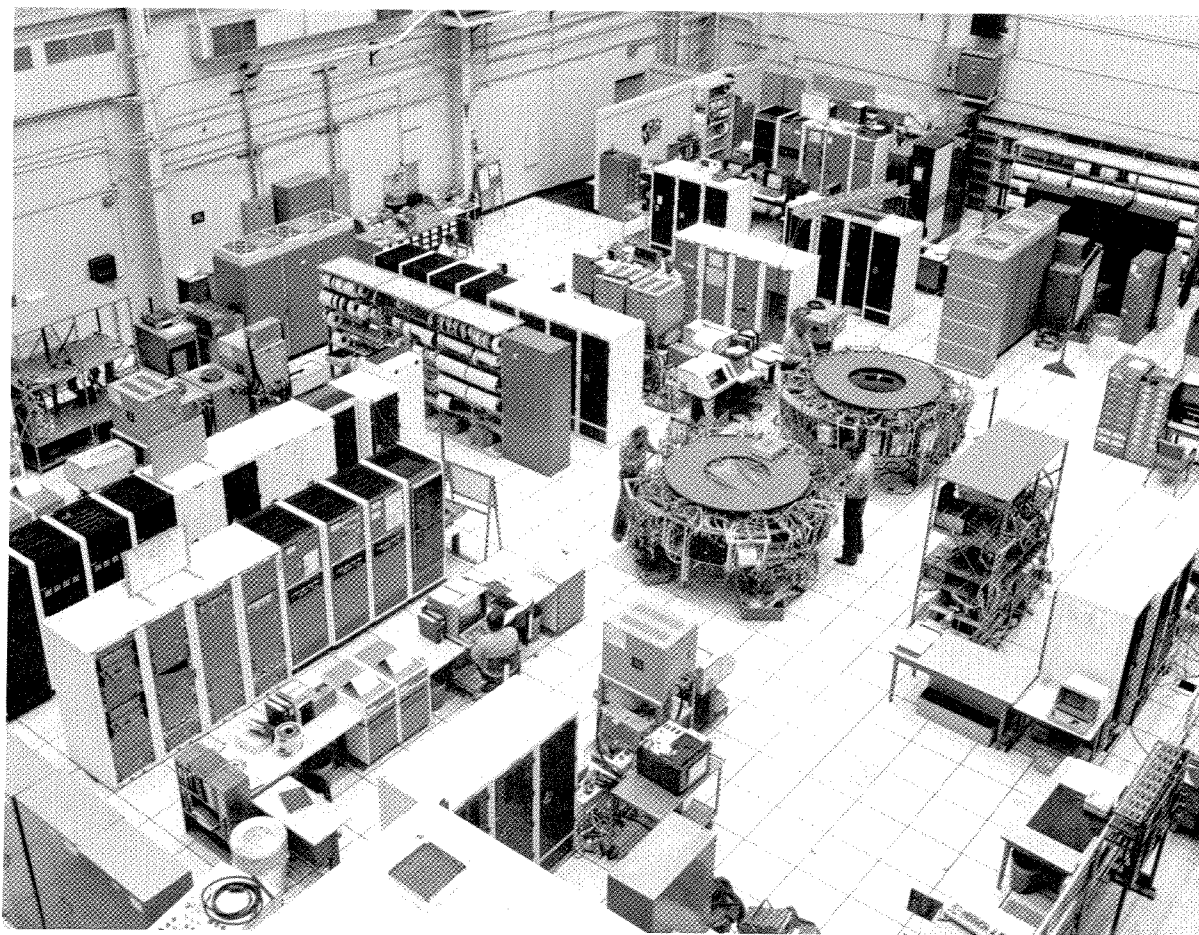
## SOFTWARE

The on-board software is composed of the operational flight software (OFS) and the mission data load (MDL) software. The mission on-board digital data load (ODDL) is the data which are entered into the IUS flight computer memory; it consists of the OFS program plus the MDL. The ODDL is transferred to the flight computer memory in one of two states: the initial load contains all mission and vehicle data plus the flight program. An update ODDL enters changes to the values of certain data parameters by defining data entries to specific memory locations.

## Operational Flight Software

The OFS is the basic IUS software which controls the IUS prelaunch/predeployment functions and its postlaunch function of placing the spacecraft into the desired orbit. The software provides the calculation and control capability for the following mission/operations functions: executive, mission sequencing, guidance, attitude control, communications, redundancy management, check out, and navigation.

IUS redundancy is controlled by the software, an IUS-unique feature. The redundancy management functional area determines the operating status of a number of IUS subsystems, and it commands reconfiguration when failures are detected. IUS hardware is monitored either by automatic systems that return status to the computers or by diagnos-



*IUS Equipment Support Section Checkout in the Boeing Aerospace Company's System's Integration Laboratory (SIL)*

tic software that exercise the hardware and monitor its performance. In the event of failure, the redundancy management function generates the necessary processing to switch to a predetermined hardware configuration; then it restarts normal processing in the new configuration.

### **Mission Data Load**

The mission and vehicle-unique data portion of the ODDL is denoted the MDL. These data include the RIMU calibration values, guidance parameters, flight control system gains, thermal maneuver references, mission sequencing event tables, etc. The MDL is in a binary load module form, with all memory address locations defined and ready for execution.

The flight operation load program for the DELCO avionics computer is developed in the IUS System Integration Laboratory (SIL) in JOVIAL J73/1 language and compiled on the DEC 10 compiler. At the time of compilation, mission and vehicle data are entered in the data base for the load program.

The load program is in absolute load module form and is placed on magnetic tape. The tape contents are loaded in the flight computer memory through the memory load unit (MLU) via an aerospace ground equipment interface with a computer input/output port. The MLU is a control unit which reads the ODDL tape, formats the data for the load, serially transfers the load data to the computer input/output port, and verifies the data transfer. The MLU is used for ODDL loads

during final mission certification in the SIL vehicle test area and for mission preparation on the launch pad. Data updates (classified and non-classified) are also possible via uplink command from the checkout station.

### **PECULIAR SUPPORT EQUIPMENT**

IUS peculiar support equipment (PSE) includes all the peculiar ground support equipment required to support IUS missions. The PSE includes IUS-unique launch site and depot ground equipment for assembly and test of the IUS vehicle, interface checkout, transportation of all configurations, loading and unloading equipment, work stands and equipment required for maintenance operations, safing, and IUS disassembly at the launch site and alternate landing sites. Computer resources within this system are the checkout station, validation and verification simulator, and the DEC 10 computer.

IUS computer programs include all of the unique software required for manufacture, test, checkout, launch and mission operations of the IUS system, and all diagnostic programs used to troubleshoot malfunctioning systems and subsystems. Computer resources within this system include the described generic operational flight software, the on-board digital data load, checkout station operational and test sequence software, validation and verification simulator operational model and test sequences software, the software design verification system, compiler software, mission operations system data formatter and post-processing software, and the DEC 10 compiler.



## CHAPTER III-IUS/SHUTTLE

The IUS extends the Space Shuttle's reach beyond the Shuttle's orbital range of 115 to 700 nautical miles. The IUS can be deployed from the Shuttle for geosynchronous orbits, elliptical orbits, higher circular orbits and deep space missions.

### LAUNCH VEHICLE DESCRIPTION

The Space Shuttle has three main units: the two solid rocket boosters, the external tank, and the orbiter.

The first unit, the two solid rocket boosters (SRB's), provide the initial ascent thrust by burning for two minutes with the main propulsion system of the orbiter. Each solid rocket booster weighs approximately 1,286,600 pounds and produces 2,650,000 pounds of thrust at sea level. The propellant grain is shaped to reduce thrust after liftoff in order to prevent overstressing the crew during the period of maximum dynamic pressure.

The second element, the external tank, feeds propellants to the orbiter's three main liquid rocket engines and serves as the Shuttle's structural backbone during launch operations.

At liftoff, the external tank holds 1,550,000 pounds of propellants, consisting of liquid hydrogen (fuel) and liquid oxygen (oxidizer). The external tank is released when the orbiter is just short of orbital velocity and the main engines have stopped. It is the only major element of the Shuttle program which is not reused.

The third and last unit is the orbiter, the aerodynamic element of the Shuttle system. It carries the crew and the payloads for the Space Shuttle. The orbiter can deliver to orbit payloads of up to 65,000 pounds in its 15-foot diameter by 60-foot long cargo bay.

### FLIGHT OPERATIONS

A typical sequence for a geosynchronous mission is depicted in Figure 3-1.

#### Ascent

Flight operations are initiated when the Space Shuttle lifts off from the launch pad. The orbiter's main engines and the boosters ignite simultaneously to propel the Shuttle, with an IUS stowed in its cargo bay, away from the launch pad.

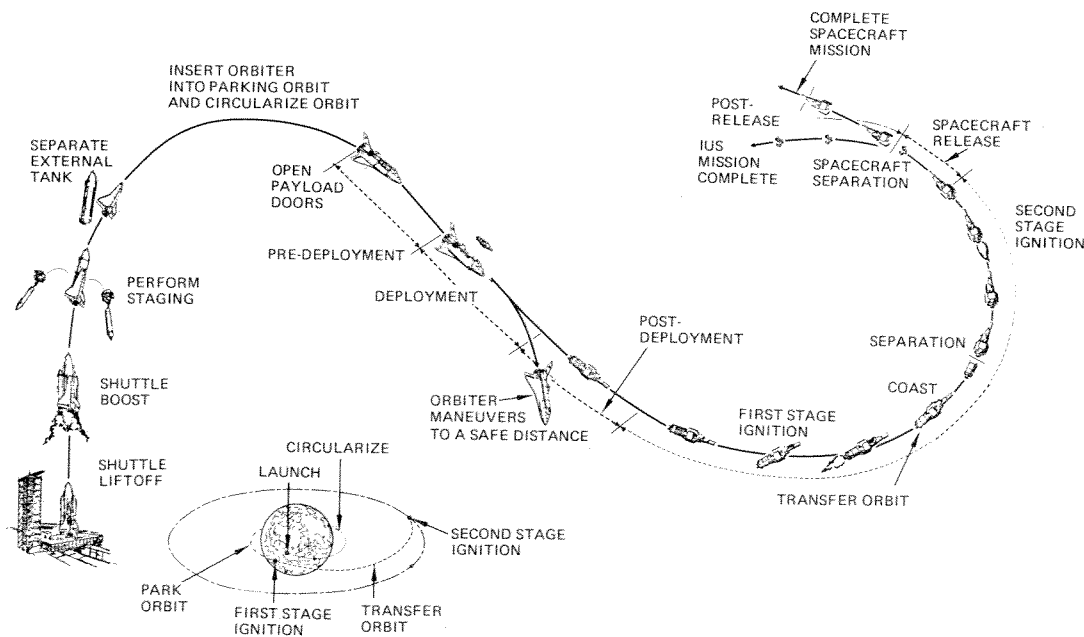


Figure 3-1: Sequence of Events for a Typical Geosynchronous Mission

At about 27 miles from earth, the two solid rocket boosters separate from the orbiter and parachute to the sea.

The three main orbiter engines continue to burn until the vehicle is just short of orbital velocity. This occurs approximately 70 miles above the earth. At this point, the engines shut down and the external tank is jettisoned. The Shuttle's small orbital maneuvering engines are used to attain the desired orbit and make any subsequent maneuvers that may be needed during the mission.

During this ascent phase of the mission, the spacecraft telemetry is interleaved with the IUS telemetry and is recorded on the orbiter's payload recorder.

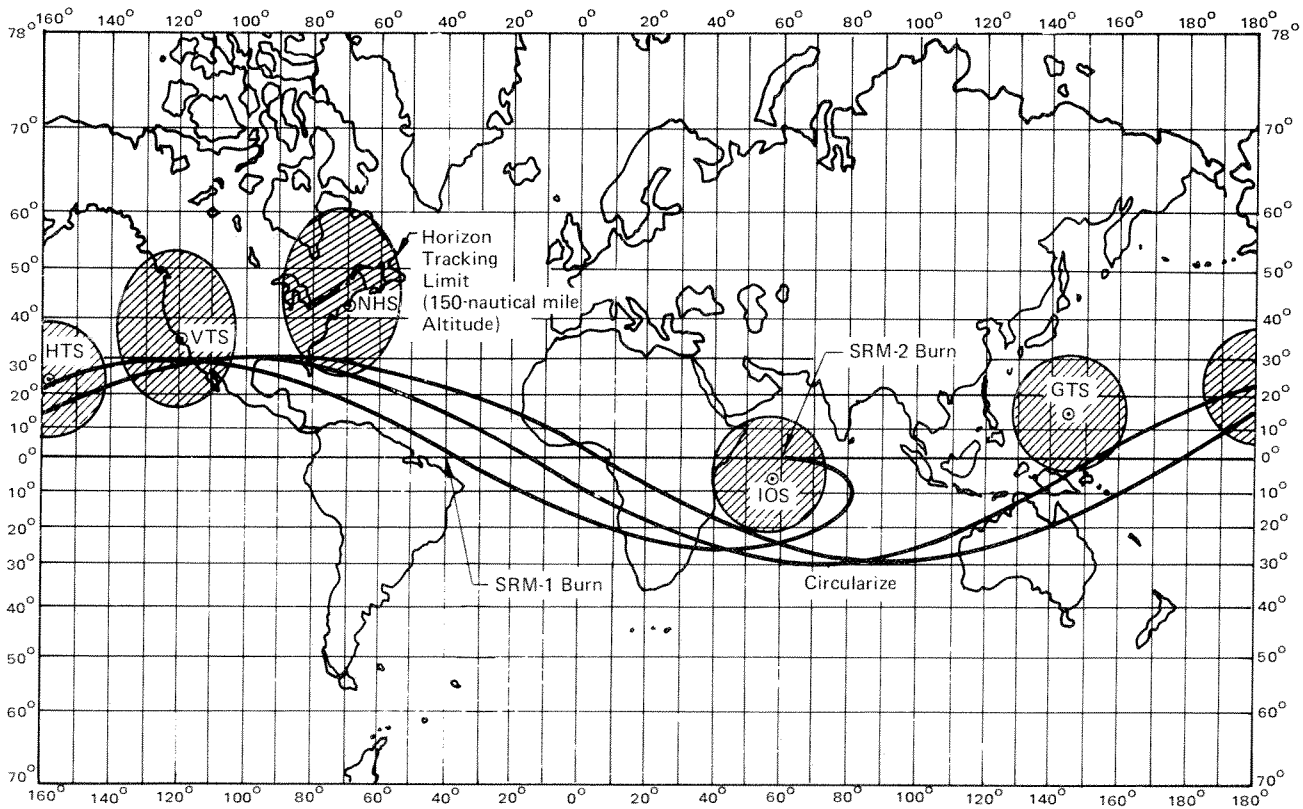
If orbiter power to the spacecraft is interrupted, the IUS airborne support equipment automatically supplies the required spacecraft power from the ASE batteries.

If high temperature limits are experienced prior to opening of the orbiter payload bay doors, the IUS will automatically power-down.

### On-Orbit

Approximately one hour after launch, the orbiter payload bay doors are opened. To fulfill payload thermal requirements and constraints, the orbiter maintains a payload bay-to-earth attitude. This allows the spacecraft to have direct communication with the ground. Spacecraft checkout is nominally scheduled during the first pass over the Hawaii Tracking Station (HTS). Figure 3-2 shows a ground track of the geosynchronous design reference mission.

The orbiter orients itself, as required, for other short term events (e.g., IUS star scan maneuvers, orbiter IMU alignments, radio frequency communications, and deployment



The ground track is typical for a generic mission's first deployment opportunity

**Figure 3-2: Geosynchronous Design Reference Mission Ground Trace**

operations). The IUS can remain on-orbit for up to seven days, being powered-up or down as required. The computer's internal batteries will maintain the computer memory.

Predeployment operations include an early IUS and spacecraft condition check (if desired), orbiter maneuvers to support a star scan by the IUS for attitude update, an RF command link check (i.e., orbiter-to-IUS and ground-to-IUS), state vector update, and a final IUS health check prior to deployment.

To support spacecraft checkout or other IUS-initiated functions, the IUS has the capability to issue a maximum of eight discrete impulses. These may be initiated manually by the orbiter crew prior to deployment from the orbiter, or automatically by the IUS mission sequencing flight software after deployment. After attitude is established (including star scan maneuver, if required), the forward end of the IUS is released from the forward ASE, and the IUS and spacecraft are erected to a 29° position in the orbiter bay. In this position, payload checkout and state vector verification are performed.

The IUS has the capability of accommodating alternate spacecraft missions. This can be accomplished with or without an RF multiword uplink command.

Prior to deployment, the orbiter crew switches the spacecraft electrical power source from the orbiter to IUS internal power. Data contained in the IUS and spacecraft telemetry verifies the spacecraft is on IUS internal power and that all of the IUS/spacecraft predeployment operations have been successfully completed. Telemetry data are evaluated by the IUS Mission Control Center for the IUS, and by the Spacecraft Control Center for the spacecraft. Analysis of the telemetry results in a GO/NO-GO decision for IUS/spacecraft deployment from the orbiter. This GO/NO-GO decision can be relayed by a voice link from the IUS Mission Control Center via the Mission Control Center in Houston to the orbiter crew.

## Deployment

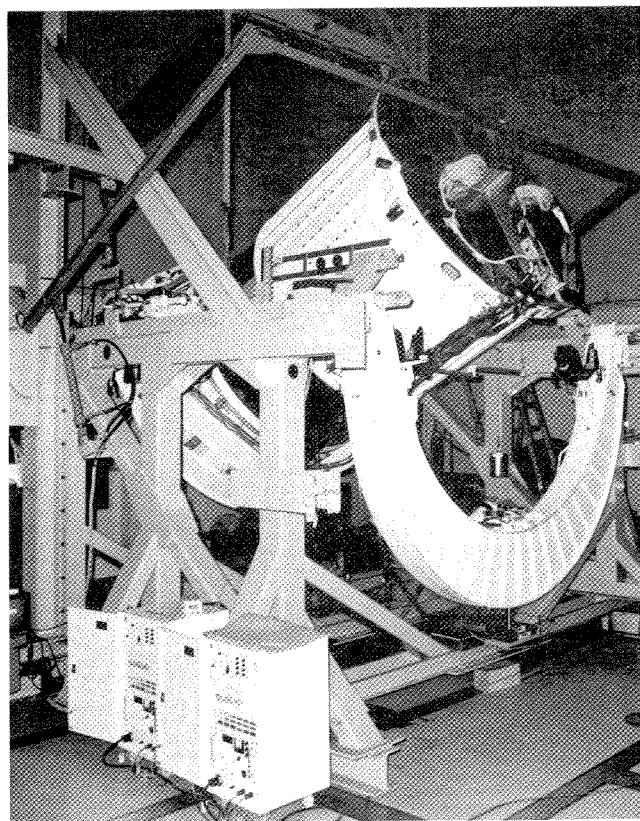
After the tilt table is in the 29° position and

the GO decision is provided from mission control, the crew activates the ordnance that separates the IUS and spacecraft umbilical cables. The crew then raises the tilt table to the 58° deployment position and activates the Super\*Zip ordnance separation device. This physically separates the IUS from the tilt table.

The IUS/spacecraft is deployed in the shadow of the orbiter or in earth eclipse. When the separation device is activated, compressed springs provide the force to separate the IUS/spacecraft from the orbiter. Ten minutes after the IUS ejects from the orbiter (at a distance of 200 feet or more), the IUS on-board computer enables the reaction control subsystem. The IUS on-board flight computer similarly sequences all subsequent operations from transfer orbit injection through spacecraft separation and IUS deactivation.

## Post-Deployment

Following RCS activation, the IUS maneuvers



*Airborne Support Equipment Deployment Test*

the spacecraft to its required thermal attitude and performs the required spacecraft thermal control maneuver. The IUS can provide pointing, toasting, roisserie, metronome and continuous roll thermal maneuvers. For more information on thermal maneuvers, see Spacecraft Services in this chapter.

Approximately 45 minutes after IUS ejection from the orbiter (now at least 10 miles away), the IUS SRM-1 ordnance inhibitors are removed. The IUS then recomputes SRM-1 ignition time and maneuvers to the proper attitude for SRM-1 burn. When the transfer orbit injection position is reached, the IUS computer enables and applies ordnance power, arms the safe and arm devices, and fires the first-stage solid rocket motor. SRM-1 burns for approximately 150 seconds to provide the thrust for the orbit transfer phase of the geosynchronous mission. During the coast phase, the IUS is capable of performing maneuvers required by the spacecraft for thermal protection or communication.

If the spacecraft requires improved accuracy, the IUS can perform a pre-SRM-2 star scan attitude update. The IUS can issue up to eight discrete impulses to the spacecraft during the mission. The IUS first stage separates from the IUS second stage just prior to reaching the apogee point of the trajectory. At apogee, the second stage motor burns for approximately 103 seconds, providing the final injection to the geosynchronous orbit. At the point of insertion, the IUS has completed the thrusting maneuvers of the main propulsion systems. The IUS provides attitude control and supports spacecraft separation. It then performs a final collision/contamination avoidance maneuver before deactivating. Table 3-1 shows a mission timeline for the geosynchronous mission.

### PERFORMANCE CAPABILITY

Performance capability of the IUS covers a wide range of earth orbit and planetary missions. Payload is defined as the total weight carried by the IUS vehicle on a particular mission and includes any spacecraft-unique adapter or other spacecraft chargeable weights.

Performance data are based on placing a 5000-pound payload in a zero-eccentricity/zero-inclination geosynchronous orbit. All performance capabilities to non-generic missions are referenced to this value. Actual capability may exceed these values. For specific capability to a defined mission orbit, please contact The Boeing Aerospace Company.

### Earth Orbit Payload Capability

The effects of orbit inclination and eccentricity on payload capability for 24-hour period (geosynchronous) orbits are shown in Figure 3-3. The zero-inclination/zero-eccentricity point corresponds to the design reference mission where a full propellant load is carried in both stages. For essentially all other cases, stage 2 propellant is offloaded to maximize payload capability. Both of the IUS SRM's can be offloaded from 0 to 50%. Offloading may be necessary to maximize payload and/or satisfy a total weight constraint.

The structure has been tested and verified for a payload of 8,000 pounds. Payload weights above 8,000 pounds will require a load reassessment to determine if structural modification is required. Studies have shown,

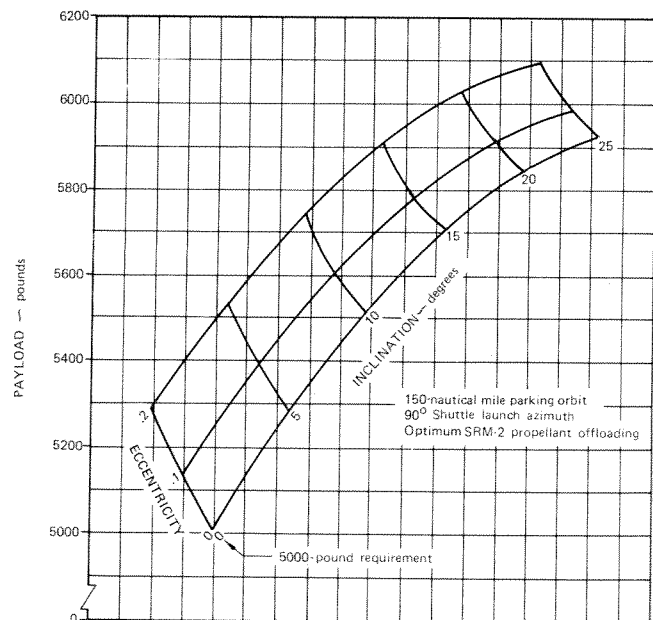


Figure 3-3: Geosynchronous Mission Payload Capability



**Table 3-1: STS/IUS Geosynchronous Mission Timeline**

Ground Elapsed Time (hr:min:sec)	Launch/Ascent and Deployment Phase Events	Event Duration (min:sec)	Ground Elapsed Time (hr:min:sec)	Launch/Ascent and Deployment Phase Events	Event Duration (min:sec)
00:00:00	Liftoff		02:28:00	Deploy IUS	:30:00
00:08:09	Main Engine Cutoff 15 x 80 n. mi. Orbit at 484 sec.		02:33:00	Configure Orbiter for RCS Attitude Control, Maneuver Orbiter to Safe Separation Distance Within 10 min. So IUS is Visible Through Upper Observation Windows	:10:00
00:08:45	Start Orbital Maneuver Subsystem	:01:55	02:38:00	Activate IUS RCS Within 10 min. from Release Unless Fewer Than 200 ft. Separates the IUS and the Orbiter.	
00:10:40	End Orbital Maneuver Subsystem 55 x 150 n. mi. 28.5° Inclination			Start Spacecraft Thermal Control Maneuvers, If Required.	
00:44:27	Start Orbital Maneuver System-2 Circularization Burn	:01:35		OR: Perform Telemetry Dipout Maneuvers Required by the Spacecraft—May Interrupt Thermal Control Maneuvers.	
00:46:02	End Orbital Maneuver System-2 Circularization Burn 150 x 150 n. mi. Orbit				
00:52:02	Reconfigure Five Orbiter General Purpose Computers	:12:00			
01:05:32	Open Payload Bay Doors (Five min. each door)	:10:00	02:53:00	Lower Tilt Table	:10:00
	Activate Ku-band Antennae within 12 min. from Payload Bay Doors Opening	:00:20	03:02:00	Perform IUS Stellar Attitude Update, if Required	
01:15:00	Transfer IUS from ASE Power to Orbiter Power	:01:00	03:25:10	Maneuver for SRM-1 Burn and Switch to 64 kbps	:02:00
01:15:32	Generate Orbiter State Vector Data	:57:00	03:27:12	SRM-1 Ignition and Burn	:02:27
01:18:00	Activate IUS and ASE Battery Heaters	:01:00	03:29:39	Correct SRM-1 Velocity Vector	:05:00
01:19:00	Checkout Communication Interface Unit and Transmit IUS/Spacecraft Telemetry	:05:00	03:34:39	Maneuver for Transfer Coast	:05:00
01:22:00	Activate IUS Rocket Engine Module Heaters	:02:00		Switch to 16 kbps	
01:25:00	Maneuver Orbiter to Deployment Attitude. Inhibit Primary Reaction Control Subsystem	:13:00	03:39:39	Start Thermal Control Continuous Roll Rate (This Maneuver Assumed for a Typical Spacecraft)	
01:43:00	Raise Tilt-Table to 29°	:10:00	04:28:00	Telemetry Dipout	:05:00
	Air Force Satellite Control Facility Coverage Desired Prior to, During, and Following Tilt Operations.		05:50:00	Telemetry Dipout, If Required by Spacecraft	:05:00
01:56:00	Transfer IUS to Internal Power (Stage 1 Batteries)	:03:00		Additional Telemetry Dipouts as Required for Spacecraft	
01:59:00	Perform IUS Pre-Deployment Checkout, (Telemetry, Tracking and Command, Data Management, Signal Conditioner Unit and Thrust Vector Control)	:02:00	08:38:55	Transfer to Stage 2 Power	:00:17
02:01:00	Transmit GO/NO-GO Telemetry Checkout Word to Mission Control Center/Air Force Satellite Control Facility	:02:00	08:39:12	Inhibit Reaction Control Subsystem and Separate Stage 1	:00:35
02:07:00	Switch IUS Telemetry Rate to 16 kbps	:02:00	08:39:19	Extend SRM-2 Nozzle (Extendible Exit Cone)	:00:17
02:09:00	Release IUS and Spacecraft Umbilicals	:02:00	08:39:47	Maneuver for SRM-2 Burn, Switch to Medium Gain Antenna and 64 kbps	:02:00
02:13:00	Raise Tilt Table to 58°	:10:00	08:41:47	SRM-2 Ignition and Burn	:01:47
	o Orbiter Reaction Control Subsystem (RCS) to Vernier Control Orbiter RCS to Manual		08:47:14	Vernier Burn to Correct SRM-2 Velocity Vector	:06:40
	o Orbiter RCS Inhibited			Switch to Omni-Antenna and 16 kbps (one sec.)	
			08:53:54	Maneuver to Separation Attitude and Stabilize	:03:20
			08:57:14	Separate	:00:04
			09:02:56	IUS Collision Avoidance Maneuver	:00:14
			09:05:00	Terminal Shutdown Sequence	

Note: Timeline assumes spacecraft does not require ASE tilt for checkout

however, that payload weights up to 16,000 pounds can be accommodated with minor structural modifications and additional qualification testing. For additional information, see Chapter 6, IUS Derivatives.

The IUS is capable of attaining many orbits of interest. Figure 3-4 presents the payload capability to low altitude circular orbits of 150 to 11,000 nautical miles, with IUS plane change varying from 10 to 40 degrees.

Payload capability data to 4, 6, 8 and 12-hour

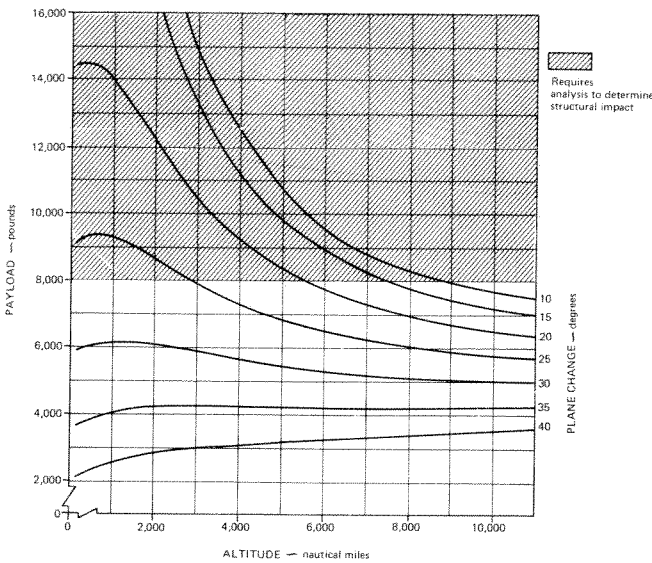


Figure 3-4: Low Altitude Payload Circular Orbit Capability

period circular orbits in Figure 3-5 is plotted as a function of final orbit inclination from 0° to 90°. The orbiter inclination range is from 28.5 to 57 degrees.

The payload capability to circular orbits of two and three times the geosynchronous radius is shown in Figure 3-6.

Figure 3-7 contains the IUS capability for a 24-hour period near polar orbit. These data apply to an Eastern Test Range orbiter launch at a 57° inclination.

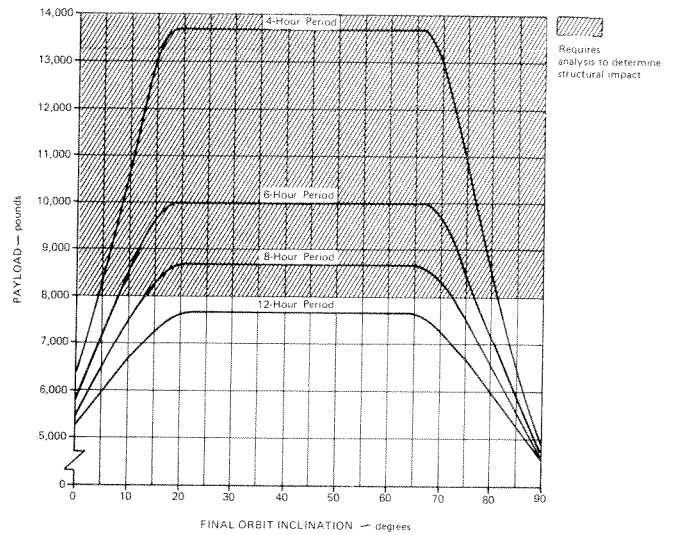


Figure 3-5: Payload Capability to Circular Orbits

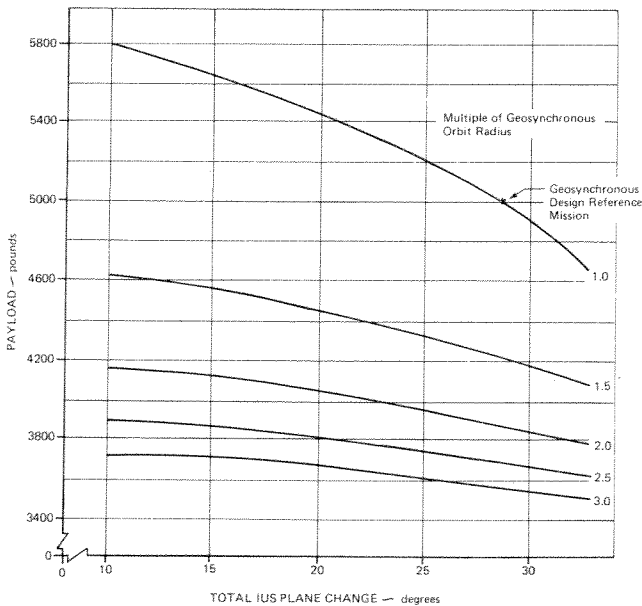


Figure 3-6: Payload Capability to High Altitudes

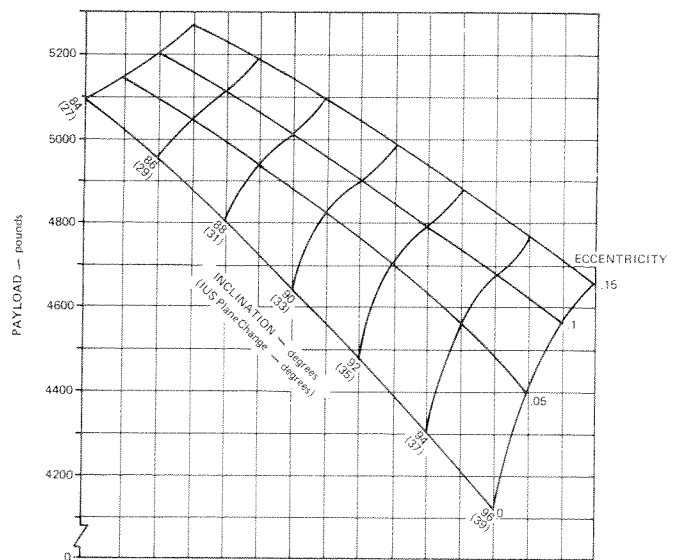


Figure 3-7: Payload Capability to 24-Hour Period Near-Polar Orbits

The data in Figure 3-8 can be used to determine the payload for orbits not shown in the previous figures. For detailed performance calculations, please contact the Boeing Aerospace Company.

Maximum payload capability is generally obtained by using Hohmann transfer trajec-

tories. This results in specific longitudes at mission orbit injection, with spacings of  $22.5^\circ$  between successive parking orbit departure opportunities. Extending the parking orbit time 12 hours and departing from either ascending or descending nodes reduces this spacing to 11.25 degrees.

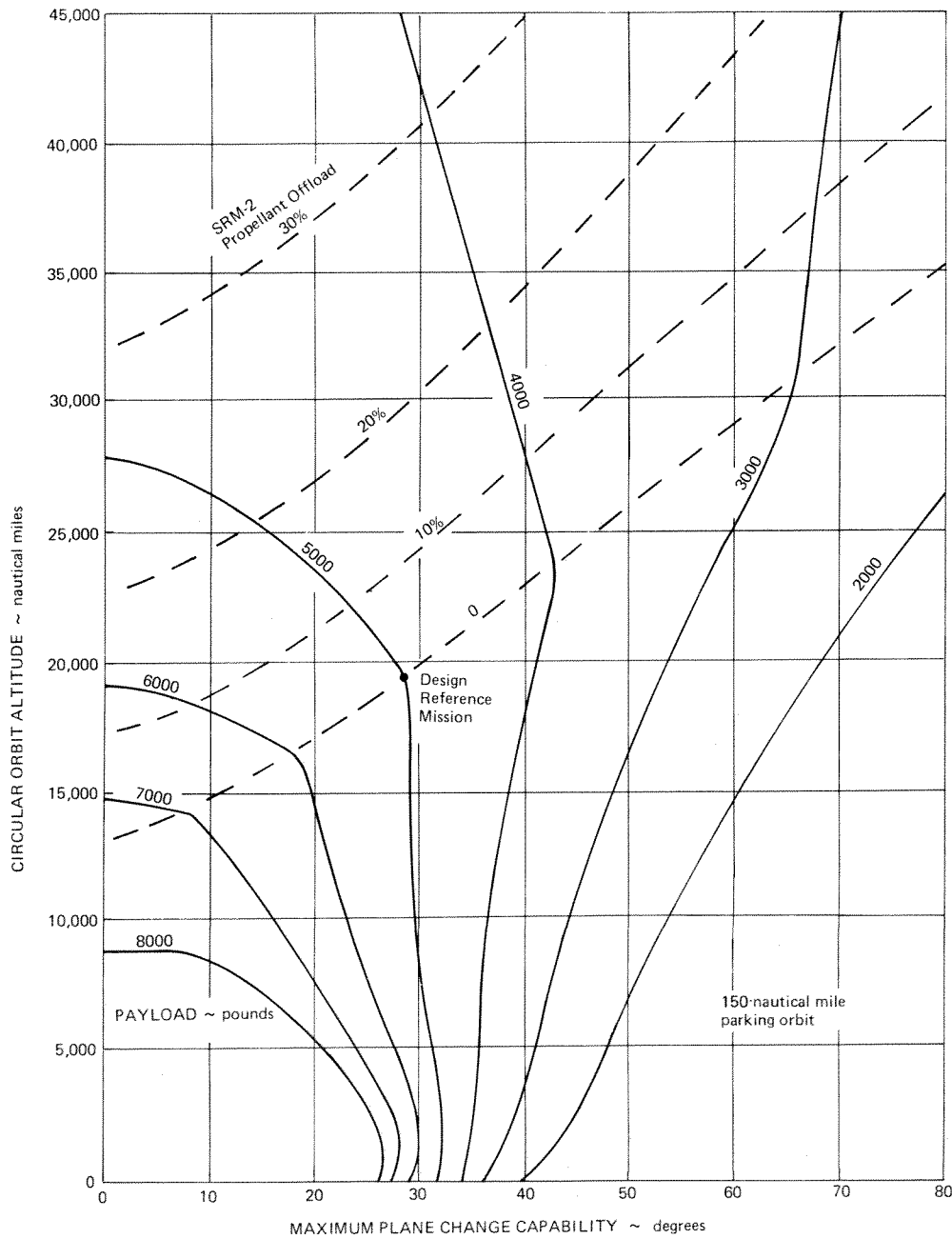


Figure 3-8: Earth Orbit Performance Capability

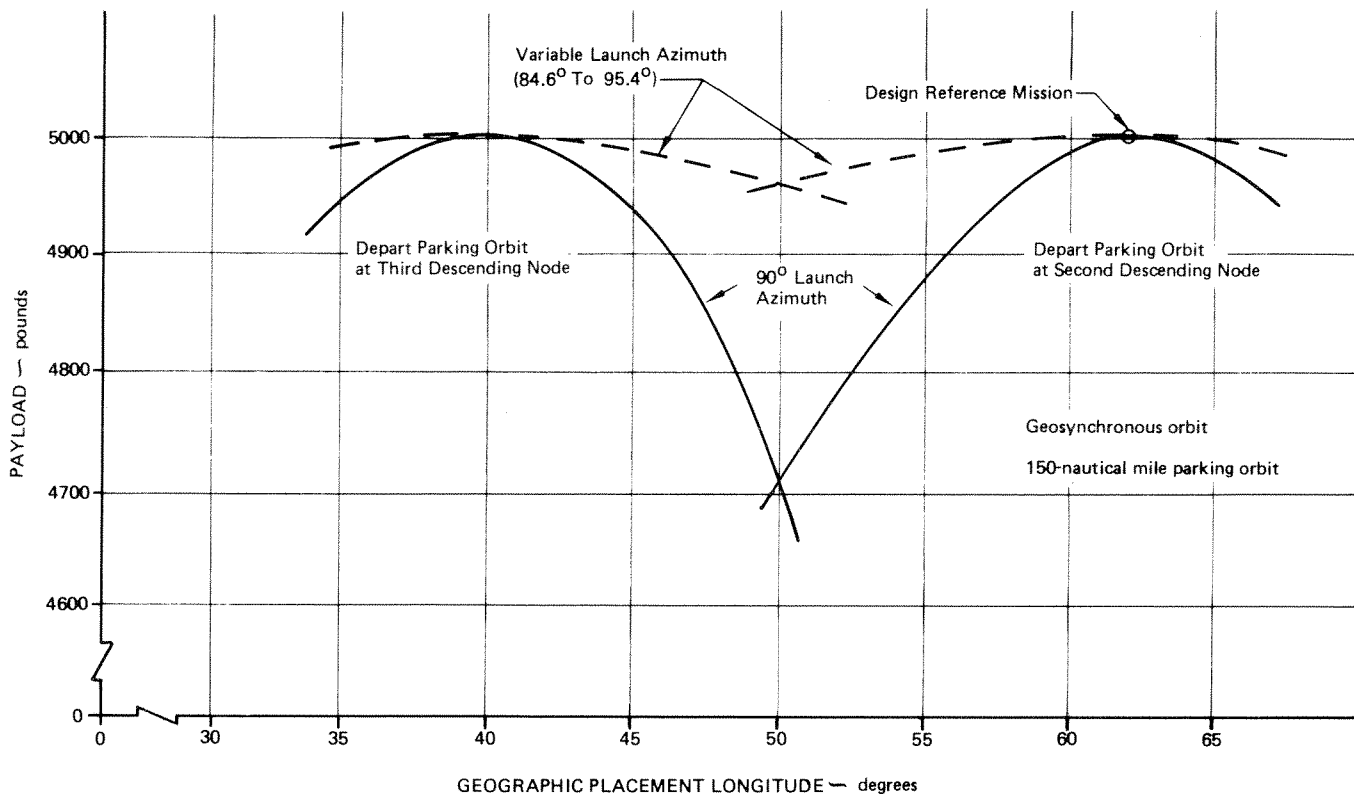


Figure 3-9. Effect of Longitude Constraint on Geosynchronous Payload Capability

Any longitude can be obtained with payload penalty as shown in Figure 3-9. Two methods are possible. A small variation in launch azimuth (from 84.6 to 95.4°) varies the nodal crossing of the parking orbit by a total of 23 degrees. The transfer trajectory is still Hohmann, but from a slightly greater parking orbit inclination, with a maximum payload penalty of 38 pounds. If the launch azimuth is constrained to 90°, then non-Hohmann transfers are used to adjust longitude. The payload penalty is greater in this case: up to 290 pounds.

The payload capability to 12-hour period elliptical orbits is plotted as a function of orbiter parking orbit inclination in Figure 3-10. The design reference mission for the IUS uses a 46.7° parking orbit inclination.

For some geosynchronous missions, longitude drift rate is required at mission orbit injection. Figure 3-11 shows the effect of drift rates on payload capability from 10°/day westerly to 10°/day easterly for both elliptical and circular drift orbits. When the elliptical drift orbit is chosen, the payload delivery occurs at apogee of the drift orbit.

This results in a performance gain relative to a circular drift orbit because plane change is more easily made at the higher altitude, i.e., lower velocity. A single velocity maneuver will arrest the drift and yield a circular orbit because one of the apsides of the elliptical drift orbit is at the geosynchronous orbit altitude.

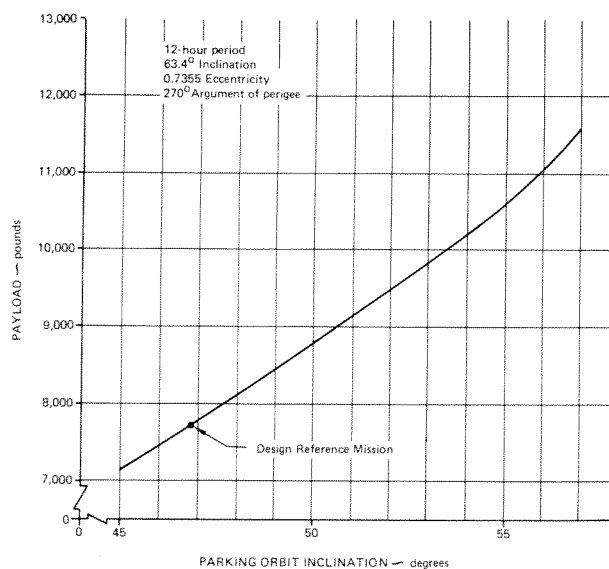
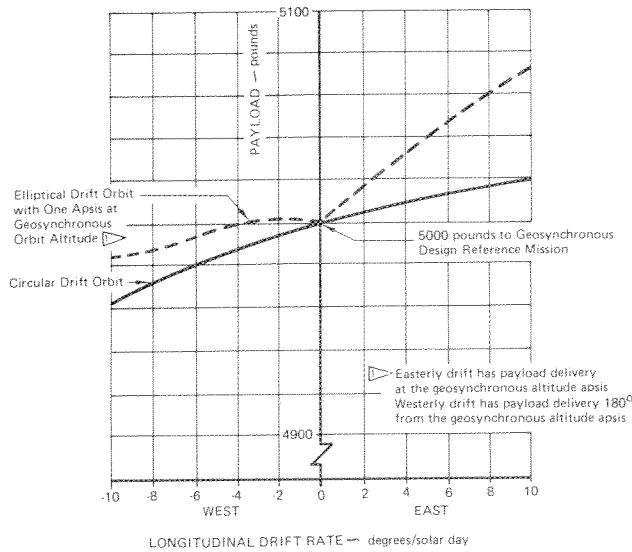


Figure 3-10: 12-Hour Period Elliptical Orbit Payload Capability



**Figure 3-11: Effect of Drift Rate on Geosynchronous Payload Capability**

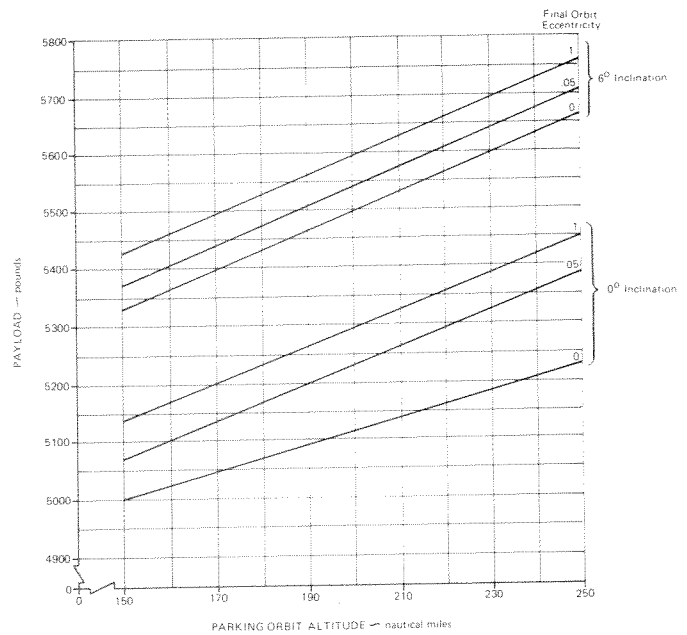
The payload capabilities discussed previously are based on a 150-nautical mile circular parking orbit. The use of higher circular parking orbits or elliptical parking orbits will enhance payload capability (Figures 3-12 and 3-13).

### Planetary Payload Capability

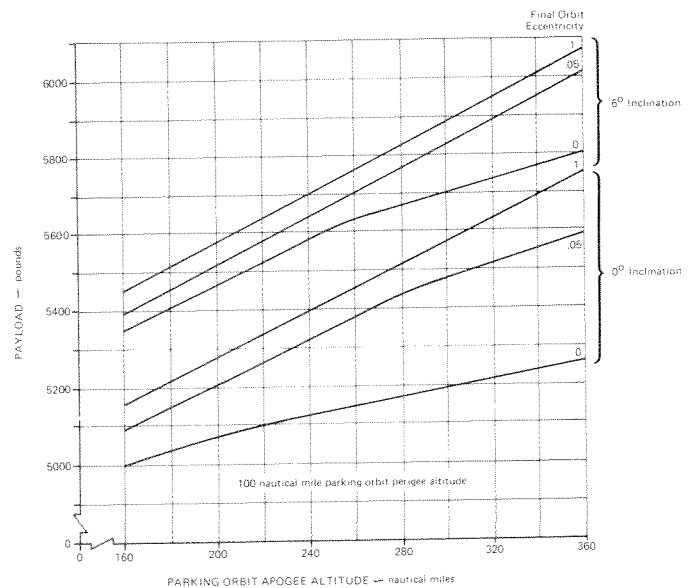
Planetary capability of the two-stage IUS is plotted as a function of earth escape energy ( $C_3$ ) in Figure 3-14. Although the IUS is designed for earth orbital use, it is applicable to low-energy earth departure missions. For increased planetary payloads or higher energy missions, the performance can be augmented by incorporating a third stage or by designing a spacecraft that has integral propulsion. Figure 3-14 also shows the performance when a three-axis-controlled kick stage is provided as a third stage to the IUS.

### Mass Properties

Flight-sequenced weights are provided in Table 3-2. They are based on "IUS Mass Properties Status Report Number 70" (MPSR 70), dated May 1, 1983. The mass properties of the IUS baseline vehicle provide the user with a basis for calculations of Shuttle-user charges.



**Figure 3-12: Effect of Higher Circular Parking Orbit on Geosynchronous Payload Capability**



**Figure 3-13: Effect of Higher Elliptical Parking Orbit on Geosynchronous Payload Capability**

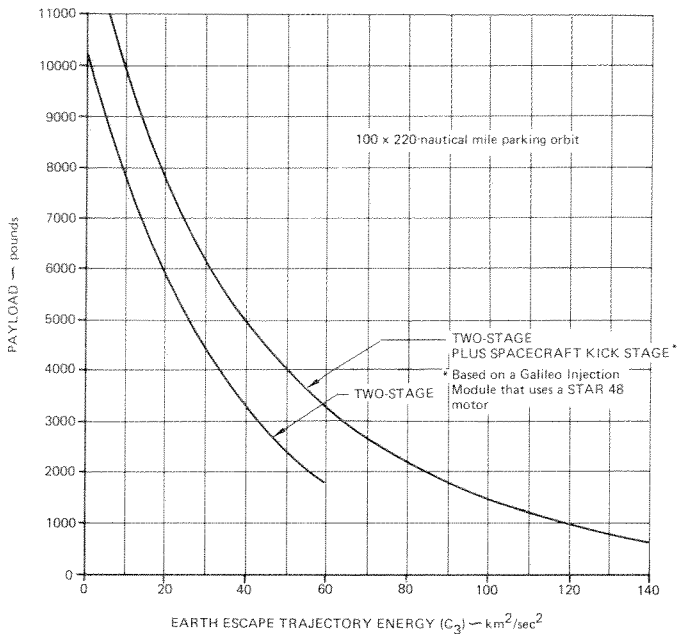
### Guidance Accuracy

The requirements and present injection error estimates for the STS geosynchronous mission

are shown in Table 3-3. These data show that the predicted accuracies are well within the specified requirements. This kind of accuracy results in a reduction in spacecraft correction requirements which translate directly into spacecraft growth or additional spacecraft life.

**Table 3-3: Predicted Accuracies**

Parameter	Requirement	Predicted
Position (n. mi.):		
Tangential (T)	66.	36.
Normal (N)	40.	18.5
Radial (R)	50.	28.4
RSS	92.	49.7
Velocity (ft/sec)		
Tangential (T)	16.	6.1
Normal (N)	12.	11.7
Radial (R)	75.	32.1
RSS	78.	34.7



**Figure 3-14: Planetary Payload Capability**

**Table 3-2: STS/Two-Stage IUS Sequential Weights**

Item	Item Weight (pounds)	Sequenced Weight (pounds)
Payload		5000
Empty Stage 2 at Burn-Out	2228.1	
RCS Propellant (Reserves & Trapped)	62.5	
RCS Propellant (Attitude Control & CAM)	7.0	
RCS Propellant to Augment SRM-2 Burn	94.5	
Stage 2 Burnout		7392.1
SRM-2 Propellant	6060.8	
Expendable Inerts	25	
Stage 2 Ignition		13,477.9
Empty Stage 1 at Burn-Out	2476.4	
RCS Propellant (Attitude Control)	33.9	
RCS Propellant to Augment SRM-1 Burn	39.7	
Stage 1 Burnout		16027.9
SRM-1 Propellant	21,403.8	
Expendable Inerts	91	
Stage 1 Ignition		37,522.7
RCS Propellant (Attitude Control)	10.2	
Total IUS + Payload		37,532.9
Boeing-Furnished ASE	5868.3	
Government-Furnished ASE	1504	
Spacecraft ASE	750	
Shuttle Cargo		45,655.2

\*Boeing Document D290-10048-70, dated 1-May-1983

**ENVIRONMENTS**

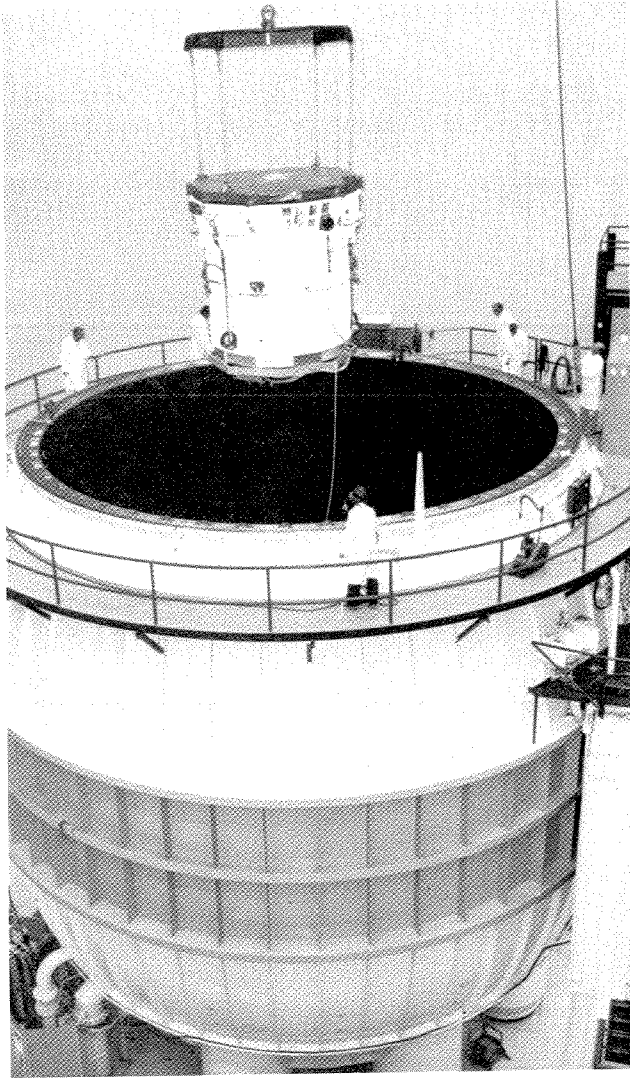
The spacecraft is subjected to the following environments: thermal, acoustic, vibration, shock, dynamic loads and pressure.

**Thermal**

Orbiter cargo bay and deep space environments combine to produce the external thermal environments. Orbiter cargo bay environmental conditions are predicted by Johnson Space Center thermal math models. Through the use of these models, the spacecraft designer can exercise some control over these environments from prelaunch through deployment.

The thermal analyses for the IUS/spacecraft interface are based on the mission timeline. Significant features of the different phases of operation are defined below. Specific spacecraft requirements will be identified in the spacecraft ICD.

**Prelaunch** - During the prelaunch period, while the IUS/spacecraft/ASE combination is in the orbiter cargo bay, the bay is purged with either air or nitrogen.



*IUS Thermal Vacuum Test*

The purge gas is thermally conditioned to the temperature selected by the spacecraft designer. The available range is from 45 to 80 degrees (F). Once this temperature is known, the cargo bay wall temperatures and heat transfer coefficients can be determined by the Orbiter Atmospheric Thermal Math Model. Typically, the wall temperatures are 10° to 20° hotter or colder than the purge gas temperature, depending on hot or cold day analyses. The convection heat transfer coefficients are about 0.5 Btu/ft<sup>2</sup>-hr-°F.

**Ascent** - At liftoff, the purge is discontinued, and the cargo bay then vents to the atmosphere.

During this period, the reduced pressure and convective heating have been accounted for

by the Orbiter Atmospheric Thermal Math Model. The model has also predicted the cargo bay wall temperatures by considering the effects of aerodynamic heating.

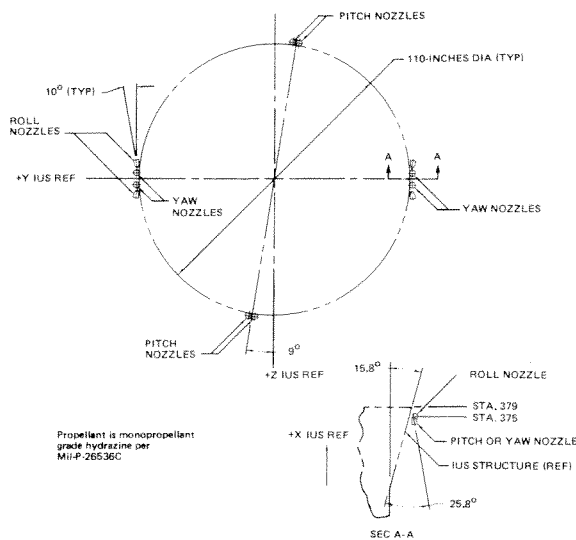
Nominal payload bay door opening time is one hour after liftoff, but off-nominal conditions can extend this time. If door opening is not completed within three hours, the mission is aborted.

**On-Orbit** - Thermal considerations require a certain orbiter orientation while on-orbit with the payload bay doors open. The spacecraft requirements are specified in the spacecraft-unique ICD. IUS has some constraints which are identified in ICD-D-E0001. The orbiter also has constraints which are dependent on many considerations. Final orbiter attitude history will be determined from the integrated thermal analyses, which considers all of the mission-peculiar requirements.

**Post-Deployment** - After deployment, during the free-flight period, the IUS is fully powered and has no thermal control orientation constraints during the first 10 hours of flight time. Missions longer than 10 hours may require thermal control maneuvers. The spacecraft designer can select preprogrammed maneuvers to satisfy spacecraft requirements. For more information on thermal control maneuvers, see Spacecraft Services section, which follows Environments.

SRM-1 and SRM-2 burns produce solid particle exhaust products. The products form a cone-shaped plane (a nominal half angle of 18.5° for SRM-2 and 20° for SRM-1) which is several hundred feet high. The particle planes are a source of radiation to the exposed low mass elements of the spacecraft. This may result in a significant additional heat source for the spacecraft.

The IUS reaction control subsystem engines are mounted near the IUS/spacecraft interface, Figure 3-15. The system uses hydrazine as propellant, and hence has exhaust products of nitrogen, hydrogen, and ammonia. The pitch and yaw engines are aft facing and do not affect the spacecraft. The roll engines, however, face laterally which may result in



**Figure 3-15: IUS Reaction Control Subsystem Nozzle Location**

plume heating of the spacecraft. The roll engines primarily duty cycle for attitude control. This is done with .010 second pulses with a 0.01% or less duty cycle. Longer burns are used only for spacecraft-requested maneuvers (e.g., when establishing a continuous roll and during SRM operation). The duty cycle during SRM operation is less than 1%.

**Acoustic**

The maximum predicted acoustic environment within the orbiter payload bay is shown in Table 3-4. An empty payload bay is the defined environment for the limiting case. The sound pressure levels are overall spatial averages.

**Vibration**

The maximum local vibration at the IUS/spacecraft interface due to boost phase acoustic environments is shown in Figure 3-16. This environment is applicable to equipment items located near this interface.

**Shock**

Figure 3-17 shows the shock environment spectra produced by the IUS at the IUS/spacecraft interface. The pyrotechnic shock at the interface, produced by the spacecraft, will not exceed the values stated in Figure 3-18.

**Table 3-4: Shuttle Payload Bay Internal Acoustic Noise**

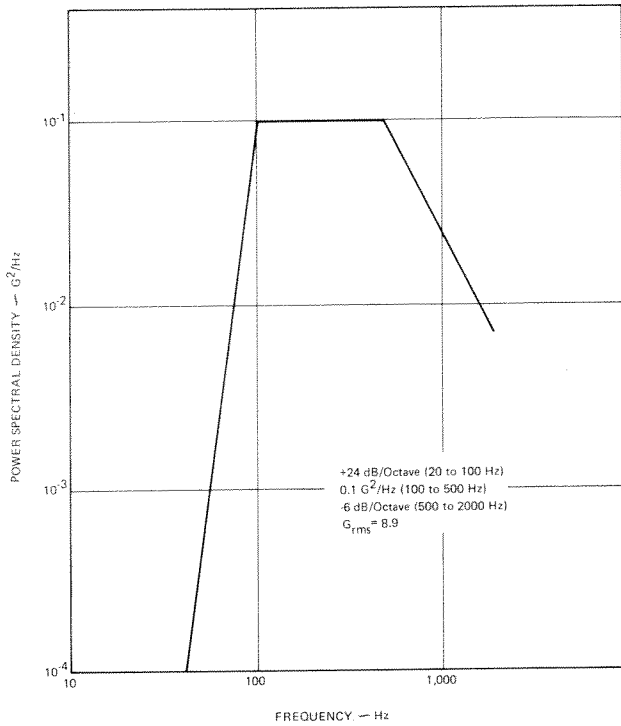
1/3 Octave Band Center Frequency (Hz)	Liftoff Sound Pressure Level (dB*)	Aeronoise Sound Pressure Level *(dB*)
31.5	122.0	112.0
40.0	124.0	114.0
50.0	126.0	116.0
63.0	127.5	118.0
80.0	129.5	120.0
100.0	130.5	122.0
125.0	132.0	124.0
160.0	133.0	125.0
200.0	133.5	126.5
250.0	134.0	127.0
320.0	134.5	127.0
400.0	134.5	127.0
500.0	134.0	127.0
630.0	133.5	125.0
800.0	133.0	123.0
1,000.0	132.0	119.0
1,250.0	131.5	118.0
1,600.0	130.0	117.0
2,000.0	129.0	115.0
2,500.0	128.0	113.0
3,200.0	126.5	111.0
4,000.0	125.0	109.0
5,000.0	124.0	107.0
6,300.0	122.5	105.0
8,000.0	121.0	102.0
10,000.0	120.0	101.0
Overall Level	145.0	137.0

(\*) Reference sound pressure =  $2.9 \times 10^{-9}$  psi  
 \*( ) Duration = 5 seconds per mission

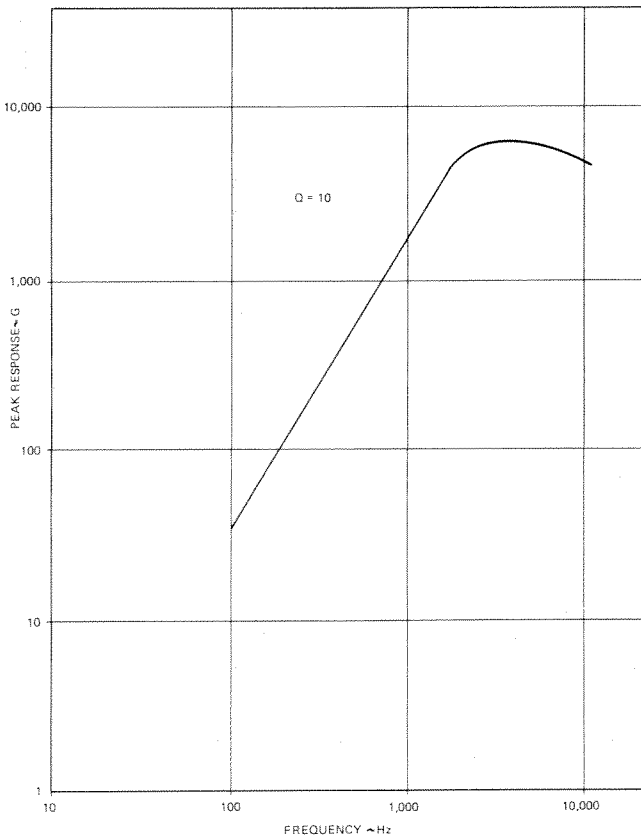
**Dynamic Loads**

NASA Safety Policy (June, 1976) and AFETRM 127-1 define the safety factors for any spacecraft flying in the Space Transportation System. In general, the spacecraft minimum factor of safety for structure is 1.4 for the flight phases where the spacecraft and IUS are attached to the orbiter. The spacecraft structural factor of safety, after deployment from the orbiter, but still attached to the IUS, is a minimum of 1.25.

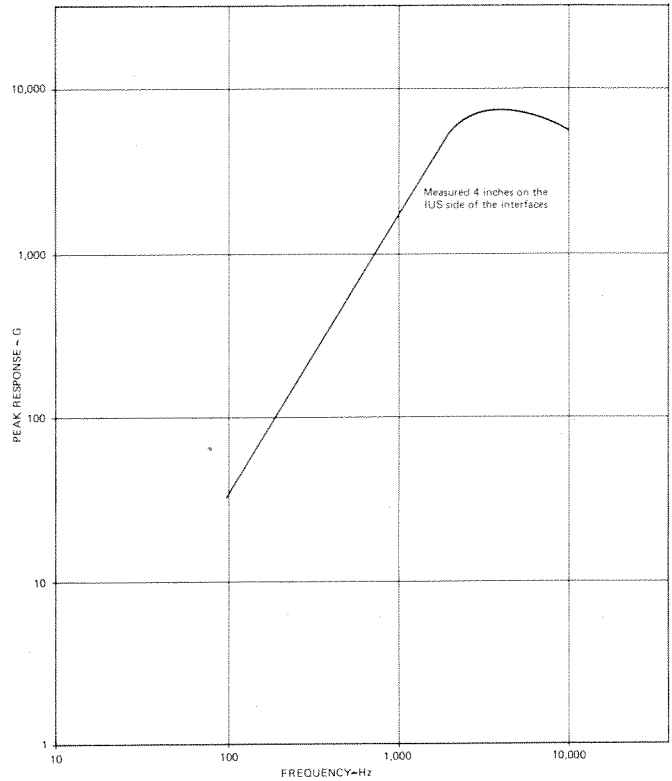




**Figure 3-16: Spacecraft Interface Maximum Vibration Environment**



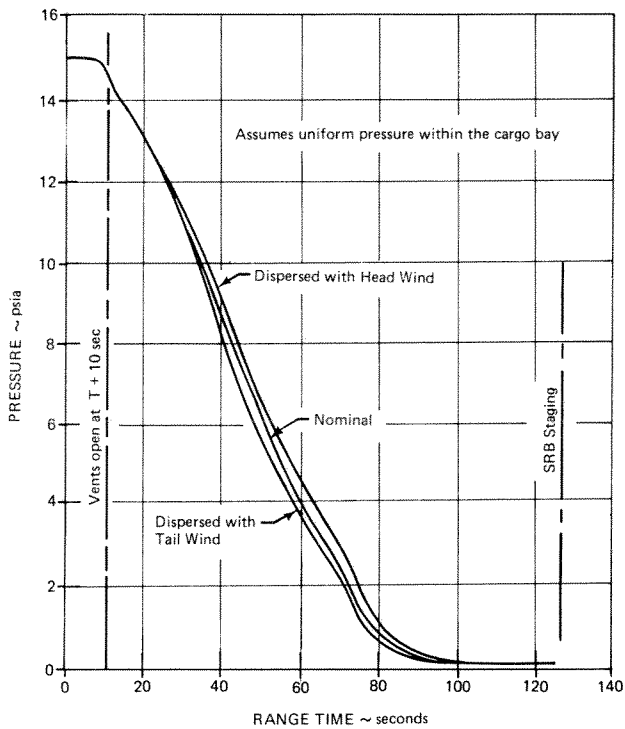
**Figure 3-17: Shock Response Spectra of the IUS/Spacecraft Interface**



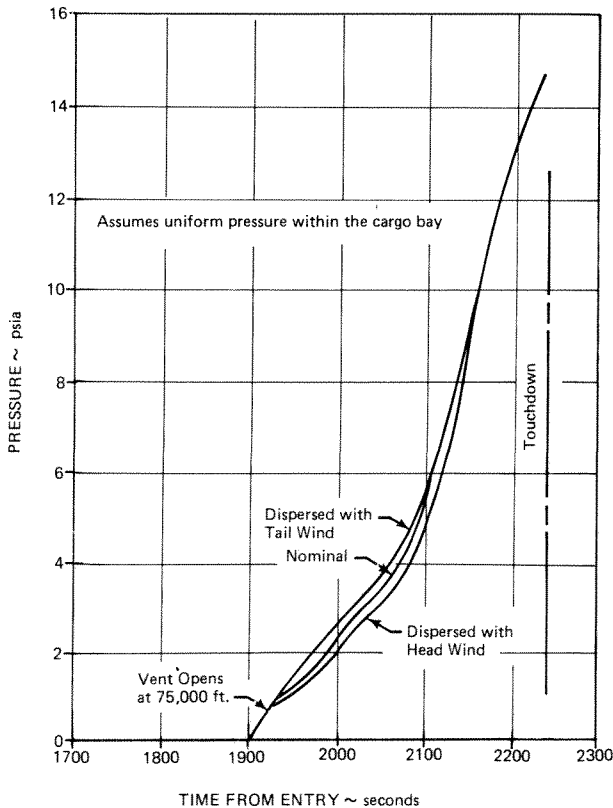
**Figure 3-18: Shock Environment Imposed by Spacecraft Disturbances**

Pressurized structures and containers have a safety factor of 2, while pressure lines and fittings have a safety factor of 4.

Design loads for the spacecraft in the IUS system are derived by static and dynamic analyses of the orbiter, IUS, and spacecraft. The Boeing Aerospace Company conducts the integrated dynamic analyses using a customer-provided spacecraft dynamic model. For the spacecraft conceptual design, the load factors presented in Table 2-2, Chapter 2, may be used. The lateral load factors are applicable to spacecraft that have cantilever frequencies of 7 Hz or greater. Lower frequency spacecraft may have significantly higher loads caused by dynamic interaction. Response accelerations of individual spacecraft components, with support frequencies below 20 Hz, may be increased substantially by local resonances and must be established by detailed dynamic analysis. The IUS interface has been designed to accommodate the limits of spacecraft weights, centers of gravity, and the load factors of Table 2-2. It has also been tested to the limits defined in Figure 2-6.



**Figure 3-19: Orbiter Payload Bay Internal Pressure History During Ascent**



**Figure 3-20: Orbiter Payload Bay Internal Pressure History During Entry**

## Pressure

Pressure histories for the orbiter payload bay are shown in Figures 3-19 and 3-20. These curves are based upon current payload bay vent opening times and assume uniform pressure distribution within the bay.

## SPACECRAFT SERVICES

Spacecraft services fall into three categories. These are: (1) generic IUS services which are available to all users at no additional cost, (2) spacecraft-unique production options which are qualified and can be added at extra cost and (3) special mission services which have been engineered and qualified for special applications and can be added without additional development or qualification cost. The following paragraphs are a discussion of these services.

It should be noted that any service provided through the IUS/spacecraft electrical interface connector must comply with the electro-magnetic compatibility (EMC) requirements of the Boeing IUS EMC Control Document (D290-10068-1) and the USAF IUS/Spacecraft Electrical and EMC Specification (SAMSO-LVR-0006). Primary power input to the IUS system from the launch vehicle or from support equipment shall be isolated from the power distribution to the spacecraft by at least 1 megohm dc resistance. When the IUS system provides a spacecraft primary power production option, the dedicated positive and negative power leads to the spacecraft shall be isolated from the IUS structure by at least 1 megohm dc resistance.

## Generic IUS Services

The standard or generic two-stage IUS provides certain spacecraft services. These include spacecraft electrical power and ordnance power availability, separation indication, telemetry transmission, discrete command initiation, orbiter/IUS/spacecraft status monitoring, and thermal control and telemetry dipout maneuver capabilities. Use of these services is at the discretion of the buyer.

**Spacecraft Electrical Power** –The IUS system supplies and/or distributes electrical power to the IUS/spacecraft interface connector for all mission phases from prelaunch to spacecraft separation. Figure 3-21 shows a typical mission profile for maximum and average power available to the spacecraft for each mission phase. Redundant switches are used to transfer spacecraft power input from among spacecraft ground support equipment (GSE), IUS ASE, and IUS vehicle battery sources.

During prelaunch checkout, spacecraft GSE power to the ASE power control unit is routed through the orbiter umbilical (T-0 umbilical) to either the spacecraft dc-to-dc converter-regulator or to the spacecraft umbilical as specified in the spacecraft-unique ICD. In addition, dedicated spacecraft power circuits are provided to directly supply spacecraft GSE power to the spacecraft umbilical (see Figure 3-22). Spacecraft power routed through the orbiter umbilical shall be current-limited within the spacecraft GSE to 22 amperes maximum per conductor pair at the umbilical. The IUS system power transfer switching is controlled by commands from the IUS GSE to the IUS ASE power control unit. Switching capability is provided in the ASE to select ground power or orbiter power input. Position status of this transfer

switch is monitored by the ASE power control panel. The spacecraft GSE shall provide an interlock circuit which interfaces with the IUS GSE so that the spacecraft GSE can inhibit or enable the IUS GSE commands that control the prelaunch spacecraft ground power transfer between the spacecraft GSE and orbiter power.

During ascent and orbiter on-orbit phases, the IUS system distributes orbiter power to the spacecraft. The IUS ASE provides redundant batteries so that uninterrupted power is supplied to the spacecraft in case orbiter power is interrupted. The IUS ASE also provides a dc-to-dc converter-regulator which isolates the orbiter power return from the spacecraft power return, provides a pre-selected output voltage between 24 and 36 vdc, and regulates the selected voltage output to  $\pm 0.3$  volts at the IUS/spacecraft connector.

Dedicated spacecraft batteries in the IUS first and second stages supply uninterrupted unregulated power to the IUS/spacecraft connector during IUS free flight. The voltage at the connector is  $28 \pm 4$  vdc with a peak voltage of 38 vdc. The IUS/spacecraft power transfer switching is shown in Figure 3-23. Power transfer switching is provided when spacecraft power is transferred from the ASE

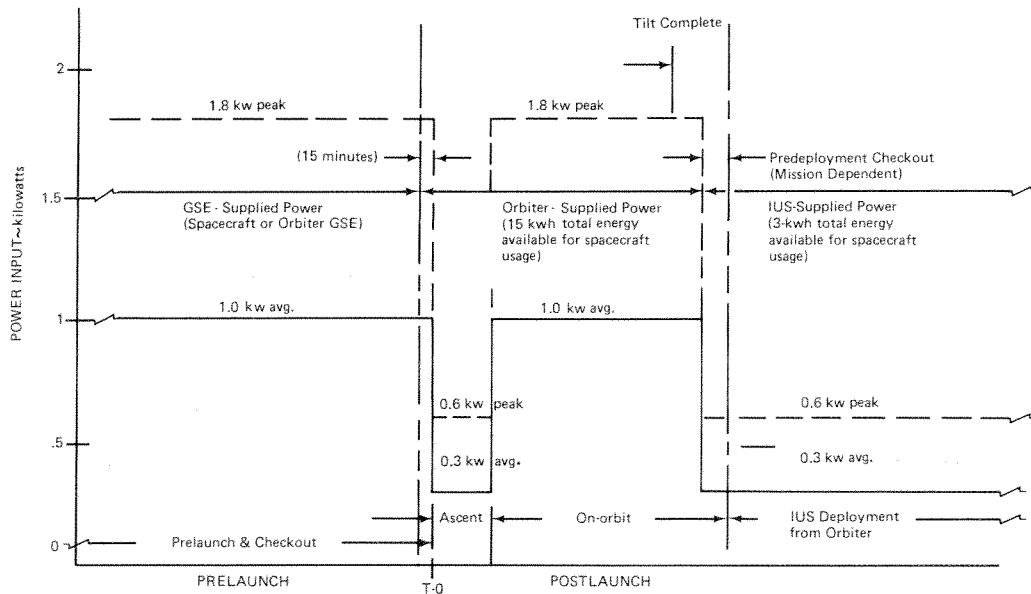
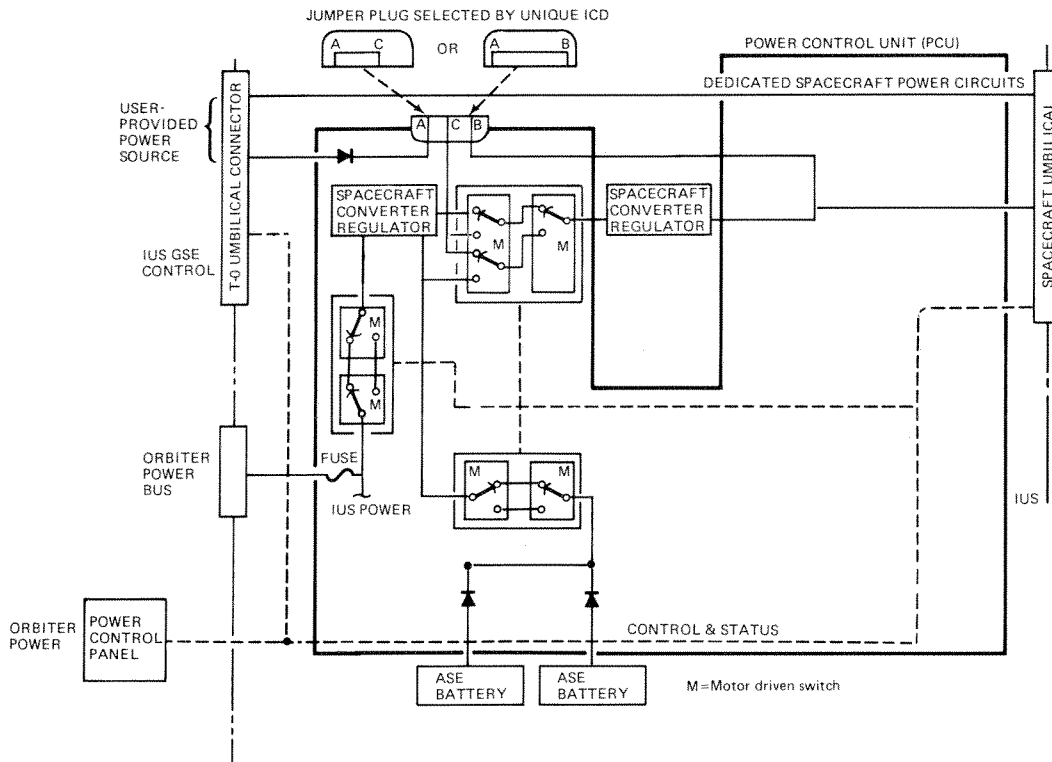
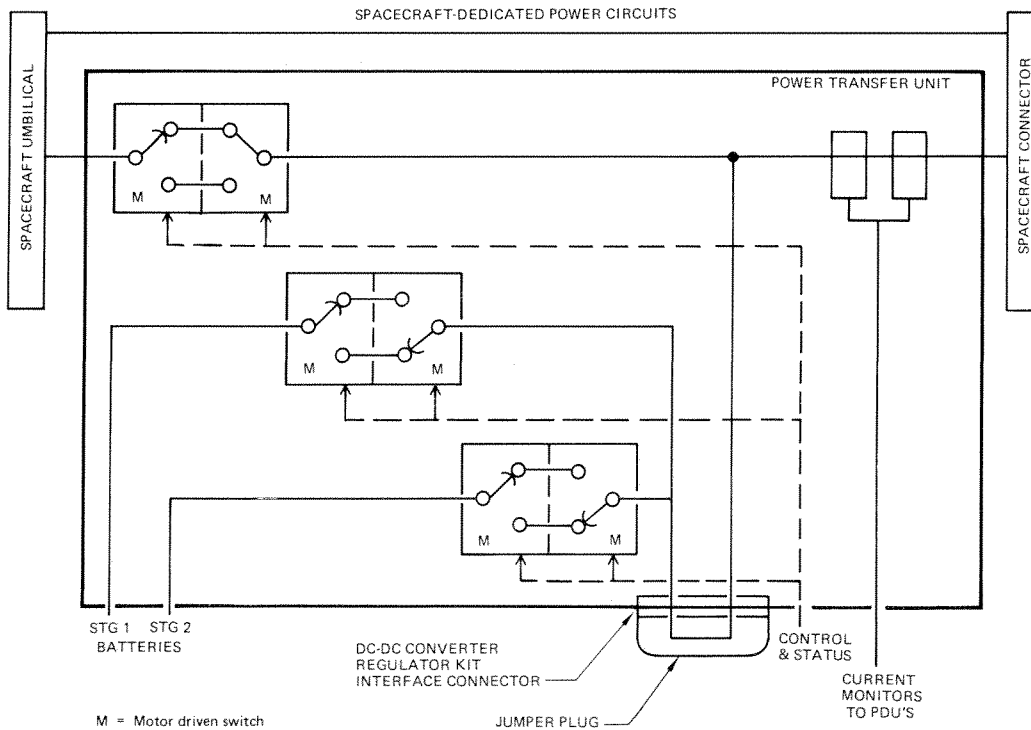


Figure 3-21: Spacecraft Power and Energy Profile



**Figure 3-22: Airborne Support Equipment Power Switching**



**Figure 3-23: Spacecraft Power Transfer Switching**

power input to the IUS first-stage dedicated spacecraft battery prior to IUS deployment, and when power from the IUS first stage spacecraft battery is transferred to the IUS second stage spacecraft battery prior to IUS stage separation. Position status of the stage 1-to-stage 2 spacecraft battery transfer switches is monitored by IUS telemetry.

Battery thermal control is designed to accommodate the specified 300-watt average spacecraft demand. The specific ICD shall identify any spacecraft average demand of less than 300 watts in order to establish if special provisions for shunts or external battery heaters are required to sustain battery temperatures during cold in-flight environments.

The IUS is required to downlink telemetry verification that 28 vdc power was delivered to the spacecraft. The spacecraft, therefore, is required to provide two divider networks of the configuration shown in Figure 3-24 on its side of the IUS/spacecraft connector.

While on-orbit in the orbiter bay, it may be necessary, under certain conditions (e.g., cargo bay doors closed after T + 1 hour, extended mission duration, etc.), for the IUS to enter a powered-down mode in order to keep the critical temperatures of the IUS equipment within allowable limits or to conserve the available orbiter electrical energy. In such a case, the spacecraft services supplied by the IUS will be reduced as shown in Table 3-5. Full power will be available for a health and status check of the IUS and the

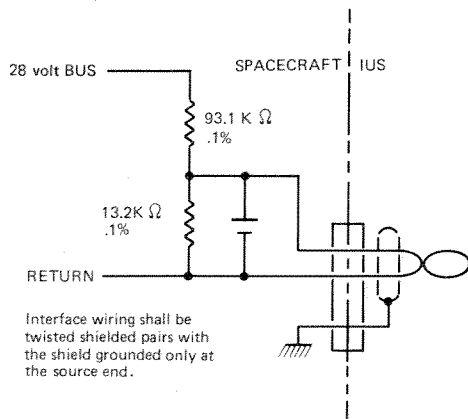


Figure 3-24: Spacecraft Bus Voltage Monitoring Divider Networks

Table 3-5: Implications of Powered-Down Mode

<u>Services to Spacecraft</u>	<u>IUS Powered-Up</u>	<u>IUS Powered-Down</u>
Spacecraft Power	Yes	Yes*
Spacecraft Data to ASE Wide Band Data Interleaver	Yes	Yes
Spacecraft Data Interleaved with IUS Data in IUS Vehicle	Yes	No
Discretes Issued by IUS	Yes	No
Discretes Initiated from Ground or Orbiter Thru IUS	Yes	No

\*To limits of orbiter energy capability

spacecraft for a short time interval after attaining orbit, and for a period of time prior to deployment from the orbiter.

**Ordnance Power** –The IUS provides and distributes ordnance power to the IUS/spacecraft interface for firing spacecraft ordnance devices in two groups of eight initiators. Interfacing ordnance circuits shall meet the requirements of the Range Safety Manual AFETRM 127-1, Volume 1, MIL-STD-1512 and D290-10068-1.

The IUS ordnance firing current supplied to the IUS/spacecraft interface for each spacecraft initiator is 5 amperes dc minimum, and the short circuit current is 11 amperes maximum. The minimum firing pulse duration is 40 milliseconds. The IUS provides the safe shorting and electrostatic protection for each IUS/spacecraft ordnance circuit interface.

The spacecraft ordnance circuit resistance measured at the IUS/spacecraft interface shall be 1.5 ohms maximum including the initiator bridgewire. The spacecraft separation ordnance initiators shall have the following characteristics:

- Resistance: 1 ohm nominal
- No fire: 1 ampere - 1.0 watt for 5 minutes
- All fire: 4.5 amperes dc maximum

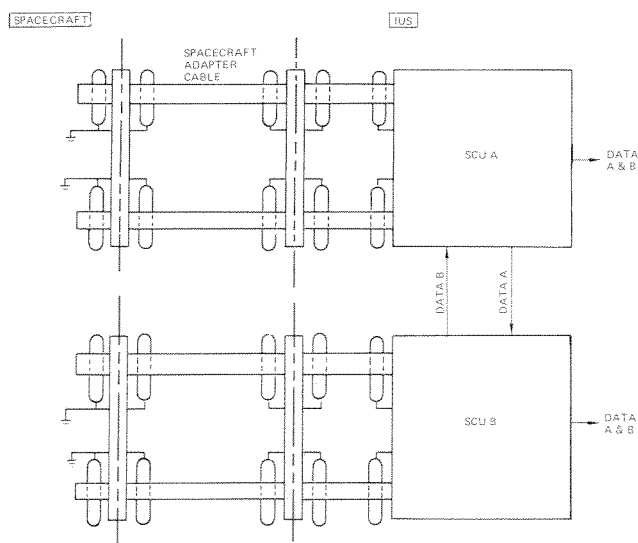
The IUS provides the current to fire the primary and backup spacecraft ordnance devices within five milliseconds of each other. The IUS ordnance control circuits are cross-strapped so that the "fire" impulse from either the A or B side will apply the fire current to both the primary and backup ordnance devices from their respective ordnance bus power sources.

The initial command to fire both groups of initiators is provided at the preselected separation time  $\pm 1$  second. A secondary firing command is provided 80 milliseconds  $\pm 10$  milliseconds after the initiation of the initial fire command.

**Separation** –The IUS provides the capability to monitor up to four separation switches or breakwires provided by the spacecraft. Figure 3-25 shows the four circuits and Figure 3-26 shows circuit details for the separation switches or breakwire alternative. Each circuit (signal and return) is monitored by the IUS telemetry system to verify spacecraft separation. Separation monitors are used only to verify spacecraft separation.

The IUS system provides the excitation voltage to the IUS/spacecraft connector interface for spacecraft separation monitor circuits having the following characteristics:

- Voltage Range: 4 to 6 vdc
- Current: 1 to 5 ma



**Figure 3-25: Spacecraft Separation Monitor Circuits**

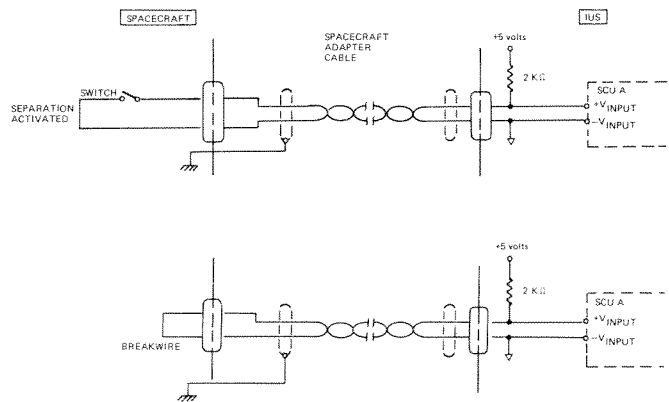
Switch or Breakwire Isolation:  
 Referenced end circuit on the IUS side  
 Isolated end circuit on the spacecraft side

**Telemetry** –The IUS has the capability to accept unencrypted spacecraft telemetry data and to interleave these data with the IUS telemetry data and transmit it to earth via the IUS S-band telemetry, tracking and command system. Spacecraft telemetry data may also be transmitted by the spacecraft directly to the ground when the IUS is in free flight.

The IUS vehicle provides the telemetry rates shown below:

Altitude	Data Rate	Bit Error Rate	Margin
19,300 nmi	16 kbps	$10^{-6}$	6 dB
19,300 nmi	64 kbps	$10^{-6}$	3 dB

These data rates are achieved with directional antennas and preferential orientation of the IUS vehicle. During SRM-2 burn at geosynchronous injection, an antenna orientation will be specified in the specific ICD to provide these rates and margins to a space ground link system (SGLS) ground station.



**Figure 3-26: Spacecraft Separation Monitors**

At the 64-kbps data rate, the IUS can accept up to 4 kbps of asynchronous unencrypted data from each spacecraft. At the 16-kbps data rate, the IUS can accept up to 1 kbps from each spacecraft. Up to four spacecraft may be accommodated by each IUS. In a single spacecraft mission, the spacecraft may use more than one port.

To assure no loss of data due to IUS/spacecraft clock tolerances, the spacecraft data rate should be no greater than 3999.5 Hz and 999.8 Hz, respectively, for the 4 kbps and 1 kbps conditions.

The redundant IUS/spacecraft serial data interface is shown in Figure 3-27. The spacecraft data and clock circuit interface and the data and clock relationship are shown in Figures 3-28 and 3-29, respectively.

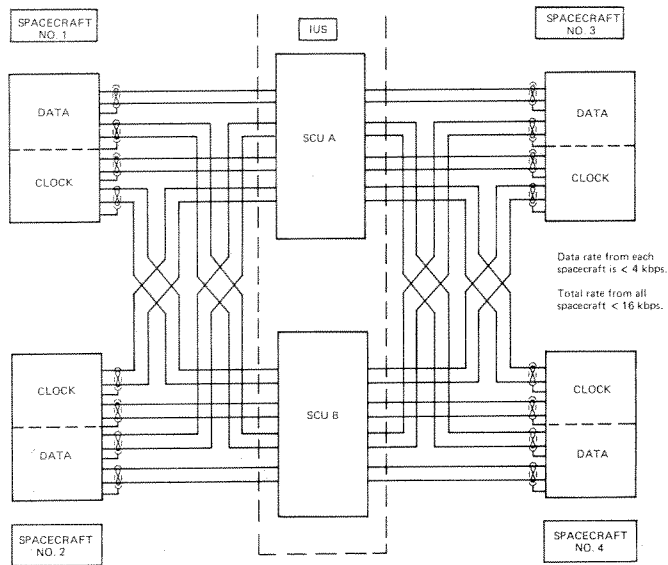
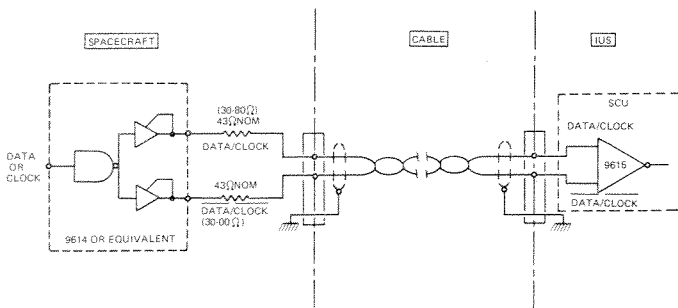


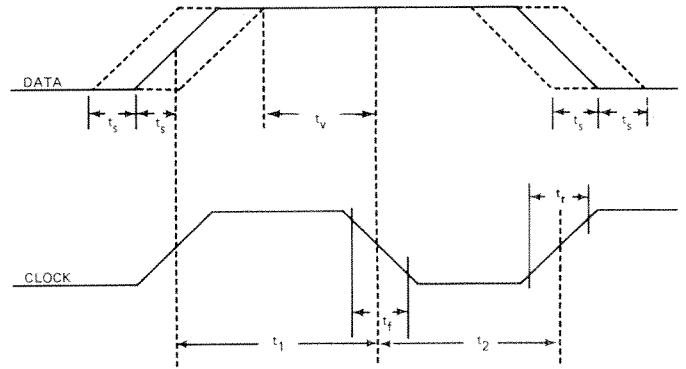
Figure 3-27: Spacecraft Telemetry Interface

**Spacecraft Commands** –The IUS has the capability to send a maximum of eight primary and eight backup discrete commands to the spacecraft for operation of solenoids, relays or switching devices.



The nomenclature for defining the signal levels (DATA and DATA) follows positive true logic.

Figure 3-28: Digital Line Receiver Interface for Telemetry Data or Clock



$$t_1 + t_2 \geq \frac{1.0}{3999.5} \text{ @ 64 kbps, IUS Telemetry Rate}$$

$$t_1 + t_2 \geq \frac{1.0}{999.8} \text{ @ 16 kbps, IUS Telemetry Rate}$$

$$\frac{t_1}{t_2} = 1.0 \pm 0.1$$

$$t_s = \text{Data-to-clock skew} \leq 50 \text{ microseconds}$$

Measured at 50% voltage levels

$$t_r \leq 10 \text{ microseconds}$$

$$t_f \leq 10 \text{ microseconds}$$

Measured between 10 and 90% voltage levels

The IUS SCU strobes data in on the falling edge of the clock.  
 $t_v > 50 \text{ microseconds}$

Figure 3-29: Spacecraft-to-SCU Data and Clock Relationship

The discrete commands are generated in the IUS computer either as an event-scheduling function (part of normal on-board automatic sequencing) or a command-processing function initiated from an uplink command. The IUS has the capability to receive uplink commands from the Air Force Satellite Control Facilities to alter the on-board event sequencing function. This permits the discrete commands to be issued at any time in the mission. Discrete command duration is a preprogrammed interval for each mission and is not alterable by external command. Uplink commands or programming of the normal on-board sequencing function permits any of the eight redundant discrete commands to be issued more than once.

**Orbiter/IUS/Spacecraft Status Monitors** –

Each IUS provides dedicated hardwires from the spacecraft through the IUS to an orbiter multiplexer-demultiplexer for subsequent display on the orbiter cathode ray tube. Parameters are displayed which require observation and which may require correction by the orbiter crew. This capability is provided until IUS/ASE umbilical separation.

**Thermal Control Maneuvers** –The IUS can perform thermal control maneuvers to reduce and control spacecraft thermal loads. Maneuvers for which the control logic has

been developed and is available include toasting, metronome, rotisserie, and continuous roll. These maneuvers are performed during the intervals spent in park and transfer orbits.

A toasting maneuver consists of periodic  $180^\circ$  rotations about the yaw axis, reversing direction from a limit position typically at 1-hour intervals (Figure 3-30).

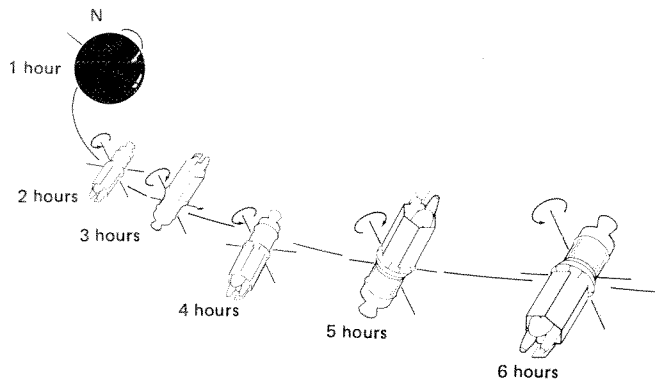


Figure 3-30: Typical Toasting Maneuver

A metronome maneuver consists of a periodic  $70^\circ$  rotation about the yaw axis, with a direction reversal typically at 1-hour intervals (Figure 3-31).

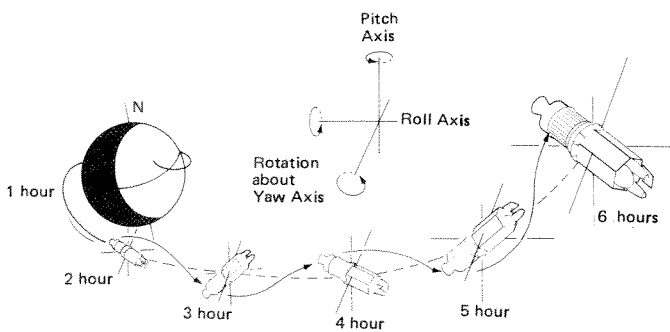


Figure 3-31: Typical Metronome Maneuver

A rotisserie maneuver, shown in Figure 3-32 consists of periodic  $230^\circ$  rotations about the roll axis, with direction reversal typically at 6-minute intervals after 1-minute dwells at limit positions.

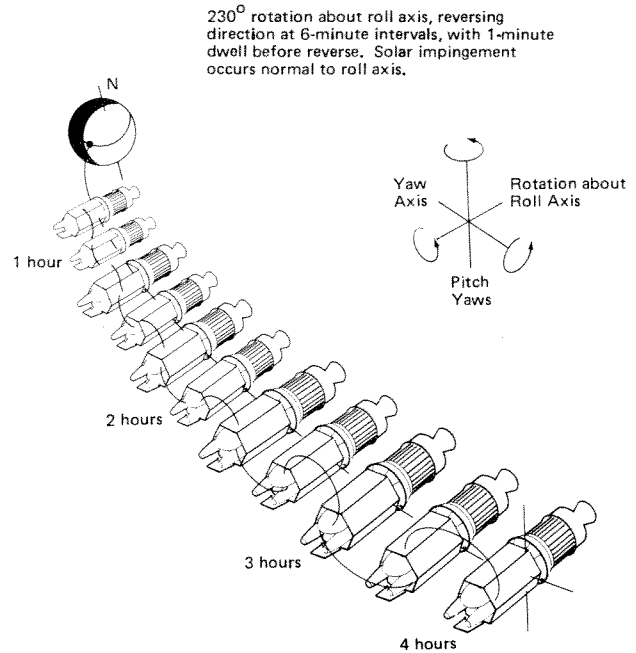


Figure 3-32: Typical Rotisserie Maneuver

A continuous roll maneuver, shown in Figure 3-33 consists of rolling about the roll axis at typical rates of  $0.6^\circ/\text{sec}$ . The maximum rate is  $6^\circ/\text{second}$ . To protect navigational accuracy, the roll direction is reversed in even increments. This maneuver may be em-

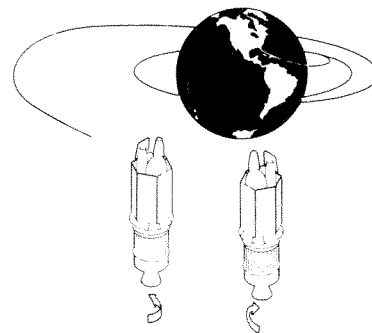


Figure 3-33: Typical Continuous Roll Maneuver



ployed at any time during a coast period and is preferred from the standpoint of having no time limitation.

**Telemetry Dipout Maneuver** —A telemetry dipout maneuver typically consists of a completely general reorientation in pitch, roll, and yaw to satisfy telemetry orientation requirements of the spacecraft. Telemetry data are transmitted continuously during all phases of flight by the IUS. Special IUS telemetry transmission maneuvers are only required during SRM-2 burn when the high data rate is employed.

### **Spacecraft-Unique Production Options**

The IUS vehicle has provisions for including certain production options to modify or augment its basic performance. The hardware components and associated software for the production options described in this section have been qualified in the full scale development program.

Additional design integration tasks may be required when a production option is added to a specific mission. These tasks consider the effects of the addition of production options and the impact of the specific mission profile on battery and reaction control propellant capacity. Capacities required above the generic capability are chargeable to the spacecraft. Design integration tasks also include mass properties analyses and equipment installation adjustments, if required, to meet center-of-gravity constraints.

Performance impacts (including weights) caused by the addition of these production options are chargeable to the spacecraft. Unless otherwise specified, the production options include wiring for power, control and instrumentation.

The spacecraft-unique production options for the Shuttle version of the IUS include the addition of dc-to-dc power conversion-regulation, increased reaction control propellant tankage, redundant telemetry, tracking and command and communication security (COMSEC) equipment.

**Power Conversion-Regulation** —Regulated dc power can be supplied to the spacecraft during IUS free flight with the addition of an in-line converter-regulator. This option provides and installs a unit in the IUS flight vehicle which supplies spacecraft power at preselected output voltage levels between 24 and 36 vdc with regulation within  $\pm 0.3$  vdc to a peak capacity output of 600 watts. The installation includes sensors for remote voltage measurement and appropriate mounting brackets. Provisions in the IUS include physical envelope, electrical power and telemetry readout reservations.

Additional losses due to the insertion of this unit into the electrical system reduce the available energy to the spacecraft from 3.0 to 2.4 kilowatt-hours.

**Reaction Control Propellant Tankage** —A third propellant tank may be added to the generic vehicle two-tank configuration for missions requiring additional capacity. This option includes the tank assembly, squib valve, filter, fill valves, mounting bracketry, and plumbing to attach to the basic reaction control subsystem manifold. Provisions in the IUS vehicle include the physical envelope and electrical power reserves and the software capability for squib firing.

**Redundant Telemetry, Tracking and Command Equipment** —This option adds SGLS-compatible equipment to convert the IUS TT&C subsystem from the standard single channel configuration to a redundant configuration. Included is an additional S-Band transponder, an additional S-Band 20-watt power amplifier, two additional log spiral antennas, and an additional medium gain antenna. Also included are a diplexer, two radio frequency switches, mounting provisions, and the required coaxial cabling and wiring. Basic vehicle provisions include the physical envelope, operational power, and software switching capability reservations.

On-board redundancy management for a dual-string TT&C includes reconfiguration of the computer in control to the telemetry channel and switching to the redundant TT&C channel

if radio frequency power output from the operating power amplifier drops below 15 watts for more than two seconds. Reconfiguration as a result of other TT&C failures cannot be done without uplink commands.

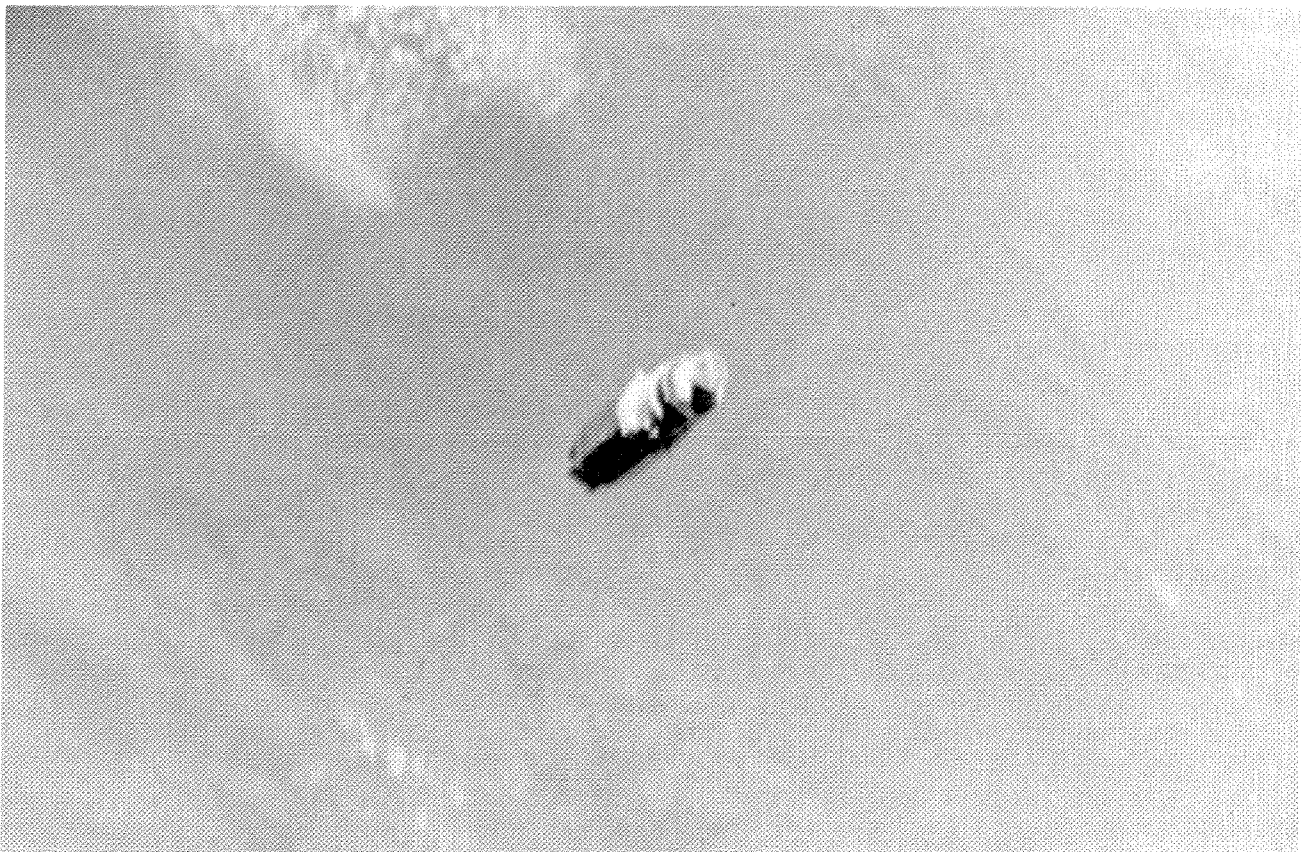
**Communication Security Equipment** –The generic two-stage vehicle contains single string COMSEC boxes. As required for redundant TT&C flights, COMSEC equipment may be added as a production option. This option includes 1 KG-46, 1 K1-23, and their mounting brackets. Battery power is augmented, as required, to supply this option. Generic vehicle provisions include the physical envelope reserve only.

### **Special Mission Services**

The following special mission services have been incorporated into the Shuttle/IUS and are available to future users at a significant

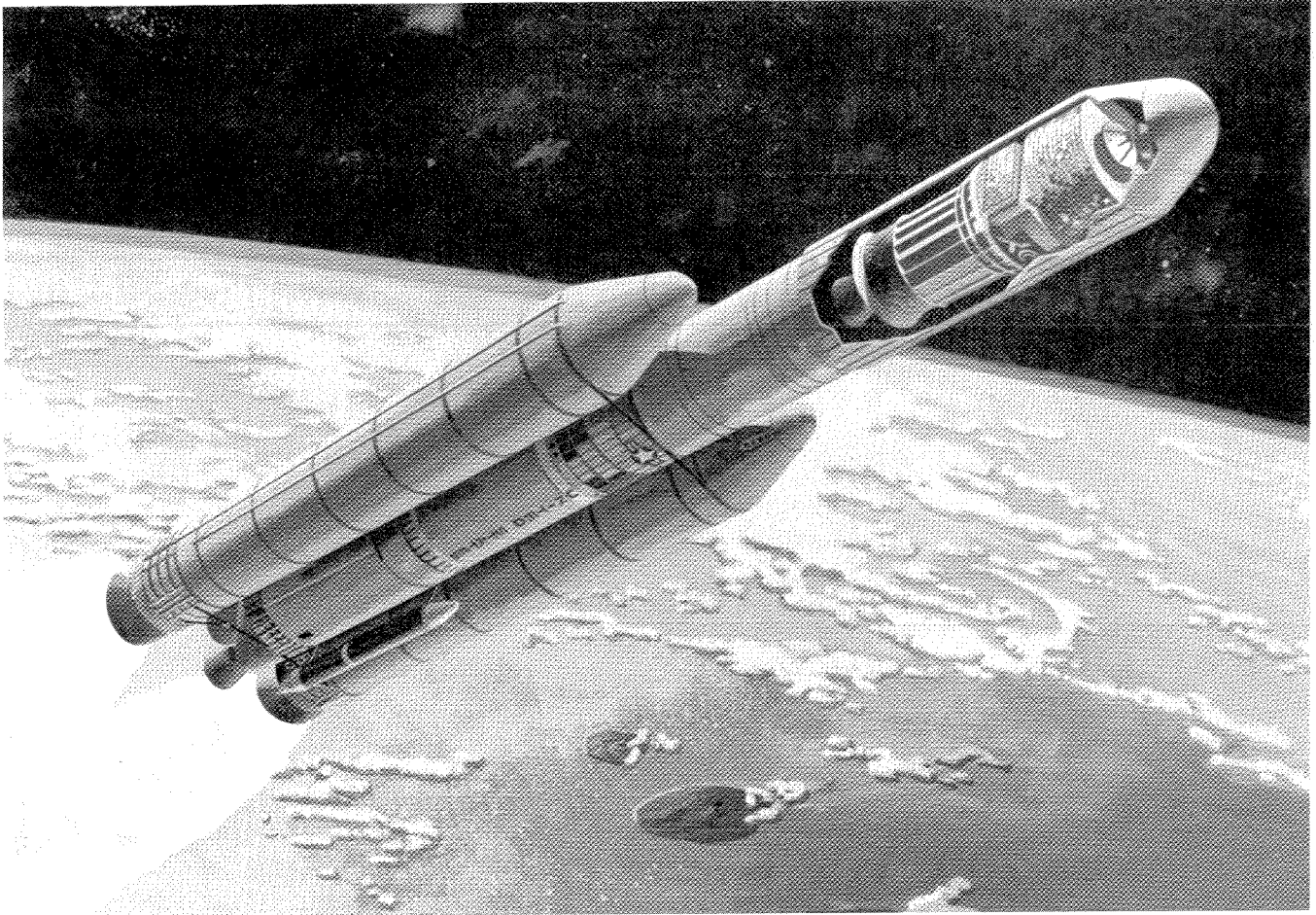
savings in engineering and development costs. These are:

- o An isolation diode assembly to provide the spacecraft with reverse current protection,
- o Remote power sensing of voltage at either side of the spacecraft power interface for voltage controlled by the dc-dc converter,
- o A single tank RCS,
- o An environmental measurement unit and accelerometers to provide flight environmental data,
- o Rocket engine module plume temperature sensors, and
- o Various battery complements.



*Inertial Upper Stage and TDRS Spacecraft in Space After Deployment From Shuttle*

## CHAPTER IV - IUS/Titan 34D



*Artist's Conception of the Titan 34D with  
IUS-2 and DSCS II/III*

The Titan 34D is the newest model of the Titan III space booster family, a family designed and manufactured by Martin Marietta of Denver, Colorado. The T34D has been developed under the management of the Air Force Systems Command, Space Division, for military applications, but it is now also available for commercial applications.

### LAUNCH VEHICLE DESCRIPTION

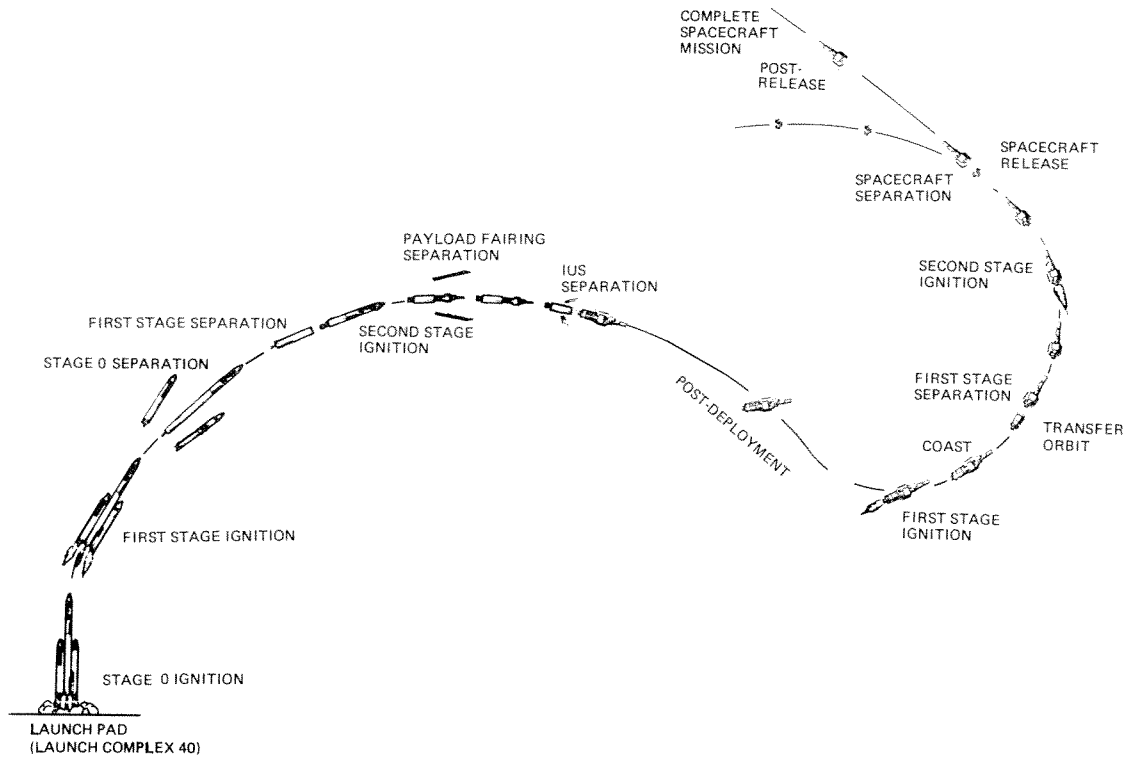
The basic T34D is 117-feet long. A fairing is used to protect the payload during ascent. This can add from 40 to 55 feet in length depending on the payload size. The total liftoff weight is 759.8 tons (excluding the payload).

The launch vehicle consists of two basic performance elements: a two-stage, liquid rocket engine core and two externally

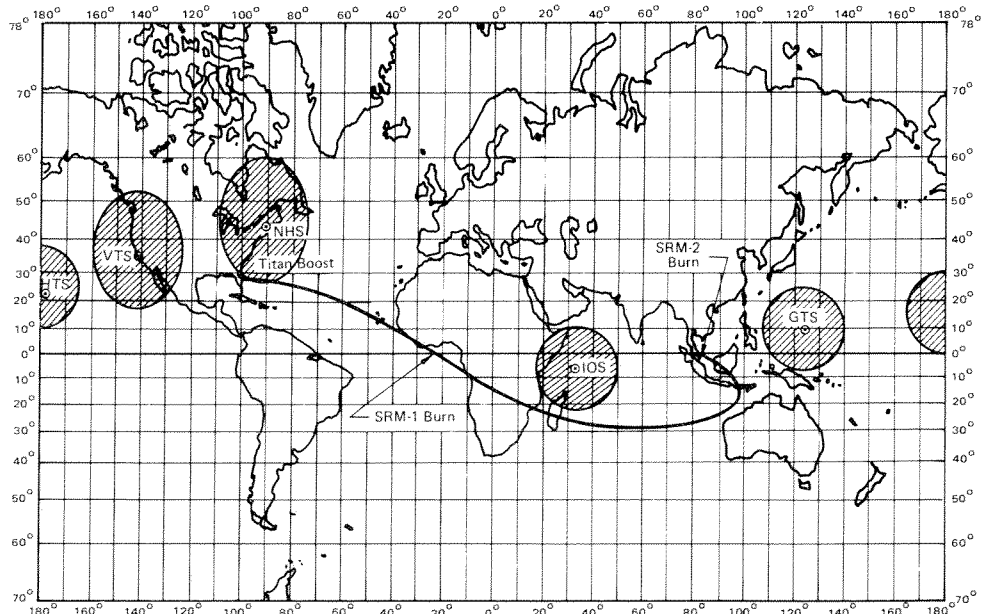
mounted solid rocket motors. The two solid rocket motors constitute stage zero. Stage 1 is fired 108 seconds into flight. At about 117 seconds, stage zero is exhausted and the motors are jettisoned. After 4.5 minutes, stage 1 drops away and the stage 2 engine is ignited. Solid propellant retrorockets are then used to separate the IUS from the spent second stage.

### FLIGHT OPERATIONS

The Titan 34D will place the IUS/spacecraft in a parking orbit, normally at an altitude of 80 x 95 nautical miles. During the powered flight described above, the T34D will operate on IUS guidance commands. Ascent events are depicted in Figure 4-1; event timelines are found in Table 4-1; and a typical ground track is shown in Figure 4-2.



**Figure 4-1: Sequence of Events for a Typical Geosynchronous Missions**



The ground track is typical for a generic mission first deployment opportunity

**Figure 4-2: Geosynchronous Design Reference Mission Ground Trace**

**Table 4-1: T34D/IUS Geosynchronous Mission Timeline**

Ground Elapsed Time (hr:min:sec)	Launch/Ascend and Deployment Phase Events	Event Duration (min:sec)
00:00:00	SRM Stage 0 Ignition (Liftoff)	
00:01:48	Stage 1 Ignition T34D	
00:04:33	Stage 2 Ignition	
00:04:34	Stage 1/2 Separation	
00:04:44	Drop Payload Fairing (PLF)*	
00:08:07	T34D/IUS Separation	:00:16
00:08:21	Initialize Guidance	:00:14
00:11:23	Start Spacecraft Thermal Control Maneuvers, if Required, or Initiate Pitch Rate to Maintain Spacecraft Horizontal to Earth, if Required.	
01:03:42	Maneuver for SRM-1 Burn	:07:00
01:10:44	SRM-1 Ignition and Burn	:01:53
01:14:17	Correct SRM-1 Velocity Vector	:05:00
01:19:17	Maneuver to Transfer Coast Attitude	:05:00
01:24:17	Start Thermal Control Roll (If Required)	
01:26:00	Switch Telemetry Rate to 16 kbps	
02:18:10	Telemetry Dipout, Transmit IUS/Spacecraft Telemetry to Remote Tracking Station	:05:00
04:10:57	Telemetry Dipout, Transmit IUS/Spacecraft Telemetry to Remote Tracking Station	:05:00
05:10:00	Switch Telemetry Rate to 64 kbps	
05:10:35	Transfer to Stage 2 Electrical Power	:00:16
05:10:51	Separate Stage 1	:00:15
05:11:06	Maneuver for SRM-2	:07:00
05:18:08	SRM-2 Ignition and Burn	:01:22
05:23:10	Correct SRM-2 Velocity Vector	:06:40
05:25:00	Switch Telemetry Rate to 16 Kbps	
05:29:50	Maneuver to Separation Attitude and Stabilize	:03:20
05:33:10	Separate Spacecraft	:00:04
05:38:52	IUS Collision Avoidance Maneuver	:00:14

\*IUS/spacecraft telemetry via IUS antenna after PLF drop (IUS switching)

At approximately seven minutes into flight, after Titan payload fairing separation, communications are switched from the Titan antenna to the IUS antenna. Eastern Test Range Stations receive and record the IUS data. At approximately eight minutes after launch, ordnance power is enabled, the Titan retro-rockets are fired, and Titan/IUS separation occurs.

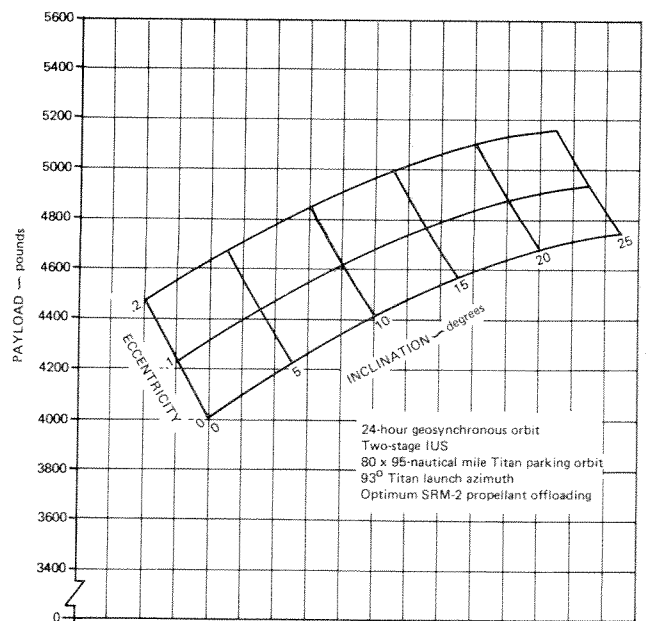
For a western injection flight, typical of communication spacecraft, the IUS/spacecraft will remain in the parking orbit about 61 minutes. Transfer orbit injection will take place over the mid-Pacific, near the first ascending node.

During coast in the parking orbit, a typical spacecraft may need thermal protection. To achieve this, the IUS orients the spacecraft to the sun. The IUS then rolls in one direction for about 26 minutes. After a short pause, the IUS rolls in the opposite direction for another 26 minutes.

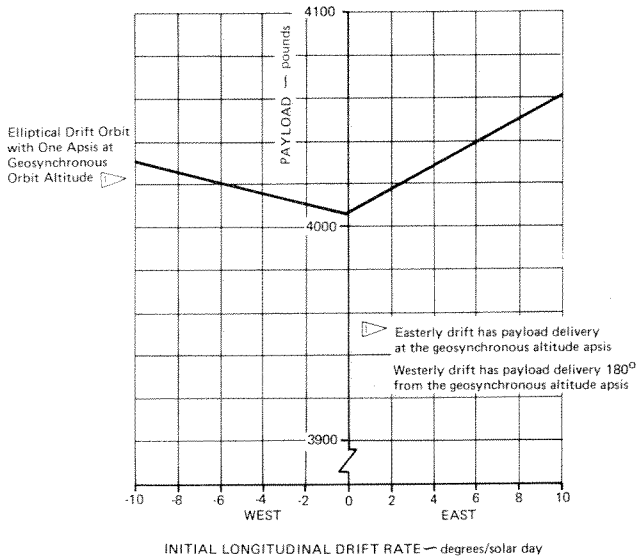
As the IUS/spacecraft approaches the SRM-1 ignition point, the IUS maneuvers to the calculated SRM-1 burn attitude. The transfer orbit and final orbit events do not differ from a nominal orbiter-launched IUS flight.

### PERFORMANCE CAPABILITY

The payload capability of the T34D/IUS to geosynchronous mission orbits is shown in Figure 4-3. This represents the capability for transfer orbit insertion at the first descending node. The effect of longitudinal drift rates is shown in Figure 4-4. Performance to other mission orbits may be obtained by contacting the Boeing Aerospace Company.



**Figure 4-3: Effect of Orbit Eccentricity and Inclination on Payload Capability**



**Figure 4-4: Effect of Drift Rate on Geosynchronous Payload Capability**

**Table 4-2: T34D/IUS Two-Stage Sequential Weights**

Item	Item Weight (pounds)	Sequenced Weight (pounds)
Payload		4006
Empty Stage 2 At Burn-Out**	2171.6	
RCS Propellant (Reserves & Trapped)	58.5	
RCS Propellant (Attitude Control & CAM)	8.6	
RCS Propellant to Augment SRM-2 Burn	58.2	
Stage 2 Burnout		11,302.9
SRM-2 Propellant	181.5	
Expended Inerts	21.1	
Stage 2 Ignition		11,505.5
Empty Stage 1	2315.7	
RCS Propellant (Attitude Control)	31.4	
RCS Propellant to Augment SRM-1 Burn	6.7	
Stage 1 Burnout		13,859.3
SRM-1 Propellant	18,866.1	
Expended Inerts	80.4	
Stage 1 Ignition		32,805.8
RCS Propellant (Attitude Control)	6.6	
Staged Aft Skirt	87.6	
Total IUS + Payload		32,900.0

\*Boeing document D290-10048-70, dated 1 May 1983  
 \*\*Includes EEC on SRM-2

### Mass Properties

A sequential weight statement for the Titan-configured IUS is shown in Table 4-2. It is based on "IUS Mass Properties Status Report Number 70" (MPSR 70), dated May 1, 1983.

The mass properties of the IUS baseline vehicle provide the user with a basis for preliminary calculations of Titan-user changes.

### Guidance Accuracy

The injection error requirements and actual results of the first Titan/IUS geosynchronous mission are shown in Table 4-3. These data show that the actual accuracies are well within the specified requirements. This kind of accuracy results in a reduction in spacecraft correction requirements, which translate directly into spacecraft growth or additional spacecraft life.

**Table 4-3: Insertion Accuracies**

DSCS II Insertion Error (EPOCH: Spacecraft Separation)				
Parameter	Target Value*	Actual**	Error	Allowable Error
Semi-Major Axis (n.mi.)	22793.	22780.	- 13	+ 120*** - 102
Eccentricity (n.d.)	0.0011	0.0013	+ 0.0002	+ .008*** - .001
Inclination (deg)	2.484	2.493	+ 0.009	+ .085*** - .074
RAAN (deg)	277.612	277.016	- 0.596	+ 2.3
Drift Rate (deg/day west)	0.5	0.567	+ .067	± 2.91
Injection Longitude (deg)	93	92.53	-.47	± 5°
DSCS III Insertion Error (EPOCH: Spacecraft Separation)				
Parameter	Target Value*	Actual**	Error	Allowable Error
Semi-Major Axis (n.mi.)	22817.	22804.	- 13.	+ 120*** - 102
Eccentricity (n.d.)	0.0000	0.0006	+ 0.0006	+ .008*** - .001
Inclination (deg)	2.483	2.502	+ 0.019	+ .085*** - .074
RAAN (deg)	277.610	276.995	- 0.615	+ 2.3

\* Boeing 6D guided nominal at separation event (includes 1 ft/sec separation impulse).  
 \*\* Orbit determination from SCF.  
 \*\*\* Orbit errors equivalent to STS-100 specification errors (in RTN Coordinates).



## ENVIRONMENTS

The spacecraft is subjected to the following induced environments: thermal, acoustic, vibration, shock, dynamic loads and pressure.

### Thermal

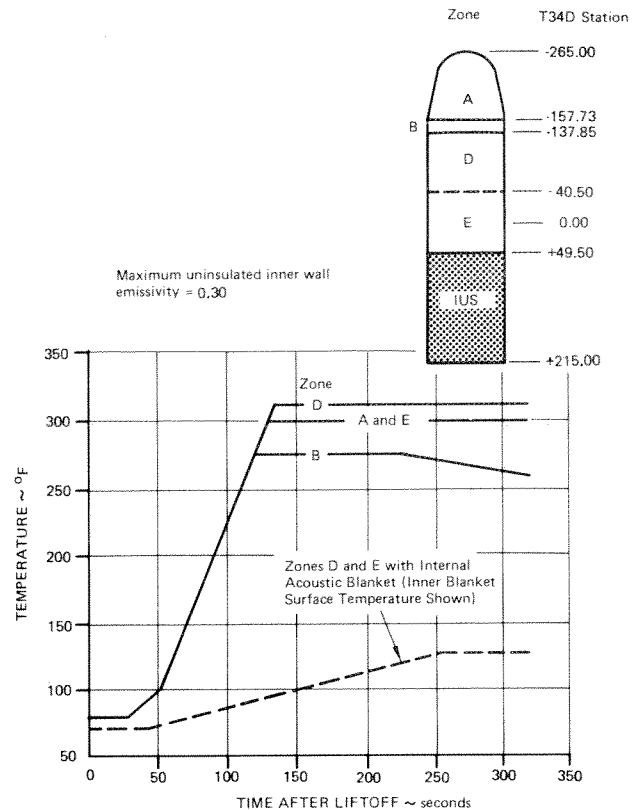
The external thermal environments are produced by the Titan payload fairing (PLF) or by the deep space environments after the fairing has been jettisoned.

**Prelaunch** — Prior to installation of the payload fairing, the spacecraft is protected by the environmental shelter (a unit within the mobile service tower at Launch Complex 40) which is air conditioned to 72°F, ±5 degrees. After installation of the payload fairing, a purge is furnished with a preselected temperature ranging from 50°F to 80°F at the fairing inlet. The spacecraft designer will specify the temperature in the spacecraft-unique ICD.

**Ascent** — During the initial portion of the ascent trajectory, the spacecraft is enclosed in the payload fairing which prevents direct exposure to the aerodynamic heating environment. Heat transfer to the spacecraft during this period is primarily by radiation from the payload fairing walls. The payload fairing wall temperature history is a function of the ascent trajectory, fairing wall heat capacity, and wall insulation.

Design ascent wall temperatures are shown in Figure 4-5. The temperatures shown represent the most severe ascent heating trajectories expected. If radiation to the spacecraft proves too great for the temperatures shown, the thermal input to the payload can be reduced by using insulation or thermal radiation shields, as indicated in Figure 4-5.

The second phase of the ascent thermal environment imposed on the spacecraft occurs at payload fairing separation. At this time, the spacecraft is exposed to the effects of free-molecular heating. The fairing is jettisoned as soon as possible for performance considerations, but not before free-molecular heating and dynamic pressure effects are acceptable to the spacecraft. Typically, this



**Figure 4-5: Payload Fairing Inner Surface Temperatures for T34D/IUS Configuration (40-foot PLF)**

occurs at altitudes of approximately 430,000 feet when free-molecular heating is around half of the solar constant (200 to 250 Btu/ft<sup>2</sup> hour). Following payload fairing separation, free-molecular heating decreases and then increases. A second peak occurs at parking-orbit injection. A third peak may be experienced near transfer orbit perigee. The relative magnitude of the latter two peaks depends on perigee altitude and transfer burn timing. If these peaks exceed allowable limits, they can be lowered by increasing parking orbit perigee.

**On-Orbit** — The on-orbit heating phase results from direct and reflected solar radiation and from direct Earth infrared radiation. These radiation levels can be controlled during the parking and transfer orbits by using either the IUS thermal control maneuvers or those of the spacecraft. Maneuvers already programmed and available for selection include toasting, metronome, rotisserie, and continuous roll. For more information on thermal control maneuvers, see Spacecraft Services section, following Environments.

**Post-Deployment** — SRM-1 and SRM-2 burns produce solid particle exhaust products. The products form a cone-shaped plane (a nominal half angle of 18.5° for SRM-2 and of 20° for SRM-1) which is several hundred feet high. The particle planes are a source of radiation to the exposed low mass elements of the spacecraft. This may result in a significant additional heat source for the spacecraft.

The IUS reaction control system engines are mounted near the IUS/spacecraft interface. The system uses hydrazine as propellant, and hence has exhaust products of nitrogen, hydrogen, and ammonia. The pitch and yaw engines are aft facing and do not affect the spacecraft. The roll engines, however, face laterally which may result in plume heating of the spacecraft. The roll engines primarily duty cycle for attitude control. This is done with .010 second pulses with a 0.01% or less duty cycle. Longer burns are used only for spacecraft-requested maneuvers (e.g., when establishing a continuous roll and during SRM operation). The duty cycle during SRM operation is less than 1%.

### Acoustic

The spacecraft experiences an acoustic environment during the boost phase until the launch vehicle/spacecraft is out of the sensible atmosphere. The internal acoustic excitation of the payload fairing has significant peaks during the first five seconds after liftoff and during transonic flight. The launch acoustic levels are lessened by channeling the Titan SRM exhaust gases through an exhaust duct until the launch vehicle begins to lift off. During liftoff, the exhaust splashes directly on the launch pad. This is the maximum acoustic excitation period. The second peak excitation period is generated by transonic aerodynamic shock waves and fluctuating dynamic pressure. This excitation lasts for approximately 55 seconds. The level is generally below liftoff levels. Acoustic noise levels for typical launch vehicles are provided in Table 4-4.

**Table 4-4: Titan 34D Payload Fairing Internal Acoustic Noise**

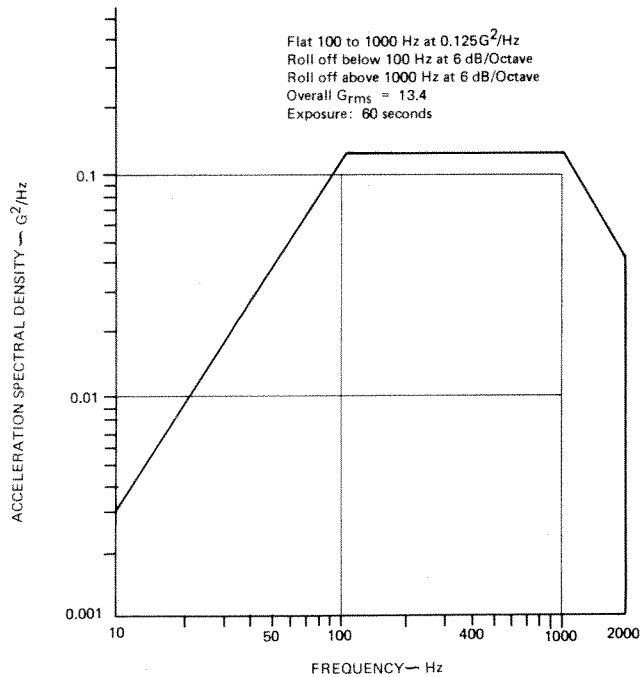
1/3 Octave Band Center Frequency (Hz)	T34D/IUS Sound Pressure Level, (dB*)
25	121
31.5	122.5
40	124
50	125.5
63	127
80	129
100	130.5
125	131.5
160	132.5
200	133.5
250	134
315	134.5
400	134.5
500	134
630	133.5
800	133
1,000	132
1,250	131
1,600	129.5
2,000	128.5
2,500	126.5
3,150	125
4,000	123
5,000	121.5
6,300	120
8,000	118
10,000	116
Overall Level	145

\*Reference sound pressure =  $2.9 \times 10^{-9}$  psi

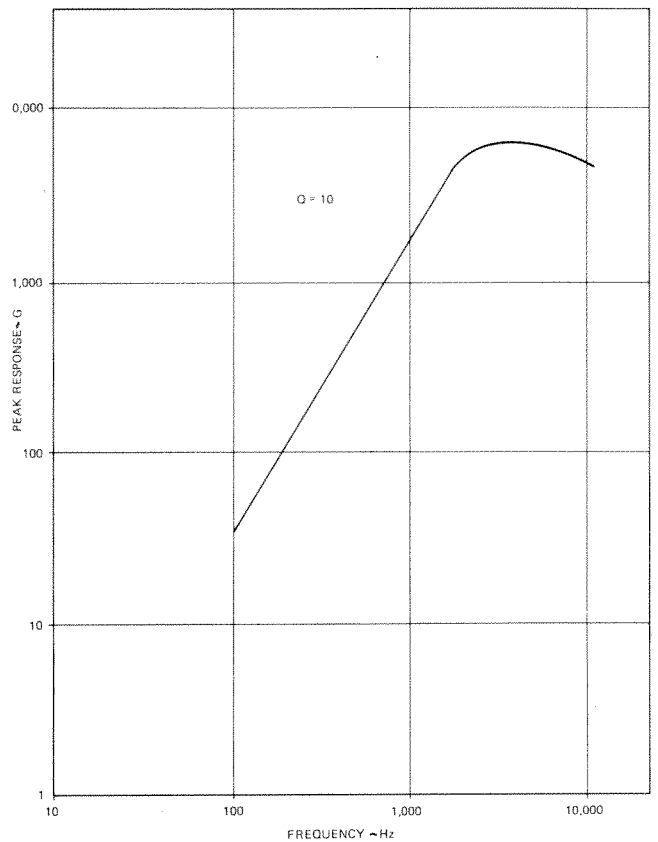
### Vibration

The random vibration environment results from the above acoustic noise fields. The same excitation period prevails when the vibration is mechanically transmitted to the spacecraft during testing. The excitation (attenuation or amplification) level at the spacecraft mounting interface is unique to each configuration. Each spacecraft program requires an analytical evaluation. A typical T34D/IUS random vibration spectrum is provided in Figure 4-6.





**Figure 4-6:** T34D/IUS Interface Random Vibration Environment for Payload Interface During Liftoff and Flight



**Figure 4-7:** Shock Response Spectra at the IUS/Spacecraft Interface

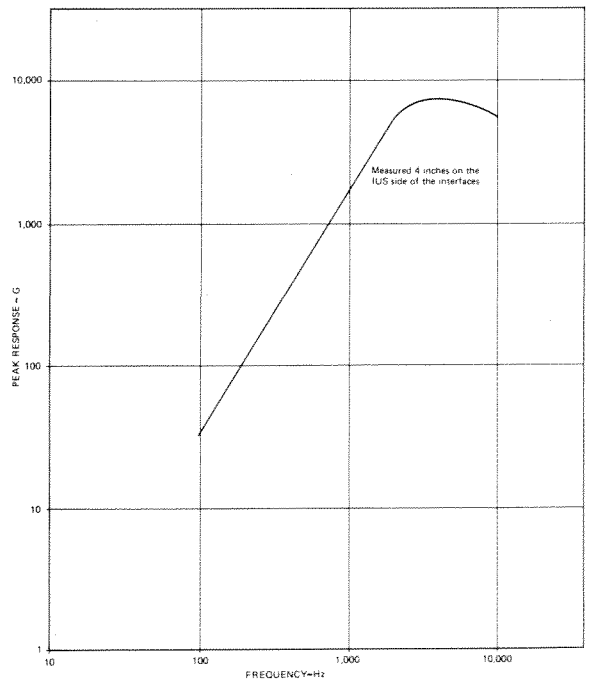
## Shock

Figure 4-7 shows the shock environment spectra produced by the IUS at the IUS/spacecraft interface. The pyrotechnic shock at the interface, produced by the spacecraft, will not exceed the values stated in Figure 4-8.

## Dynamic Loads

Table 4-5 shows load factors for initial sizing and/or evaluation of spacecraft primary structure. Spacecraft response is a function of spacecraft weight, stiffness, and lateral/axial coupling as well as booster configuration. These load factors are intended to provide a conservative design envelope that includes a variation range in these parameters. Transient load analyses performed during spacecraft development will be used to provide the detailed member loads required for complete design and evaluation of the structure.

Total axial load factors can be obtained from the table by adding the steady state and dynamic components. Lateral and torsional



**Figure 4-8:** Shock Environment Imposed by Spacecraft Disturbances

**Table 4-5: T34D/IUS Spacecraft Center-of-Gravity Load Factors**

Event and Axis	Steady-State Acceleration, G	Dynamic Acceleration, G or G/inch for Torsional
Liftoff		
- Axial	+1.5	+1.5
- Lateral	-	+5.0
- Torsional	-	+0.05
Maximum Airloads		
- Axial	+2.0	+1.0
- Lateral	-	+2.5
- Torsional	-	+0.05
Stage 1 Shutdown		
- Axial	0 to +4.5	+4.0
- Lateral	-	+2.0
Stage 2 Shutdown		
- Axial	0 to +2.5	+4.0
- Lateral	-	+2.0

Note: Separable payload weight is assumed to be about 4000 pounds.

load factors at the extremities of the spacecraft may exceed center-of-gravity load factors. Load distributions as a function of longitudinal distance (x) from the spacecraft/launch vehicle interface may be estimated using the equation below where  $G_{CG}$  is the center-of-gravity load factor from the load factor table and  $x_{CG}$  is the distance from the spacecraft center-of-gravity to the spacecraft/launch vehicle interface:

$$G_x = (3/5)G_{CG} \cdot (x/x_{CG}) + (2/5)G_{CG}$$

The following frequency criteria are recommended to minimize vehicle/spacecraft interaction.

Lateral frequencies from 2.0 to 10.0 Hz:

- Relatively high acceleration responses are observed during liftoff.
- Levels above these fundamental frequencies result in lower dynamic loads during liftoff as well as lower spacecraft loads and displacements during maximum airloads.

- Levels below 6Hz can affect launch vehicle control system performance.

Axial frequencies from 12 to 24 Hz:

- Spacecraft respond to stage 1 engine shutdown.
- Spacecraft respond to stage 1 engine thrust pulsation.

### Pressure

The pressure change within the payload fairing during flight is a function of the trajectory, atmospheric condition, free volume within the fairing, and openings in the fairing and adjacent launch vehicle compartment. Specific compartment pressure differentials cannot be quoted without knowledge of the spacecraft volume, and leakage areas, etc., associated with the spacecraft/fairing combination. Typical pressures encountered on previous missions are within one psi of ambient pressure. Similarly, the pressure change within the payload fairing has not exceeded 1.1 psi/second.

### SPACECRAFT SERVICES

Spacecraft services fall into three categories. These are: (1) generic IUS services which are available to all users at no additional cost, (2) spacecraft-unique production options which are qualified and can be added at extra cost and (3) special mission services which have been engineered and qualified for special applications and can be added without additional development or qualification cost. The following paragraphs are a discussion of these services.

It should be noted that any service provided through the IUS/spacecraft electrical interface connector must comply with the electro-magnetic compatibility (EMC) requirements of the Boeing IUS EMC Control Document (D290-10068-1) and the USAF IUS/Spacecraft Electrical and EMC Specification (SAMSO-LVR-0006). Primary power input to the IUS system from the launch vehicle or from support equipment shall be isolated from the power distribution to the spacecraft by at least 1 megohm dc resistance. When the IUS system provides a spacecraft primary power production option, the dedicated posi-

tive and negative power leads to the spacecraft shall be isolated from the IUS structure by at least 1 megohm dc resistance.

### Generic IUS Services

The standard or generic two-stage IUS provides certain spacecraft services. These include ordnance power availability, separation indication, telemetry transmission, discrete command initiation, and thermal control maneuver capabilities. Use of these services is at the discretion of the buyer.

**Ordnance Power** —The IUS provides and distributes ordnance power to the IUS/spacecraft interface for firing spacecraft ordnance devices in two groups of eight initiators. Interfacing ordnance circuits shall meet the requirements of the Range Safety Manual AFETRM 127-1, Volume 1, MIL-STD-1512 and D290-10068-1.

The IUS ordnance firing current supplied to the IUS/spacecraft interface for each spacecraft initiator is 5 amperes dc minimum, and the short circuit current is 11 amperes maximum. The minimum firing pulse duration is 40 milliseconds. The IUS provides the safe shorting and electrostatic protection for each IUS/spacecraft ordnance circuit interface.

The spacecraft ordnance circuit resistance measured at the IUS/spacecraft interface shall be 1.5 ohms maximum including the initiator bridgewire. The spacecraft separation ordnance initiators shall have the following characteristics:

Resistance:	1 ohm nominal
No fire:	1 ampere - 1.0 watt for 5 minutes
All fire:	4.5 amperes dc maximum

The IUS provides the current to fire the primary and backup spacecraft ordnance devices within five milliseconds of each other. The IUS ordnance control circuits are cross-strapped so that the fire impulse from either the A or B side will apply the fire current to both the primary and backup ordnance devices from their respective ordnance bus power sources.

The initial command to fire both groups of initiators is provided at the preselected separation time  $\pm 1$  second. A secondary firing command is provided 80 milliseconds  $\pm 10$  milliseconds after the initiation of the initial fire command.

**Separation** —The IUS provides the capability to monitor up to four separation switches or breakwires provided by the spacecraft. Figure 4-9 shows the four circuits and Figure 4-10 shows circuit details for the separation switches or breakwire alternative. Each circuit (signal and return) is monitored by the IUS telemetry system to verify spacecraft separation. Separation monitors are used only to verify spacecraft separation.

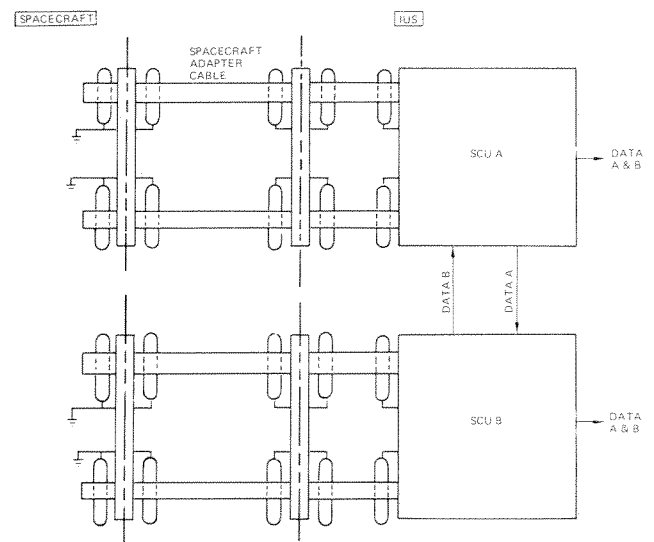


Figure 4-9: Spacecraft Separation Monitor Circuits

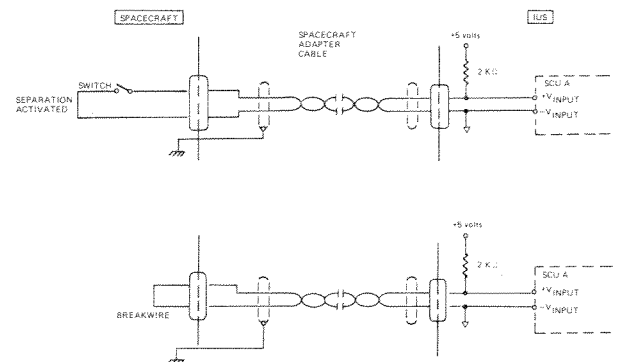


Figure 4-10: Spacecraft Separation Monitors

The IUS system shall provide the excitation voltage to the IUS/spacecraft connector interface for spacecraft separation monitor circuits having the following characteristics:

Voltage Range: 4 to 6 vdc

Current: 1 to 5 ma

Switch or Breakwire Isolation:

Referenced end circuit on the IUS side

Isolated end circuit on the spacecraft side.

**Telemetry** –The IUS has the capability to accept unencrypted spacecraft telemetry data and to interleave these data with the IUS telemetry data and transmit to earth via the IUS S-band telemetry, tracking and command system. Spacecraft telemetry data may also be transmitted by the spacecraft directly to the ground when the IUS is in free flight.

The IUS vehicle provides the telemetry rates shown below:

Altitude	Data Rate	Bit Error Rate	Margin
19,300 nmi	16 kbps	$10^{-6}$	6 dB
19,300 nmi	64 kbps	$10^{-6}$	3 dB

These data rates are achieved with directional antennas and preferential orientation of the IUS vehicle. During SRM-2 burn at geosynchronous injection, an antenna orientation will be specified in the specific ICD to provide these rates and margins to a space ground link system (SGLS) ground station.

At the 64-kbps data rate, the IUS can accept up to 4 kbps of asynchronous unencrypted data from each spacecraft. At the 16-kbps data rate, the IUS can accept up to 1 kbps from each spacecraft. Up to four spacecraft may be accommodated by each IUS. In a single spacecraft mission, the spacecraft may use more than one port.

To assure no loss of data due to IUS/spacecraft clock tolerances, the spacecraft data rate should be no greater than 3999.5 Hz and 999.8 Hz, respectively, for the 4 kbps and 1 kbps conditions.

The redundant IUS/spacecraft serial data interface is shown in Figure 4-11. The

spacecraft data and clock circuit interface and the data and clock relationship are shown in Figures 4-12 and 4-13, respectively.

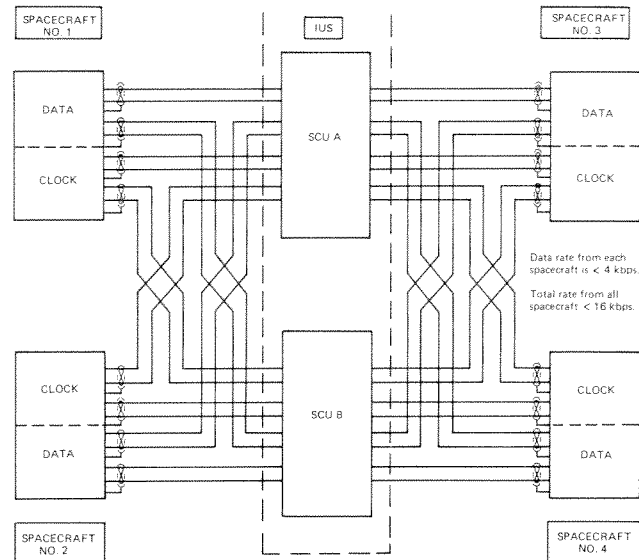
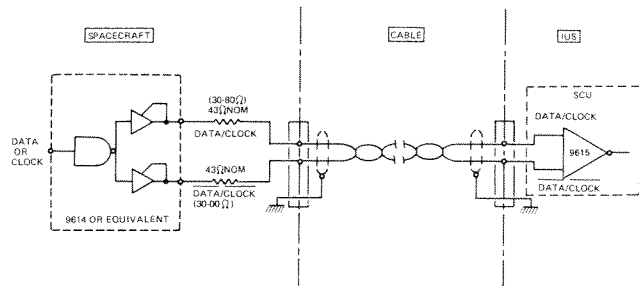
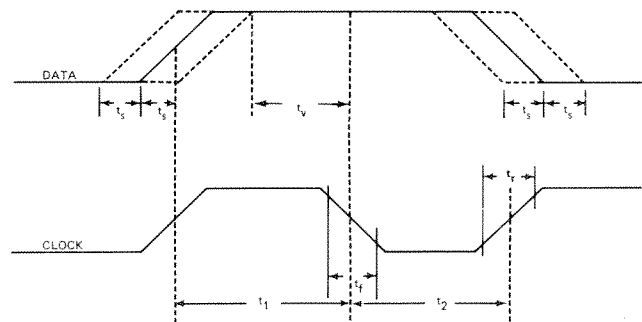


Figure 4-11: Spacecraft Telemetry Interface



The nomenclature for defining the signal levels (DATA and CLOCK) follows positive true logic.

Figure 4-12: Digital Line Receiver Interface for Telemetry Data or Clock



$$t_1 + t_2 \geq \frac{1.0}{3999.5} \text{ @ 64 kbps, IUS Telemetry Rate}$$

$$t_1 + t_2 \geq \frac{1.0}{999.8} \text{ @ 16 kbps, IUS Telemetry Rate}$$

$$\frac{t_1}{t_2} = 1.0 \pm 0.1$$

$$t_3 \leq 50 \text{ microseconds}$$

Measured at 50% voltage levels

$$t_4 \leq 10 \text{ microseconds}$$

$$t_5 \leq 10 \text{ microseconds}$$

Measured between 10 and 90% voltage levels

The IUS SCU strobes data in on the falling edge of the clock.

$$t_6 > 50 \text{ microseconds}$$

Figure 4-13: Spacecraft-to-SCU Data and Clock Relationship

**Spacecraft Commands**—The IUS has the capability to send a maximum of eight primary and eight backup discrete commands to the spacecraft for operation of solenoids, relays or switching devices.

The discrete commands are generated in the IUS computer either as an event-scheduling function (part of normal on-board automatic sequencing) or a command-processing function initiated from an uplink command. The IUS has the capability to receive uplink commands from the Air Force Satellite Control Facilities to alter the on-board event sequencing function. This permits the discrete commands to be issued at any time in the mission. Discrete command duration is a preprogrammed interval for each mission and is not alterable by external command. Uplink commands or programming of the normal on-board sequencing function permits any of the eight redundant discrete commands to be issued more than once.

**Thermal Control Maneuvers**—The IUS can perform thermal control maneuvers to reduce and control spacecraft thermal loads. Maneuvers for which the control logic has been developed and is available include toasting, metronome, rotisserie, and continuous roll. These maneuvers are performed during the intervals spent in park and transfer orbits.

A toasting maneuver consists of periodic  $180^\circ$  rotations about the yaw axis, reversing direction from a limit position typically at 1-hour intervals (Figure 4-14).

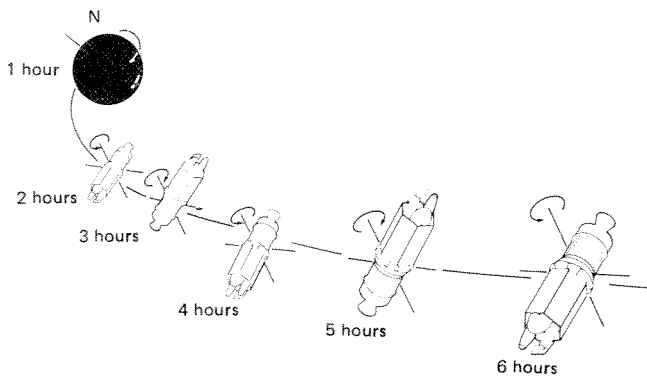


Figure 4-14: Typical Toasting Maneuver

A metronome maneuver consists of a periodic  $70^\circ$  rotation about the yaw axis, with a direction reversal typically at 1-hour intervals (Figure 4-15).

A rotisserie maneuver, shown in Figure 4-16 consists of periodic  $230^\circ$  rotations about the roll axis, with direction reversal typically at 6-minute intervals after 1-minute dwells at limit positions.

A continuous roll maneuver, shown in Figure 4-17 consists of rolling about the roll axis at typical rates of  $0.6^\circ/\text{second}$ . The maximum rate is  $6^\circ/\text{second}$ . To protect navigational

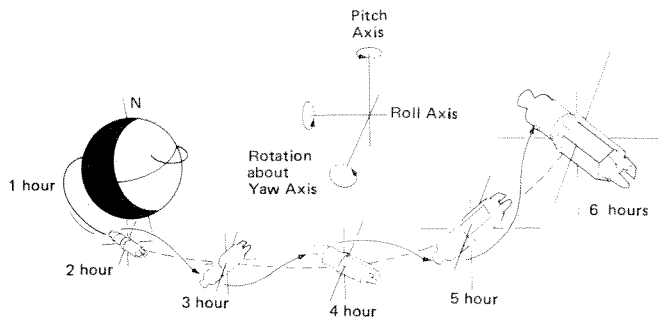


Figure 4-15: Typical Metronome Maneuver

$230^\circ$  rotation about roll axis, reversing direction at 6-minute intervals, with 1-minute dwell before reverse. Solar impingement occurs normal to roll axis.

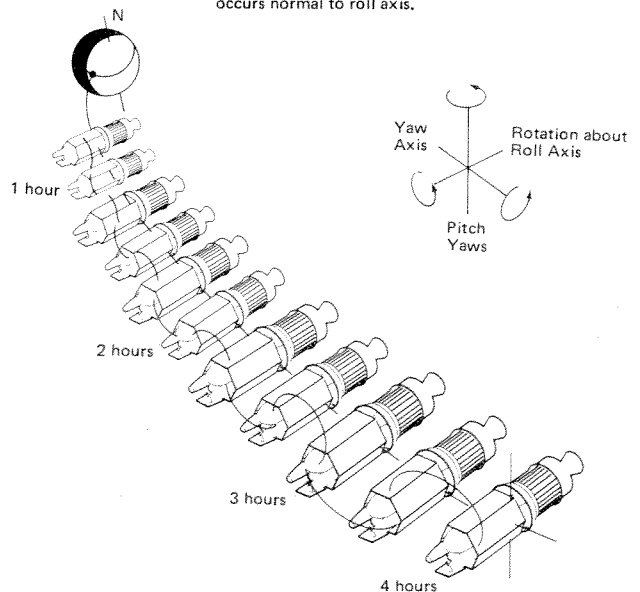
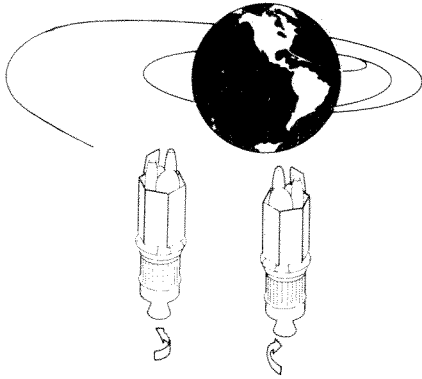


Figure 4-16: Typical Rotisserie Maneuver



**Figure 4-17: Typical Continuous Roll Maneuver**

accuracy, the roll direction is reversed in even increments. This maneuver may be employed at any time during a coast period and is preferred from the standpoint of having no time limitation.

**Telemetry Dipout Maneuver**—A telemetry dipout maneuver typically consists of a completely general reorientation in pitch, roll, and yaw to satisfy telemetry orientation requirements of the spacecraft. Telemetry data are transmitted continuously during all phases of flight by the IUS. Special IUS telemetry transmission maneuvers are only required during SRM-2 burn when the high data rate is employed.

**Spacecraft-Unique Production Options**

The IUS vehicle has provisions for including certain production options to modify or augment its basic performance. The hardware components and associated software for the production options described in this section have been qualified in the full scale development program.

Additional design integration tasks may be required when a production option is added to a specific mission. These tasks consider the effects of the addition of production options and the impact of the specific mission profile on battery and reaction control propellant capacity. Capacities required above the ge-

neric capability are chargeable to the spacecraft. Design integration tasks also include mass properties analyses and equipment installation adjustments, if required, to meet center-of-gravity constraints.

Performance impacts (including weights) caused by the addition of these production options are chargeable to the spacecraft. Unless otherwise specified, the production options include wiring for power, control and instrumentation.

The spacecraft-unique production options for the T34D version of the IUS include the addition of spacecraft electrical power availability, dc-to-dc power conversion-regulation, increased reaction control propellant tankage, redundant tracking, telemetry and command, a star scanner and an extendible exit cone.

**Spacecraft Electrical Power**—This production option adds batteries on stages 1 and 2, a power transfer unit on stage 2 and the interconnect wiring harnesses to supply spacecraft power. Up to 3.0 kilowatt-hours of battery electrical power can be provided to the IUS/spacecraft interface. IUS provisions include physical envelope and electrical connector interface provisions.

The IUS power distribution system will, during IUS flight, supply spacecraft power with the following characteristics:

Avg. Power (watts)	300*
Peak Power (watts)	600
Total Energy (kwh)	3.0
Voltage Range	24-36 vdc
Peak Voltage	28 vdc

\* Battery thermal control is designed to accommodate the specified 300-watt average spacecraft demand. The specific ICD shall identify a spacecraft average demand of less than 300 watts in order to establish a requirement for special provisions for shunts or external battery heaters to sustain required battery temperatures during cold in-flight environments.

Note: For multiple spacecraft mounted on a single IUS, this is total available power.

The spacecraft shall provide telemetry monitor signals to the IUS from the spacecraft side of the IUS/spacecraft interface to verify the voltage of the IUS-supplied spacecraft power. The IUS is required to telemeter verification that 28-volt power was delivered to the spacecraft. To support this voltage monitoring requirement, the spacecraft shall provide two divider networks of the configuration shown in Figure 4-18.

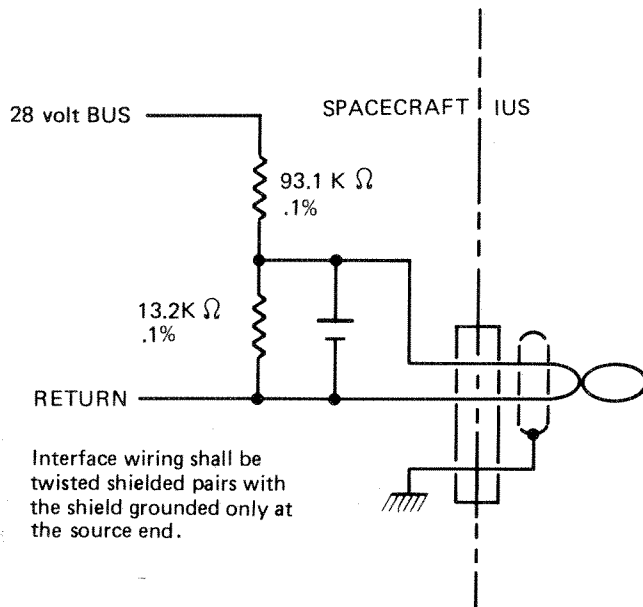


Figure 4-18. Spacecraft Bus Voltage Monitoring Divider Networks

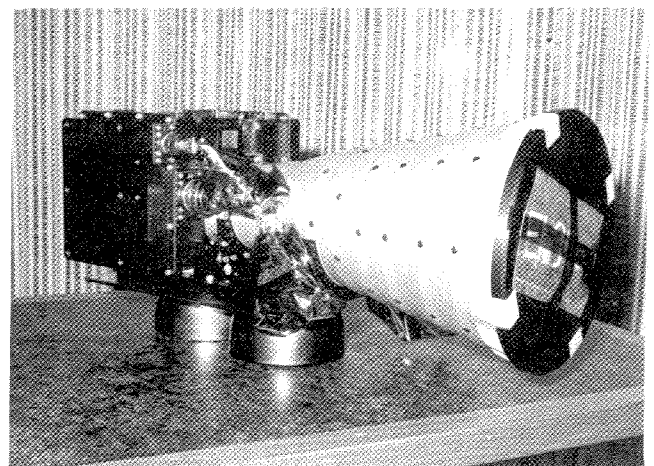
**dc-to-dc Power Conversion Regulation** —Regulated dc power can be supplied to the spacecraft during IUS free flight with the addition of an in-line converter-regulator. This option provides and installs a unit in the IUS flight vehicle which supplies spacecraft power at preselected output voltage levels between 24 and 36 vdc with regulation within  $\pm 0.3$  vdc to a peak capacity output of 600 watts. The installation includes sensors for remote voltage measurement and appropriate mounting brackets. Provisions in the IUS include physical envelope, electrical power and telemetry readout reservations.

Additional losses due to the insertion of this unit into the electrical system reduce the available energy to the spacecraft from 3.0 to 2.4 kilowatt-hours.

**Reaction Control Propellant Tankage** —A third propellant tank may be added to the generic vehicle two-tank configuration for missions requiring additional capacity. This option includes the tank assembly, squib valve, filter, fill valves, mounting bracketry, and plumbing to attach to the basic reaction control system manifold. Provisions in the IUS vehicle include the physical envelope and electrical power reserves and the software capability for squib firing.

**Tracking, Telemetry and Command (TT&C) Equipment** —This option adds SGLS-compatible equipment to convert the IUS TT&C subsystem from the standard single channel configuration to a redundant configuration. Included is an additional S-Band transponder, an additional S-Band 20-watt power amplifier, two additional log spiral antennas, and an additional medium gain antenna. Also included are a diplexer, two radio frequency switches, mounting provisions, and the required coaxial cabling and wiring. Basic vehicle provisions include the physical envelope, operational power, and software switching capability reservations.

On-board redundancy management for a dual-string TT&C includes reconfiguration of the computer in control to the telemetry channel and switching to the redundant TT&C channel if radio frequency power output from the operating power amplifier drops below 15 watts for more than two seconds. Reconfiguration as a result of other TT&C failures cannot be done without uplink commands.



Star Scanner

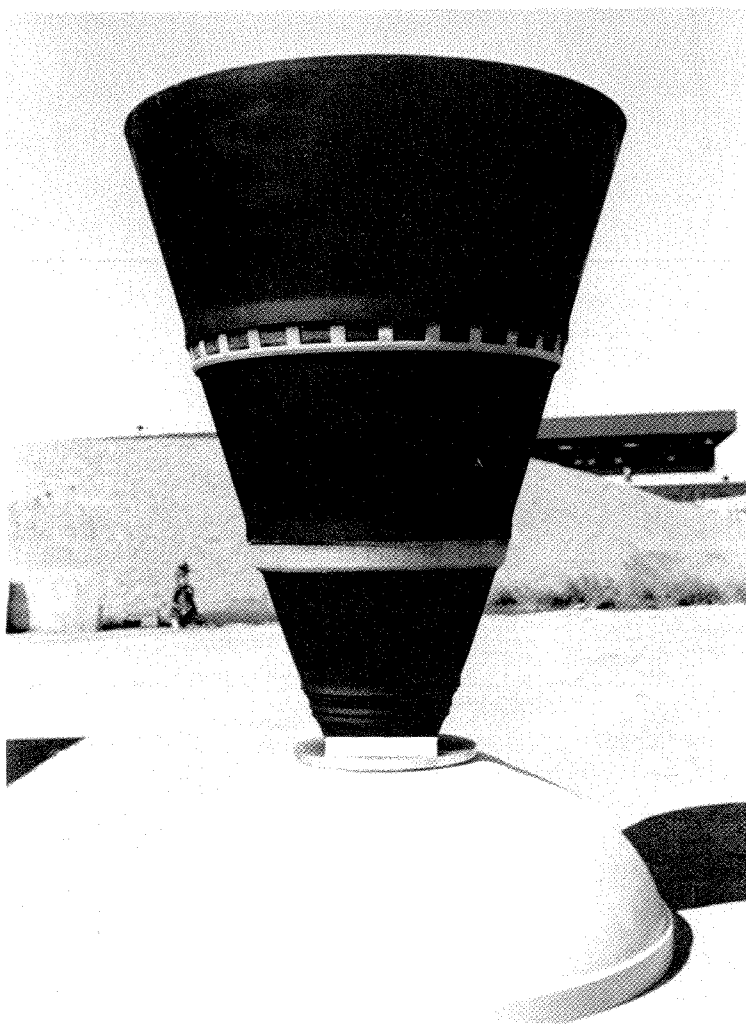
**Star Scanner**—A star scanner may be added to supply signals for the IUS on-board computers which, in combination with the software programming available with this option, provide position updates to the IUS guidance system. Basic spacecraft provisions include physical envelope and software switching capability reservations.

**Extendible Exit Cone**—This option adds a set of nested two-dimensional carbon/carbon extendible exit cones to the IUS stage-2 SRM

basecone along with the associated deployment mechanisms, activation power source, necessary wiring, and thermal protection. The benefit of this option is a nominal SRM-2 specific impulse increase of 14.4 seconds (to 303.5 seconds) at a nominal chargeable weight increase of about 96 pounds.

#### **Special Mission Services**

There are no special mission services available for the T34D/IUS.



*Extendible Exit Cone Deployed from the SRM-2 Basecone*



# CHAPTER V - SPACECRAFT INTEGRATION

Integration of a new spacecraft with the IUS and the Shuttle or the Titan 34D typically consists of four general phases: concept, definition, verification and operations.

For a repeat mission where the spacecraft to be integrated is within the same spacecraft family as one previously launched, integration typically consists of only the verification and operations phases.

The Boeing Integration and Launch Services (I&LS) available to the spacecraft agency include all IUS/spacecraft integration tasks for the four phases and the analyses from concept through post launch. A brief description of the objective of each phase

follows. Figure 5-1 is a schedule of all the phases with an outline of their tasks.

## CONCEPT PHASE

The objectives of the concept phase are early identification of spacecraft interface requirements, assessment of spacecraft compatibility with respect to IUS capability, and development of data for more detailed studies. To accomplish the objectives, the following tasks are performed: (1) interface compatibility analyses; (2) a preliminary launch vehicle/IUS/spacecraft integrated loads analysis to define spacecraft loads for design resolution, including development of

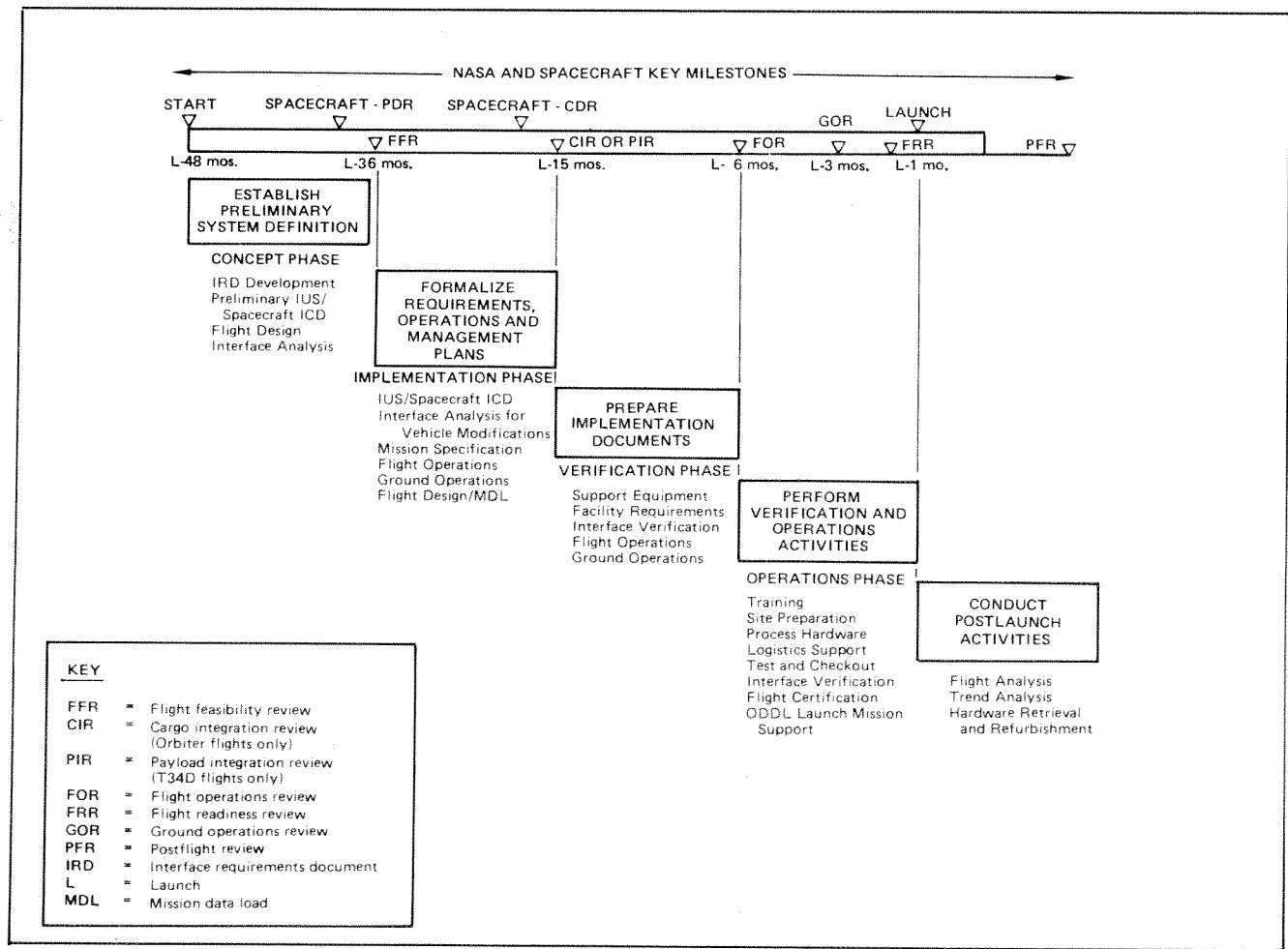


Figure 5-1: Available Spacecraft Integration Services

IUS 3-degree-of-freedom planning trajectories; (3) a baseline mission; (4) a preliminary thermal analysis; and (5) a baseline ground operations flow from IUS/spacecraft mate through launch.

The concept phase typically culminates with the spacecraft preliminary design and one flight feasibility review.

## DEFINITION PHASE

The objectives of the definition phase are to establish the launch vehicle/IUS modifications that are required to meet specific spacecraft requirements, to initiate longlead hardware and software activities, to support the launch date and to establish ground and flight operations requirements. To fulfill these objectives the following studies and data are provided: (1) the interface design analyses to support the spacecraft critical design review (CDR); (2) the pre-CDR spacecraft design based on IUS/spacecraft integrated-design loads analysis; (3) the pre-operational and operational mission design and mission data load which includes the simulated mission 6-degree-of-freedom trajectories which envelope the IUS/spacecraft mission; (4) the development of timelines for ground operations; and (5) the development of flight support requirements.

For the Shuttle flow, the definition phase is completed with the spacecraft CDR and the cargo integration review. For the Titan, the payload integration review is the culmination.

## VERIFICATION PHASE

The objectives of the verification phase are: (1) to implement the baseline flight and ground support requirements, which are determined at the large integration review or the payload integration review; (2) to complete launch vehicle/IUS/spacecraft interface verification activities; and (3) to assure flight readiness. To realize these objectives, flight and ground operations implementation documents are examined at the reviews and verification analyses are provided to support the final flight readiness review. As a minimum, the final verification includes the integrated IUS/spacecraft flight loads and associated

structural assessments, and flight control stability. Thermal and EMI/EMC environmental assessments are completed, with other verification analyses and assessments being performed as required. This phase will usually overlap the operations phase.

## OPERATIONS PHASE

The objectives of the operations phase are: (1) perform the hands-on ground processing of the spacecraft and IUS; (2) integrate the spacecraft and IUS and integrate the IUS/spacecraft with the launch vehicle; (3) supply real time launch and mission support; and (4) provide a post-flight analysis. In order to fulfill the objectives of this phase, the detailed plans, operating procedures, and schedules for launch preparations, launch, and mission support are completed. These data support the IUS and spacecraft processing through final checkout, final interface verification, launch vehicle integration, and launch preparation at the Eastern Launch Site (ELS) and at the DOD and NASA mission control centers. Launch and mission simulations are conducted at this time with the support teams prior to launch.

### Ground Operations

Transportation facilities at ELS for the receiving and shipping of spacecraft and/or support equipment include a choice of air, rail, water or land transport. The transporter/cannister is available for on-site transportation of the IUS/spacecraft cargo in a controlled and recorded environment.

Generally, spacecraft can be accommodated at ELS with little or no facility modification. Any facility modifications need to be identified early in the program to assure that the requested facilities are ready to accept the spacecraft during the operation phase.

Spacecraft interface verification can be completed at the spacecraft facilities using IUS simulation for electrical and mechanical interface verification, if required. The IUS-to-spacecraft electrical interface is verified prior to IUS/spacecraft mating during acceptance testing.

Typical operations and flow options for ground and launch operations at ELS are briefly described below and are shown in Figure 5-2. IUS assembly, service and checkout are done in the east low bay of the Solid Motor Assembly Building (SMAB). In the DOD-Shuttle flow, the IUS is assembled and checked and then is transported from the east bay to the Shuttle Payload Integration Facility (SPIF) in the SMAB west bay. The spacecraft is mated to, and integrated with, the IUS in the SPIF. The IUS/spacecraft is tested as a unit, and cargo-to-orbiter interface checks are made with orbiter simulation. The IUS/spacecraft is transported from the SPIF to Launch Complex 39A as a unit, using the environmentally controlled transporter/cannister. The IUS/spacecraft cargo is installed in the rotating service structure (RSS) at Launch Complex 39A for additional checks and servicing before being installed

into the orbiter. Final checkout and servicing of the spacecraft, IUS and orbiter are done at pad 39.

In the non-DOD Shuttle flow, the IUS is transported from the SMAB east bay to the Vertical Processing Facility (VPF) using the transporter. The spacecraft is integrated with the IUS in the VPF. The IUS/spacecraft-to-orbiter interface is checked with the orbiter simulator at the VPF. The IUS/spacecraft plus other cargo (if applicable) are transported from the VPF to Launch Complex 39A as a Shuttle cargo load using the environmentally controlled transporter/cannister. The cargo load is installed in the rotating service structure for additional checks and servicing before installation into the orbiter. Final checkout and servicing of the spacecraft, IUS, and orbiter are also done at this complex.

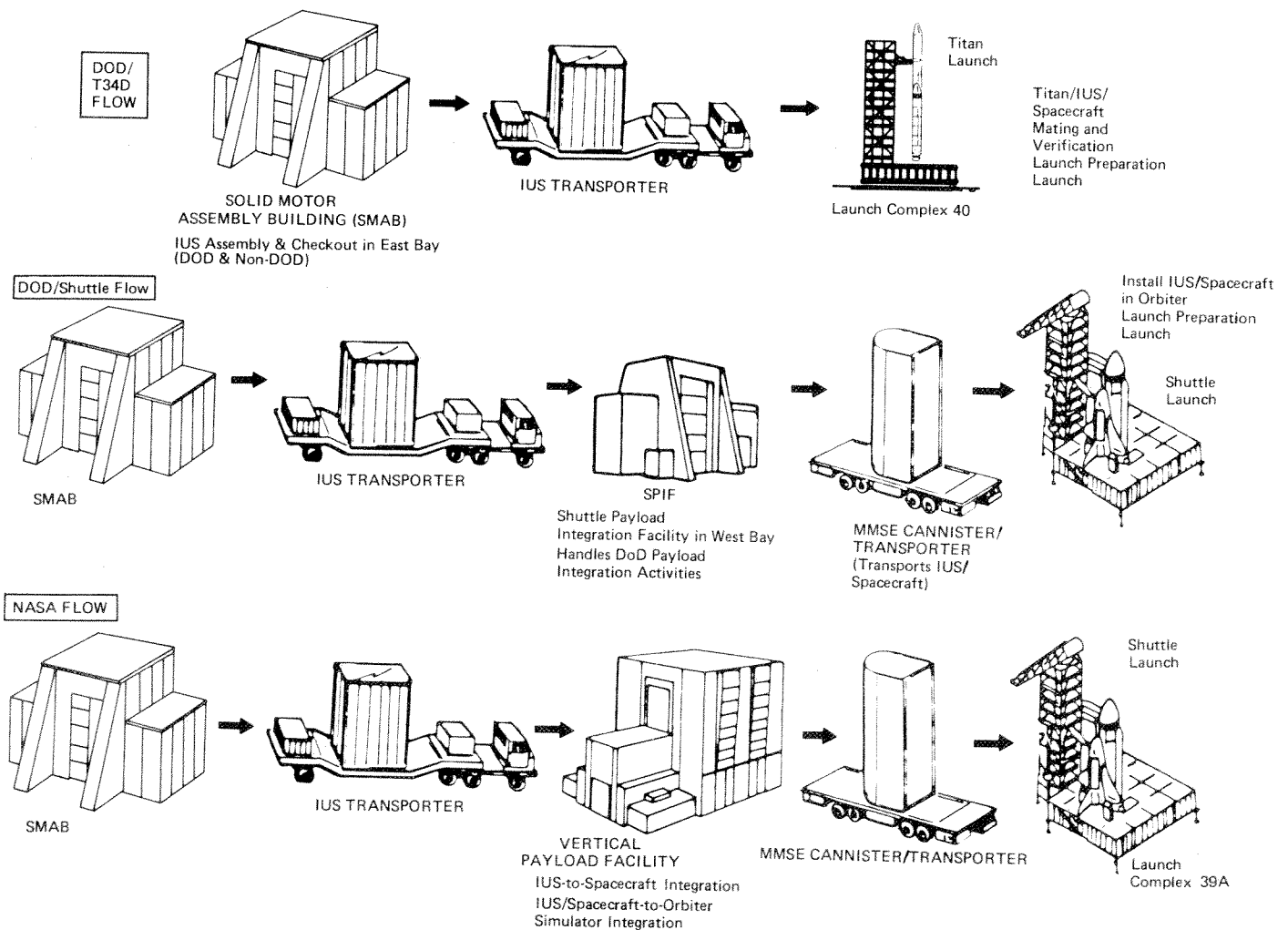
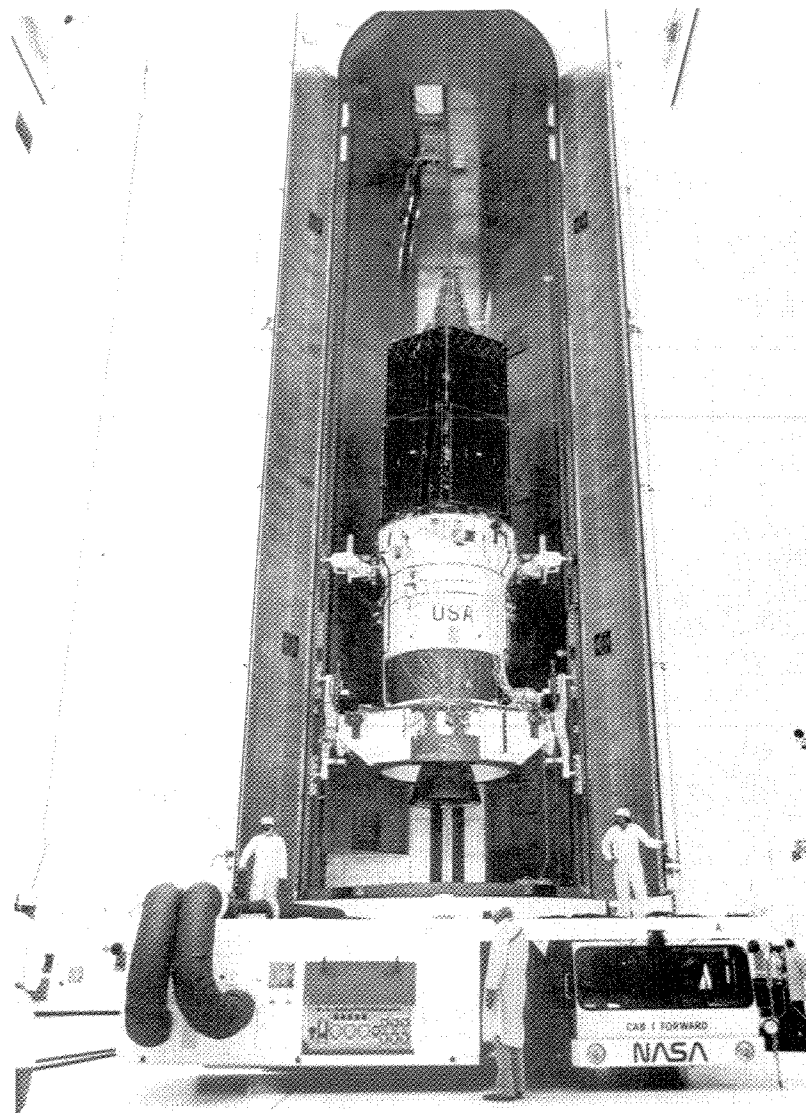


Figure 5-2: Eastern Launch Site Ground Support Operations



*The Multi-Mission Support Equipment Transporter/Cannister with IUS-1/TDRS-A*

A DOD flow using the expendable Titan launch vehicle (T34D) is another option. In this flow, all integration and servicing activities—IUS, spacecraft and T34D—are completed at Launch Complex 40. A mobile service tower, which contains an environmentally controlled section that encloses the IUS/spacecraft, is used for prelaunch integration and servicing. The mobile service tower is mounted on rails and is rolled back out of the way just prior to launch.

**Support Equipment Interfaces** —The transporter/cannister is primarily used for transportation and handling of complete orbiter cargos during integration and processing operations. The cannister, loaded or un-

loaded, may be transported and handled in either the vertical or horizontal position. The cannister duplicates the orbiter payload bay. Pickup points/retention fittings are similar in type, quantity and location to those in the orbiter. Access doors are along the top of the container and operate exactly like the orbiter doors with reference to allowable envelopes and clearances.

The transporter/cannister instrumentation set monitors and records the acceleration, temperature, and humidity experienced by the cargo during transit.

A towable environmental conditioning unit provides an air purge to the cannister at a flow rate of 0-94 lb./min., maintains the

temperature at  $70^{\circ} + 5^{\circ}\text{F}$ , provides cleanliness (nominally class 100), supplies the guaranteed class 500 (FED STD 209A) air with 15 ppm or less hydrocarbons, and maintains the humidity between 30% and 50%.

Intersite transportation equipment is available to carry a cargo between the spacecraft manufacturing center and the launch site. The intersite transportation equipment can be moved on the road, with special permits and escorts, or by air. Tie-downs and sling set are available for use with the intersite transportation equipment. The intersite transportation equipment payload bay is 15 feet in diameter. A transport environment monitoring system senses shock, vibration, temperature, humidity and power levels during transit. An alarm warns the driver if any critical parameters are out of tolerance.

Other payload servicing equipment is available from past programs. This equipment consists of such items as cryogenic charging carts, portable vacuum pumps, hydrazine servicing carts, and battery chargers. The condition and availability of any payload service equipment must be determined prior to use.

**Facility Interfaces and Services**—Early determination of spacecraft facility requirements is necessary for efficient operations at ELS. This determination should be initiated in the conceptual phase of the project planning. Boeing I&LS are available to provide this planning activity. Shuttle users are assigned areas that are adequate for the required task. If no such facility exists or if modifications are required, a facility is selected which best suits the task and requires the least modification.

There are numerous facilities and services available at ELS for ground and launch operations. Some of these may be of special interest to spacecraft contractors and are listed below. For the different facilities locations see Figure 5-3.

**DOD Titan/Shuttle Facilities**—The *Satellite Assembly Building (SAB)* supports processing i.e., integration and checkout of DOD payloads. The satellite assembly building has a high bay area with overhead crane, screen room, roof-mounted antennas, controlled environment, security provisions, large equipment access door, storage and office space,

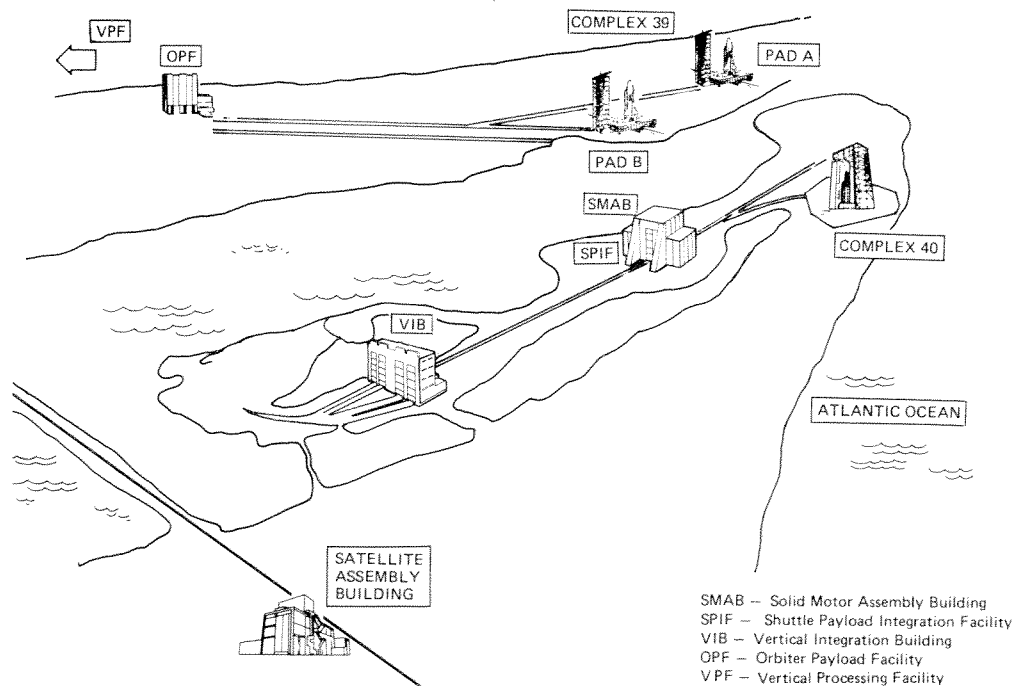


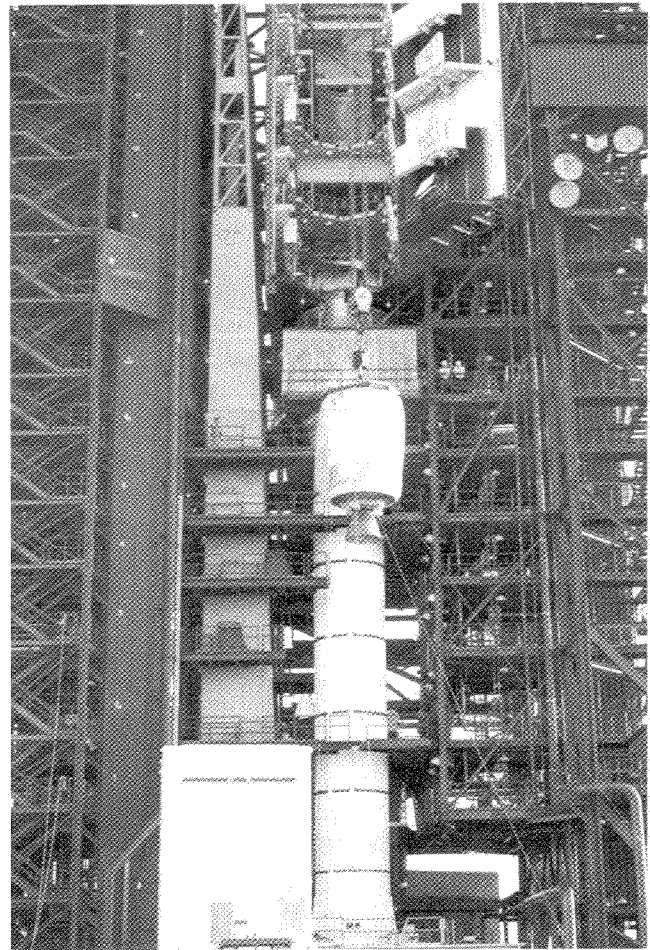
Figure 5-3: Eastern Launch Site Facility Locations

etc., to support stand alone processing of DOD T34D and STS-launched spacecraft.

The *Solid Motor Assembly Building (SMAB)* has undergone design changes to provide integration/separation facilities. The resulting facility supports a number of diverse operations: (1) The east bay contains two test stands for final assembly and checkout of the IUS. (2) The west bay, referred to as the *Shuttle Payload Integration Facility (SPIF)*, handles DOD payload integration activities. Two integration cells provide contamination isolation to support spacecraft testing, mating with the IUS, fueling, and payload interface verification testing. Equipment is provided to verify the mechanical and electrical interfaces of the payload to the orbiter. The transporter/cannister interfaces with the SPIF for the transfer of the cargo to the rotating service structure at Launch Complex 39A. (3) The center high bay of the SMAB is used for buildup, checkout and mating of the Titan 34D solid rocket motors.

The fourth floor of the *Vertical Integration Building (VIB)* has been modified to provide a checkout station for the IUS. The VIB checkout station supports the IUS in the Solid Motor Assembly Building, Vertical Processing Facilities, and Launch Complex 39A or 40. Also, it is in cell 2 of this facility that the Titan 34D core is inspected, erected and checked.

*Launch Complex 40 (LC-40)* is the Titan-dedicated launch area. The launch complex is comprised of a launch pad, a mobile service tower (MST) with environmental shelter, an umbilical tower, and the gas storage and propellant holding areas. The MST is Launch Complex 40's primary payload support facility. It is used to integrate the IUS/spacecraft with the Titan 34D and later serves as a station to check and service the integrated system. Work platforms at 14 levels provide access to the Titan 34D, IUS, spacecraft and the payload fairing. A controlled environment shelter is housed within the MST for the IUS/spacecraft combination. The environmental shelter has provisions for RF communications with the satellite assem-



*IUS/T34D Assembly in Mobile Service Tower.*

bly building and for the electrical and air conditioning umbilical system for the Titan 34D and the spacecraft. This area allows 360° access to the IUS/spacecraft and payload fairing by using fixed and folding platforms at six levels.

**Non-DOD Shuttle Facilities** –The *Vertical Processing Facilities (VPF)* building is designated as a major non-DOD facility for off-line processing of Shuttle vertical cargo. The major uses of this facility are the integration of non-DOD spacecraft to the IUS and the integration of payloads utilizing other upper stages. The vertical processing facilities consist of an environmentally controlled high bay and air lock, and a single-story support structure. Activities typically performed at the facility are: receiving and inspection; IUS/spacecraft mate; electrical and mechanical



systems test and checkout, including simulated payload-to-orbiter interface verification using the cargo integration test equipment; total cargo integration and fueling; ordnance installation; and cargo stowage in the payload transporter/cannister.

The *Orbiter Processing Facility (OPF)* is the facility designated for orbiter preparation including prelaunch installation and postflight removal of the ASE. Horizontally installed payloads may also utilize the orbiter processing facility.

*Launch Complex 39A (LC-39A)* is for the Shuttle. The major payload support facility here is the rotating service structure. The payload changeout room, a part of the rotating service structure, has provisions for IUS/spacecraft mating, interface verification, servicing, systems checkout and payload installation and/or removal. The rotating service structure is designed for vertical

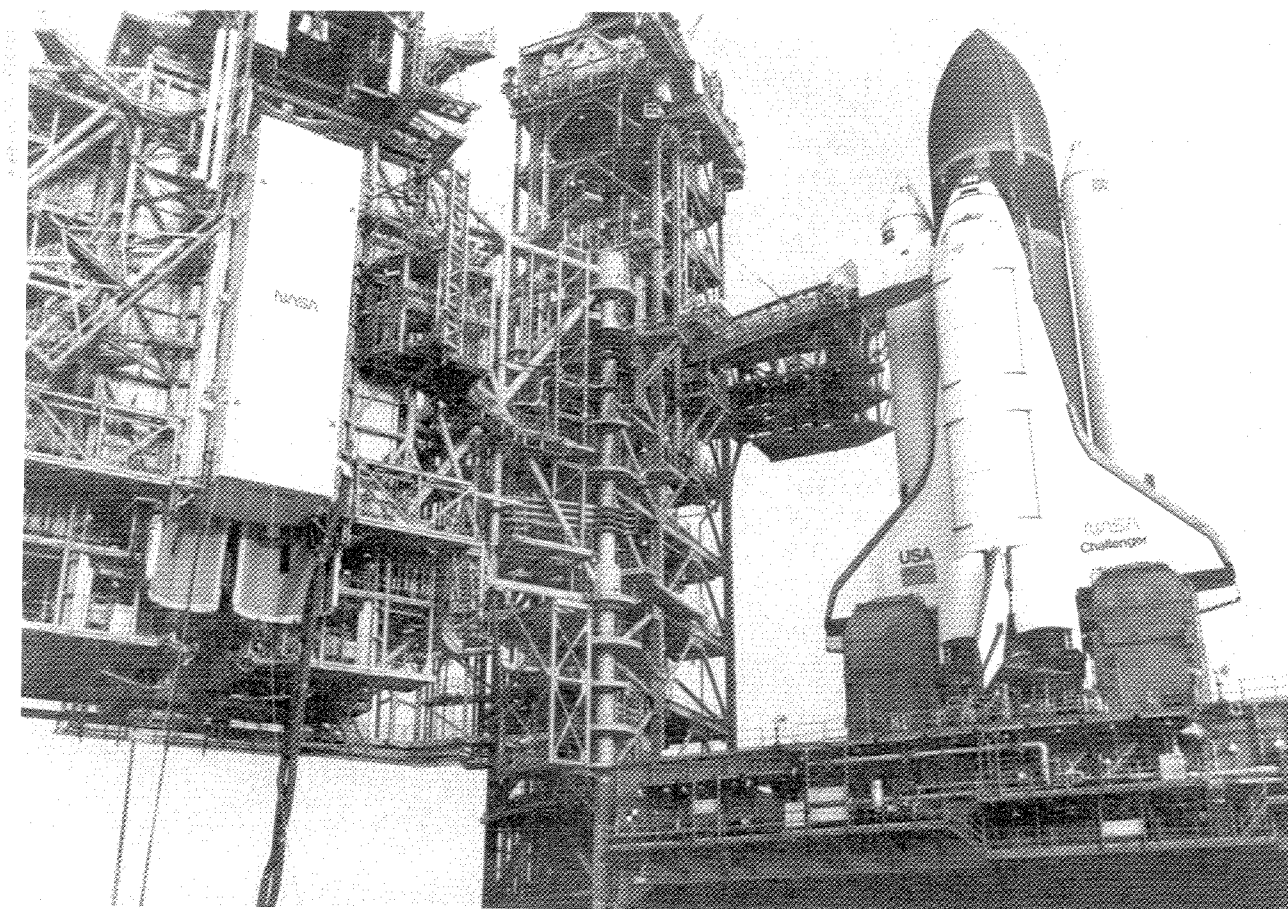
transfer of payloads into the orbiter while maintaining a controlled payload environment. Transfer is accomplished by using the payload ground handling mechanism. Provisions exist for storable propellant loading of less than 1200 pounds, using payload-provided ground support equipment.

The *Explosive Safe Area (ESA-60)* is a contamination-controlled facility which is used for reaction control system propellant (hydrazine) loading and servicing.

The *Liquid Fuel Storage Area* has hydrazine fuel stored in tanks or drums.

The *Ordnance Checkout and Storage Facility* is used for receiving/inspection, checkout and storage of ordnance devices such as electrical explosive and safe/arm devices.

The *Propellant Laboratory* provides spacecraft propellant transfer from storage to



*The Shuttle "Challenger" with the Rotating Service Structure*

service module. Facilities include propellant quality verification provisions, a clean room, and supplies of gaseous nitrogen, gaseous helium, and alcohol.

### **Responsibilities**

All operations at ELS require coordination and support from the ELS agencies/contractors. Typical operational responsibilities for an IUS mission when considering only the flight hardware and support equipment are noted below.

Spacecraft checks, preparation for integration and stand alone operations are the responsibility of one agency/contractor, usually the spacecraft supplier.

A single agency/contractor will usually have the overall integration responsibility for the mating and integration of the spacecraft and the IUS. Support will be provided by the respective IUS and spacecraft contractors. In the past, Boeing I&LS has been the cargo integration contractor for the IUS/spacecraft. As the IUS/spacecraft is integrated with the launch vehicle, the launch vehicle agency/contractor has the overall integration responsibility with support from the cargo integration contractor.

Thus the individual spacecraft and IUS agencies/contractors will usually be responsible throughout the prelaunch ground operations processing at ELS.

### **Contingency Abort Operations**

Abort operations that occur prior to launch will be processed in the reverse order of

buildup, i.e., the IUS/spacecraft are removed from the orbiter bay and returned to the Shuttle payload integration facility or vertical processing facility for demating and disposition.

When the mission is aborted after launch of the Shuttle, and the orbiter lands at ELS, the orbiter will first be safed. After safing operations are completed, the orbiter will be moved into the orbiter operations processing facility. The IUS/spacecraft will be removed from the orbiter and placed in the transporter/cannister in the horizontal position. The transporter/cannister will then be moved for rotation of the cannister from the horizontal to the vertical position on the transporter. The cannister (with the IUS/spacecraft) is transported to the Shuttle payload integration facility or the vertical processing facility where the IUS/spacecraft is removed from the cannister and installed in the facility. The IUS/spacecraft is demated and dispositioned by the responsible agencies.

If the orbiter lands at a secondary or contingency landing site, the orbiter agency will safe the orbiter and remove the IUS/spacecraft from the payload bay. The IUS/spacecraft will be positioned on the cradle. The IUS and spacecraft will be demated, with each of the concerned agencies/contractors responsible for their respective craft. Any special support equipment and/or handling equipment will be airlifted to the landing site.



## CHAPTER VI-IUS IMPROVEMENTS AND DERIVATIONS

The flight-proven elements of the IUS provide an unparalleled capability in upper stage performance. They also provide a versatility in derivative designs that enable a variety of needs to be satisfied with minimal development cost. The activities which could add to development costs, i.e., component and systems tests, orbiter payload integration, development of new ground support and test equipment, establishment of space, logistics and integration facilities, etc., are minimized. Because of their capabilities, versatility, and minimal development cost, the IUS derivatives should be considered when new user requirements are proposed.

### PERFORMANCE IMPROVEMENTS

Performance improvement studies have considered both the benefits of launch system capability increases and the basic IUS performance improvements.

#### Launch System Improvements

Launch system improvements can offer cost-effective performance improvement. Titan 34D improvements are available which provide slightly better performance than the current Shuttle/IUS performance. This is achieved by enabling a fully loaded IUS to be placed in a highly elliptical parking orbit. For more specific information, the Titan 34D Users Guide written and published by the Martin Marietta Company is a recommended reference.

The Shuttle can also provide higher circular or elliptical parking orbits. These higher energy orbits provide payload increases of 200+ pounds to geosynchronous orbit, but make mission planning less flexible. This improvement restricts the opportunity to share the cost of the Shuttle launch, and, in the case of the elliptical orbits, limits the opportunity to successfully complete the mission. These are not viable options for a generic IUS and should only be considered on a mission-by-mission basis.

### IUS Improvements

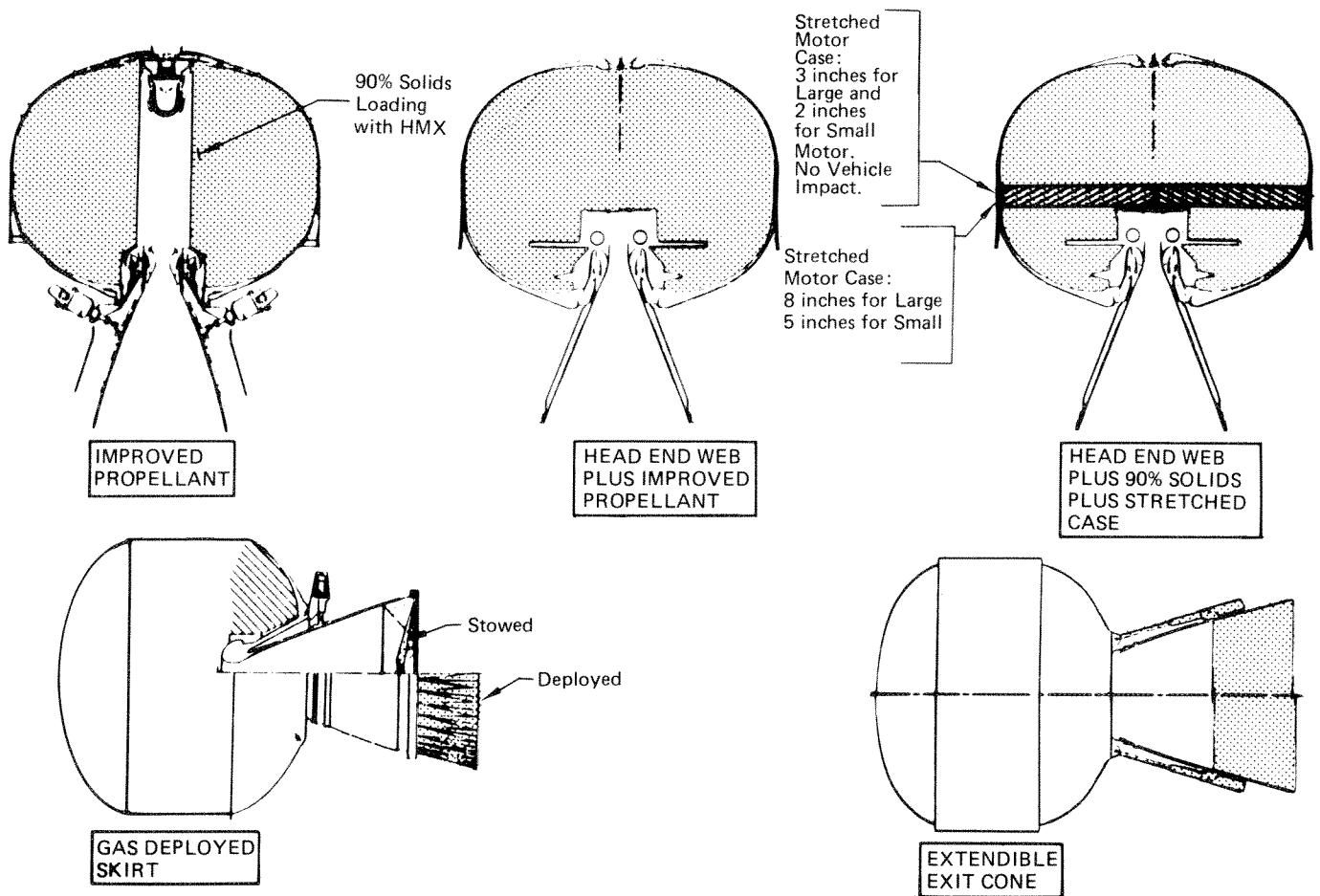
Propulsion system improvements and weight reduction items have been studied to identify the cost-effective approaches that provide increased payload capability. For appreciable performance improvement, propulsion system improvements are the most cost-effective. Weight reduction items offer a small increase in payload capability while retaining the large expense per pound. They are, therefore, best treated as producibility changes or as part of a technology program, e.g., lithium batteries.

Improvements to the motor are the results of recent development programs and/or altering the core geometry. The Kevlar cases, carbon/carbon nozzles, Tech-Roll seal joint and small motor extendible exit cone are advanced features of the basic motor which do not lend themselves to appreciable performance improvement.

Advanced propellants with a 90% solids loading and HMX have been tested and verified on several programs and do offer improvement for the IUS. A head end web grain design with advanced propellants in Kevlar cases in conjunction with carbon/carbon nozzles has also been demonstrated on at least three development programs.

These two changes, the advanced propellant and the advanced propellant with head end web grain, are two of the candidate changes shown in Figure 6-1. Other options exist to stretch the motor cases so that more propellant can be added. Nozzle-extension devices, such as the extendible exit cone (EEC) or the gas-deployed skirt can also be added to the large motor. (The stretched motor cases include the advanced propellant and head end web grain.)

A 2-inch stretch of the small motor and a 3-inch stretch of the large motor are designs that can be accommodated with no system impact to IUS, but the 5-inch stretch of the



**Figure 6-1: Candidate Propulsion Improvements**

small motor and the 8-inch stretch of the large motor will impact the IUS system because each adds 4 inches of length to the IUS. The four-inch increments can be accommodated with minimum system impact.

The 5-inch small motor stretch can afford a cost-effective means to implement the graphite epoxy composite motor support cone, interstage, and other weight reduction items that are not cost-effective as stand alone changes.

The addition of an EEC on the large motor is straightforward. The same deployment mechanisms from the small motor are used, and the cone is virtually identical to those being tested for the MX missile. The gas-deployed skirt is an alternate to the EEC. An Air Force Rocket Propulsion Lab development program has already shown this to be a valid concept.

Gas-deployed skirt benefits are not shown in Figure 6-2 because they are typically about 70% of the EEC performance benefit and, like the EEC, can be included with any of the other modifications. As a stand alone change, the EEC or gas-deployed skirt are not cost effective. The advantage of a gas-deployed skirt is its simple, reliable operation and lower recurring cost.

Development programs for each of these motor improvements, in both the large and the small motor have been analyzed for cost and schedule estimates. The performance of the candidate options has also been determined, both individually and in combination. Figure 6-2 shows the results. The columns represent the various small motor improvements; the rows represent the large motor options; the baseline motors are included for comparison. The top row (baseline large

SRM-1	SRM-2 with EEC		BASILINE	90% SOLIDS	90% SOLIDS PLUS HEAD END WEB	2 in. STRETCH WITH 90% SOLIDS AND HEAD END WEB	5 in. STRETCH WITH 90% SOLIDS AND HEAD END WEB
	SCHEDULE MONTHS ATP TO FIRST FLIGHT			33 months	33 months	40 months	40 months
BASELINE			PAYLOAD				
			5109	5175	5154	5232	5234
90% SOLIDS	without EEC	34 months	5376	5607			
	with EEC	34 months	5509	5781			
90% SOLIDS PLUS HEAD END WEB	without EEC	34 months	5530		5962	6158	
	with EEC	34 months	5642		6132	6510	
PLUS 3 in. STRETCH with 90% SOLIDS AND HEAD END WEB	without EEC	41 months	5649		6143	6535	
	with EEC	41 months	5757		6287	6761	
OR 8 in. STRETCH with 90% SOLIDS AND HEAD END WEB	without EEC	41 months	5857		6421	6945	7401
	with EEC	41 months	5951		6538	7093	7615

- 1 The 8-inch SRM-1 or the 5-inch SRM-2 each add 4 inches to IUS length. Orbiter attach fittings are available for 4 and 8-inch IUS length increase.
- 2 The schedules are based on phased developments. Add 6 months contingency if simultaneous developments are planned.
- 3 Performance assumes that modified interstage (due to stretch of SRM-2) and motor support cone are graphite epoxy composites.

**Figure 6-2: Geosynchronous Orbit Payload Performance for Motor Improvement Options**

motor) reveals that an improvement to the small motor by itself is not much of a benefit. In order to achieve the benefits shown, propellant has to be offloaded. Alternately, the first column shows that large motor improvements result in significant benefits. Further study reveals that once a large motor improvement is brought on-line, a small motor improvement will provide a significant benefit.

Planning is underway with the Air Force Space Division to implement a development program for the 8-inch stretched SRM-1. If this program is implemented, the capability will be available in 1986.

### COST REDUCTION

Studies have been conducted to identify changes to the IUS which enable appreciable recurring cost reduction with an acceptable non-recurring cost for implementation. Since the redundant avionic system represents 60% of the recurring cost, it is an obvious area in which to concentrate a cost reduction study.

A definite design concept has emerged with design activities now initiated. The design eliminates all of the redundant avionics boxes. The resultant reliability is 0.969. This is due to the built-in redundancy in the inertial measurement unit, the high quality SAMSO standard 73-2C piece parts and retention of all MIL-STD-54D test requirements.

The elimination of the redundant units provides a significant weight reduction which also translates into performance improvement. This performance improvement allows the small motor extendible exit cone to be deleted. The overall result is a recurring cost reduction of 20% with a payload-to-geosynchronous orbit increase of about 350 pounds.

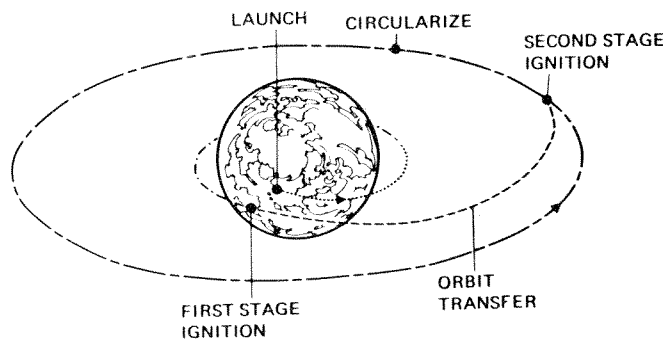
### HEAVY PAYLOAD CAPABILITY

Current IUS application studies show an increasing need for a heavier payload capability for future space missions. IUS two-stage vehicle product development activities have been conducted to study and assess the

Space Transportation System's upper stage vehicle requirements for payload configurations approaching 16,000 pounds.

The structural test vehicle (STV) was statically tested to demonstrate an 8,000-pound spacecraft capability. This testing at Boeing's Kent Space Center qualified the IUS structure to STS-specified ultimate loads with no detrimental deformation or yielding.

A typical heavy payload mission configuration relies on the IUS first and second stage solid rocket motor burns ("bang-bang" sequence) to leave the orbiter parking orbit. An integral propulsion system in the spacecraft then completes the transfer orbit and establishes the final orbit placement as shown in Figure 6-3.



**Figure 6-3: Typical Heavy Payload IUS/Spacecraft Mission**

The IUS two-stage vehicle capability can be improved by increasing the gage of the equipment support section (ESS) skin panels and stiffeners, and by increasing the cross-section of the longeron elements. Maximum capability is determined by the location of the payload center-of-gravity above the IUS interface.

Preliminary analyses verified the IUS structural capability to support the Galileo spacecraft (11,897 pounds) and the VOIR spacecraft (14,760 pounds). The IUS required no structural modifications to support Galileo and required only the modifications discussed above to support the VOIR spacecraft. Both spacecraft configurations did incorporate a third stage motor to reach final escape velocity for planetary missions after the IUS first and second stage burns.

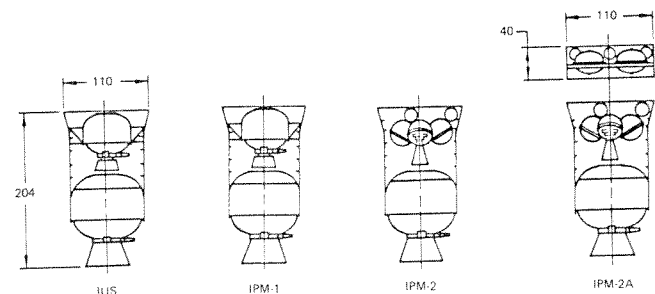
On-going analyses of possible IUS/spacecraft configurations have continued to confirm that the present IUS two-stage configuration is usable "as-is" for many heavy payload applications.

## INTEGRATED PROPULSION MODULE

Simplified, lower-cost IUS derivatives are under study for spacecraft customers who desire an integrated propulsion module. These are designated as IPM vehicles, and all are two-stage rockets. The IPM vehicles rely on the navigational and computational capabilities of the spacecraft, but use as many of the IUS components as practical to minimize development cost.

There are presently three versions (Figure 6-4): an all solid propellant vehicle; a hybrid, solid propellant first stage with a liquid propellant second stage; and an extended capability hybrid vehicle. They are designed to use IUS and other existing space-proven hardware while maintaining the IUS external configuration so that they are compatible with the IUS airborne support equipment in the Shuttle.

The IPM family can provide the energy to lift a 5,500 to 6,000-pound spacecraft from the Shuttle parking orbit to geosynchronous orbit. With an improved SRM-1, these capabilities



**Figure 6-4: The IPM Family**

can be increased to the 6,700 to 8,000-pound range. Specific payload capability is a function of the particular spacecraft and IPM configuration.

## PERIGEE STAGE VEHICLES

The IUS solid rocket motors are logical candidates for use in perigee stage designs. Preliminary design concepts were prepared for

each of these motors. Spinning and three-axis-control concepts were evaluated for both motors.

The small motor versions are comparable to PAM-DII in performance. Users in this size range, however, do not want the three-axis control, and for that reason the activities on small motor perigee stages have been suspended.

For the large motor perigee stages, the opposite is true. For this spacecraft weight range, three-axis control is preferred to spinning. Hence, the effort has been concentrated on a three-axis-controlled large solid rocket motor perigee stage.

The performance predictions for this perigee stage are already available and are shown in Figure 6-5. Also shown is the growth version's performance which uses the proposed 8-inch stretched SRM-1. The performance shown is the payload-to-geosynchronous transfer orbit as a function of the plane

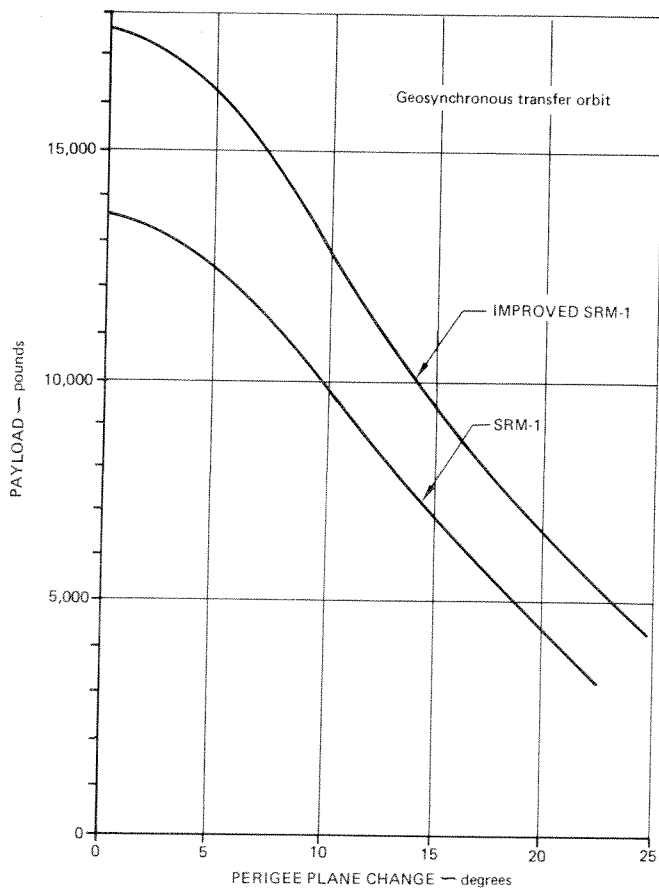


Figure 6-5: Transfer Orbit Stage Payload-Plane Change Capability

change accomplished during the perigee burn. Preliminary mission accuracy requirements for this system are  $\pm 610$  n.mi. at apogee radius,  $\pm 10$  n.mi. at perigee radius,  $\pm 0.7^\circ$  inclination and  $\pm 1.5^\circ$  argument of perigee.

The perigee stage or transfer orbit stage (TOS) design concept as conceived by Boeing utilizes as many of the IUS space-proven components as practical to minimize the development cost. Its external configuration consists of the IUS first stage and a modified IUS equipment support section which contains the TOS equipment complement (Figure 6-6).

The TOS equipment complement features a state-of-the-art avionics module for the guidance, navigation and control functions, and an inertial sensor unit. A microprocessor

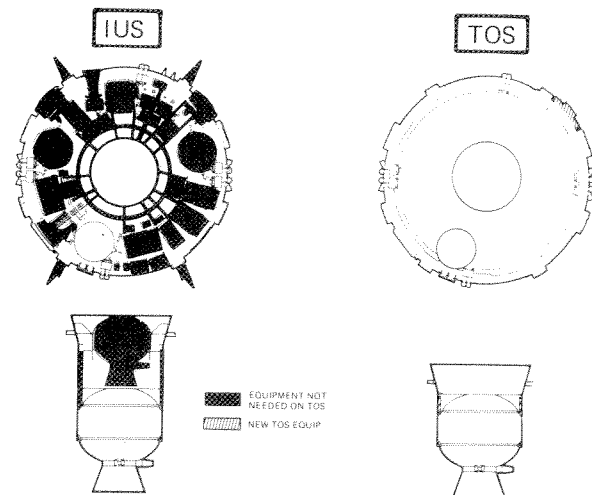


Figure 6-6: IUS/TOS Derivation

in the avionics module executes the TOS flight program. The software is modular in form which easily accommodates future growth and adaptability.

The following are some of the spacecraft services that the TOS is designed to supply: the TOS uses a commercial version of the IUS ASE which supports the TOS/spacecraft while the payload combination is in the orbiter; the TOS has dual spacecraft launch capability; and the TOS furnishes 16 discrete relay closures to return spacecraft-provided power to spacecraft discrete signals (8 signals per spacecraft for a dual launch). These relay closures are capable of being issued at any time in the TOS operational sequence. Fol-

lowing deployment, the TOS can perform attitude heading, thermal maneuvers, and post-burn spin-up, if necessary, for those spacecraft which require spin stabilization.

### TWIN-STAGE AND THREE-STAGE VEHICLES

The twin stage is the name given to a vehicle configuration which uses two large motors. The vehicle design has already been through a preliminary design review. The earth orbital performance for the vehicle is depicted in Figure 6-7.

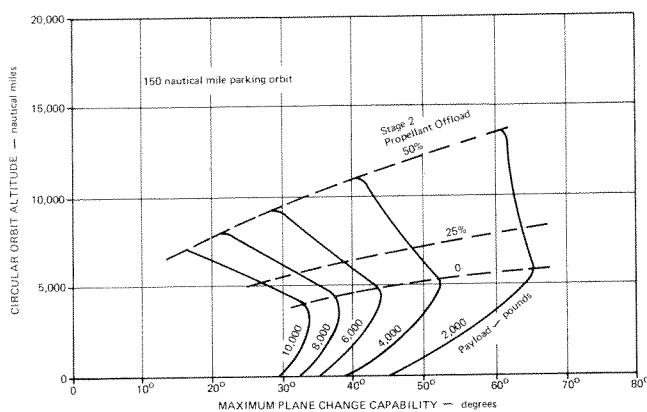


Figure 6-7: IUS Twin-Stage Earth Performance Capability

A three-stage vehicle has also been through a preliminary design review. This concept uses a twin-stage vehicle with a third stage. Two types of third stages were studied: a spinner and a three-axis-controlled vehicle.

Figure 6-8 shows the planetary performance of the two-stage IUS, the twin-stage and the three-stage vehicles as a function of orbit altitude. The three-stage vehicle performance represents the spinner version, but the performance for the three-axis-controlled vehicle is virtually identical.

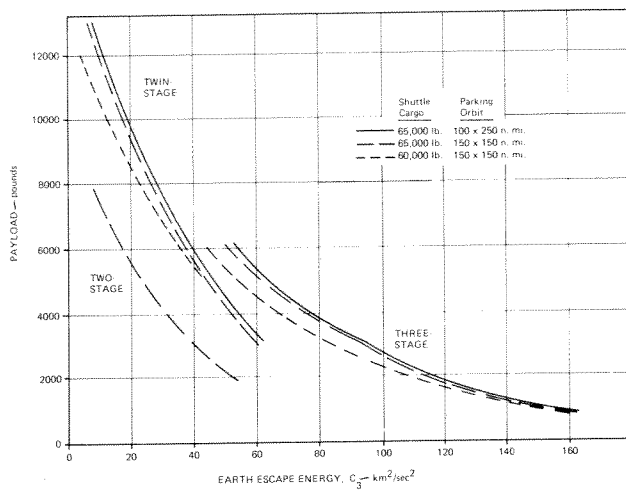


Figure 6-8: Planetary Payload Capability

## GLOSSARY

AFETRM	Air Force Eastern Test Range Manual	EMC	Electromagnetic Compatibility
AFTA	Aft Frame Tilt Actuator	EPS	Electrical Power Subsystem
ATP	Authority to Proceed	ESS	Equipment Support Section
Bi-Ø	Bi-Phase	EVA	Extra-vehicular Activity
Btu	British Thermal Unit	F	Fahrenheit
C	Centigrade	FSD	Full Scale Development
CCAFS	Cape Canaveral Air Force Station	fps	feet per second
CDR	Critical Design Review	FSK	Frequency Shift Key
CG	Center of Gravity	G	Gravitational Level of Acceleration
CLK	Clock	GFE	Government-Furnished Equipment
CMD	Commands	GNS	Guidance and Navigation System
COMSEC	Communication Security Equipment	GN <sub>2</sub>	Gaseous Nitrogen
CPU	Central Processing Unit	GPC	General Purpose Computer
C <sub>3</sub>	Launch Vehicle Energy (km <sup>2</sup> /second <sup>2</sup> )	GSE	Ground Support Equipment
dB	Decibels	GSO	Geosynchronous Orbit
dc	direct current	GTS	Guam Tracking Station
DIA	Diameter	h	hour
DMS	Data Management Subsystem	HEPA	High Efficiency Particulate Air (Filter Unit)
DRM	Design Reference Mission	HMX	Cyclotetrometholene Tetronitromine
DSCS	Defense Satellite Communications System	HTS	Hawaii Tracking Station
EEC	Extendible Exit Cone	Hz	Hertz
EED	Electro-explosive Device	I&LS	Integration and Launch Services
ELS	Eastern Launch Site	ICD	Interface Control Document

IMU	Inertial Measurement Unit	OPF	Orbiter Processing Facility
IOS	Indian Ocean Station	PAM-DII	Payload Assist Module–Delta
IUS	Inertial Upper Stage	PCU	Power Control Unit
kbps	kilobits per second	PDU	Power Distribution Unit
kg	kilogram	PLF	Payload Fairing
kHz	kiloHertz	PN	Pseudo-noise
kw	kilowatt	ppm	parts per million
lb	pound	PRLA	Payload Retention Latch Actuator
LRU	Line Replaceable Unit	psia	pounds per square inch absolute
ma	milliampere	PSK	Phase Shift Key
MAD-COM-E	Command Software for IUS (Air Force Satellite Control Facility)	Q	Dynamic Pressure
MDL	Mission Data Load	RAAN	Right Ascension of the Ascending Node
MECO	Main Engine Cutoff	RCS	Reaction Control System
MF	Mainframe	REM	Rocket Engine Module
MLU	Memory Load Unit	RF	Radio Frequency
MMSE	Multi-mission Support Equipment	RIMU	Redundant Inertial Measurement Unit
MPSR	Mass Properties Status Report	rms	root mean square
MSFC	Marshall Space Flight Center	rss	root sum squared
MST	Mobil Service Tower	RSS	Rotating Service Structure
NASA	National Aeronautics and Space Administration	RTN	Radial Tangential Normal
NHS	New Hampshire Station	SAB	Satellite Assembly Building
n.d.	no dimension	SAMSO	Space and Missile Systems Organization
n.mi.	nautical miles	SCF	Satellite Control Facility
ODDL	On-board Digital Data Load	SCU	Signal Conditioner Unit
OFS	Operational Flight Software		



SD	Space Division	STS	Space Transportation System (Shuttle)
SGLS	Space-to-Ground Link Subsystem	TDRSS	Tracking and Data Relay Satellite System
SINC	Satellite Integration Support Contractor	TM	Telemetry
SIU	Signal Interface Unit	TT&C	Telemetry, Tracking, and Command
SMAB	Solid Motor Assembly Building	TVC	Thrust Vector Control
SMTH	Standard Mix Table Harness	USAF	United States Air Force
SPIF	Shuttle Payload Integration Facility	V	Velocity
SPO	System Program Office	Vdc	Volts direct current
SRB	Solid Rocket Booster	VPF	Vertical Processing Facility
SRM	Solid Rocket Motor	VTS	Vandenburg Test Station
STDN	Space Tracking and Data Network		

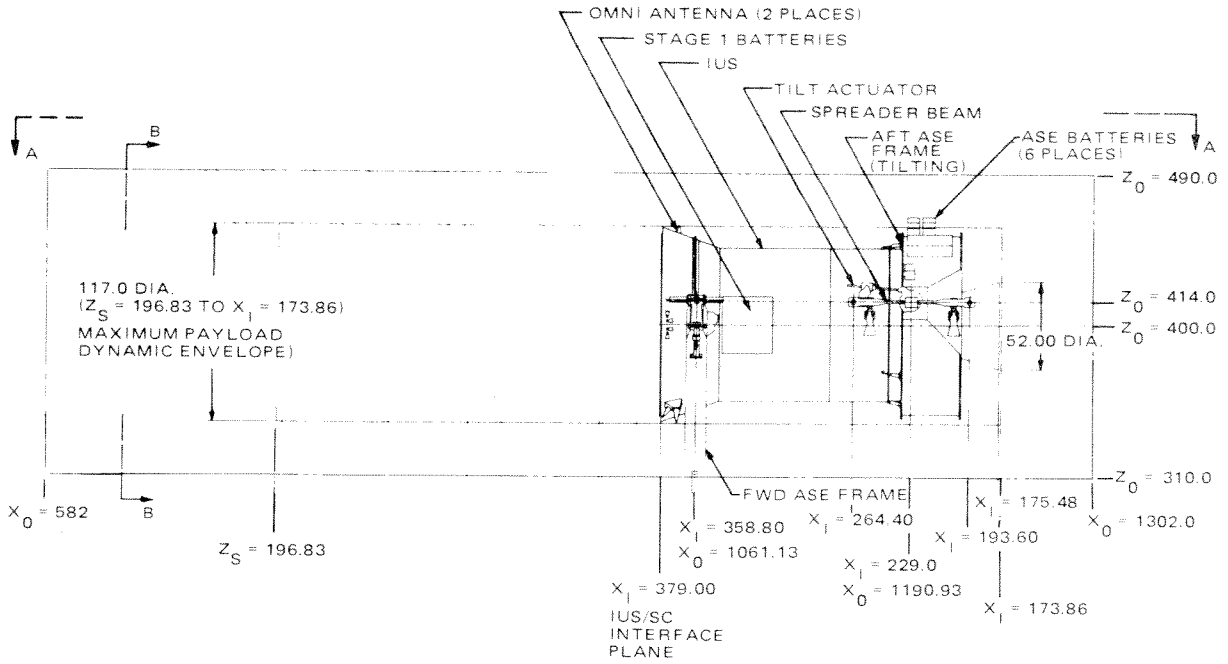


**APPENDIX A  
ASE  
STRUCTURAL/MECHANICAL  
DESIGN DATA**

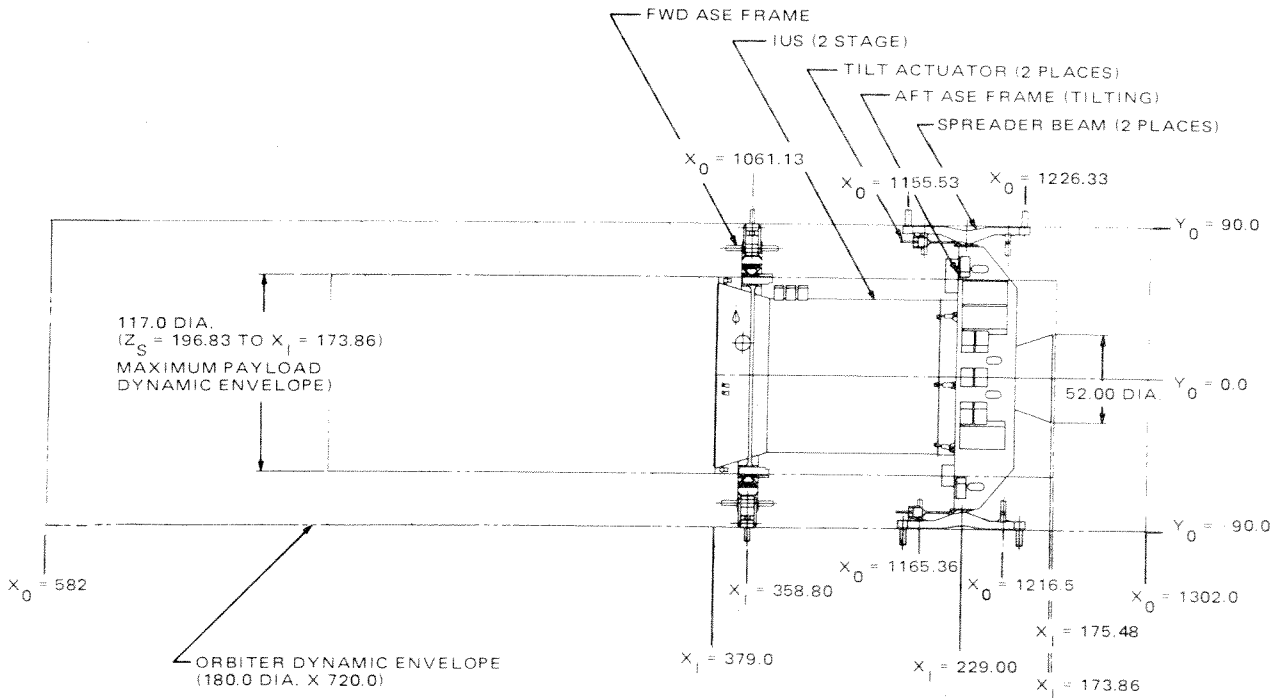
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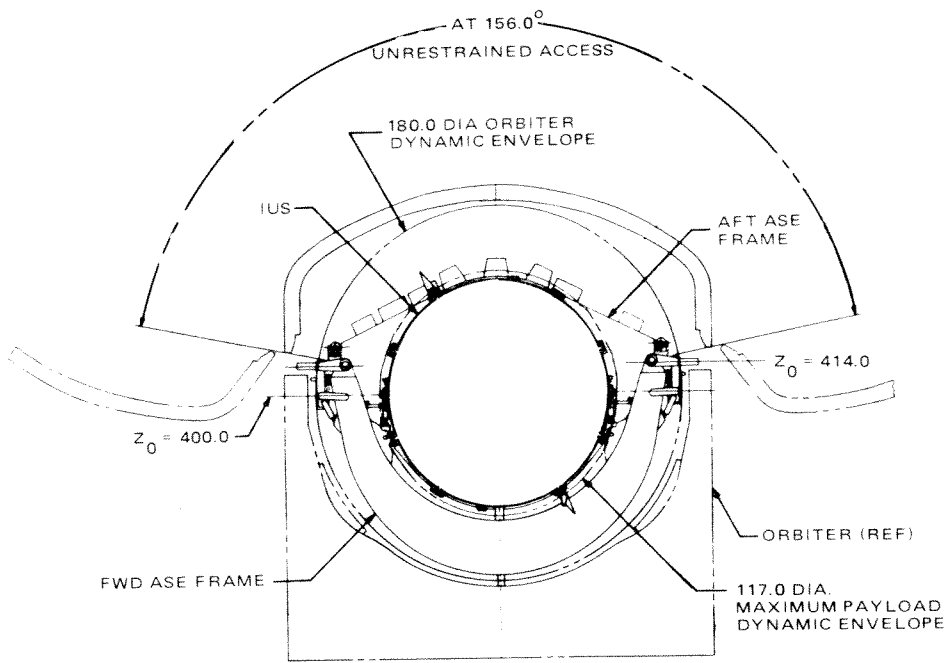
# INTEGRATED ASE/IUS/PAYLOAD CONFIGURATION



**SIDE VIEW**



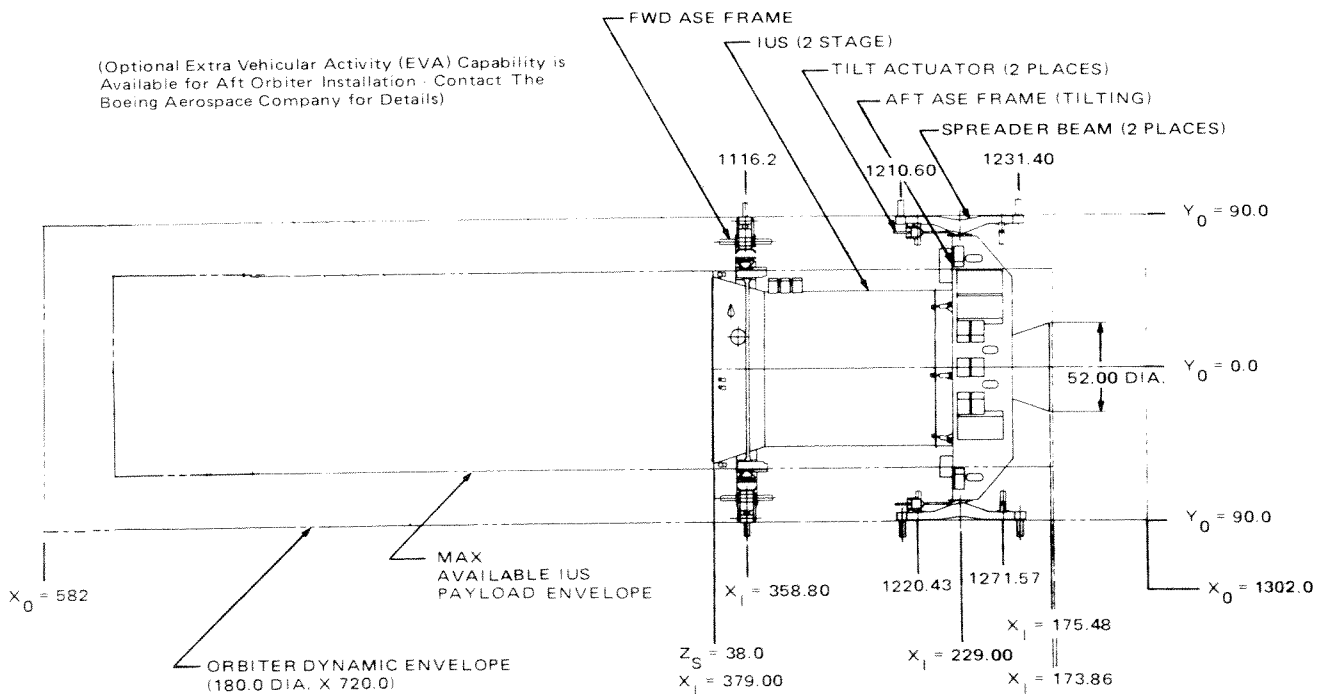
**VIEW A-A**



VIEW B-B

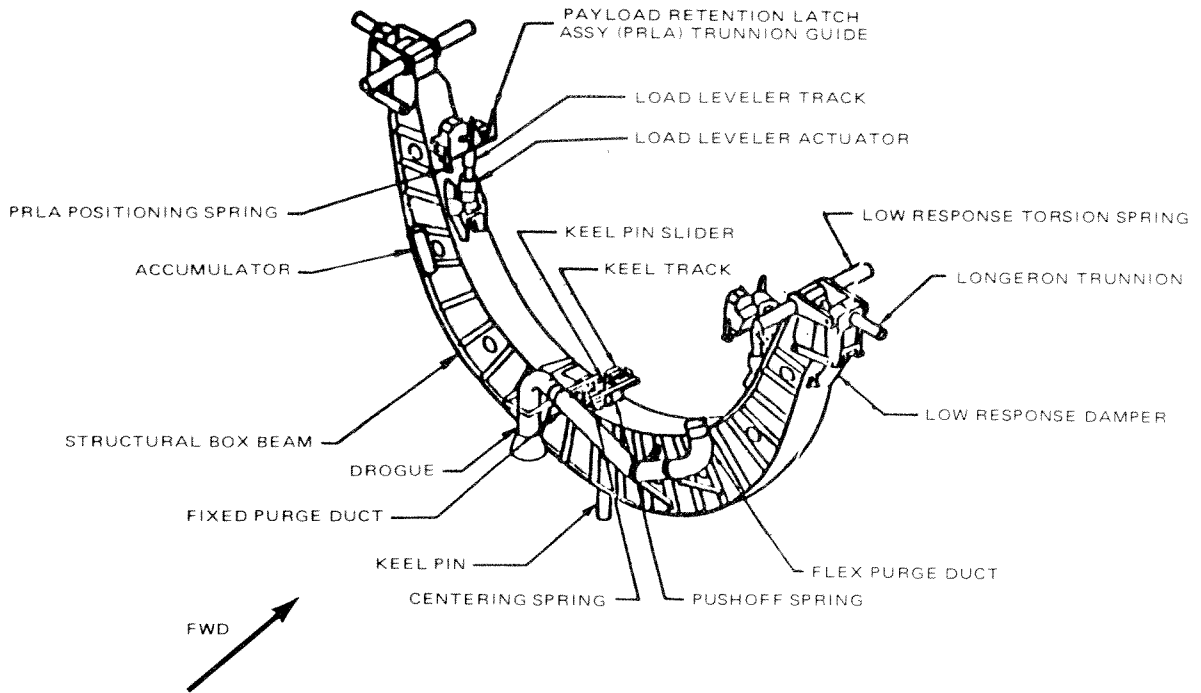
## OPTIONAL INTEGRATED ASE/IUS/PAYLOAD CONFIGURATION

(Optional Extra Vehicular Activity (EVA) Capability is Available for Aft Orbiter Installation - Contact The Boeing Aerospace Company for Details)



PLAN VIEW

# FORWARD ASE FRAME

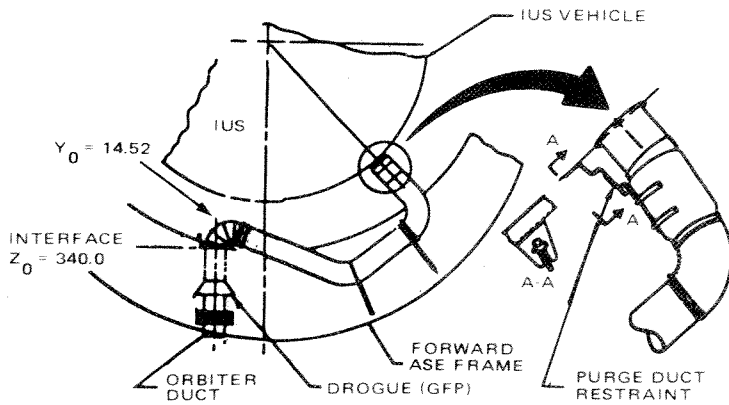


• Forward ASE Frame Stiffness

Z-axis = 24,000  $\begin{matrix} +2500 \\ -2000 \end{matrix}$  lb/in

• Forward ASE Weight 1575 lb.

## PURGE DUCT

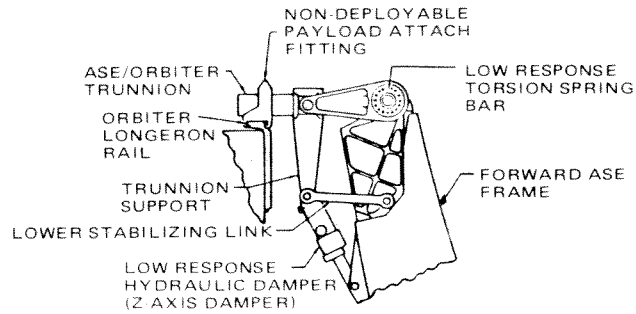


VIEW FACING FWD.

	REQUIREMENT	TESTED TO
FLOW RATE	68 lbs/min	77.2 lbs/min
PRESSURE	0.72 psi	1.2 psi

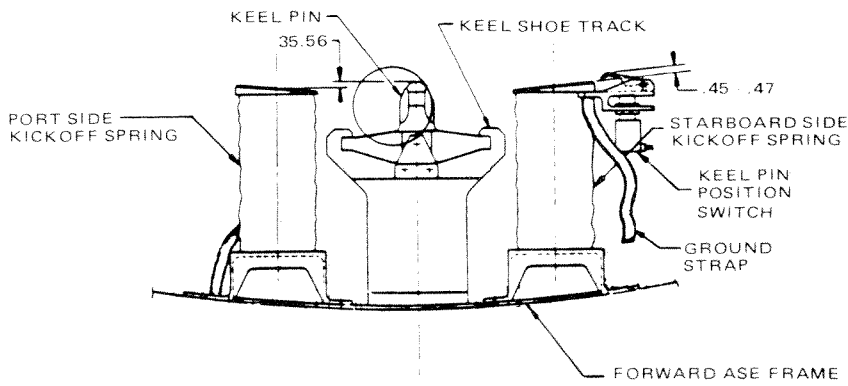
# FORWARD ASE FRAME DETAILS

## FORWARD ASE FRAME LOW RESPONSE SYSTEM

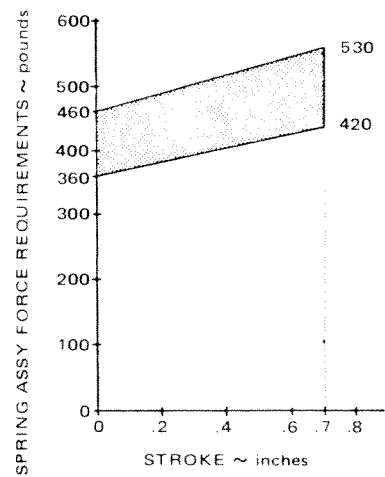


- Torsion Bar/Tube Assembly Stiffness  
Z Axis - 15,300 pound/inch per side

## IUS VEHICLE FORWARD KICKOFF SPRINGS

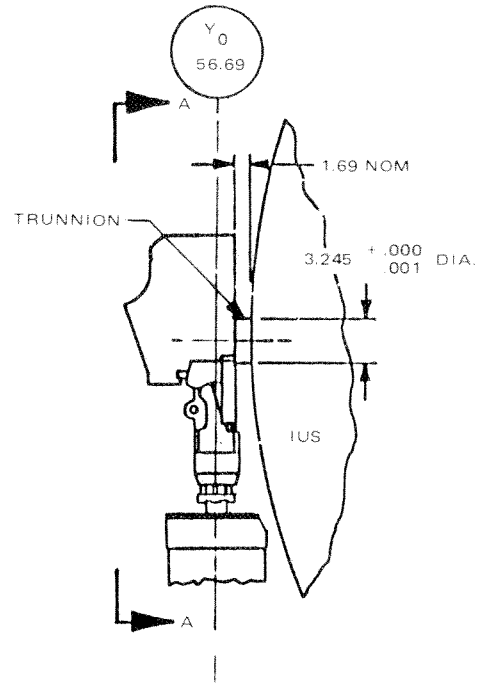
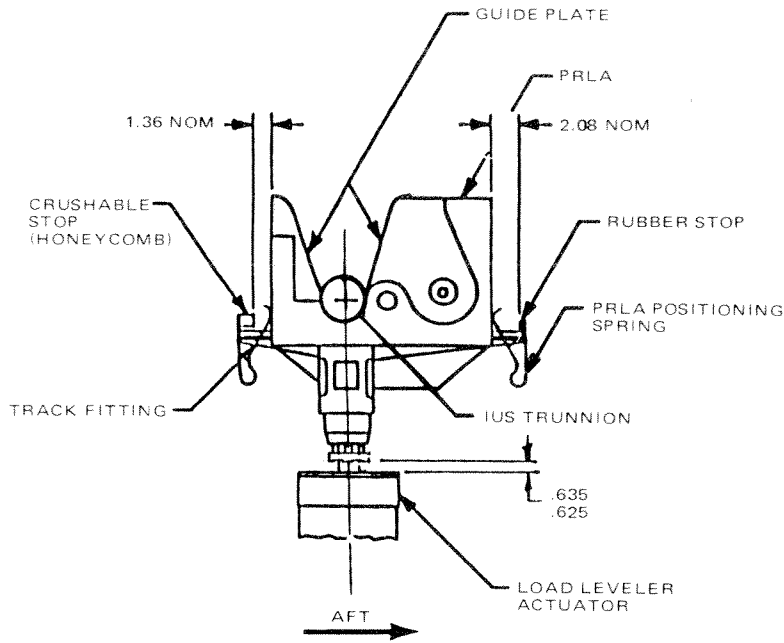


## KICK-OFF SPRING ASSY (SINGLE) ACCEPTANCE REQUIREMENT





# LOAD LEVELER/PAYLOAD RETENTION LATCH ASSEMBLY INSTALLATION (PRLA)

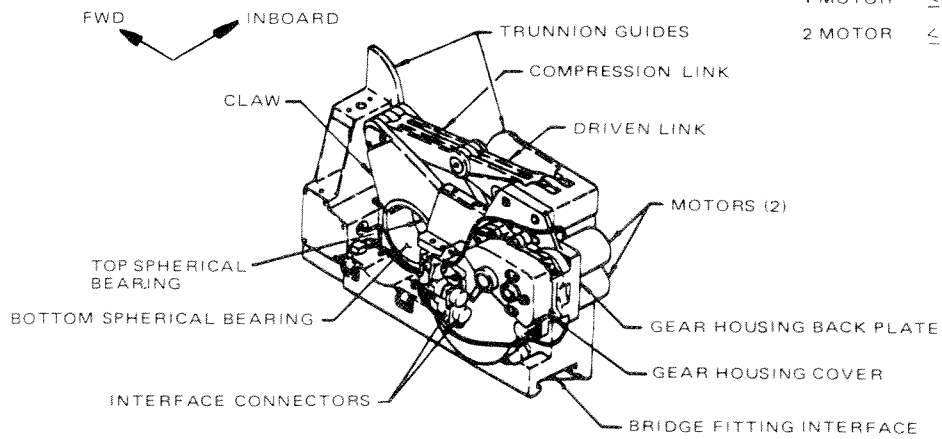


**VIEW A-A  
(LOOKING INBOARD)**

## PRLA

### PRLA OPERATING TIME (seconds)

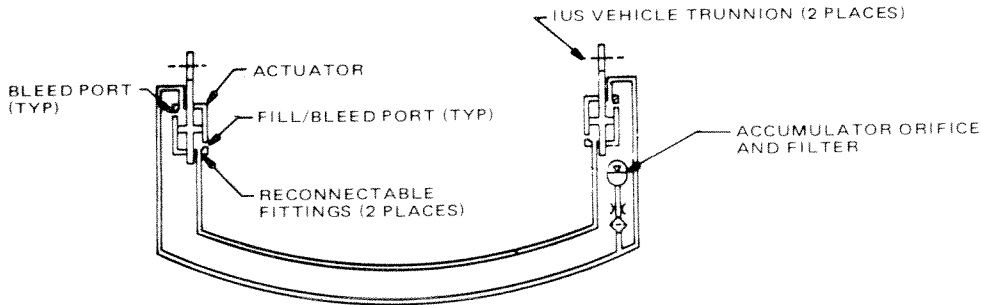
	SPEC	NOMINAL
1 MOTOR	≤ 60	40
2 MOTOR	≤ 30	20



### PRLA POSITION STATUS: (A6 PANEL)

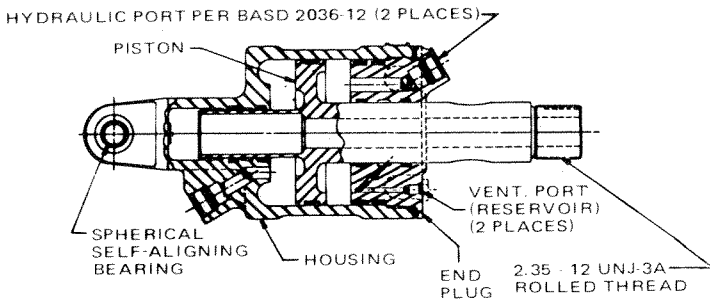
- Unlatched
- Ready to latch (Trunnion .7 to .9 inch from stowed position)
- Latched

## LOAD LEVELER FUNCTIONAL SCHEMATIC LOAD



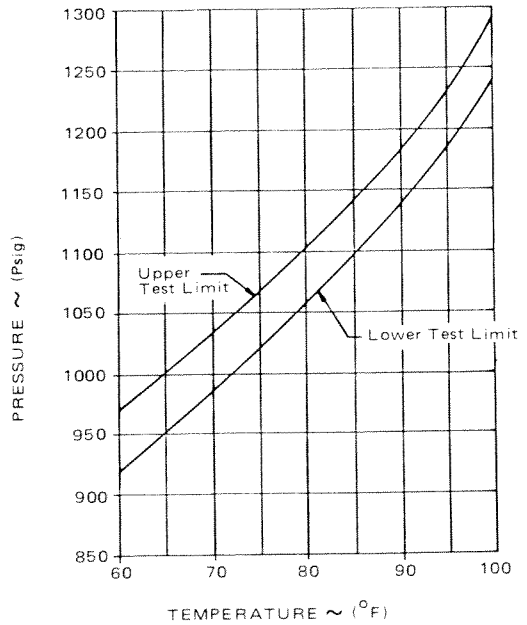
- Hydraulic fluid used on both sides of the piston
- Welded hydraulic system except as noted
- Dual rod seals

## LOAD LEVELER ACTUATOR

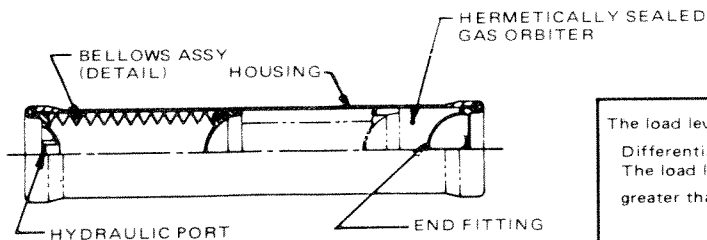


- Bleed/Fill valve two places (not shown) Located 90° from hydraulic ports
- Seals per MIL-R-25988/1 dynamic  
Seals per MIL-R-25988/4 static
- Stroke = ± 1.00 inch  
(Stroke limited to ± .625 when installed)

## LOAD LEVELER ACCUMULATOR RANGES



## FORWARD ASE LOAD LEVELER ACCUMULATOR

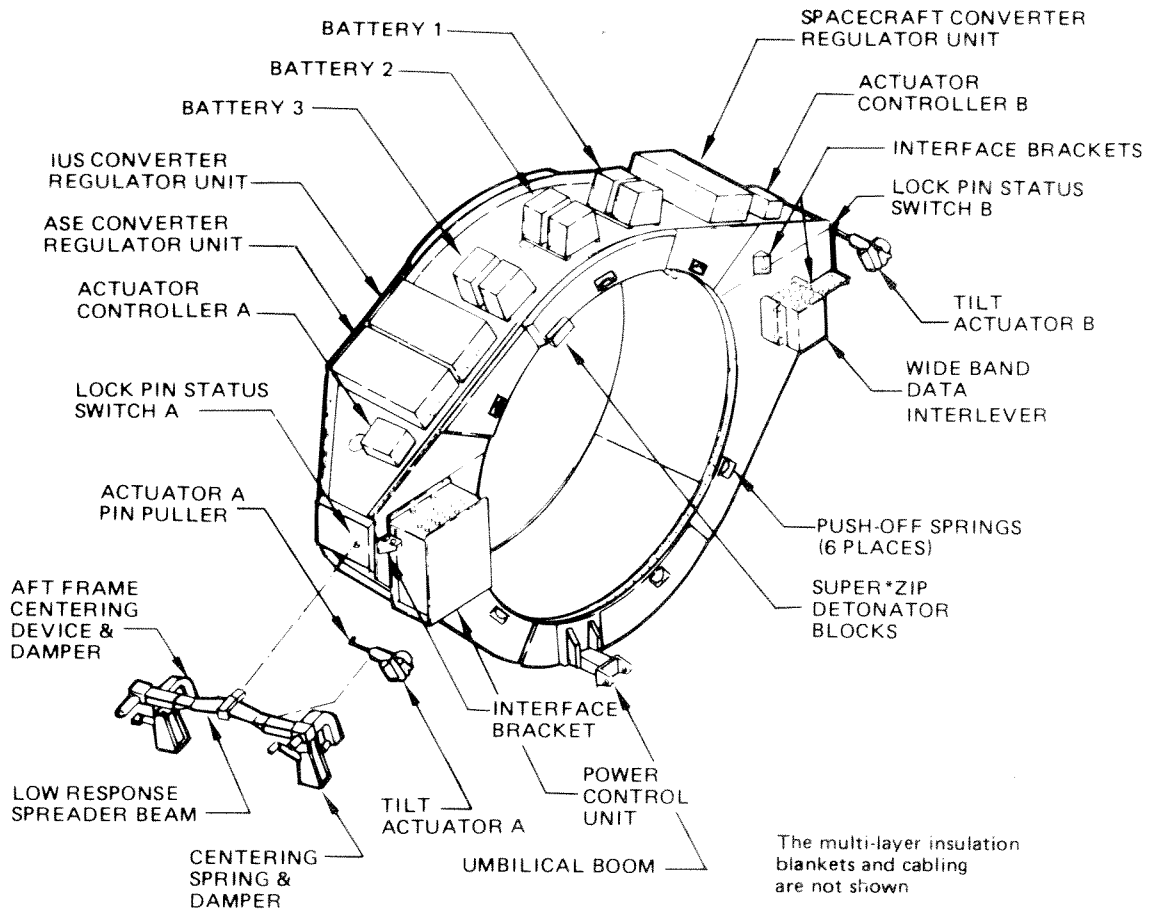


- Hydraulic accumulator assy  
Volume: 60 cubic in.
- Operating pressure: 3000 psi max

The load leveler operational characteristics shall be as follows:  
 Differential load due to friction must be less than 450 pounds.  
 The load leveler system shall provide a damping coefficient of no greater than  $\frac{275 \text{ lb} \cdot \text{sec}^2}{\text{in}^2}$  for differential piston velocity.

The load leveler operations shall not exceed a deflection of ± 0.60 inches. The load leveler shall be operated in such a manner that the maximum piston velocity shall not exceed 14.0 inches per second or the maximum differential load shall not exceed 14000 pounds.

# AFT ASE FRAME

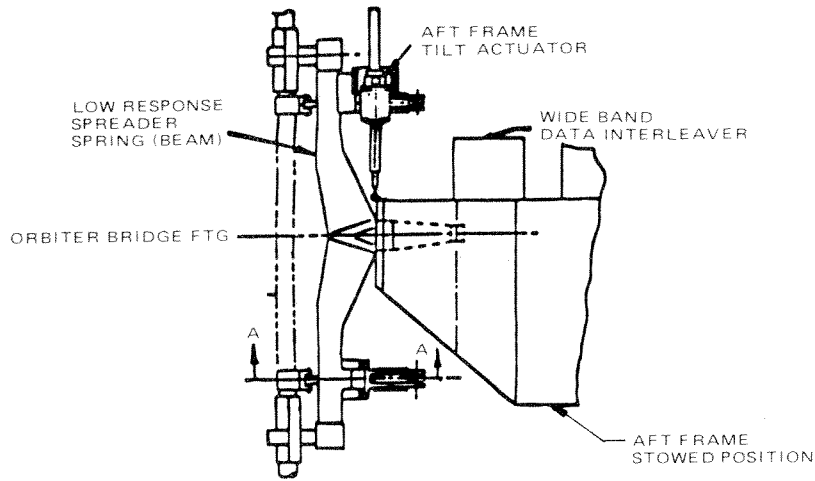


- Tiltable Aft Frame 2610 pounds
- Aft Frame Tilt, Centering and Low Response Mechanisms 1441 pounds
- Aft ASE Frame Stiffness:

Z-axis 16,000 + 2500 - 2000 pounds/inch

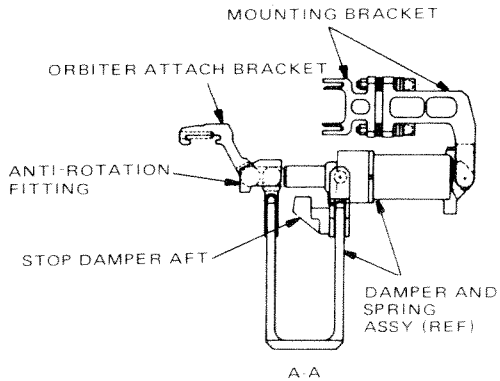
# AFT ASE FRAME DETAILS

## AFT ASE FRAME LOW RESPONSE SYSTEM



- Low response Spreader Beam Stiffness:  
Z axis: 9,200 pounds/inch per side

## SELF-CENTERING DEVICE

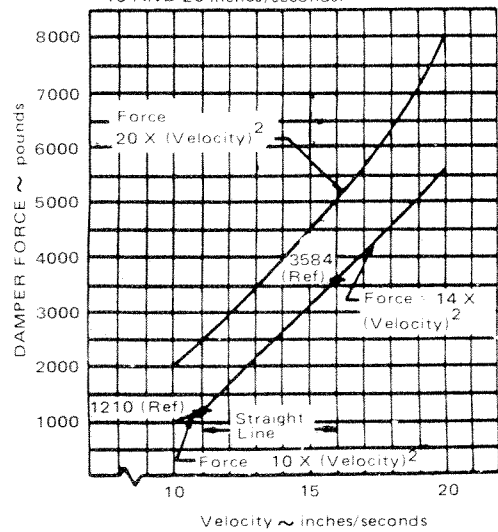


- Self-Centering Spring Stiffness:  
2470 + 270  
- 370 pound/inch (Per spring)

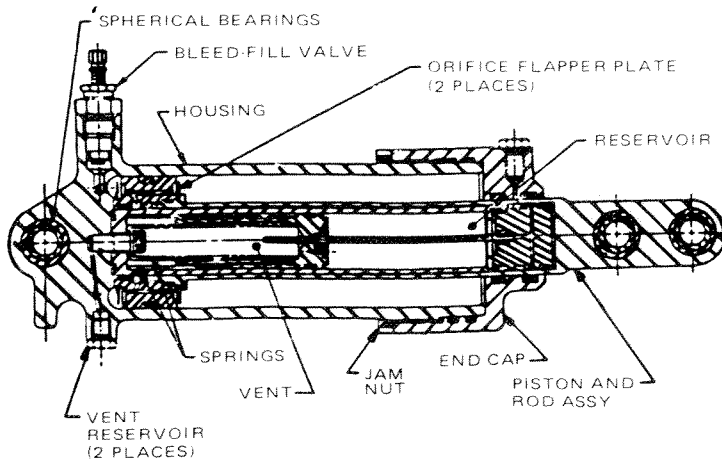
## AXIAL LIMIT LOADS

COMPRESSION 15000 pounds  
TENSION 15000 pounds

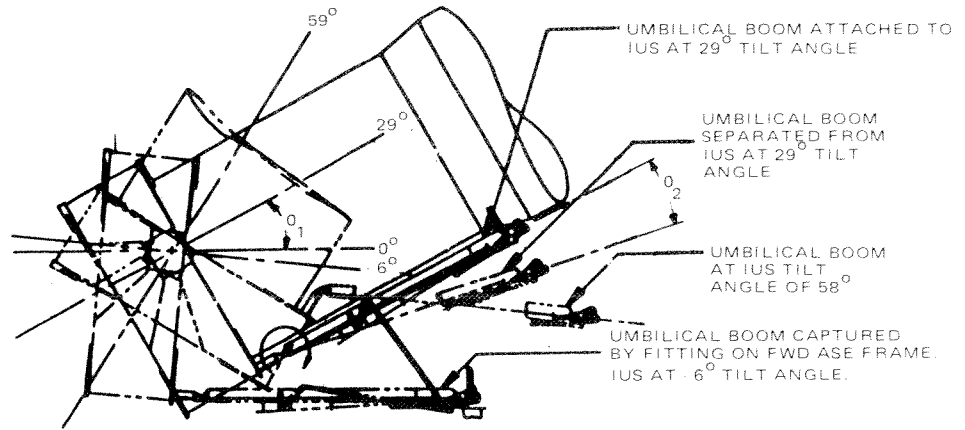
FORCE AS SHOWN FOR A FREQUENCY OF 3 Hz AND VELOCITIES BETWEEN 10 AND 20 inches/seconds.



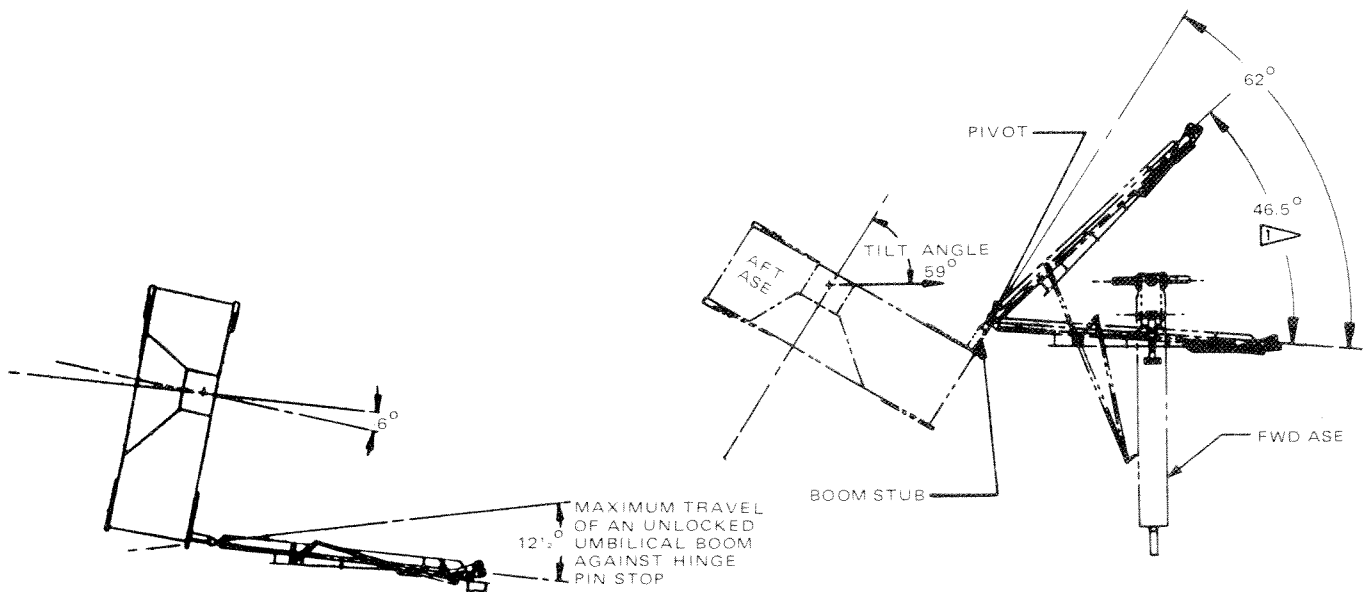
## AFT FRAME Y-DAMPER ASSEMBLY



# UMBILICAL BOOM OPERATION



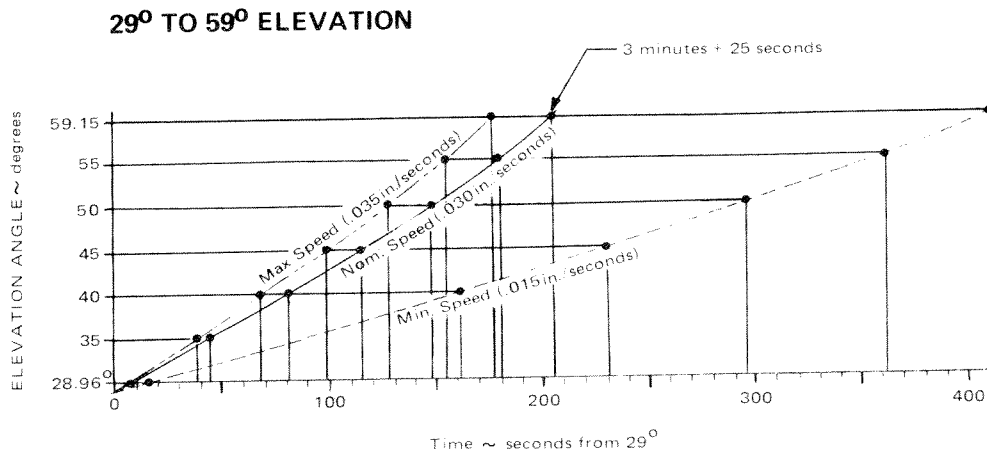
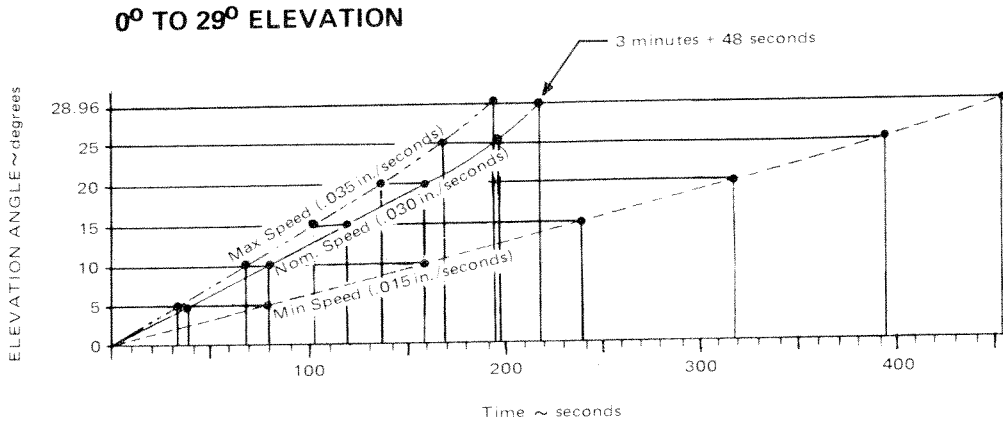
VEHICLE TILT ANGLE 0 <sub>1</sub>	BOOM SEPARATION ANGLE		TORSION SPRING HINGE MOMENT in/lbs
	ANGLE 0 <sub>2</sub>	CLEARANCE inch	
29°	5°	11.5	155
39°	19.3°	36.7	139
59°	62°	113.3	82



▷ PHYSICAL STOP WITH ARMS IN TENSION

- Max vehicle tilt angle with boom latched (AFTA stallout) 50°
- Vehicle tilt angle during backup umbilical plug release 35° to 39°
- Umbilical boom weight ~ 86 pounds

# DEPLOYMENT CHARACTERISTICS



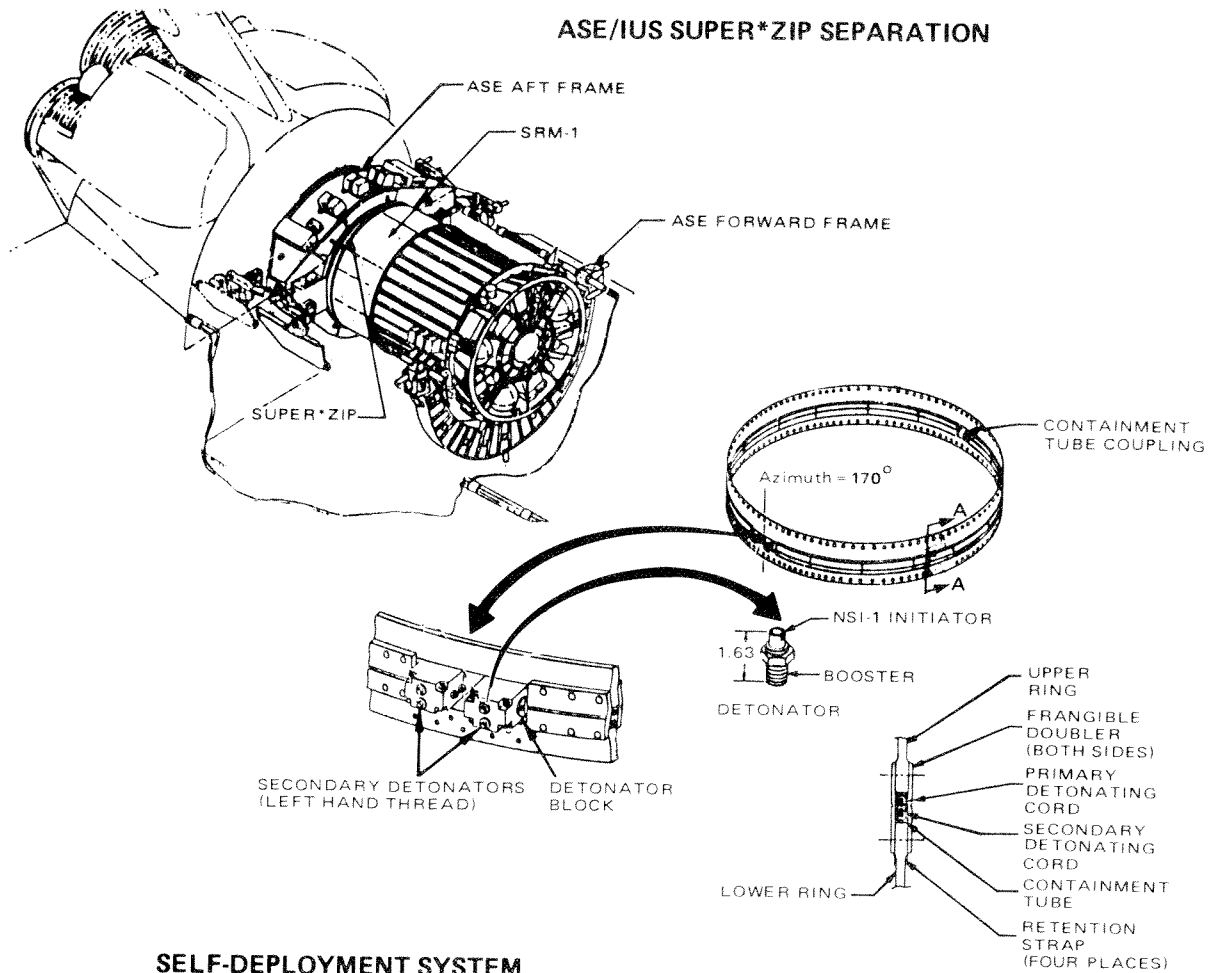
## VALUE FOR DEPLOYMENT ANALYSIS

AFT FRAME TILT ANGLE Degrees	0°	5°	10°	15°	20°	25°	28.96°	30°	35°	40°	45°	50°	55°	59.15°
AFTA LENGTH - inches	24.503	25.689	26.884	28.077	29.261	30.426	31.334	31.569	32.683	33.761	34.801	35.796	36.746	37.495
MOMENT ARM - inches	13.511	13.663	13.698	13.632	13.475	13.240	13.004	12.935	12.567	12.144	11.670	11.151	10.591	10.098
MIN ANGULAR RATE - deg/min	3.82	3.77	3.76	3.78	3.83	3.89	3.97	3.99	4.10	4.25	4.42	4.62	4.87	5.11
NOM ANGULAR RATE - deg/min	7.63	7.55	7.53	7.57	7.65	7.79	7.93	7.97	8.20	8.49	8.84	9.25	9.74	10.21
MAX ANGULAR RATE - deg/min	8.90	8.81	8.78	8.83	8.93	9.09	9.25	9.30	9.57	9.91	10.31	10.79	11.36	11.91
ANGULAR POSITION TOL* - degrees							± .43°							± 1.17°

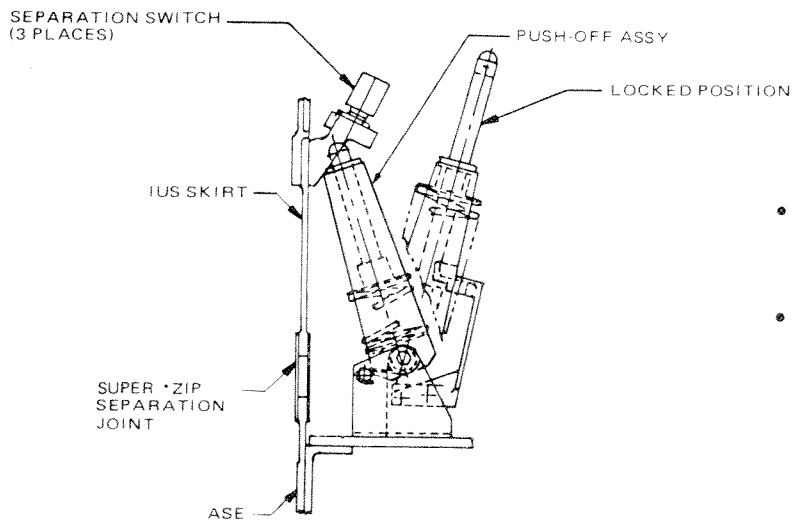
\* System Dead Band = ± .04 plus AFTA tolerance = ± .058 at 29° and = ± .135 at 58°.

# SEPARATION AND RESTOW

## ASE/IUS SUPER\*ZIP SEPARATION

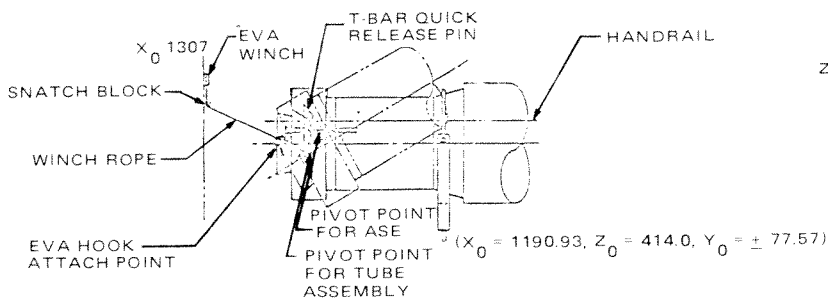
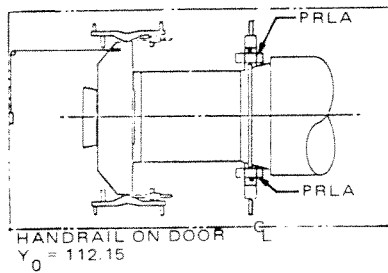


## SELF-DEPLOYMENT SYSTEM



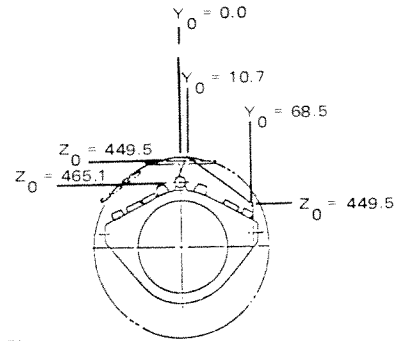
- Six self-deployment devices located at azimuths  $5.63^\circ$ ,  $64.13^\circ$ ,  $129.38^\circ$ ,  $185.63^\circ$ ,  $244.13^\circ$  and  $304.88^\circ$ .
- Minimum separation velocity = 0.35 ft/sec  
Initial force = 147 lb  $\pm$  12%  
Residual force = 46 lb  $\pm$  12% (per spring)

# EVA IUS/ASE MANUAL RESTOW\*

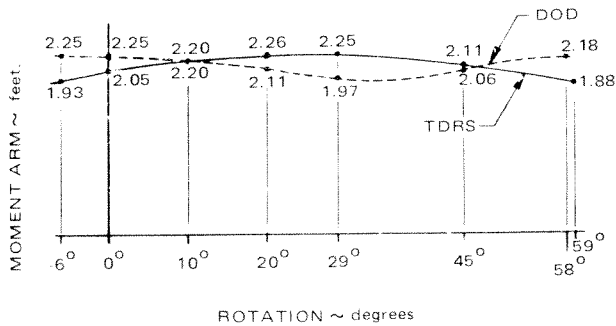


DOD X<sub>0</sub> = 1268.00 Y<sub>0</sub> = 69.5 Z<sub>0</sub> = 398.15

TDRS X<sub>0</sub> = 1212.93 Y<sub>0</sub> = 69.5 Z<sub>0</sub> = 398.15

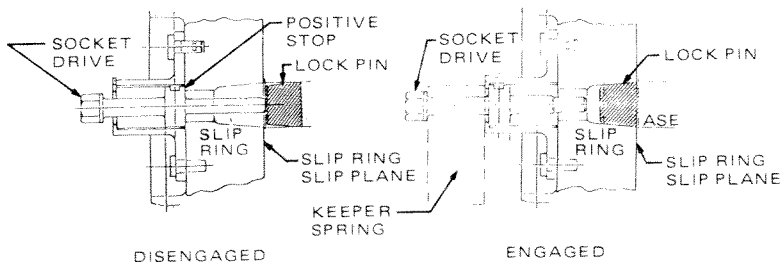


## WINCH TUBE MOMENT ARM



EVA WINCH CAPABILITY - 435 pounds

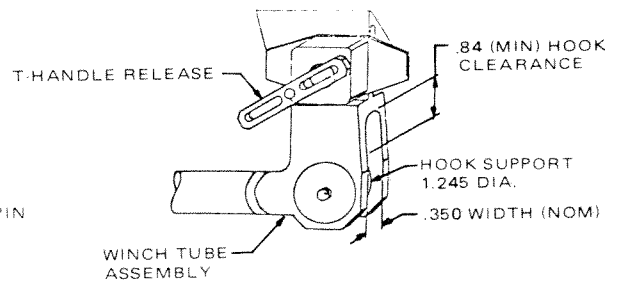
## SLIP RING DECOUPLER



- Torque to turn bolt - 9½ in/lbs minimum
- Turns to seat - 10¼ turns

\* (Optional Extra Vehicular Activity (EVA) Capability is Available for Aft Orbiter Installation - Contact Boeing Aerospace Company for Details)

## TILT ROD



- Force on "T" handle to release ~ 6.5 lbs
- Force on rod to re-engage fittings ~ 12 lbs.