ATLAS

MISSION PLANNER'S GUIDE

GENERAL DYNAMICS
Commercial Launch Services

MISSION PLANNER'S GUIDE FOR THE ATLAS LAUNCH VEHICLE FAMILY

Revision 3
APRIL 1992

This document supersedes the Mission Planner's Guide
for the Atlas Launch Vehicle Family — Revision 2, 10 July 1990

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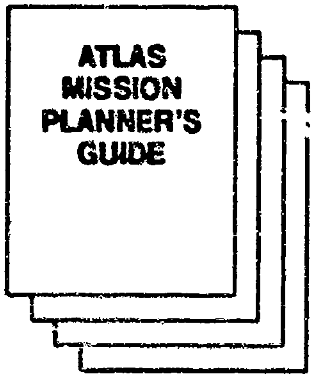
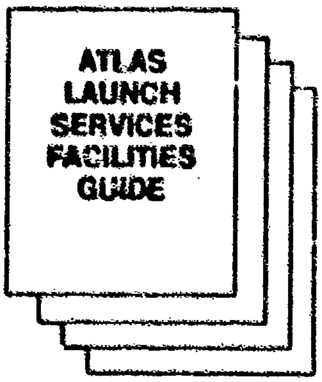


ATLAS LAUNCH VEHICLE FAMILY PERFORMANC
SPACECRAFT:
- FAIRING OPTIONS
- ADAPTER OPTIONS

INTERFACES
ENVIRONMENTS
MANAGEMENT
HISTORY



ATLAS LAUNCH SERVICES FACILITIES
SPACECRAFT GROUND PROCESSING
LAUNCH FACILITIES
SPACECRAFT TRANSPORTATION
SPACECRAFT SERVICES

ATLAS
MISSION PLANNER'S GUIDE

Revision Date	Revision Number	Change Description	Approval
21 March 1989	1		
10 July 1990	2		
April 1992	3-1	• Section 2 is a complete update of vehicle performance • Section 3: major updates include: — Structural interface capability — PLF acoustics environment — Separation system shock — Interface random vibration • Section 4: updated structural capabilities • Section 7, Enhancement Options, is new • Section 8, West Coast Atlas, is new • Significant technical changes in sections other than 2, 7, and 8 are identified by bars in margins	KE Kobit

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GENERAL DYNAMICS
Commercial Launch Services

This updated Atlas Mission Planner's Guide presents new information on the performance capabilities of Atlas. Higher performance is now offered as a result of a successful development program and an ongoing enhancement program. A greater range of vehicle configurations and performance levels is also offered to allow a closer match to customer requirements and to lower cost. The performance data is presented in sufficient detail for preliminary assessment of the General Dynamics vehicle family for your missions.

This guide includes essential technical and programmatic data for preliminary mission planning and preliminary spacecraft design. Interfaces are in sufficient detail to assess a first order compatibility. A brief description of the Atlas vehicles and the launch facilities is also given. See the companion Atlas Launch Services Facility Guide for spacecraft processing and launch services at Space Launch Complex 36.

This guide is subject to changes and will be revised periodically. In proposals or specific contracts, baseline information will be supplemented and updated to address specific customer program requirements.

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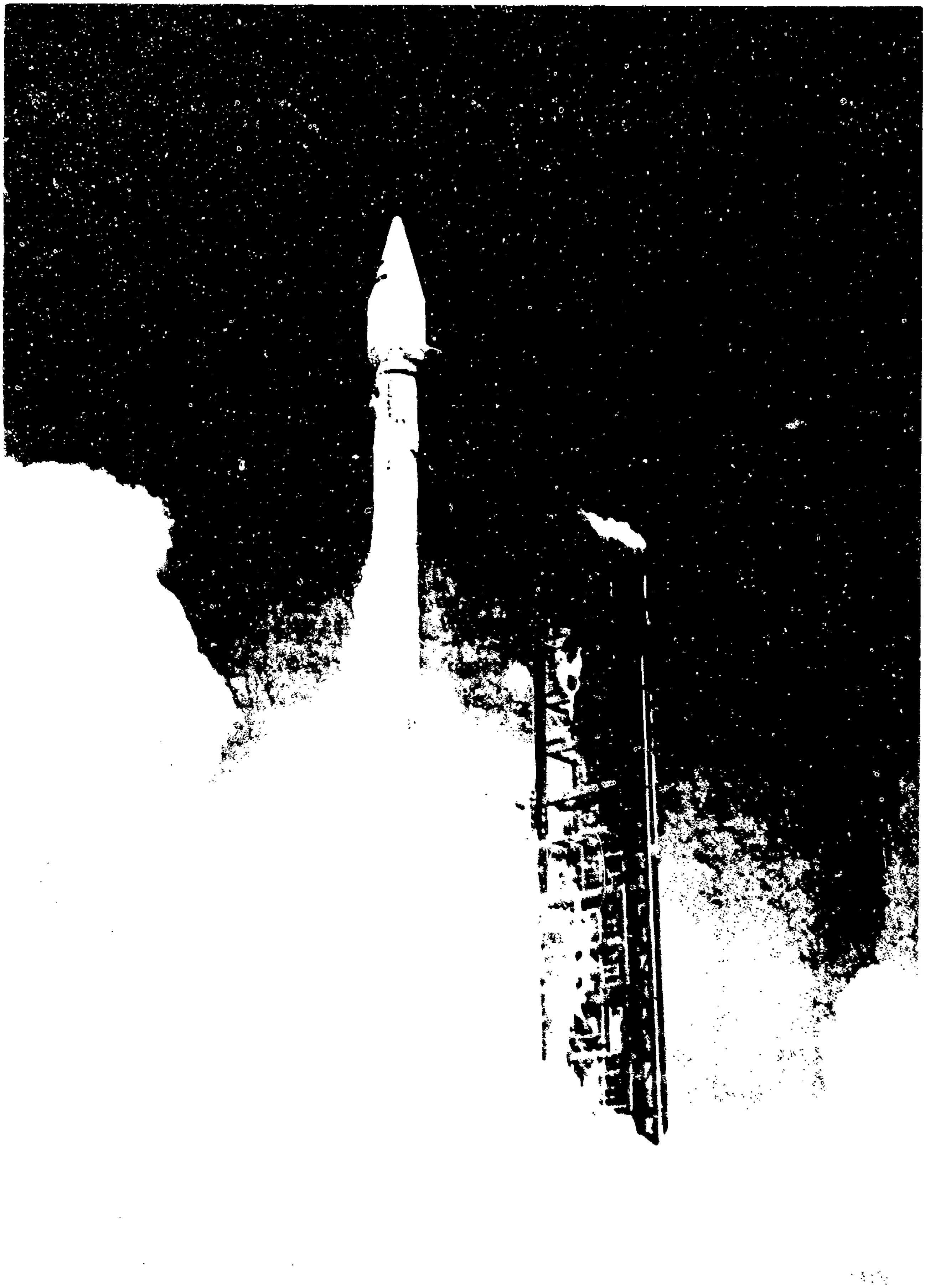


of the Atlas G/Centaur vehicle. It is designed to fly with the large payload fairing (14-foot diameter), which increases the payload volume to accommodate today's larger spacecraft (Figure 2).

Atlas II builds on the Atlas I configuration to provide increased performance capability. Atlas II upgrades include increased booster engine thrust, and lengthening of the propellant tanks. In addition, a new state-of-the-art guidance and navigation avionics suite, the inertial navigation unit (INU), has been added.

Atlas IIA is similar to the Atlas II except that the Centaur's Pratt & Whitney RL10 propulsion system has been uprated and the avionics has been upgraded with the addition of the remote control unit (RCU).

Atlas IIAS is similar to the Atlas IIA except for the addition of four Castor IVA solid rocket motors.

SPACECRAFT ACCOMMODATIONS

General Dynamics offers two payload fairing configurations and seven spacecraft adapters (five with separation systems) to accommodate a wide range of spacecraft requirements. Figure 3 illustrates the envelopes for both the large payload fairing (LPF) and the medium payload fairing (MPF). Both fairings are compatible with each of the Atlas vehicle configurations.

The Type A, A1, B, B1, C, C1, and D adapters provide industry standard mechanical interfaces (Figure 4). Each adapter may be used with any of the payload fairings and Atlas vehicles. The separation systems provided with the standard Type A, A1, B, B1,

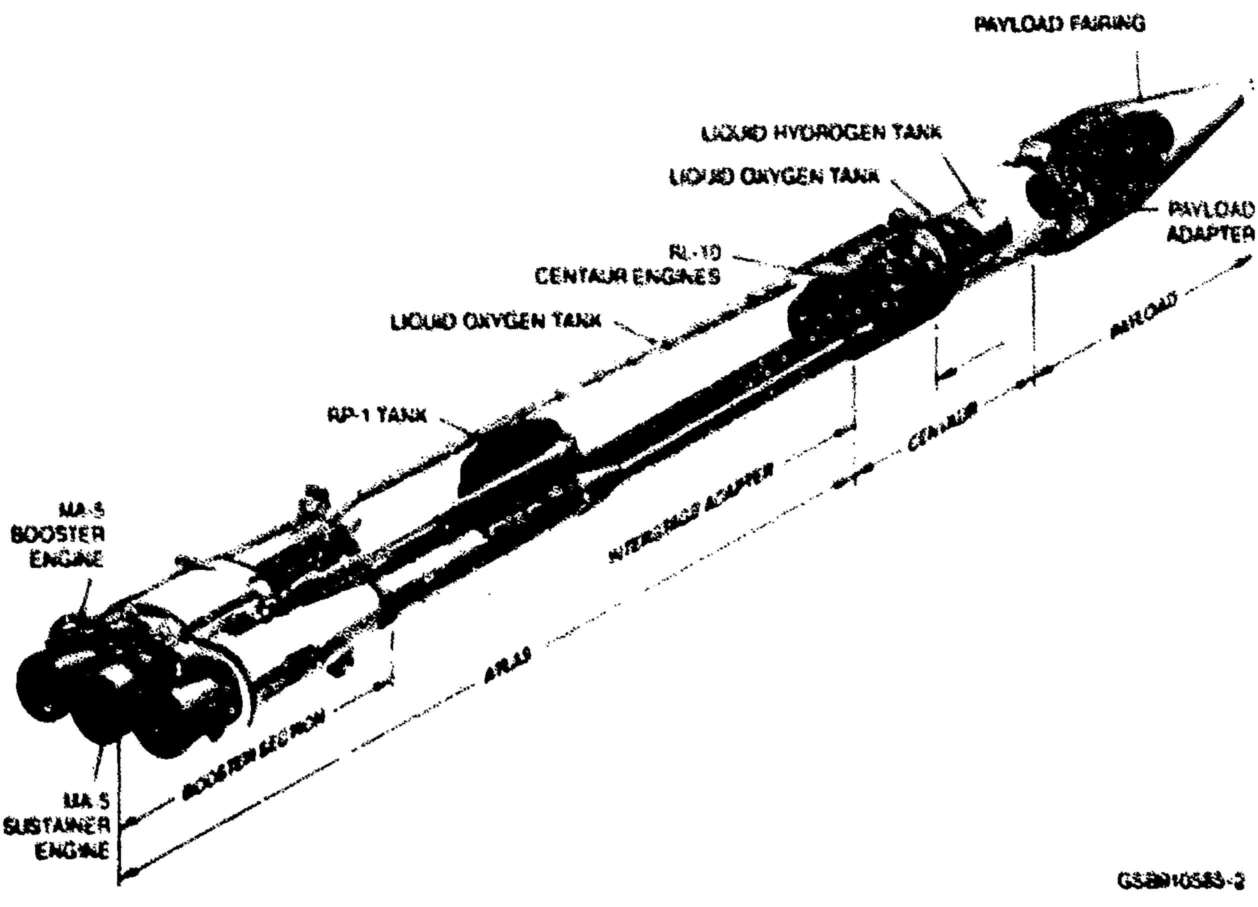


Figure 2. Atlas launch vehicle.

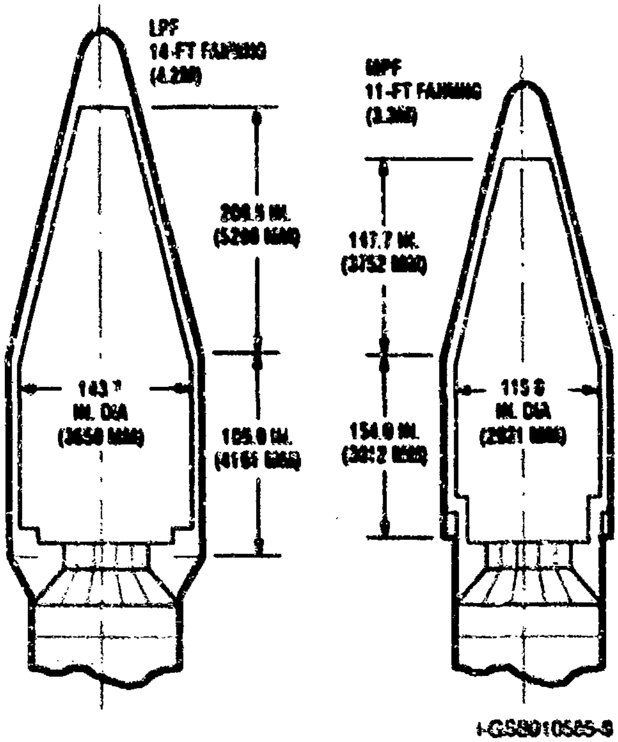


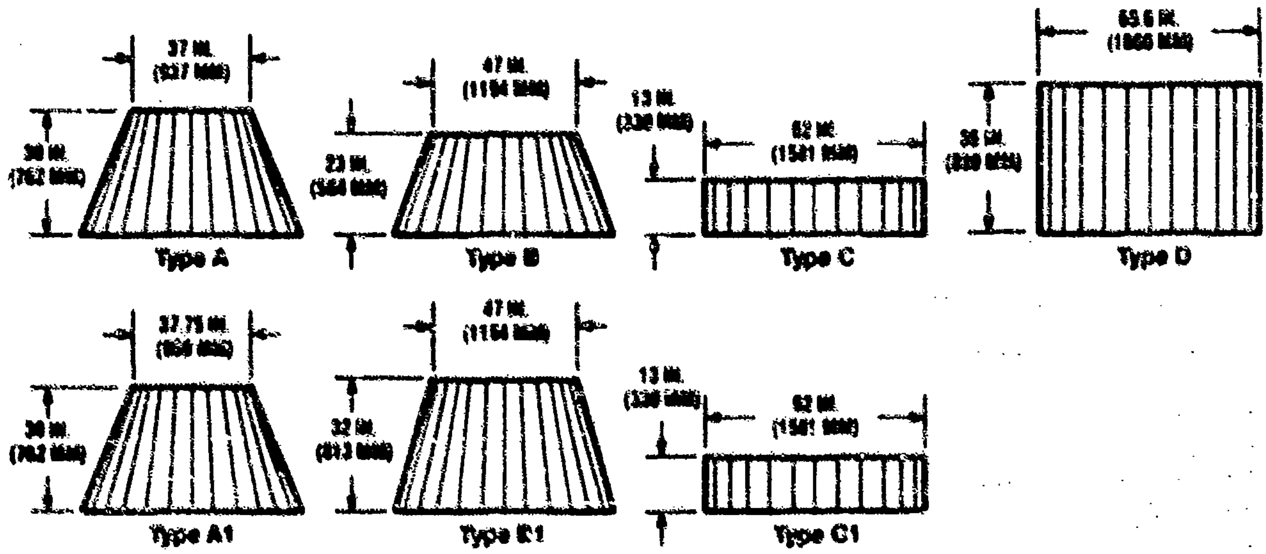
Figure 3. Payload static envelopes for the large and medium payload fairings.

and D adapters are similar. Each separation system consists of a clamp band set and separation springs to give the necessary separation energy after the clamp band is released. The Type C and C1 adapters provide bolted interfaces for spacecraft-provided adapters or mission-peculiar requirements. Section 4.1 defines the specifics for each interface.

ATLAS AND CENTAUR HERITAGE

Atlas and Centaur have played a major role in the U.S. space program since the launch of the world's first communications satellite (SCORE) on Atlas 10B in December 1958. Some of those historic events include:

- First American in orbit (Mercury) (Atlas 109D)
- First launch of a liquid hydrogen stage (Centaur)
- First lunar mission (Surveyor)



ADAPTER TYPE	APPROXIMATE[1] FORWARD INTERFACE	COMPATIBILITY
TYPE A TYPE A1	37 IN. (937 MM)	[illegible]
TYPE B TYPE B1	47 IN. (1194 MM)	1194A
TYPE C[2] TYPE C1	62 IN. (1581 MM)	61.25 IN. DIA BOLT CIRCLE 62.010 IN. DIA BOLT CIRCLE
TYPE D	65 IN. (1666 MM)	1666A

[1] SEE SECTION 4 FOR SPECIFIC FORWARD INTERFACE DIMENSIONS

[2] TYPE C, C1 ADAPTER INTERFACE ONLY TO MISSION-PECULIAR OR SPACECRAFT-PROVIDED ADAPTERS

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Figure 4. Industry standard spacecraft adapters.

- All United States planetary missions launched on ELVs (Atlas and/or Centaur).
- The 500th Atlas launch, in 1991 (Atlas 53E)

As launch vehicle needs have changed, Atlas vehicles have evolved to meet the new mission requirements. Figure 5 illustrates some of the specific vehicle configurations flown over the years.

RELIABILITY

Atlas space launch vehicles flown primarily with Centaur and Agena upper stages have a demonstrated reliability of 96% (123 successes out of 128 launch attempts).

Centaur D-1, flown since 1973, has an outstanding 96% success record (43 successes out of 45 flight trials).

The current family of Atlas/Centaur launch vehicles has a demonstrated reliability of 94% using the widely accredited Duane methodology.

COMMERCIAL LAUNCH SERVICES

General Dynamics Commercial Launch Services offers a full launch service, from spacecraft integration, processing and encapsulation, through launch operations and verification of the orbit. Our launch service includes:

a. Launch vehicle
b. Launch operations services
c. Mission-peculiar equipment design, test, and production
d. Technical integration and interface design between the launch vehicle and spacecraft
e. Program management
f. Launch facilities and support provisions
g. PPF and HPF facilities
h. Spacecraft support at CCAFS
i. Mission program management
j. Validation of spacecraft separation sequence and orbit
k. Range safety interface.

We provide administrative guidance and assistance, when necessary, for import/export licenses, permits, and clearances from government and political entities.

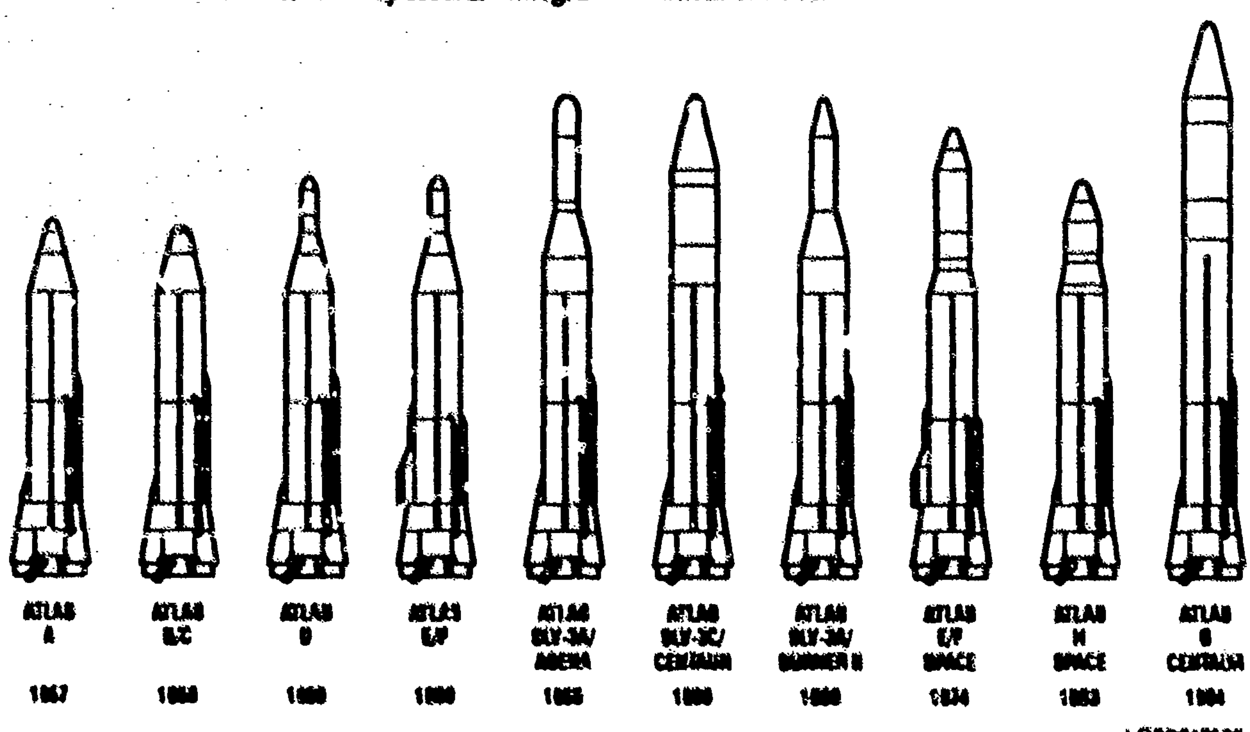

Figure 5. Atlas heritage.

The General Dynamics Commercial Launch Services organization provides a single point of contact, the mission manager, to the customer. The mission manager is responsible for program development, integration through launch, and contract completion. The mission manager chairs key meetings and reviews, and participates in milestone events at the launch site. He works closely with his Atlas vehicle manager counterparts. Figure 6 shows the interface management concept.

Launch vehicle and mission-peculiar design and development, as well as technical integration and management, are accomplished in San Diego, California.

Launch operations are performed at Cape Canaveral Air Force Station (CCAFS), Florida (Figure 7). Payload processing is normally performed at the Astrotech facilities in Titusville, Florida (Figure 8). NASA and USAF facilities at Kennedy Space Center and CCAFS are available, if required. The launch operations manager directs the General Dynamics team during spacecraft processing and launch. Customer/spacecraft on-site personnel work directly with the mission manager and launch operations team.

All government agreements required for commercial launches have been approved. The Kennedy Space Center and Cape Canaveral Air Force Station agreements covering payload and Atlas launch vehicle processing facilities, services, and range support are complete.

ADVANTAGES OF SELECTING ATLAS

Our Atlas vehicles and services provide the following key advantages:

1. Dedicated launch pad to ensure commercial launch schedules and maintain commitments
2. Single-payload manifesting to ensure launch service dedication and responsiveness
3. A mature launch service, in both launch operations and vehicle design
4. An experienced team that has launched over 40 communications satellites
5. Moderate payload launch environments (shock, vibration, acoustic, thermal, etc.) that are generally lower than those of other launch vehicles
6. Mission design flexibility demonstrated in a diverse array of mission types, including most U.S. planetary missions and numerous geostationary transfer orbit missions
7. Flexible mission design capability provides maximum spacecraft on-orbit lifetime through optimized use of spacecraft and Centaur propulsion systems

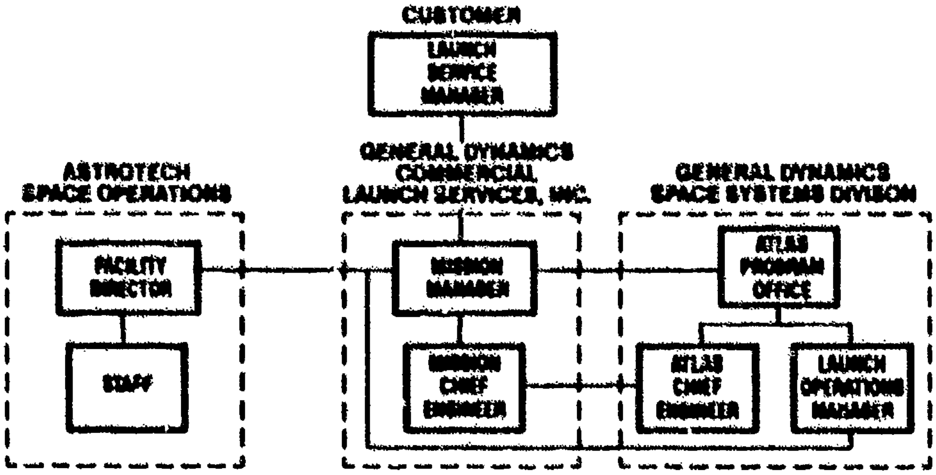

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Figure 6. Communications interface.

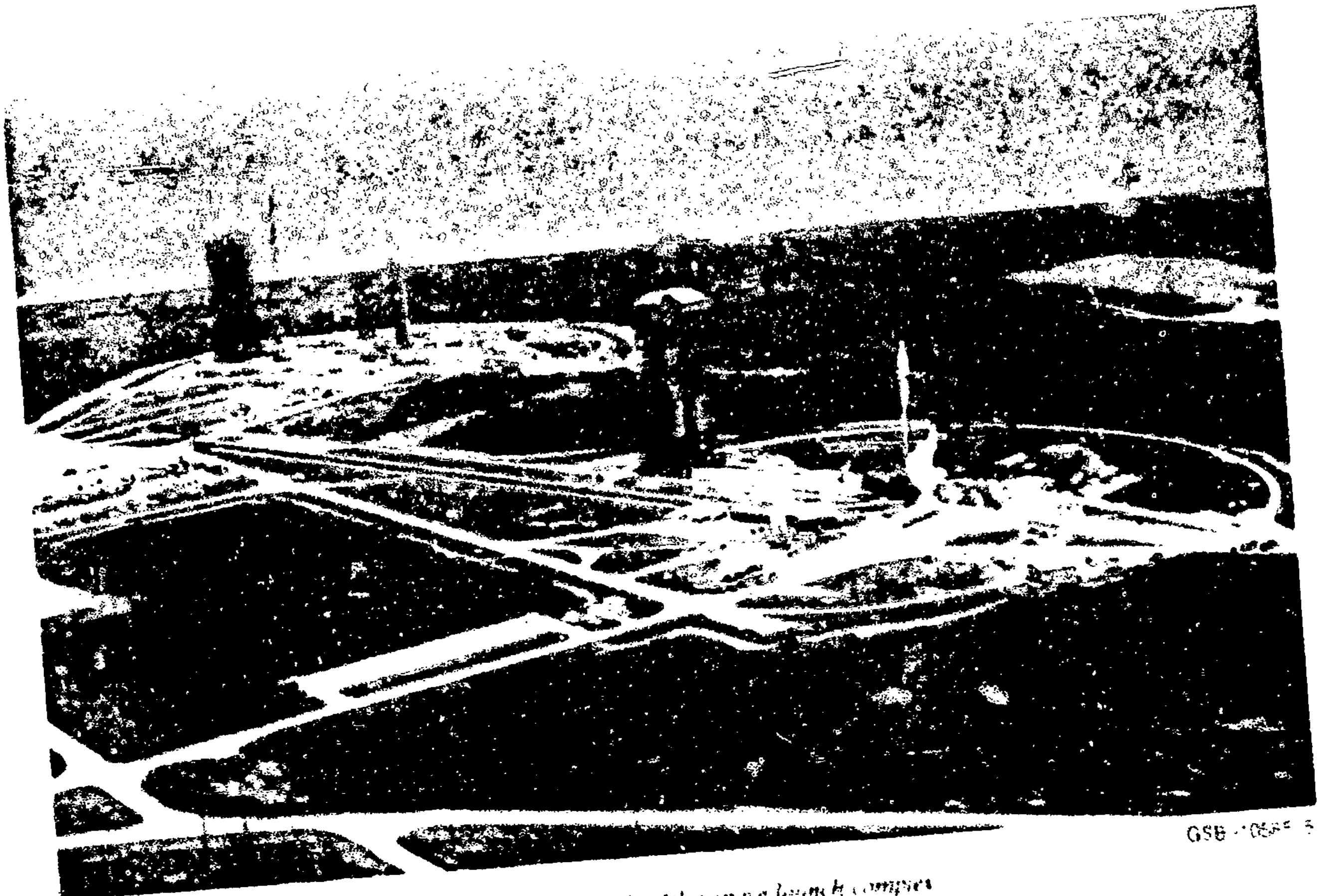


Figure 7. SLC-36 Atlas space launch complex

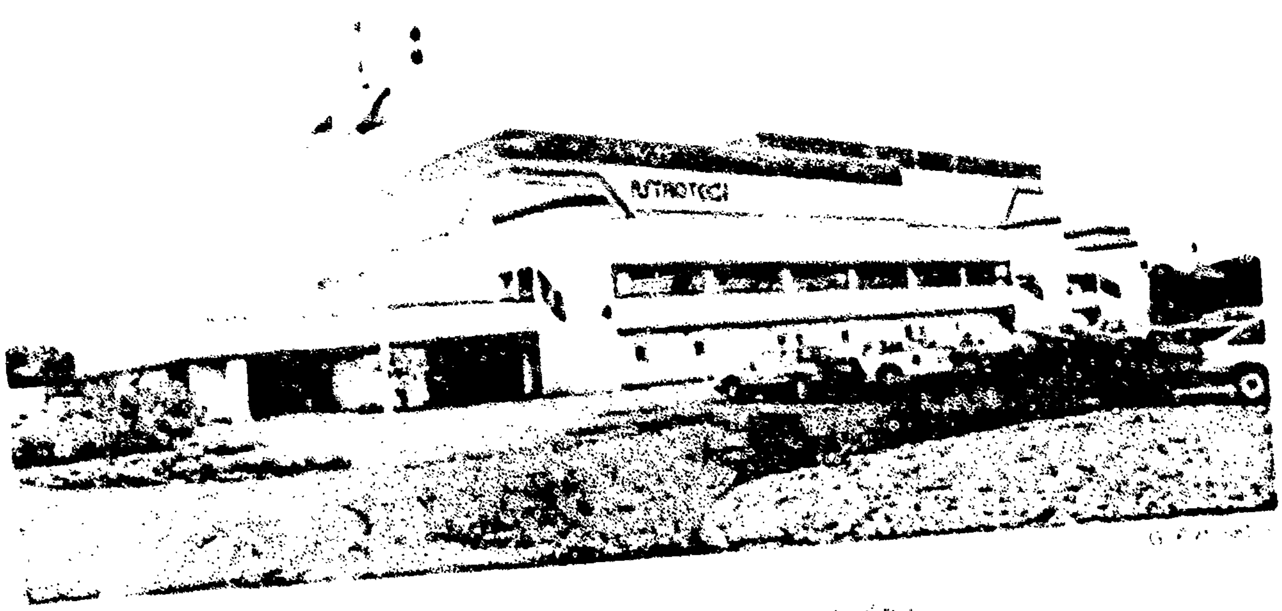
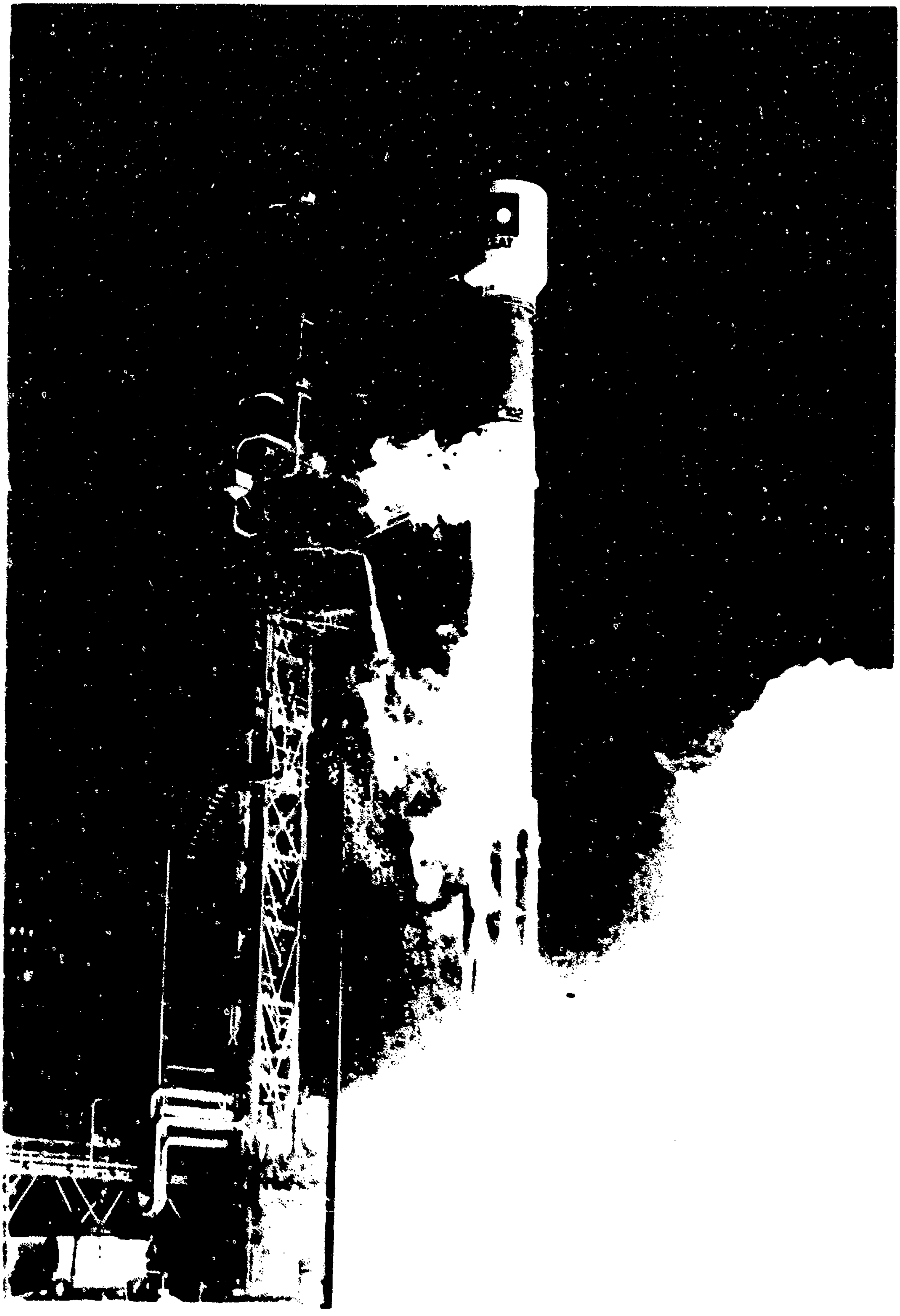


Figure 8. Astrotech spacecraft processing facilities



1 • THE ATLAS LAUNCH VEHICLE

1.1 DESCRIPTION

The Atlas launch vehicle system consists of the Atlas booster, the Centaur upper stage, and the payload fairing. Figure 1-1 gives a summary description of Atlas subsystems and characteristics.

ATLAS — Atlas was originally developed by the U.S. Air Force as an intercontinental ballistic missile. Early in development, Atlas made the transition to become a versatile and highly reliable space booster. John Glenn made his historic flight on this vehicle. It has since undergone a series of improvements, including tank lengthening, engine performance increases, and system modernization.

CENTAUR — Centaur was developed as the world's first high-energy, liquid oxygen-liquid hydrogen propellant stage. Since its first launch, it has gone through several performance and reliability upgrades, particularly in the areas of electronics and software.

1.2 ATLAS MAJOR CHARACTERISTICS

The Atlas booster is of thin-wall, fully monocoque, corrosion-resistant stainless steel construction. The fuel tank, which contains RP-1, and the oxidizer tank, which contains liquid oxygen, are separated by an ellipsoidal intermediate bulkhead. Structural integrity of the tanks is maintained in flight by the pressurization system and on the ground by either internal tank pressure or by application of mechanical stretch.

The Atlas booster is controlled by the Centaur avionics system, which provides guidance, flight control, and vehicle sequencing functions. An external equipment pod houses Atlas systems including Atlas flight termination, propellant utilization, pneumatics, and instrumentation.

Atlas booster propulsion is provided by either the Rocketdyne MA-5 (Atlas I) or MA-5A (Atlas II, IIA, IIAS) engine system, which includes the sustainer, two vernier (Atlas I), and one booster engine (two thrust chambers). All engines are ignited prior to liftoff and monitored until 70% thrust is achieved, after which a controlled release occurs. The booster engine package is jettisoned during ascent and sustainer-powered flight continues ("sustainer phase") until propellant depletion.

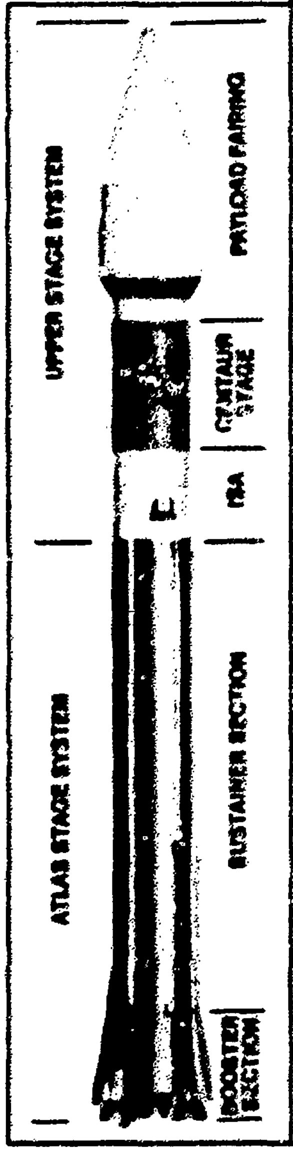
Solid Rocket Booster (SRB) — The Atlas IIAS uses four Thiokol Castor IVA SRBs. This motor was selected based on its performance and the excellent reliability record (99.94% success in over 1,790 flights) of the Castor solid motor family. Each Castor IVA SRB is 37 feet long and 40 inches in diameter. Two are ignited at liftoff and the remaining two are air-lit.

Structural modifications required to attach the SRBs include a redesign of the Atlas thrust section, an increase in Atlas skin gauges, and the addition of an attachment ring in the fuel tank (see Figure 1-2).

The Atlas is integrated with the Centaur vehicle by the interstage adapter. This aluminum skin/stringer frame structure provides the structural link between the Atlas and Centaur vehicles. The Atlas vehicle is separated from the Centaur vehicle by a pyrotechnic flexible linear shaped charge system attached to the forward ring of the interstage adapter.

1.3 CENTAUR MAJOR CHARACTERISTICS

The Centaur propellant tanks, like those of the Atlas lower stage, are constructed of thin-wall fully monocoque, corrosion-resistant stainless steel. Centaur employs high-energy liquid hydrogen and liquid oxygen propellants separated by a double-wall, vacuum-insulated intermediate bulkhead. Tank stabilization is maintained at all times by either internal pressurization or application of mechanical stretch.



	ATLAS I	ATLAS II	ATLAS IIA	ATLAS IIAS
LARGE PAYLOAD FAIRING SIZE:	13.75 FT DIA x 40.1 FT LONG (4.19 M x 12.22 M)	COMMON →	COMMON →	COMMON →
MEDIUM PAYLOAD FAIRING SIZE: STRUCTURE:	10.5 FT DIA x 34.1 FT LONG (3.2 M x 10.4 M) ALUMINUM SKIN & STRINGER CONSTRUCTION			
CENTAUR UPPER STAGE SIZE:	10.0 FT DIA x 30.0 FT LONG (3.05 M x 9.14 M)	10.0 FT DIA x 33.0 FT LONG (3.05 M x 10.05 M)	COMMON →	COMMON →
PROPELLANT:	30.4 KLB LH_2 & LO_2 (13.8 METRIC TONS)	37.0 KLB LH_2 & LO_2 (16.8 METRIC TONS)		
GUIDANCE:	INERTIAL PLATFORM	STRAPDOWN RING LASER GYRO[4]	COMMON →	COMMON →
PROPULSION:	PRATT & WHITNEY RL10A-3-3A (TWO) THRUST VACUUM 33KLB (146.8 KN) (TOTAL) Isp VACUUM 444.4 SEC	COMMON →	PRATT & WHITNEY RL10A-4 (TWO)* THRUST VACUUM 41.6 KLB (185.85KN) (TOTAL) I_{sp} VACUUM 449.5 SEC	COMMON →
AVIONICS:	• MECHANICAL SWITCHING	COMMON →	SOLID STATE SWITCHING	COMMON →
	• TELEMETRY/RANGE SAFETY • ELECTRICAL POWER	COMMON →	COMMON →	COMMON →
INTERSTAGE ADAPTER SIZE: STRUCTURE:	10 FT DIA x 13.0 FT LONG (3.05 M x 3.96 M) ALUMINUM SKIN & STRINGER CONSTRUCTION	COMMON →	COMMON →	COMMON →
ROLL CONTROL:		ATLAS ROLL CONTROL MODULE	COMMON →	COMMON →
ATLAS BOOSTER SIZE:	10.0 FT DIA x 72.7 FT LONG (3.05 M x 22.16 M)	10.0 FT DIA x 81.7 FT LONG (3.05 M x 24.90 M)	COMMON →	COMMON →
PROPELLANT:	303.0 KLB LO_2 & RP-1 (137.4 METRIC TONS)	344.77 KLB LO_2 & RP-1 (156.38 METRIC TONS)		
GUIDANCE:	FROM UPPER STAGE	COMMON →	COMMON →	COMMON →
PROPULSION:	ROCKETDYNE MA-5 BOOSTER ENGINE PACKAGE – THRUST 377.5 KLB (1679 KN) – I_{sp} 259.1 SEC	ROCKETDYNE MA-5A BOOSTER ENGINE PACKAGE – THRUST 414 KLB (1841 KN) – I_{sp} 263.8 SEC	COMMON →	COMMON →
	SUSTAINER ENGINE – THRUST 60.5 KLB (269 KN) – I_{sp} 220.4 SEC	COMMON →	COMMON →	THIOKOL CASTOR 4A (4) SRMs – WEIGHT 25.5 KLB EACH (11.56 METRIC TONS) – THRUST 97.5 KLB (433.8 KN) – BURN DURATION: 53 SEC
	VERNIER ENGINES (2) – THRUST 1,230 LB (5500N) – I_{sp} 196.7 SEC	DELETED →	DELETED →	
AVIONICS:	• FLIGHT CONTROL • TELEMETRY/RANGE SAFETY • ELECTRICAL POWER	COMMON →	COMMON →	COMMON →

* OPTION RL10A-4-1 (TWO) THRUST VACUUM 44.6 KLB (198.4 KN) (TOTAL) I_{sp} VACUUM 451.5 SEC

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Figure 1-1. Summary of Atlas subsystems and their characteristics.

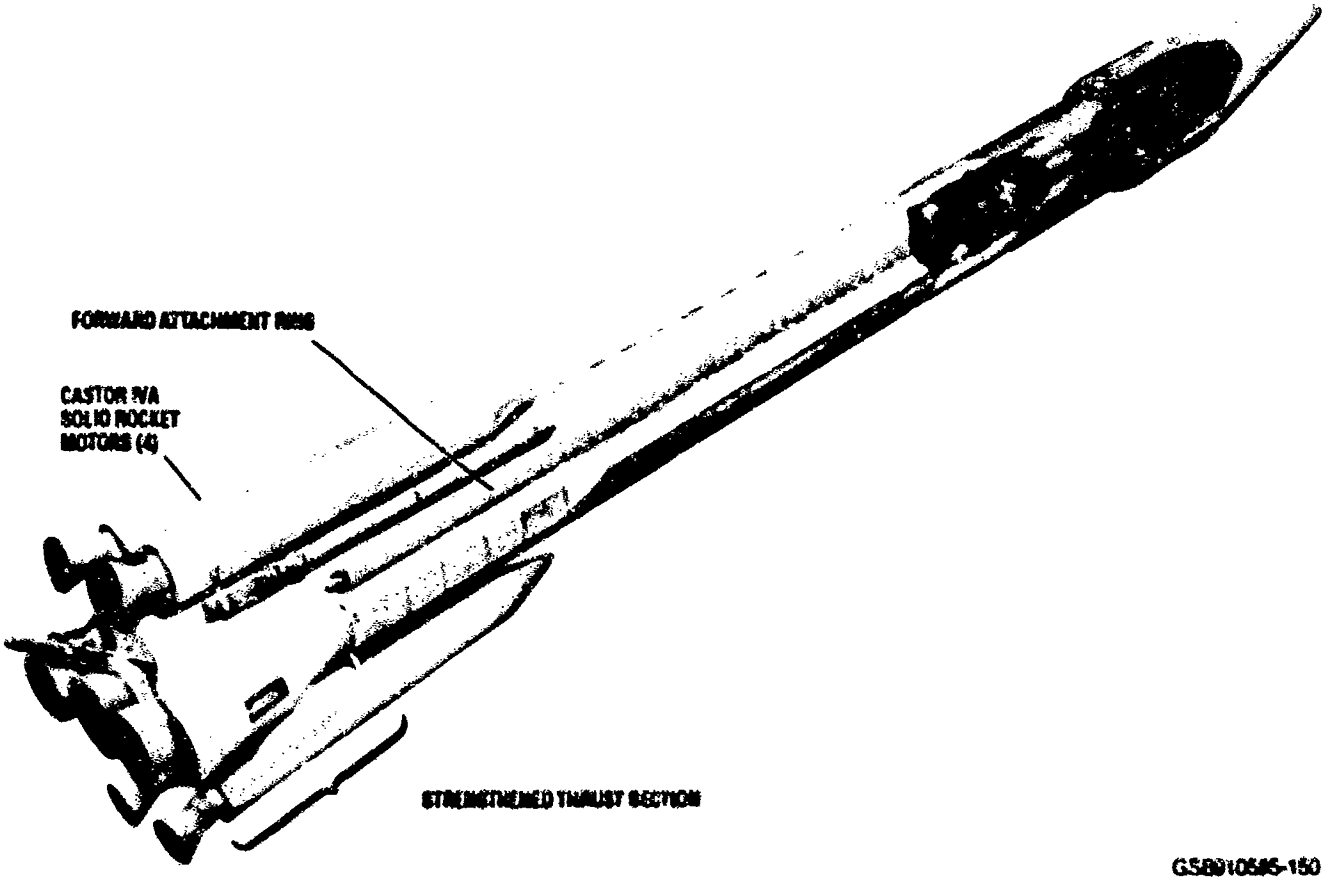

Figure 1-2. Atlas IIAS vehicle.

The stub adapter and equipment module are attached to the forward end of the Centaur. The stub adapter is bolted to the forward ring of the Centaur tank and supports the equipment module, payload fairing, and spacecraft adapter. The equipment module attaches to the forward ring of the stub adapter and provides mounting arrangements for the Centaur avionics packages and the spacecraft adapter (Figure 1-3).

Centaur avionics packages mounted on the equipment module provide control and monitoring of all vehicle functions. The inertial navigation unit (INU) performs the inertial guidance and attitude control computations for both Atlas and Centaur phases of flight and also provides control for Atlas and Centaur tank pressures and propellant use.

Centaur RL10 Engine — The Centaur upper stage utilizes two Pratt & Whitney RL10 engines. The RL10A-3A on an Atlas I and Atlas II is rated at 16,500 lb thrust. The RL10A-4 on the Atlas IIA and Atlas IIAS is rated at 20,500 lb (91.1 kN) thrust without an extendible nozzle, and at 20,800 lbf (92.5 kN) with an extendible nozzle. The engine is a gimballed, turbopump-fed, regeneratively cooled, single-chamber rocket engine consisting of a fixed primary nozzle and an optional secondary extendible nozzle (Figure 1-4). The extendible nozzle provides enhanced engine performance through an increase in expansion ratio.

1.4 PAYLOAD VOLUME

The usable payload volume is dependent upon the spacecraft adapter and payload fairing employed.

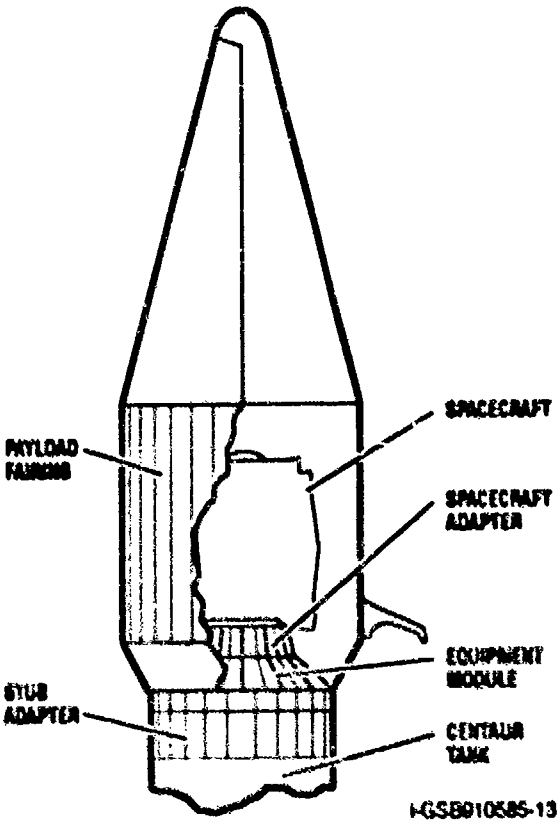

Figure 1-3. The spacecraft is mated to the Centaur upper stage via an adapter attached to the equipment module.

Numerous payload compartment arrangements are available with the Type A through D spacecraft adapters and medium and large payload fairings.

1.4.1 PAYLOAD FAIRINGS — The payload fairing protects the spacecraft from time of encapsulation through atmospheric ascent. The Atlas user has a choice between the large or medium payload fairing configuration.

For either fairing, a thermal shield or acoustic blanket can be added should the spacecraft require more benign environments. If additional payload fairing volume is required, General Dynamics has investigated lengthening the large payload fairing (reference Section 7.4).

Both the large and medium fairings are discussed in greater detail in Section 4.

1.4.2 SPACECRAFT ADAPTERS/SEPARATION SYSTEM — The spacecraft is mounted to the launch vehicle using a spacecraft adapter. The Type A, A1, B, B1, and D adapters employ a pyrotechnic V-band clamp system. Adapters and separation systems are discussed in detail in Section 4.

Each Atlas vehicle configuration is compatible with each of the payload fairings, adapters, and separation systems. Because of structural and propulsion system differences between the vehicle configurations, there are minor differences in spacecraft environments. A summary of vehicle structural compatibility and the differences in environments is provided in Table 1-1.

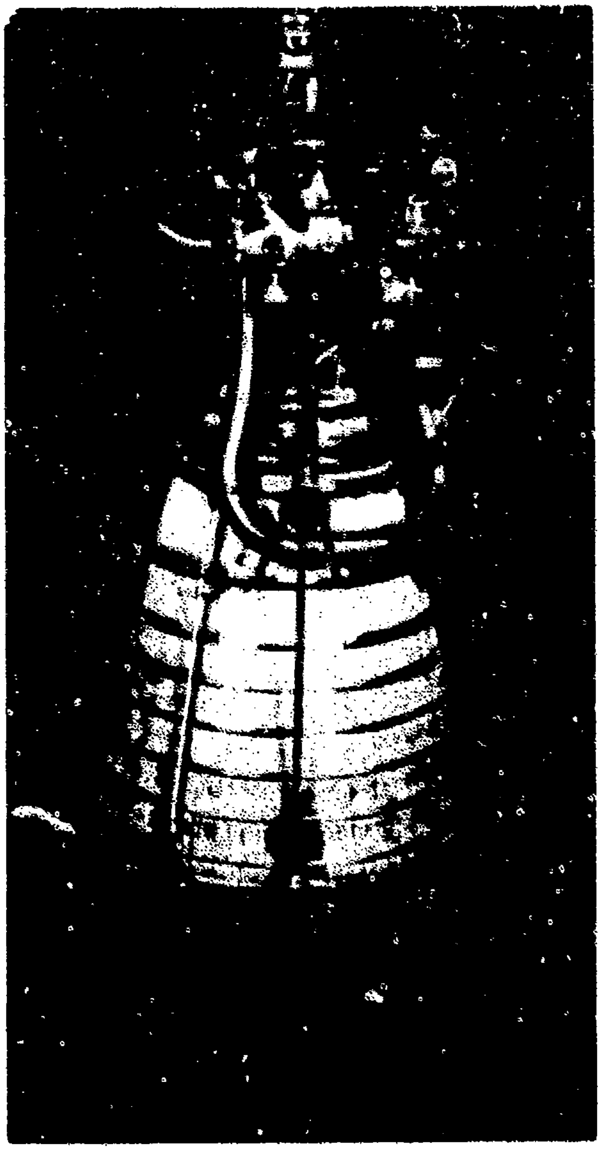

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Figure 1-4. Extendible nozzles and higher thrust provide additional performance.

Table 1-1. Spacecraft common accommodations for the Atlas family.

<table>
<tr><td rowspan="2"></td><td colspan="4">ATLAS VEHICLE</td></tr>
<tr><td>I</td><td>II</td><td>IIA</td><td>IIAS</td></tr>
<tr><td>PAYLOAD
• FAIRINGS
11-, 14-FT DIA
• ADAPTERS
A, A1, B, B1, C, C1, D</td><td colspan="4">COMMON AND INTERCHANGEABLE USEAGE
(REFERENCE SECTION 4)</td></tr>
<tr><td rowspan="2">INTERFACES
• MECHANICAL
• ELECTRICAL</td><td colspan="4">COMMON FOR ANY STANDARD ADAPTER UTILIZED</td></tr>
<tr><td colspan="4">COMMON
(REFERENCE SECTION 4.1.3)</td></tr>
<tr><td>ENVIRONMENTS
• LIMIT DESIGN LOADS</td><td colspan="3">COMMON
(TABLE 3-2)</td><td>TABLE 3-3</td></tr>
<tr><td>• ACOUSTICS</td><td colspan="3">COMMON
(FIGURE 3-8A, B, D)</td><td>FIGURE 3-8C & D</td></tr>
<tr><td>• ALL OTHER ENVIRONMENTS</td><td colspan="4">COMMON
(REFERENCE SECTION 5)</td></tr>
</table>

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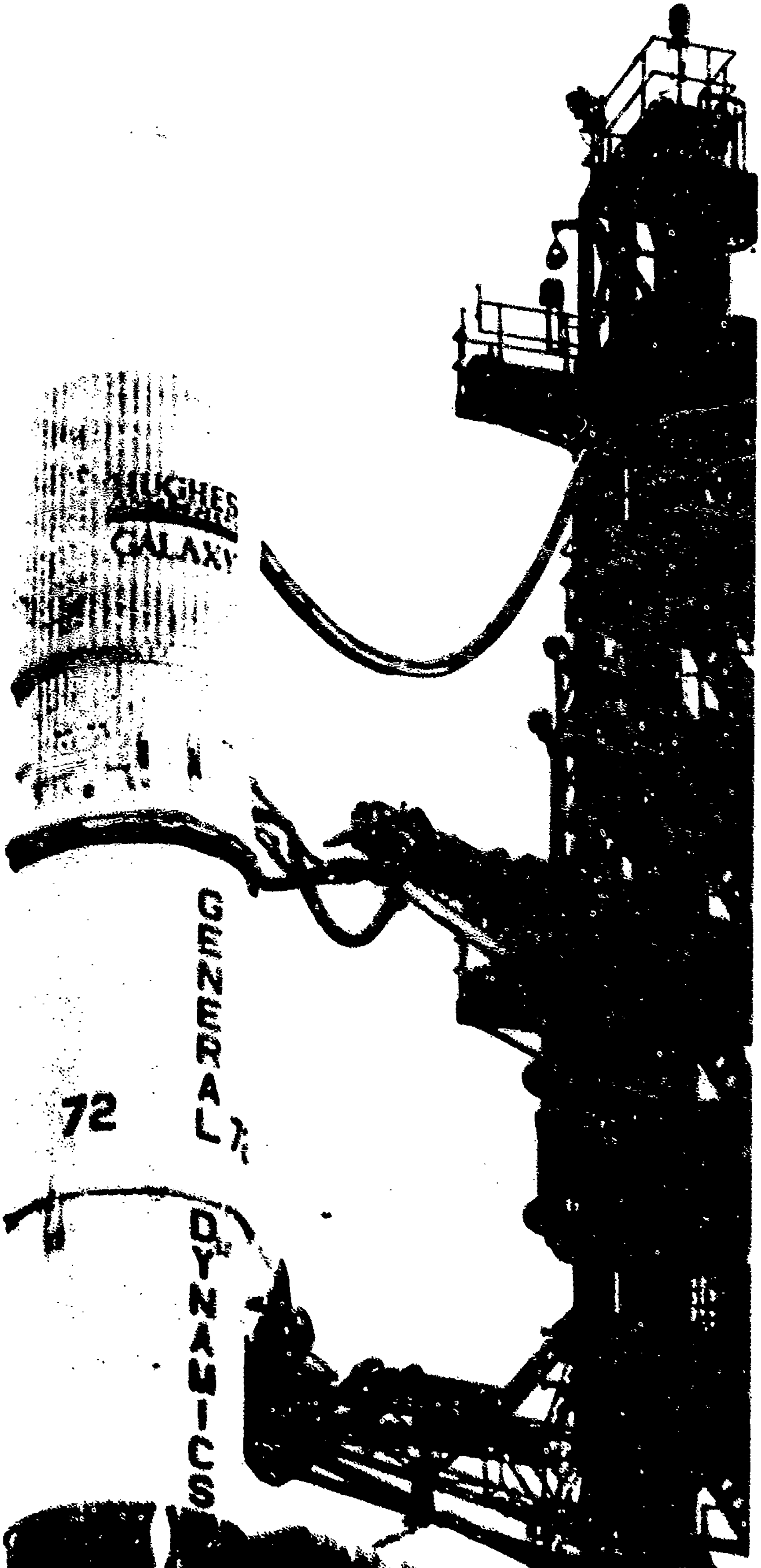
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GENERAL
DYNAMICS
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2 • ATLAS MISSION DESIGN AND PERFORMANCE

Over the last three decades, the Atlas and Centaur stages have flown together as the Atlas/Centaur and with other stages (e.g., Atlas/Agena and Titan/Centaur) to deliver a variety of commercial, military, and scientific payloads to their target orbits. Based on our experiences with over 500 Atlas launches, performance for each launch vehicle is determined by engineering analysis of developed and new hardware with emphasis placed on conservative performance prediction to ensure each vehicle meets design expectations. With Atlas I and Atlas II operational, and Atlas IIA nearing its initial launch capability, engineering estimates of Atlas family performance capabilities have been revised to reflect flight-qualified hardware performance characteristics and our improved knowledge of developed hardware. In addition, several vehicle enhancement options are now being developed under contract. With this information, Atlas family performance offerings have been improved and expanded. Table 2-1 illustrates the new performance capabilities of the Atlas family.

The standard GTO performance option provides a reduced performance level by constraining the mission design to a standard ascent profile.

The custom performance option provides a performance level that is slightly higher than in the previous Mission Planner's Guide.

The Block 1 performance option provides a significant increase in performance through an uprate of the Centaur propulsion system and a series of vehicle structural and software enhancements.

As suggested by the table, Atlas is capable of being launched from Cape Canaveral Air Force Station (CCAFS) in Florida and is now planned to be launched from Vandenberg Air Force Base (VAFB) in California. This section further describes Atlas I, II, IIA, and IIAS mission and performance options available with a Florida launch. Additional mission and performance data with launch from VAFB is provided in Section 8 of this document.

2.1 MISSION DESCRIPTIONS

Atlas is a reliable and versatile launch system, capable of delivering payloads to a wide range of low and high circular orbits, elliptical transfer orbits, and Earth-escape trajectories. Each Atlas launch vehicle, available with either a medium (MPF) or large payload fairing (LPF), is dedicated to a single payload. The trajectory design for each mission is

Table 2-1. Summary of Atlas performance capabilities

	Payload Systems Weight Capabilities, lb (kg)				
	Atlas I	Atlas II	Atlas IIA	Atlas IIAS	Fairing
Geosynchronous Transfer					
[90 x 19,324 nmi (167 x 35788 km), i = 27.0°, ω_p = 180°, 99% confidence level]					
Block 1 Enhanced	--	--	6,970 (3160)	8,450 (3830)	MPF
	--	--	6,710 (3045)	8,150 (3700)	LPF
Custom	5,240 (2375)	6,500 (2950)	6,700 (3040)	8,150 (3700)	MPF
	4,970 (2255)	6,200 (2810)	6,400 (2900)	7,850 (3560)	LPF
Standard GTO	--	5,750 (2610)	6,050 (2745)	7,450 (3379)	LPF
Low Earth Orbit from CCAFS					
[100 nmi (185 km) circular orbit, i = 28.5°, 99.87% confidence level]					
	--	14,500 (6580)	16,050 (7280)	19,050 (8640)	LPF
Low Earth Orbit from VAFB					
[100 nmi (185 km) circular orbit, i = 90°, 99.87% confidence level]					
	--	12,150 (5510)	13,600 (6170)	16,100 (7300)	LPF

MPF: medium [11-ft (3.3-m)] payload fairing; LPF: large [14-ft (4.2-m)] payload fairing

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specifically tailored to optimize the mission's critical performance parameter (maximum satellite orbit lifetime, maximum weight to transfer orbit, etc.) while satisfying satellite and launch vehicle constraints.

Atlas mission ascent profiles are developed using one or more Centaur upper stage main engine burns. Each mission profile type is suited for a particular type of mission.

Direct Ascent Missions — With a one-Centaur-burn mission design the Centaur main engines are ignited just after Atlas/Centaur separation and the burn is continued until the Centaur and spacecraft are placed into the targeted orbit. Centaur/spacecraft separation occurs shortly after the burn is completed. Direct ascents are primarily used for low Earth circular orbits and elliptic orbits with orbit geometries (i.e., arguments of perigee and inclinations) easily reached from the launch site. Orbits achievable with little or no launch vehicle yaw steering and those that can be optimally reached without coast phases between burns are prime candidates for the direct ascent mission design. Atlas/Centaur has previously flown 15 missions using the direct ascent design.

Parking Orbit Ascent Missions — Parking orbit ascent, used primarily for geosynchronous transfer missions, is the most widely used Atlas trajectory design. Performance capabilities are based on two Centaur burns injecting Centaur and the satellite into a transfer orbit selected to satisfy mission requirements. The first Centaur burn commences just after Atlas/Centaur separation and is used to inject the Centaur/spacecraft into a mission performance-optimal parking orbit. After a coast to the desired location for transfer orbit injection, the second Centaur main engine burn provides the impulse to place the satellite into the transfer and/or final orbit. If targeted to an elliptic transfer orbit, the satellite then uses its own propulsion system to achieve the final mission orbit. Missions requiring circular final orbits will use the second Centaur burn to circularize the satellite at the desired altitude and orbit inclination. Fifty-eight Atlas/Centaur R&D and operational missions have flown using the parking orbit ascent mission profile.

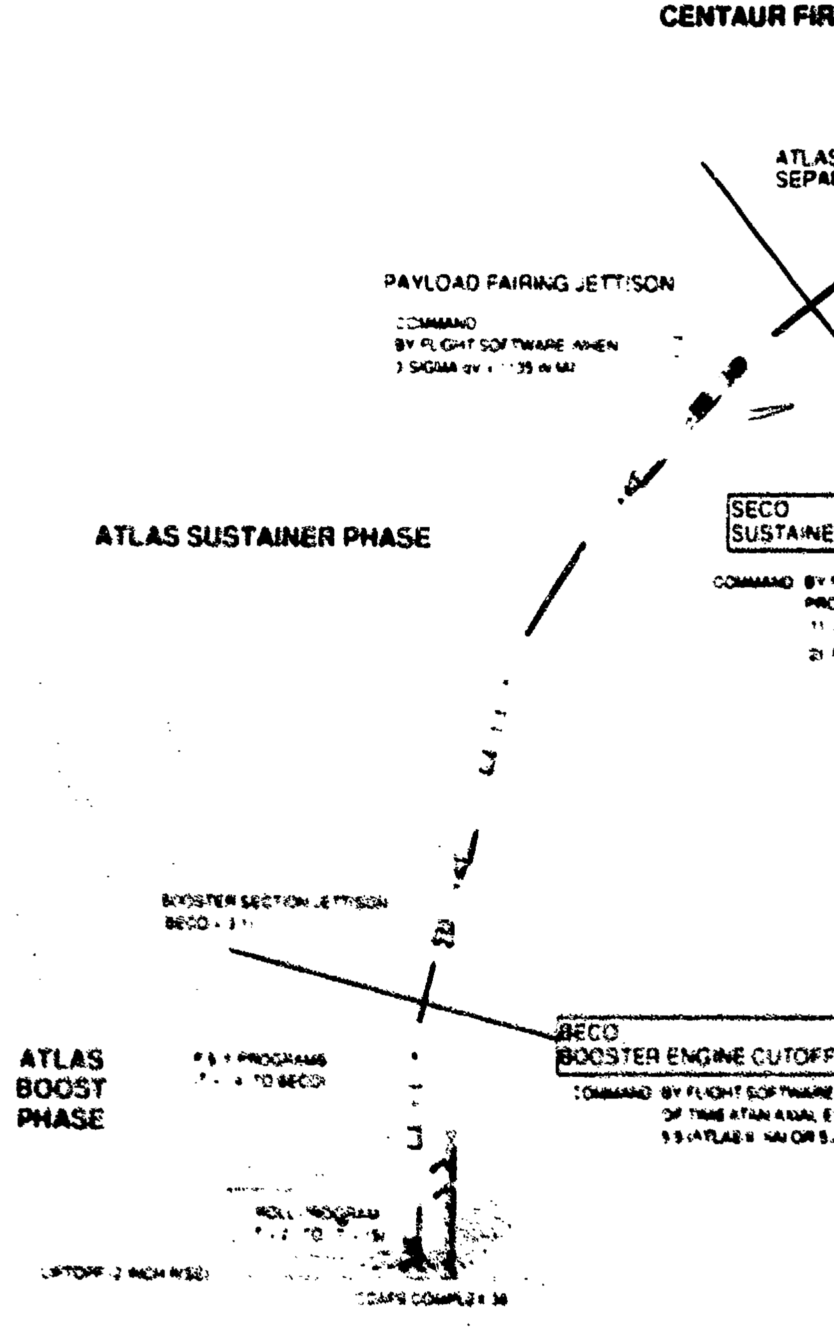
2.2 ATLAS ASCENT PROFILE

To familiarize users with Atlas and Centaur mission sequences, information is provided in the following paragraphs regarding the direct and parking orbit ascent mission designs. Figure 2-1 shows sequence of events data for a typical parking orbit ascent mission. Table 2-2 shows typical mission sequence data for each Atlas vehicle for a typical geosynchronous transfer mission. This data is representative; actual sequencing will vary to meet the requirements of each mission.

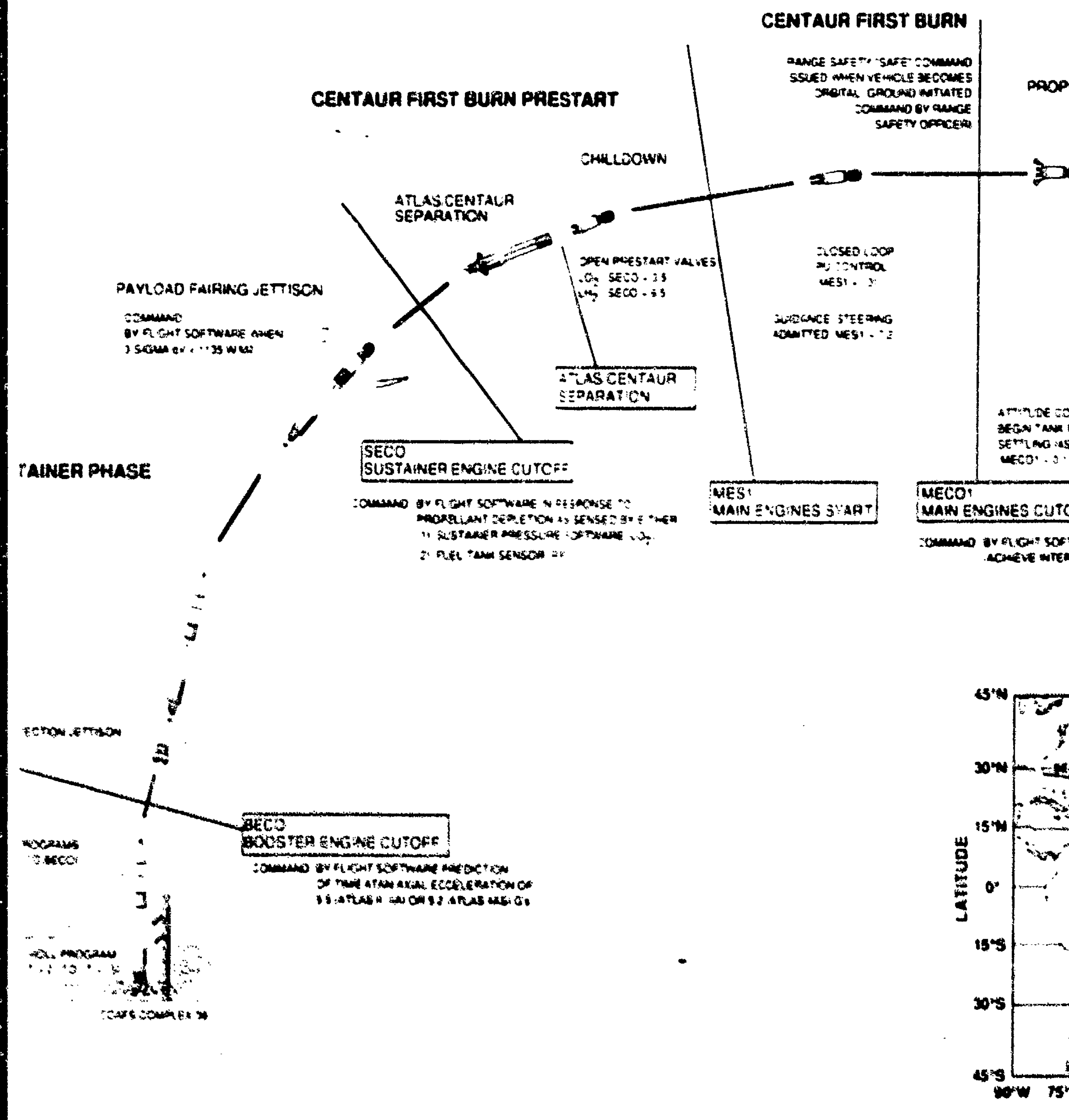
2.2.1 BOOSTER PHASE — Atlas can be launched at any time of day to meet spacecraft mission requirements. At liftoff, the booster ascent phase begins with ignition of the Rocketdyne MA-5/MA-5A engine system and, for Atlas IIAS, the first pair of Thiokol Castor IVA solid rocket boosters (SRBs).

During the short-vertical rise away from the pad, the vehicle rolls from the launch pad azimuth to the appropriate flight azimuth. At a vehicle-dependent altitude between 700 ft (215 m) and 1,000 ft (305 m), the vehicle begins pitching over into the prescribed ascent profile. At approximately 8,000 ft (2438 m), the vehicle enters a nominal zero pitch and yaw angle of attack phase to minimize aerodynamic loads.

For Atlas IIAS, the first pair of SRBs burn out at approximately 54 seconds into flight. Ignition of the second pair is governed by structural loading parameters. First pair jettison occurs when range safe-

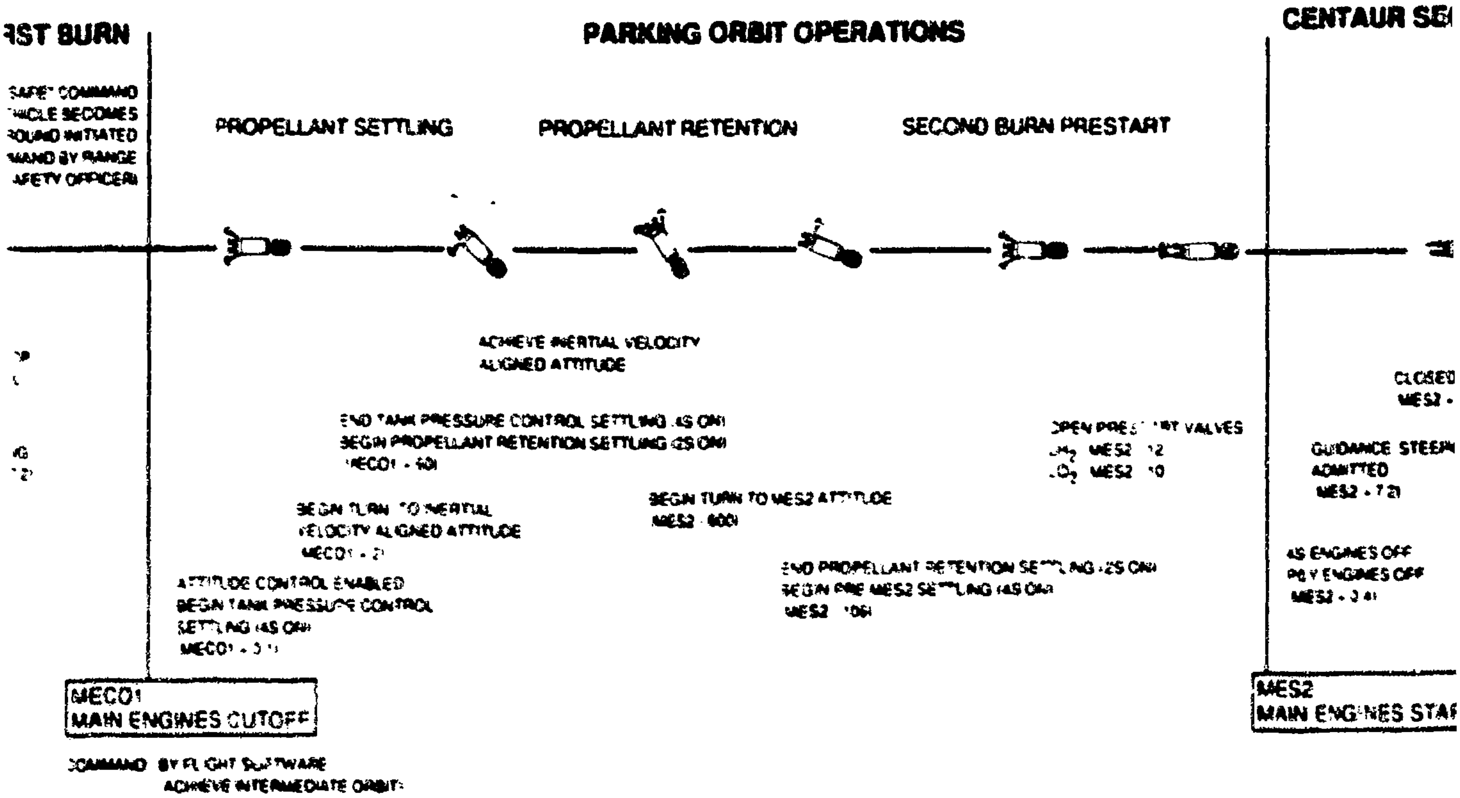

CENTAUR FIR
PAYLOAD FAIRING JETTISON
ATLAS SUSTAINER PHASE
SECO
SUSTAINE
BOOSTER SECTION JETTISON
BECO
BOOSTER ENGINE CUTOFF
ATLAS
BOOST
PHASE

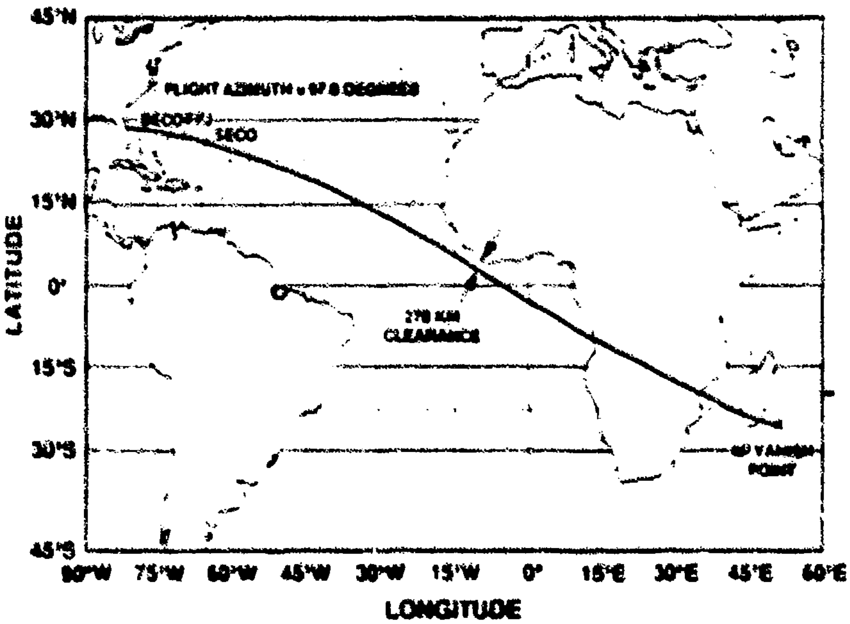
1

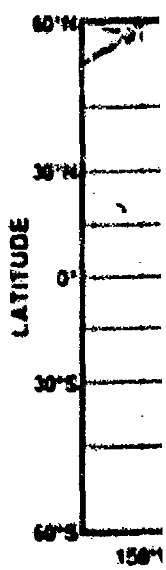


CENTAUR FIRST BURN
RANGE SAFETY 'SAFE' COMMAND ISSUED WHEN VEHICLE BECOMES ORBITAL (GROUND INITIATED COMMAND BY RANGE SAFETY OFFICER)
PROP
CENTAUR FIRST BURN PRESTART
CHILLDOWN
ATLAS/CENTAUR SEPARATION
OPEN PRESTART VALVES
CLOSED LOOP PU CONTROL
GUIDANCE STEERING ADMITTED
PAYLOAD FAIRING JETTISON
COMMAND BY FLIGHT SOFTWARE WHEN
ATLAS CENTAUR SEPARATION
SECO
SUSTAINER ENGINE CUTOFF
TAINER PHASE
MES1
MAIN ENGINES START
MECO1
MAIN ENGINES CUTO
COMMAND BY FLIGHT SOFTWARE IN RESPONSE TO PROPELLANT DEPLETION AS SENSED BY EITHER
2) FUEL TANK SENSOR
BECO
BOOSTER ENGINE CUTOFF
COMMAND BY FLIGHT SOFTWARE PREDICTION
45°N
30°N
15°N
0°
15°S
30°S
45°S
90°W
LATITUDE

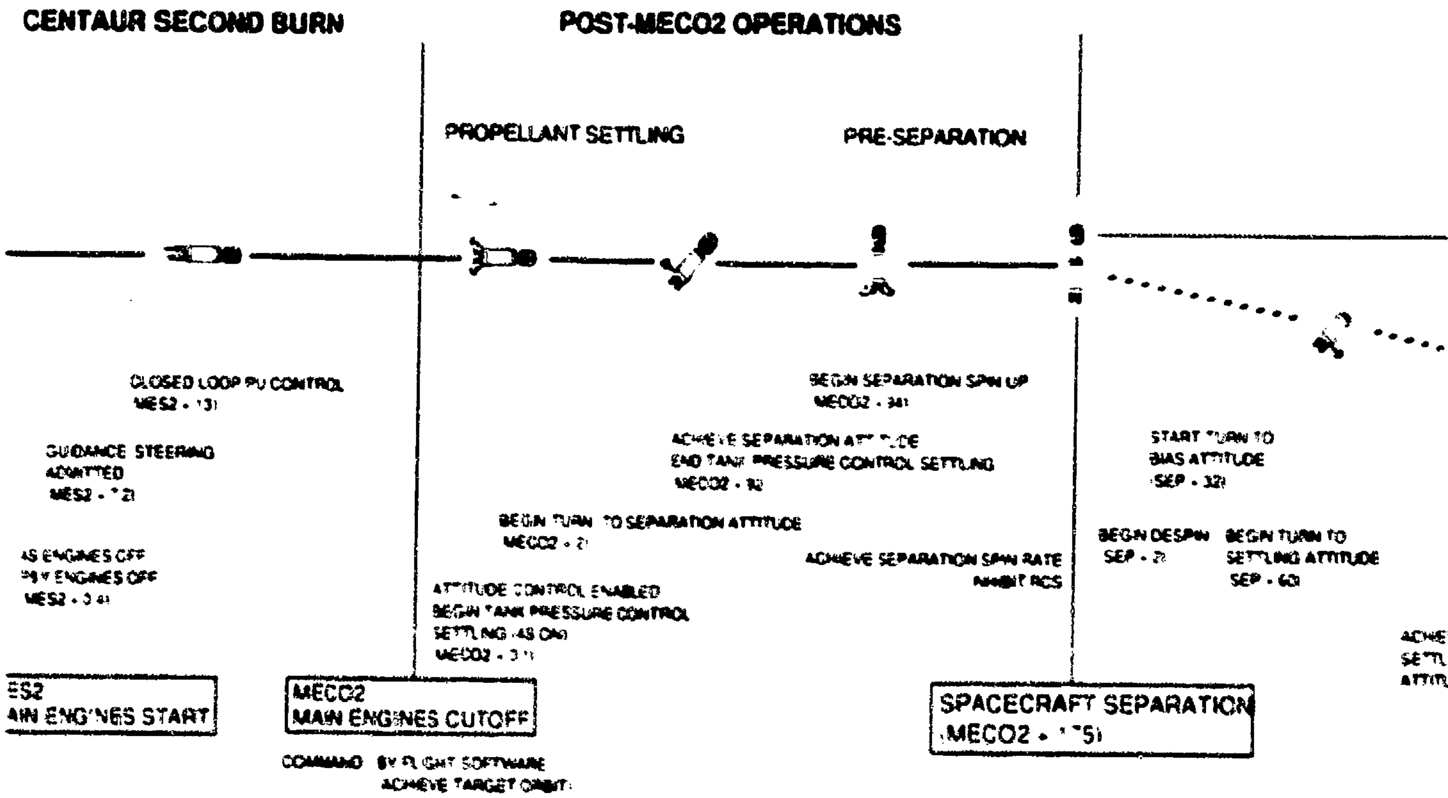
2

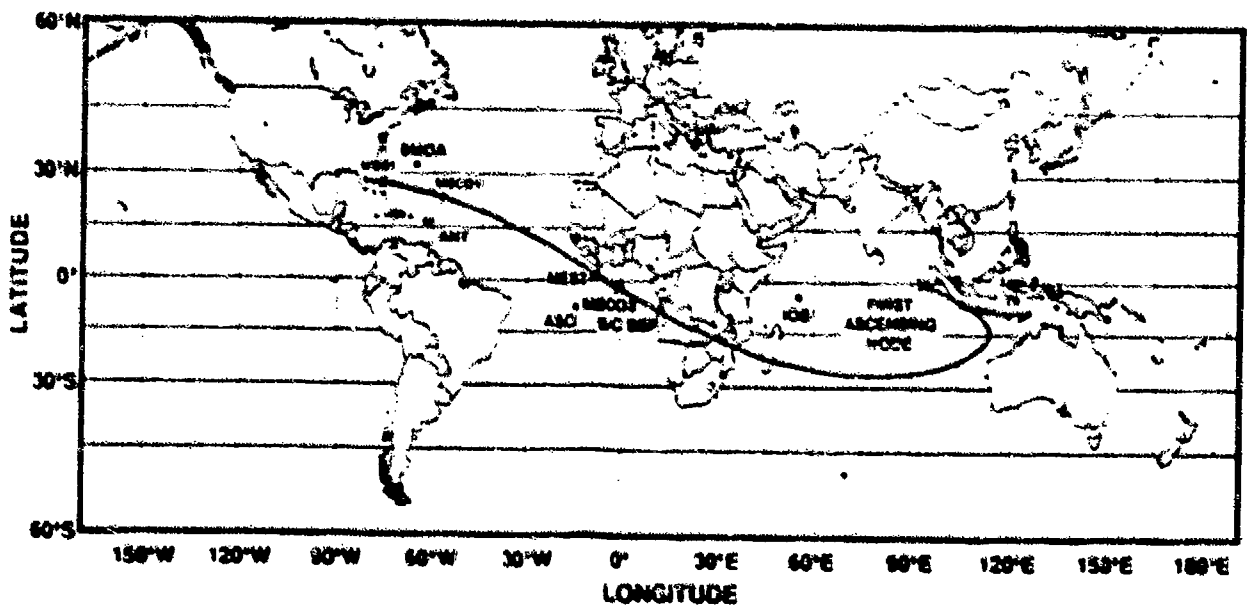






INSTANTANEOUS IMPACT POINT TRACE





SUB-VEHICLE TRACE

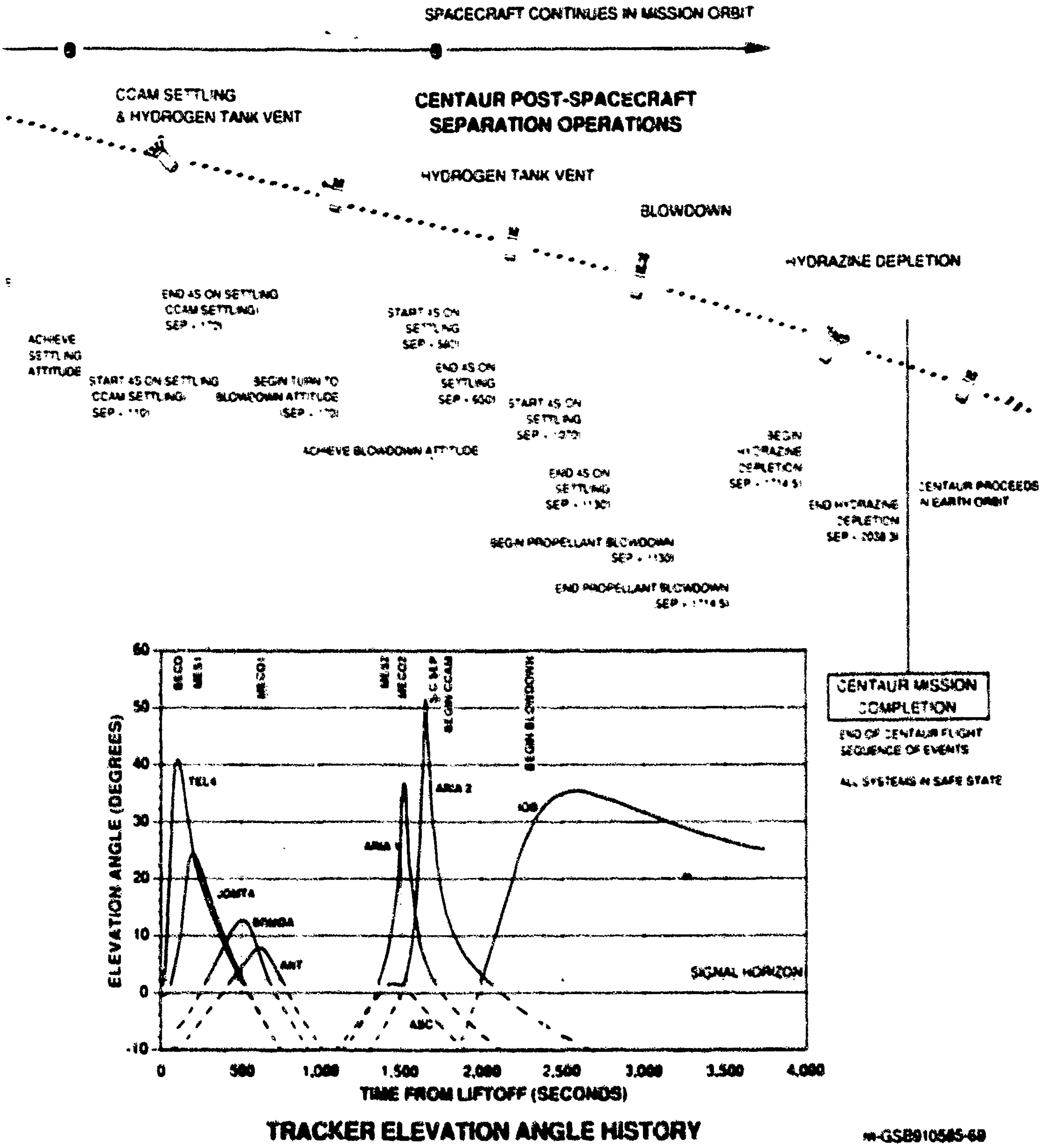


Figure 2-1. Typical Atlas ascent profile.

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Table 2-2. Typical GTO mission launch vehicle sequence data.

Event (time in seconds)	Atlas I	Atlas II	Atlas IIA	Atlas IIAS
Guidance Go-Inertial	-11.0	-11.0	-11.0	-11.0
1st SRB pair ignition (Atlas IIAS)	--	--	--	-0.5
Liftoff	0.0	0.0	0.0	0.0
1st SRB pair burnout (IIAS)	--	--	--	54.1
2nd SRB pair ignition (IIAS)	--	--	--	57.5
1st SRB pair jettison (IIAS)	--	--	--	88.1
2nd SRB pair burnout (IIAS)	--	--	--	112.1
2nd SRB pair jettison (IIAS)	--	--	--	114.1
Atlas booster engine cutoff (BECO)	155.9	168.4	168.7	167.9
Booster package jettison (BPJ)	159.0	171.5	171.8	171.0
Insulation panels jettison (Atlas I)	180.9	--	--	--
Payload fairing jettison (PFJ)	221.9	222.1	229.9	220.6
Atlas sustainer engine cutoff (SECO)	265.8	276.1	274.3	277.4
Atlas/Centaur separation	267.8	278.0	276.2	279.4
Begin extendible nozzle deployment (IIA, IIAS)	--	--	277.8	280.9
Centaur main engine start (MES1)	278.3	288.6	286.8	290.4
Centaur main engine cutoff (MECO1)	580.3	678.6	576.1	567.5
Start turn to MES2 attitude	840.3	872.8	856.0	821.0
Centaur main engine start (MES2)	1440.3	1472.8	1456.0	1421.0
Centaur main engine cutoff (MECO2)	1533.4	1578.9	1541.3	1517.4
Start alignment to separation attitude	1535.4	1580.9	1543.3	1519.4
Begin spinup	1590.4	1672.9	1635.3	1611.4
Separate spacecraft (SEP)	1675.4	1753.9	1716.3	1692.4
Start turn to CCAM attitude	1794.4	1813.9	1776.3	1752.4
Centaur end of mission	3058.8	3552.2	3513.4	3490.7

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ty parameters are met. The second pair is jettisoned wo seconds after burnout.

.storically, a nominal zero total angle of attack been maintained for Atlas/Centaur missions .om 8,000 ft (2438 m) through booster engine cutoff BECO). General Dynamics is implementing an alpha-bias angle-of-attack steering profile late in the booster phase. Atlas IIA missions will use this alpha-bias steering technique from approximately 80,000 ft (24 380 m) through BECO to further reduce gravity and steering losses.

Booster staging occurs when the desired axial acceleration is attained. For Atlas I, II, and IIA GTO missions, BECO typically occurs at an axial acceleration of 5.5g. For Atlas IIAS, BECO occurs at 5.2 g. Earlier staging for reduced maximum axial acceleration and/or optimum mission design is easily accomplished with minor associated changes in performance. The booster phase steering profile is implemented through our launch day ADDJUST wind steering programs, which enhance launch availability by controlling wind-induced flight loads. Following jettison of the booster engine and associated thrust structure, flight continues in the sustainer phase. For Atlas I, the fiberglass insulation panels are jettisoned approximately 25 seconds after BECO. Sustainer engine cutoff (SECO) occurs when all available sustainer propellants are consumed. Selected atmospheric ascent data from liftoff through first upper stage burn accompanies the performance data for each launch vehicle at the end of this section (see Figures 2-7, 2-10, 2-16, and 2-22).

For typical Atlas missions, the payload fairing is jettisoned prior to SECO, when 3-sigma free molecular heat flux falls below 360 Btu/ft^2-hr (1135 W/m^2). For sensitive spacecraft, payload fairing jettison can be delayed later into the flight with some performance loss.

2.2.2 CENTAUR PHASE — Centaur main engine start (MES or MES1) occurs 10.5 seconds after the Atlas stage is jettisoned. For direct ascent missions, the Centaur main burn injects the spacecraft into the targeted orbit and then performs a series of pre-separation maneuvers. With parking orbit ascent missions, the Centaur first burn (typically the longer of the two) injects the spacecraft into an elliptic performance-optimal parking orbit. Following first burn main engine cutoff (MECO1), the Centaur and spacecraft enter a coast period. During the coast period (approximately 13 minutes for a typical geosynchronous transfer mission), the Centaur normally aligns its longitudinal axis along the velocity vector. Because typical parking orbit coasts are of short duration, most spacecraft do not require special pointing or roll maneuvers. Should a spacecraft require attitude maneuvers during coast phases, Centaur can accommodate all roll axis alignment requirements and provide roll rates up to 1.0 ± 0.5 degrees per second in either direction during nonthrusting periods. Greater roll rates can be evaluated on a mission-peculiar basis. Prior to second Centaur burn main engine start (MES2), the vehicle is aligned to the ignition attitude and the engine start sequence is initiated.

At a guidance-calculated start time, the Centaur main engines are reignited and the vehicle is guided to the desired orbit. Upon reaching the target, the main engines are shut down (MECO2) and Centaur begins its alignment to the spacecraft separation attitude. Centaur can align to any attitude for separation. Preseparation spinups to 5.0 ± 0.5 rpm can be accommodated.

After Centaur/spacecraft separation, Centaur conducts its collision and contamination avoidance maneuver (CCAM) to prevent recontact and minimize the contamination of the spacecraft.

2.3 PERFORMANCE GROUND RULES

Atlas performance ground rules for various missions with launch from Cape Canaveral Air Force Station in Florida are described in this section.

2.3.1 PAYLOAD SYSTEMS WEIGHT DEFINITION — Performance capabilities quoted throughout this document are presented in terms of payload systems weight. ***Payload systems weight (PSW) is defined as the total mass delivered to the target orbit, including the separated spacecraft, the spacecraft-to-launch vehicle adapter, and all other hardware required on the launch vehicle to support the payload (a payload flight termination system, harnessing, etc.).*** Table 2-3 pro-

Table 2-3. Performance effects of spacecraft-required hardware.

Performance effect of payload adapter masses [lb (kg)]:

Type A 105 (48)	Type B 109 (49)	Type C 54 (25)	Type D 143 (65)
Type A1 113 (51)	Type B1 130 (59)	Type C1 51 (23)	

Performance effect of other spacecraft-required hardware [lb (kg)]:

Standard Package	PLF Acoustic blankets	PLF thermal shield	Environment verification package [telepak, instruments]
16 (7)	25 (11)	8 (4)	20 (9)

Standard package consists of flight termination system (FTS) on Centaur plus two standard access doors, a re-radiating antenna, and a customer logo on the payload fairing

Other hardware:

Centaur hardware affects performance at 2.2 lb (1 kg) mass to 2.2 lb (1 kg) performance ratio

Payload fairing hardware affects performance at ~19.8 lb (9 kg) mass to 2.2 lb (1 kg) performance ratio

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vides weights for our standard payload adapters (see Section 4.1.2 for payload adapter details). Data is also provided estimating the performance impact of various mission-peculiar hardware requirements. As a note, performance effects shown are approximate. The launch vehicle trajectory, spacecraft mass, and mission target orbit can effect the performance contributions of each mission-peculiar item.

2.3.2 PAYLOAD FAIRINGS — All performance in this document is based on use of the 14-ft (4.2-m)-diameter large payload fairing. GD also offers an 11-ft (3.3-m)-diameter medium payload fairing. Higher performance is available for those payloads that fit in the medium fairing. Performance gains are vehicle configuration and trajectory design-dependent, but for GTO missions the gain is approximately 300 lb (135 kg).

For spacecraft that require greater volume than the standard large payload fairing, a 3-ft (1-m) stretch to our large fairing has been evaluated. Performance will degrade approximately 150 lb (65 kg) with its use. Figure 2-2 illustrates the three fairing options. Additional information appears in Section 7.4.

2.3.3 LAUNCH VEHICLE PERFORMANCE CONFIDENCE LEVELS — Atlas missions are targeted to meet the requirements of each user. Historically, Atlas and most U.S.-launched missions have been designed with a performance confidence level of 3-sigma (99.87%). With the flexibility of Atlas/Centaur hardware and flight software, performance confidence levels can be set based on the requirements of each mission. The minimum residual shutdown (MRS) performance option, discussed later in this section, takes full advantage of this concept. All data in this document, with the exception of the elliptical transfer orbit performance data, is based on the 3-sigma confidence level. That is, performance shown will be attained or exceeded with a 99.87% probability.

For the elliptical transfer orbit data, General Dynamics has baselined a 99% confidence level performance reserve. Because many of today's communications satellites can benefit from reduced launch vehicle confidence levels (and the associated nominal performance increases), minimum residual shutdown data is also discussed. General Dynamics will respond to any desired performance confidence level requirement needed by the user.

2.3.4 CENTAUR SHORT-BURN CAPABILITY — For low Earth orbit (LEO) mission applications, General Dynamics has evaluated the launch vehicle requirements for short-duration Centaur second burns. With missions requiring short-duration second burns (15–30 seconds), minor hardware and sequencing changes may be required. Propellant residuals will be biased to ensure proper engine propellant inlet conditions at main engine start. Centaur main engine burns as short as 15 seconds are

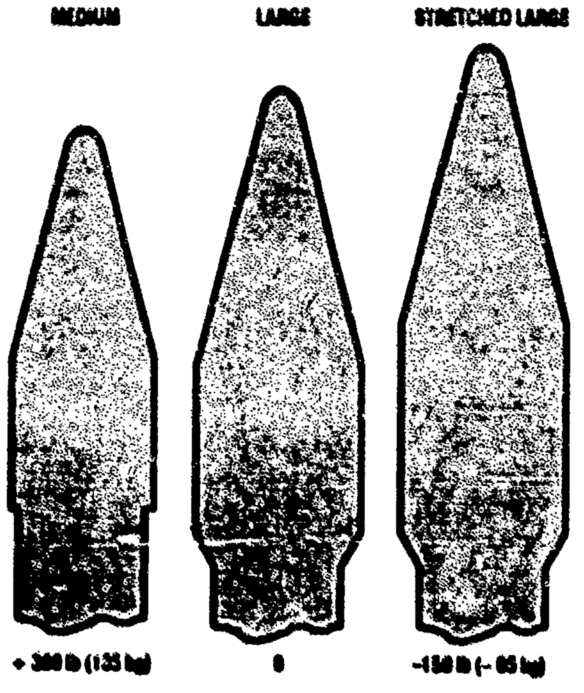

Figure 2-2. Atlas payload fairing options.

possible. All performance data shown using short-duration burns include the performance effects of hardware and sequencing modifications to Centaur.

2.3.5 CENTAUR LONG-COAST CAPABILITY — A Centaur extended-mission kit is currently in development to support long-duration Centaur parking orbit coasts. Coasts of up to 90 minutes in duration are manageable, constrained by helium pressurant and hydrazine RCS propellant capacities. The long-coast kit consists of a larger vehicle battery, shielding on the Centaur aft bulkhead, additional helium capacity, and an additional hydrazine bottle. Performance estimates using long parking orbit coasts include the effect of an extended-mission kit. See Section 7.2 for additional details.

2.3.6 HEAVY-PAYLOAD LIFT CAPABILITY — The Centaur equipment module and payload adapters have been optimized for geosynchronous transfer missions. To manage the larger payload masses Atlas is capable of delivering to low Earth orbits (typically greater than 9,000 lb (4000 kg), two heavy-payload interfaces have been identified. Figure 2-3 illustrates the interfaces with associated performance penalties. In both cases the user must account for the mass of a spacecraft-to-launch vehicle adapter in addition to the stated performance penalty. See Section 7.3 for additional details.

2.4 GEOSYNCHRONOUS LAUNCH MISSION TRAJECTORY AND PERFORMANCE OPTIONS

Through Centaur's flexible flight software, a number of trajectory designs are possible. Depending on the mission requirements, the total satellite weight, the dry mass to propellant mass ratio, and the type of satellite propulsion (liquid or solid) system, one of the following trajectory design options will prove optimal.

- Geosynchronous transfer (and reduced inclination transfers)
- Supersynchronous transfer
- Subsynchronous transfer/perigee velocity augmentation

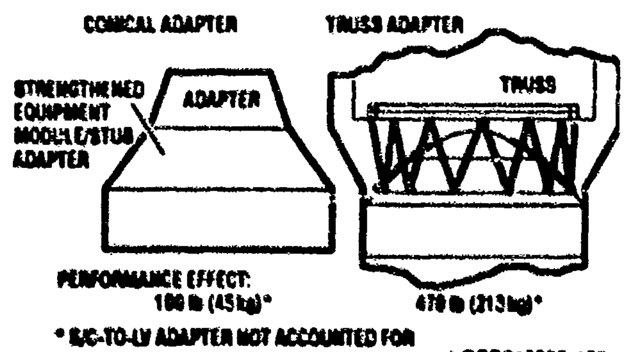

Figure 2-3. Heavy payload interfaces

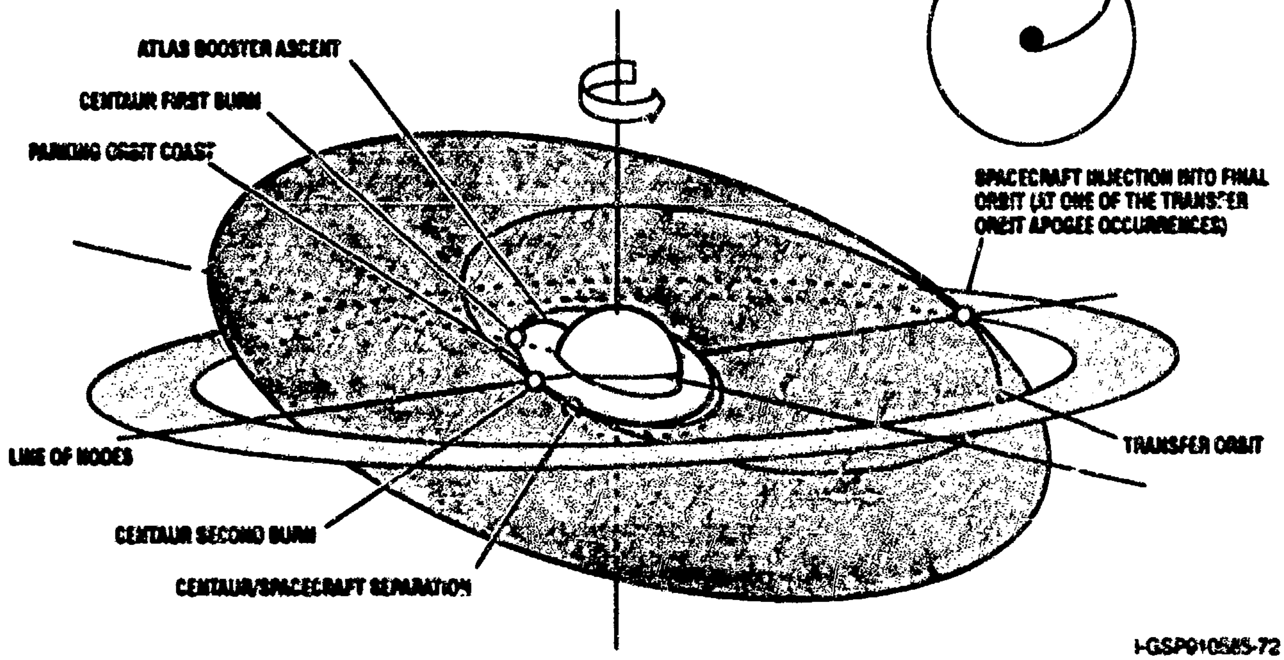

Figure 2-4. The geosynchronous transfer orbit mission trajectory profile.

2.4.1 GEOSYNCHRONOUS TRANSFER — The geosynchronous transfer orbit mission is the standard mission design for communications satellite launches. Figure 2-4 illustrates the orbital mission profile involved. The transfer orbit inclination achieved depends upon launch vehicle capability, satellite launch mass, and the performance characteristics of both systems. Based on the performance of the current Atlas family and enhanced capabilities of today's liquid apogee engine (LAE) subsystems, General Dynamics is finding that a 27-degree inclination is optimal for maximizing satellite beginning-of-life mass given an optimally sized satellite propulsion system. The 300-plus second specific impulses of current LAEs have resulted in a shift in optimum inclination from 26.5 to 27 degrees. With satellites weighing less than the GTO capability of the launch vehicle, excess performance can be used to further reduce inclination or raise perigee.

Although the GTO design is intuitively the standard launch option, single-payload manifesting allows the option of alternate designs that can extend geostationary satellite lifetimes. Supersynchronous transfers, subsynchronous transfers, and other mission enhancement options can enhance lifetime with satellites that use common sources of liquid propellant for both orbit insertion and on-orbit stationkeeping.

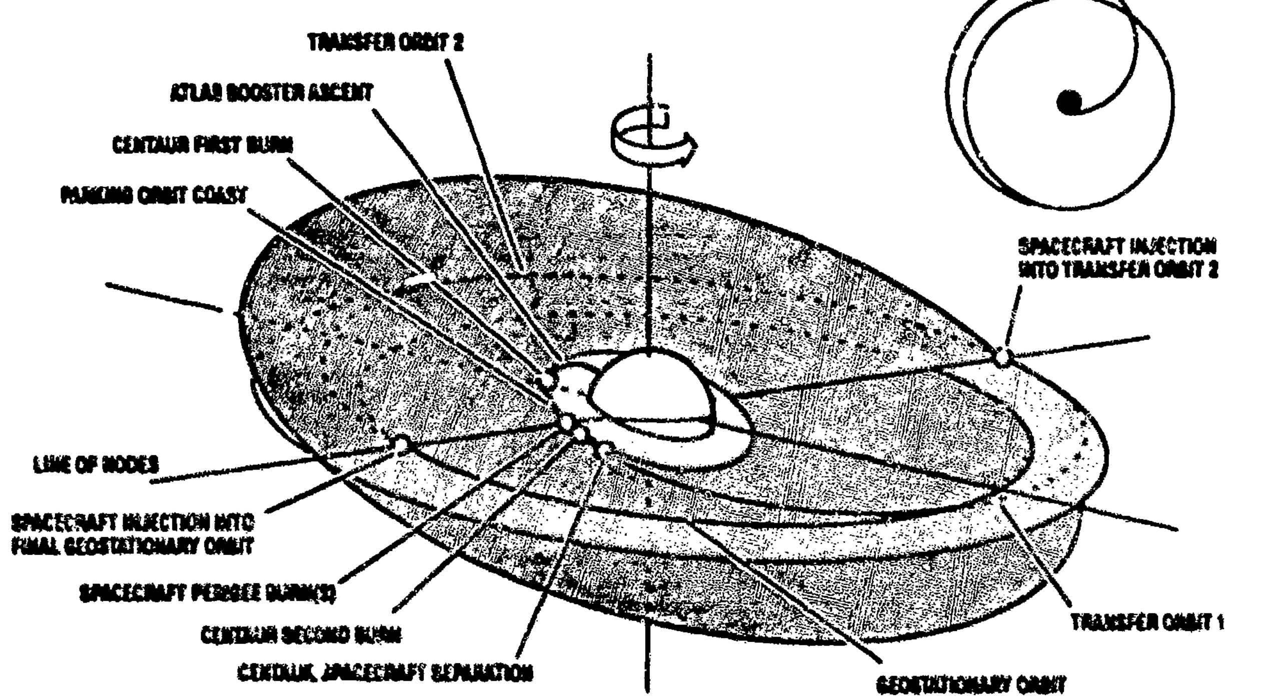

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Figure 2-5. The supersynchronous transfer orbit mission trajectory profile.

2.4.2 SUPERSYNCHRONOUS TRANSFER — The supersynchronous trajectory design offers an increase in beginning-of-life propellants by minimizing the delta-velocity required of the satellite for orbit insertion. The Atlas injects the satellite into an intermediate transfer orbit with an apogee well above geosynchronous altitude. If the apogee altitude capability exceeds the satellite maximum allowable altitude, the excess launch vehicle performance can be used to lower orbit inclination. At supersynchronous altitudes, the decreased inertial velocity allows the satellite to make orbit plane changes more efficiently. The satellite makes the plane change and raises perigee to geosynchronous altitude in one or more apogee burns. It then coasts to perigee and circularizes into final geostationary orbit. The total delta velocity in this supersynchronous transfer design is less than would be required to inject from an equivalent performance reduced inclination geosynchronous transfer, resulting in more satellite propellants available for on-orbit operations. Figure 2-5 illustrates the supersynchronous trajectory mission profile. Table 2-4 quantifies the potential mission gains with the supersynchronous mission for a 4,078 lb (1850 kg) satellite. The Atlas II/EUTELSAT II (AC-102) mission launched 7 December 1991 successfully used the supersynchronous transfer optimization strategy.

Table 2-4. Mission benefits of supersynchronous transfer

	GTO	Supersynch
S/C mass [kg]	1850	1850
Transfer orbit parameters		
Perigee altitude [km]	391	183
Apogee altitude [km]	35,786	50,000
Orbit inclination [deg]	19.6	21.2
Argument of perigee [deg]	180	180
Final orbit	GSO	GSO
S/C ΔV required for GSO insertion [m/sec]	1643	1607
Estimated mission lifetime [years]	6.7	7.2

S/C gains propellant for an additional ½ year of lifetime

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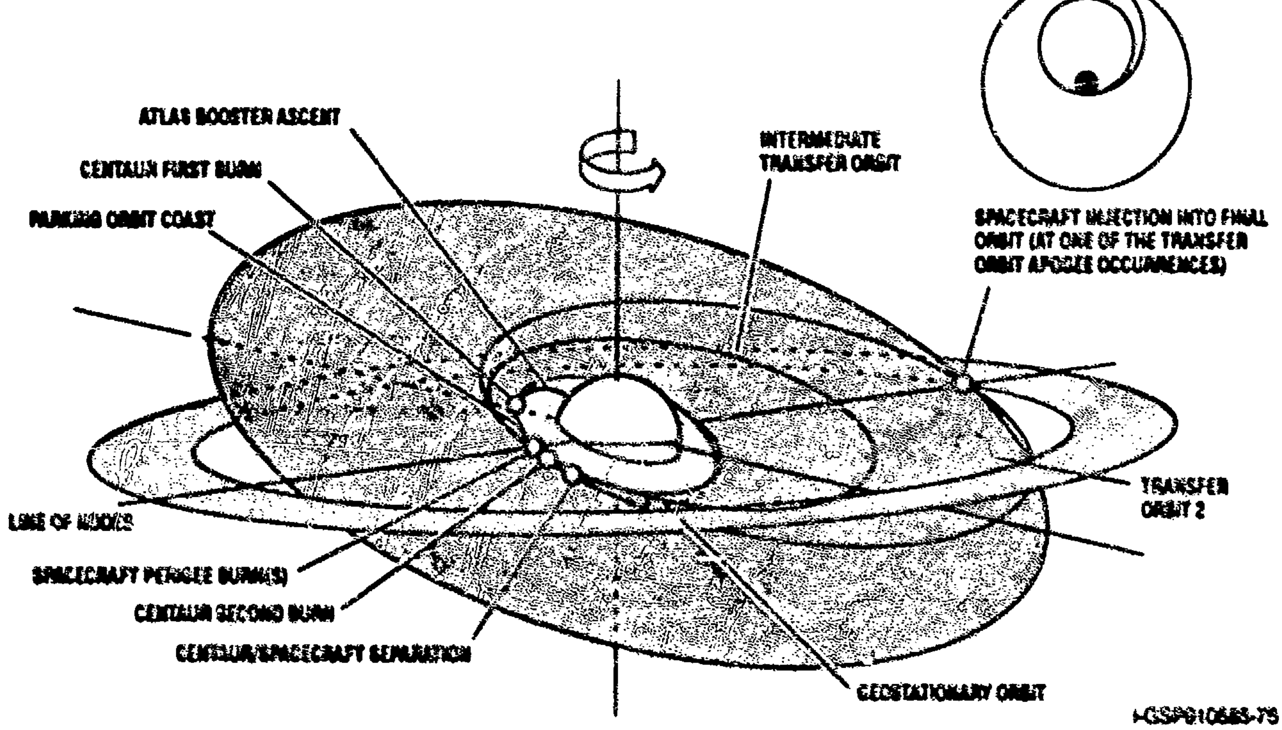

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Figure 2-6. The subsynchronous transfer orbit mission trajectory profile.

2.4.3 SUBSYNCHRONOUS TRANSFER/PERIGEE VELOCITY AUGMENTATION — The perigee velocity augmentation (PVA) trajectory design, compared with the standard GTO design, can provide increased propellant mass at beginning of life on geostationary orbit. This is beneficial when propellant tank capacity is large with respect to the dry mass. The Atlas delivers the satellite to a subsynchronous intermediate transfer orbit (apogee less than geosynchronous) with an inclination of approximately 27 degrees because the satellite weight exceeds the GTO launch capability. The separated satellite coasts to subsequent transfer orbit perigee(s), where the satellite supplies the required delta-velocity for insertion into geosynchronous transfer. At apogee, using one or more burns, the satellite lowers inclination and circularizes into geostationary orbit. As illustrated in Table 2-5, mass at beginning of life is enhanced. The orbit profile is shown in Figure 2-6.

2.4.4 MISSION PERFORMANCE LEVEL PHILOSOPHY — As Table 2-1 illustrates, General Dynamics has expanded the performance offerings for the Atlas family of launch vehicles. With development of our Block 1 performance enhancement package and the current family of vehicles, multiple perform-

Table 2-5. Mission benefits of subsynchronous transfer

	GTO	Subsynch
S/C mass (kg)	3654	3654
S/C offload to meet GTO launch capability (kg):	-100	0
Transfer orbit parameters:		
Perigee altitude (km):	167	167
Apogee altitude (km):	35,786	30,000
Orbit inclination (deg):	27.0	27.0
Argument of perigee (deg):	180	180
Final orbit:	GSO	GSO
S/C ΔV required for GSO insertion (m/sec):	1806	1913
S/C mass at beginning of life (kg):	1916	1933
Estimated mission lifetime (years):	12.1	12.6
S/C gains propellant for an additional ½ year of lifetime		

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ance levels are available. In addition, GD can also meet performance requirements by customizing (or standardizing) the mission and trajectory designs to meet specific mission desires. To meet evolving commercial satellite mission launch requirements, GD intends to offer performance capability levels (as opposed to explicit hardware configurations) as part of its standard commercial launch services package.

Besides offering stated performance levels that are associated with each vehicle, other performance requirements can be met in several ways. With satellite missions that may require less performance than a specific configuration may offer, additional mission constraints will be employed that will use excess performance to benefit the launch services customer. Ascent trajectory designs can be shaped to include coverage of the Centaur second burn and spacecraft separation from Ascension Island tracking station as opposed to pre-positioning advanced range instrumentation aircraft (ARIA). The ascent profile can be standardized to reduce mission integration analyses and/or schedules. These options can allow a more cost-effective solution for cases where maximum vehicle performance is not required.

Standard GTO Performance Option — The standard GTO performance option is a highly constrained version of the reduced performance concept. The standard GTO mission is, as stated, constrained to be a geosynchronous transfer mission using a standardized atmospheric ascent profile. As Atlas is currently integrating satellites from most of the world's manufacturers, integration analyses can be limited to those that are required to support the specific mission payload on the satellite bus.

Custom Performance Option — The custom performance level option is closely related to the performance levels quoted in the previous issue of this guide. With the first flights of Atlas I and Atlas II, the upcoming launch of Atlas IIA, and continuing development of Atlas IIAS, portions of unallocated design reserves have been released for mission use. The Custom version of Atlas IIA and Atlas IIAS will nominally use the Centaur RL10A-4 engine. Ultimate vehicle configuration, given the expected phase-in of enhanced hardware, will inevitably depend on timeframe of proposed launch and specific performance requirements.

Block 1 Performance Option — The Block 1 performance option is currently on contract for Atlas IIAS, and primarily involves an enhancement to the Centaur RL10A-4 engine system. Thrust rating and specific impulse are increased. The enhanced system is designated the RL10A-4-1 engine. This enhancement alone accounts for more than a 130 lb (60 kg) increase in performance for GTO missions. In addition, seven-degree cant nozzle Castor IVA SRBs will replace the current 11-degree cant air-lit pair. A set of other minor enhancements (weight reduction, sequencing improvements) complete the Block 1 package.

Several of the upgrades under contract for Atlas IIAS are also directly applicable to the Atlas IIA vehicle and are being incorporated. The Block 1 package is available to support a 1993 launch capability.

2.5 MISSION PERFORMANCE DATA

Detailed performance curves are provided, for each launch vehicle, at the end of this section. Data is shown for several types of launch missions and is based on our current estimate of each vehicle's lift capability. The performance ground rules are shown on each curve and additional information is provided in the following paragraphs.

As a note, because of the limited number of Atlas I vehicles, performance data shown in this document is limited to geosynchronous transfer elliptical orbit capabilities. This data is shown primarily to illustrate the flight-verified capability of the vehicle and to provide a point of reference for the Atlas II family of vehicles.

2.5.1 ELLIPTICAL TRANSFER CAPABILITY — The optimum trajectory profile for achieving elliptical transfer orbits is the parking orbit ascent. Atlas performance capability for 27-degree inclined orbits is shown in Figures 2-8, 2-11, 2-17, and 2-23. The 27-degree inclined orbit missions are launched at flight azimuths that have been approved and flown for many missions. The 27-degree inclination is near-optimal for geostationary transfer missions. The Centaur second burn is executed near the first descending node of the parking orbit (near the equator). Performance is shown for both the 99% confidence level and minimum residual shutdown (MRS). Tabular performance data is provided in Tables 2-9 through 2-12, at the end of each launch vehicle performance data section.

For some missions, an ascending node injection into the transfer orbit may offer advantages. The performance degradation and mission constraints associated with this mission type need to be analyzed on a mission-peculiar basis.

2.5.2 REDUCED INCLINATION ELLIPTICAL TRANSFER CAPABILITY — The inclination effect on payload systems weight capability for a geosynchronous transfer orbit design for inclinations between 29 and 20 degrees is shown in Figures 2-9, 2-12, 2-18, and 2-24. Performance degrades as inclination drops below 28.5 degrees in part because the launch vehicle instantaneous impact point (IIP) trace is constrained by range safety requirements to remain a minimum of 150 nmi (278 km) off the Ivory Coast of Africa. This requirement dictates that yaw steering be implemented in the ascent phase to meet range safety requirements while also attempting to lower park orbit inclination toward the desired transfer orbit target value. The remainder of the inclination is completed with yaw steering in the Centaur second burn. Data is shown at the 99% confidence level and minimum residual shutdown for transfer orbit apogee altitudes of 19,324 nmi (35 788 km) and 53,996 nmi (100 000 km).

2.5.3 EARTH-ESCAPE PERFORMANCE CAPABILITY — Earth-escape mission performance is shown in Figures 2-13, 2-19, and 2-25. Centaur's heritage as a high-energy upper stage makes it ideal for launching spacecraft into Earth-escape trajectories. Performance data shown uses the parking orbit ascent design and a near-planar ascent to an orbit that contains the outgoing asymptote of the escape hyperbola with a ~700-second coast time between the upper stage burns. The reference performance curves were developed using a 180-degree argument of perigee target, and coast times reflect this constraint. The actual coast time necessary to achieve the desired departure asymptote will be determined by specific mission requirements. Our quoted performance assumes that 3-sigma (99.87% confidence level) flight performance reserves are held.

Additional performance data is shown for an optional vehicle configuration that uses the parking orbit ascent design with a third stage, a near-optimum size solid propellant orbit insertion stage (OIS), the Thiokol STAR 48B (TEM-711-18) motor. This vehicle configuration is advantageous for missions that require a very high-energy Earth departure, cases in which vehicle staging effects make it more efficient for a third stage to provide an additional energy increment. The reference performance mission was targeted similarly to the no-OIS case with the STAR 48B burn occurring just after Centaur/STAR 48B separation. This vehicle configuration is similar to the STAR 37-based Atlas configuration flown for the AC-27 and AC-30 Pioneer 10 (F) and Pioneer 11 (G) missions.

2.5.4 LOW EARTH ORBIT CAPABILITY — Atlas can launch payloads into a wide range of low Earth orbits from Cape Canaveral using either the direct ascent or parking orbit ascent mission profiles. LEO capabilities typically require heavy-payload modifications.

Direct Ascent to Circular Orbit — Figures 2-14, 2-20, and 2-26 show circular orbit payload systems weight capability to LEO using the one Centaur burn mission profile. The maximum capability available is with planar ascent to a 28.5-degree inclination orbit. As shown, inclinations from 28.5 degrees up to 55 degrees are possible with the direct ascent. Given known range safety constraints, direct ascent performance to inclinations greater than 55 degrees are not possible due to land overflight constraints up the eastern seaboard of the United States and Canada. Direct ascent peformance to reduced inclination orbits (down to ~22 degrees) are also possible, but at the expense of substantial performance due to range safety overflight constraints over the Ivory Coast of Africa.

Direct Ascent to Elliptical Orbit — Figures 2-14, 2-20, and 2-26 also show elliptical orbit performance capability using the direct ascent with perigee altitude at 100 nmi (185 km). Similar range safety and orbital mechanics constraints limit inclinations available with a Florida launch.

Parking Orbit Ascent to Circular Orbit — Payload delivery to low-altitude circular orbit can be accomplished by two or more upper stage burns. The first Centaur burn is used to inject the Centaur and payload into an elliptic parking orbit. A park orbit perigee altitude of 80 nmi (148 km) is assumed for our reference cases. Expected parking orbit coast durations will require use of the Centaur extended mission kit. The second Centaur burn will circularize the spacecraft into the desired orbit altitude.

Circular orbit performance capabilities for altitudes between 160 nmi (300 km) and 1,000 nmi (2000 km) are shown for each vehicle in Figures 2-14, 2-20, and 2-26. Data is shown for 28.5-, 55-, and 63.4-degree inclinations. With high-inclination orbits (inclinations greater than 55 degrees), range safety requirements require that Atlas meet instantaneous impact constraints up the eastern seaboard of the United States and Canada. Additional inclination is added in the later stages of the Centaur first burn and with the second Centaur burn. As desired orbit inclination increases, performance degradations become more pronounced. High- inclination orbit performance capabilities are further discussed in Section 8.

2.5.5 INTERMEDIATE CIRCULAR ORBITS — Performance data is shown in Figures 2-15, 2-21, and 2-27 for altitudes between 5,000 nmi (10 000 km) and 11,000 nmi (20 000 km). Similar ground rules apply to the intermediate circular orbit data as to the LEO circular orbit data except with respect to heavy payload requirements. The lower performance capabilities associated with the higher energy circular missions should allow use of standard payload adapters.

2.6 MISSION OPTIMIZATION AND ENHANCEMENT

Atlas trajectory designs are developed using a detailed integrated trajectory simulation executive and a state-of-the-art optimization algorithm (sequential quadratic programming, SQP). The optimization capability shapes the trajectory profile from liftoff through spacecraft injection into final orbit.

The SQP program uses up to 50 independent design variables chosen to maximize a performance index and satisfy specified constraints. Typical control variables include boost phase initial pitch and roll (launch azimuth) maneuvers, Atlas sustainer steering, and Centaur steering for all burns. In addition, space vehicle pitch and yaw attitudes, ignition times, etc. can be included as part of the total optimization process. The optimization program is formulated with up to 40 equality and inequality constraints on variables such as dynamic pressure, tracker elevation angle, and range safety.

With General Dynamics' experience with launch of interplanetary and scientific spacecraft, an additional trajectory analysis tool is available to assist in the mission design/mission optimization process. The N-BODY trajectory simulation program is used, in concert with our optimization capability, in the development of launch vehicle missions requiring precision inertial targeting in which perturbation effects of other celestial bodies are required to be considered.

These trajectory analysis tools, along with our extensive guidance and targeting capabilities, enable Atlas to optimize the mission based on spacecraft characteristics. The most widely used mission enhancement options are as follows:

- Inflight retargeting
- Minimum residual shutdown
- Explicit right ascension of ascending node

2.6.1 INFLIGHT RETARGETING — The software capability of the Centaur upper stage makes it possible to evaluate Atlas performance in flight and then target for an optimal injection condition that is a function of the actual performance of the booster stage. Centaur can be retargeted to a variable transfer orbit inclination, apogee, perigee, argument of perigee, or any combination of the above. Inflight retargeting can provide a dual performance benefit. First, the nominal launch vehicle flight performance reserve (FPR) is reduced when the FPR contribution due to Atlas dispersions is eliminated. Second, whether Atlas performance is high, nominal, or low, the retargeting logic is calibrated to devote all remaining propellant margin to benefit the mission. With inflight retargeting, any desired level of confidence of a guidance shutdown can be implemented. Inflight retargeting was successfully executed for the Atlas II/EUTELSAT II mission.

2.6.2 MINIMUM RESIDUAL SHUTDOWN — Centaur propellants may be burned to minimum residuals for a significant increase in nominal performance capability. When burning to minimum residuals, the flight performance reserve (FPR) propellants are eliminated to nominally gain additional delta-velocity from the Centaur.

It is practical for Centaur to burn all its propellants when the satellite has a liquid propulsion system that is capable of correcting for variations in launch vehicle performance. This option is particularly attractive and appropriate when the trajectory design includes a supersynchronous or subsynchronous (PVA) transfer orbit. Satellites using solid propellant (fixed impulse) orbit insertion stages require FPR propellants to ensure that the Centaur injection conditions will match the capability of the fixed impulse stage.

When Centaur burns all propellants to minimum residuals, the liquid propellant satellite corrects for the effects of launch vehicle dispersions. These dispersions primarily affect apogee altitude. Variations in other transfer orbit parameters are minor. The performance variation associated with MRS can also be quantified as an error in injection velocity which can be approximated as a dispersion in transfer orbit perigee velocity. Table 2-6 documents 3-sigma MRS perigee velocity injection variations for the Atlas family. MRS was successfully executed for the Atlas I/CRRES mission launched 25 July 1990.

2.6.3 RIGHT ASCENSION OF ASCENDING NODE CONTROL — Some satellite mission objec-

Table 2-7. Typical injection accuracies at spacecraft separation.

	Orbit at Centaur/ Spacecraft Separation		± Three-sigma Errors				
Mission	Apogee [nmi (km)]	Inclination (deg)	Apogee [nmi (km)]	Perigee [nmi (km)]	Inclination (deg)	Argument of Perigee (deg)	RAAN (deg)
GTO	19,287 (35720)	27.0	57.4 (106)	1.3 (2.4)	0.02	0.16	0.18
GTO	19,411 (35949)	22.1	58.9 (109)	1.2 (2.2)	0.02	0.19	0.21
GTO (Supersynchronous)	22,189 (41094)	19.3	77.3 (143)	1.2 (2.2)	0.02	0.21	0.23
LEO (circular)	216 (400)	60.0	2.4 (4.4)	2.6 (4.8)	0.06	NA	0.06

NA = Not Applicable

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tives may require launch-on-time placement into transfer and/or final orbit. For Earth orbital missions, this requirement typically manifests itself as a right ascension of ascending node (RAAN) target value or range of values. Centaur's heritage of meeting the inertial orbit placement requirements associated with planetary missions makes it uniquely capable of targeting to an orbit RAAN (or range of RAANs dictated by actual launch time in a launch window) in addition to the typical target parameters. With GTO missions, some satellite mission operational lifetimes can be enhanced by controlling RAAN of the targeted transfer orbit. A satellite intended to operate in a non-zero degree geosynchronous final orbit can benefit with proper RAAN placement. A drift toward a zero-degree inclination orbit can help reduce the typical north-south stationkeeping budget of the satellite, thereby increasing the amount of time the satellite can remain in an operational orbit.

Control of the node is obtained by varying the argument of perigee of the transfer orbit and the satellite burn location with respect to the equator. The difference between the inclination of the transfer orbit and final orbits, and the latitude of the satellite burn, determines the amount of nodal shift between the transfer orbit and the mission orbit. Control of this shift is used to compensate for off-nominal launch times, keeping the inertial node of the final orbit fixed throughout a long-launch window. Centaur software can be programmed to control the argument of perigee (satellite burn location) as a function of time into the launch window to obtain the desired final orbit inertial node.

Table 2-6. Three-sigma performance variations with MRS.

Perigee velocity dispersions		
Atlas I		300 ft/sec (91.4 m/sec)
Atlas II		285 ft/sec (86.9 m/sec)
Atlas IIA	(Custom)	285 ft/sec (86.9 m/sec)
	(Block 1)	290 ft/sec (88.4 m/sec)
Atlas IIAS	(Custom)	275 ft/sec (83.8 m/sec)
	(Block 1)	280 ft/sec (85.3 m/sec)

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2.7 INJECTION ACCURACY AND SEPARATION CONTROL

Atlas' combination of precision guidance hardware with flexible guidance software provides accurate payload injection conditions for a wide variety of missions. The minimal data required to specify targeted end conditions provides for rapid preflight retargeting in response to changing mission requirements. These functional capabilities have been demonstrated on many low Earth orbit, geosynchrononous orbit, lunar, and interplanetary missions.

Accuracy for a variety of GTO and LEO missions is displayed in Table 2-7 and are typical of the three-sigma accuracies following final upper stage burn. On all missions to GTO (more than 40 to date), Atlas has met all mission injection requirements.

Past lunar and interplanetary mission accuracy requirements and achievements are shown in Table 2-8. On some of the planetary missions, the guidance requirement included orientation of a solid rocket kick stage to achieve the proper final planetary intersect conditions. The major error source on these missions was the uncertainty of solid rocket impulse.

Table 2-8. Lunar and interplanetary mission accuracy.

	Figure of Merit (FOM) (m/sec)	
Mission	Mission Requirement	Guidance System 1σ
Surveyor	< 50	7.0
Mariner Mars	< 13.5	3.5
Mariner Venus Mercury	< 13.5	2.4
Pioneer 10	< 39	39*
Pioneer 11	< 36	36*
Viking 1	< 15	3.6
Viking 2	< 15	3.6
Voyager 1	< 21	16.2*
Voyager 2	< 21	17.6*
Pioneer Venus 1	< 7.5	2.3
Pioneer Venus 2	< 12.0	3.2

* Major error source - solid kickstage motor

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2.7.1 ATTITUDE ORIENTATION AND STABILIZATION — During coast phases, the guidance, navigation, and control (GN&C) system can orient the spacecraft to any desired attitude. The guidance system can reference an attitude vector to a fixed inertial frame or a rotating orthogonal frame defined by the instantaneous position and velocity vector. The reaction control system (RCS) autopilot incorporates three-axis-stabilized attitude control for attitude and hold and maneuvering. In addition to a precision attitude control mode for spacecraft preseparation stabilization, Centaur can provide a stabilized spin rate to the spacecraft while maintaining roll axis orientation prior to separation. The Centaur system can accommodate spin rate up to 5.0 ± 0.5 rpm, subject to some limitation due to space vehicle mass property misalignments. A detailed analysis for each Centaur/spacecraft combination will determine the maximum achievable spin rate.

The extensive capabilities of the GN&C system allow the upper stage to satisfy a variety of spacecraft orbital requirements including thermal control maneuvers, sun-angle pointing constraints, and telemetry transmission maneuvers.

2.7.2 SEPARATION POINTING ACCURACIES — Pointing accuracy just prior to spacecraft separation is a function of guidance system hardware, guidance software, and autopilot attitude hold capabilities. In the non-spinning precision pointing mode, the system can maintain attitude errors less than 0.6, 0.6, and 1.6 degrees, and attitude rates less than 0.2, 0.2, and 0.5 deg/sec about the pitch, yaw, and roll axes, respectively (prior to separation) (see Table 2-9). For a mission requiring preseparation spinup, these sources combine with any tipoff ef-

Table 2-9. Summary of guidance and control capabilities.

Coast phase attitude control:	
Roll axis pointing (deg, half angle)	≤ 1.6
Passive thermal control rate (deg/sec)	1.5 ± 0.5 (clockwise or counterclockwise)
Centaur separation parameters at separation command (no spin requirement):	
Roll axis pointing (deg, half angle)	≤ 0.7
Body axis rates (deg/sec)	
Pitch	±0.2
Yaw	±0.2
Roll	±0.5
Spacecraft separation parameters following separation (with spin requirement)	
Nutation (deg, half angle)	≤ 5.0
Momentum pointing (deg, half angle)	≤ 3.0
Spin rate (deg/sec)	≤ 30.0 ± 3.0

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fects induced by the separation system and any spacecraft principal axis misalignments to produce postseparation momentum pointing and nutation errors. Here, nutation is defined as the angle between the actual space vehicle geometric spin axis and the spacecraft momentum vector. Although dependent on the actual spacecraft mass properties (including uncertainties) and the spin rate, momentum pointing and maximum nutation errors following separation are typically less than 3.0 and 5.0 degrees, respectively.

2.7.3 SEPARATION VELOCITY — The relative velocity between the spacecraft and the Centaur is a function of the mass properties of the separated vehicles and the separation mechanism. Our separation systems provide a minimum relative velocity of 1.0 ft/sec (0.3 m/sec) and are designed to preclude recontact between the spacecraft and the Centaur.

ATLAS I PERFORMANCE

- Elliptical transfer orbit
- Reduced-inclination elliptical transfer orbit
- PSW versus transfer orbit apogee altitude

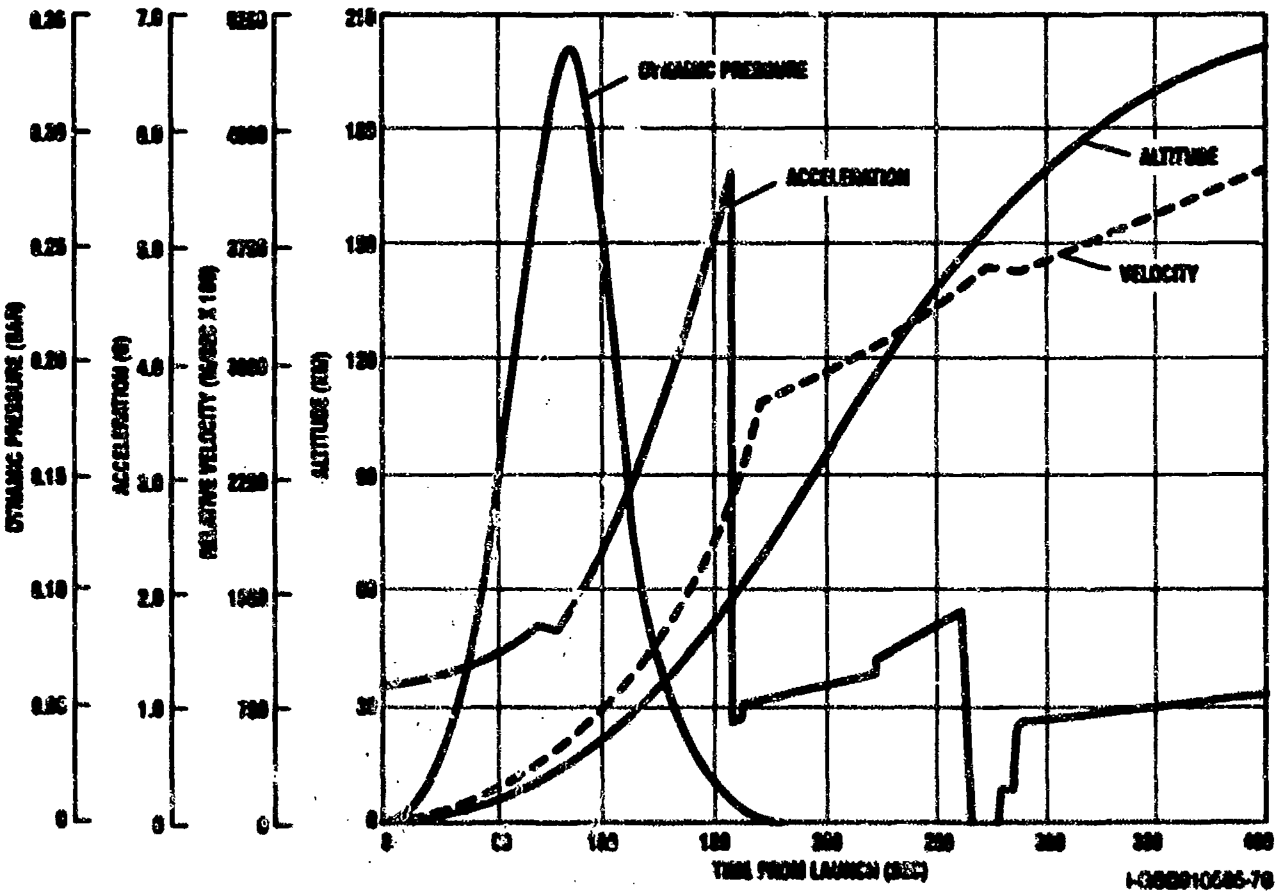

Figure 2-7. Atlas I nominal ascent data.

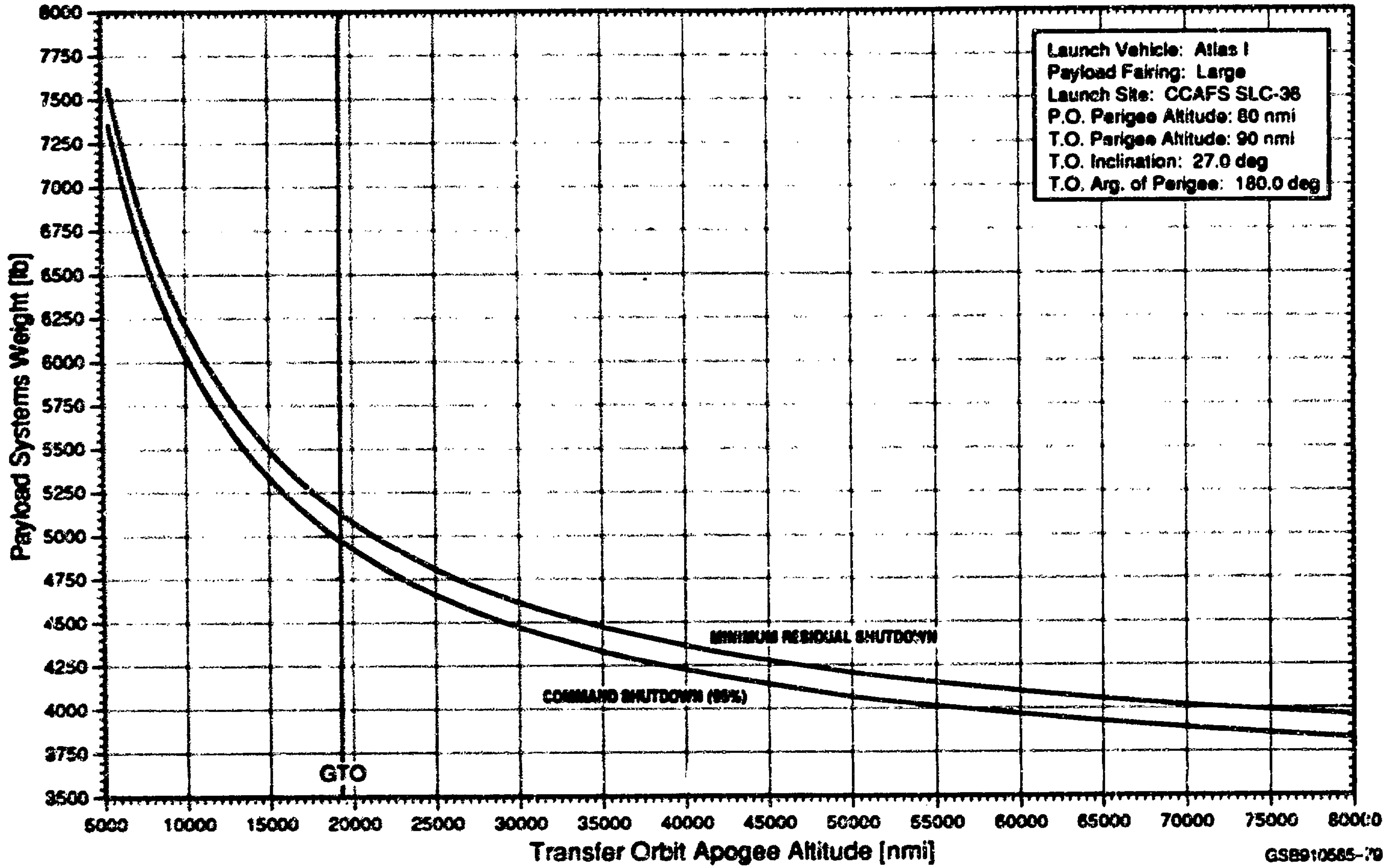

Figure 2-8a. Atlas I performance to elliptical transfer orbit.

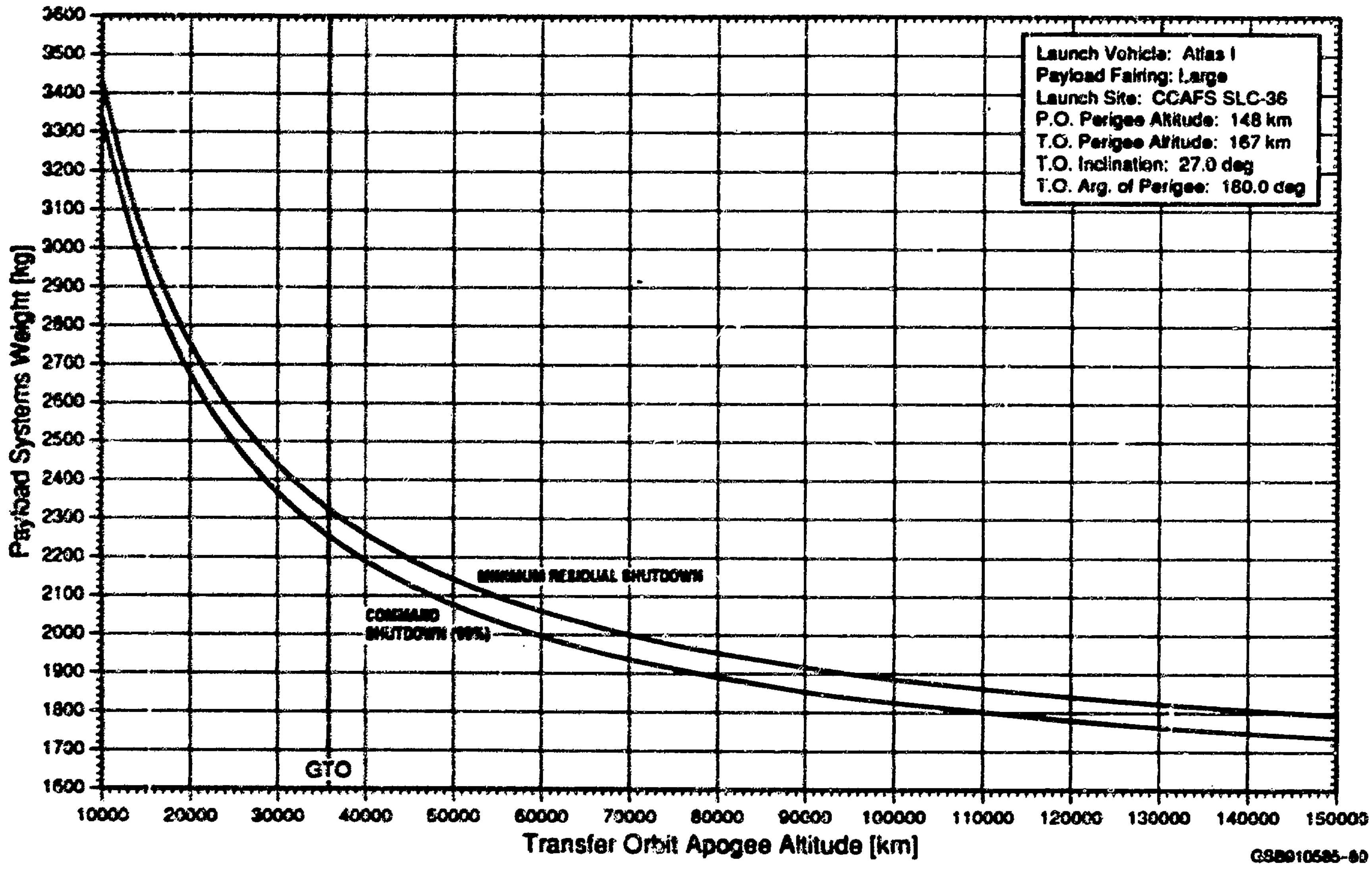

Figure 2-8b. Atlas I performance to elliptical transfer orbit (metric).

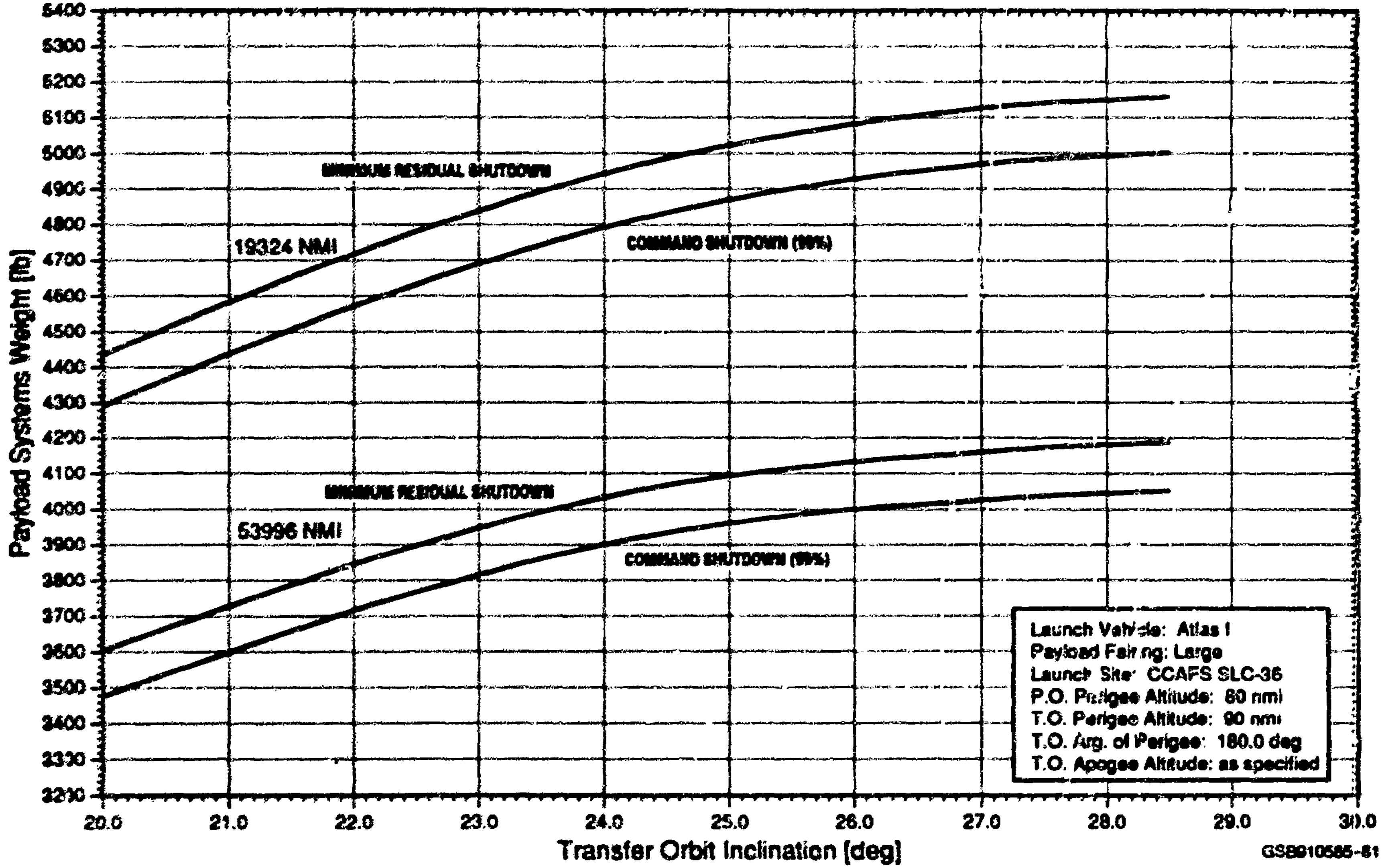

Figure 2-9a. Atlas I reduced inclination elliptical transfer orbit performance.

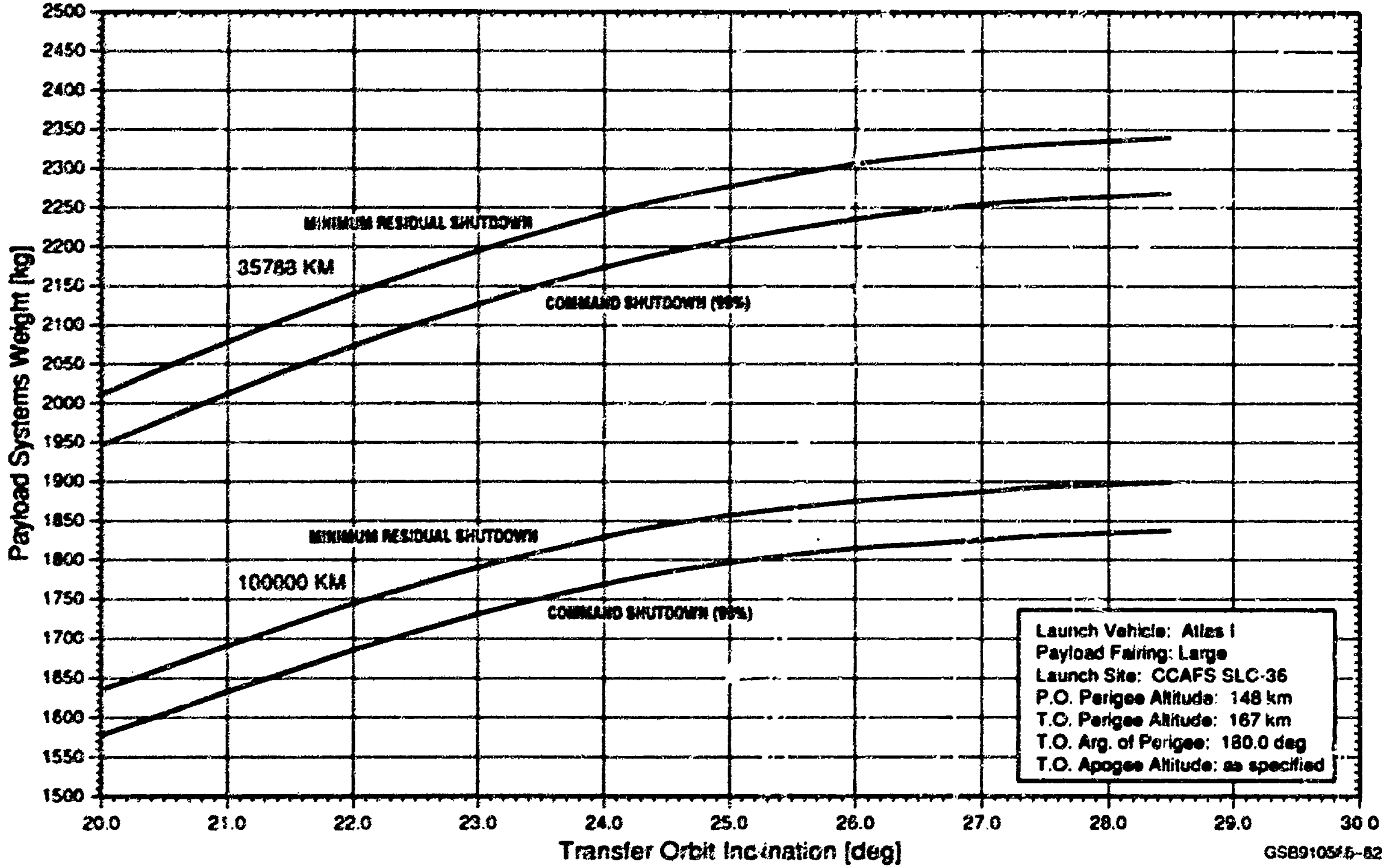

Figure 2-9b. Atlas I reduced inclination elliptical transfer orbit performance (metric).

Table 2-10. Atlas I performance — PSW vs transfer orbit apogee altitude.

Apogee Altitude [nmi (km)]		Payload Systems Weight [lb (kg)]			
		MRS		99%	
80,994	(150000)	3,956	(1794)	3,826	(1735)
67,495	(125000)	4,038	(1832)	3,907	(1772)
53,996	(100000)	4,160	(1887)	4,025	(1826)
48,596	(90000)	4,226	(1917)	4,089	(1855)
43,197	(80000)	4,307	(1953)	4,170	(1891)
40,497	(75000)	4,355	(1975)	4,217	(1913)
37,797	(70000)	4,409	(2000)	4,270	(1937)
35,097	(65000)	4,472	(2028)	4,330	(1964)
32,397	(60000)	4,543	(2061)	4,400	(1996)
31,048	(57500)	4,583	(2079)	4,439	(2014)
29,698	(55000)	4,624	(2098)	4,481	(2033)
28,348	(52500)	4,671	(2119)	4,528	(2054)
26,998	(50000)	4,722	(2152)	4,577	(2076)
25,648	(47500)	4,778	(2167)	4,631	(2101)
24,298	(45000)	4,839	(2195)	4,690	(2127)
22,948	(42500)	4,906	(2225)	4,756	(2157)
21,598	(40000)	4,979	(2258)	4,829	(2190)
20,248	(37500)	5,061	(2296)	4,910	(2227)
19,324	(35786)	5,124	(2324)	4,970	(2254)
18,898	(35000)	5,154	(2338)	4,998	(2267)
17,549	(32500)	5,258	(2385)	5,101	(2314)
16,199	(30000)	5,375	(2438)	5,217	(2366)
14,849	(27500)	5,511	(2500)	5,349	(2426)
13,499	(25000)	5,666	(2570)	5,502	(2496)
12,149	(22500)	5,850	(2654)	5,681	(2577)
10,799	(20000)	6,066	(2752)	5,894	(2673)
9,449	(17500)	6,327	(2870)	6,150	(2790)
8,099	(15000)	6,647	(3015)	6,464	(2932)
6,749	(12500)	7,049	(3197)	6,858	(3111)
5,400	(10000)	7,569	(3433)	7,361	(3339)

60 nmi (148 km) perigee parking orbit; 90 nmi (167 km) perigee transfer orbit; 27-degree inclination; 180-degree argument of perigee; large payload fairing

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ATLAS II PERFORMANCE

- Elliptical transfer orbit
- Reduced-inclination elliptical orbit
- Earth escape
- Low Earth orbit
- Intermediate circular orbit
- PSW versus transfer orbit apogee altitude

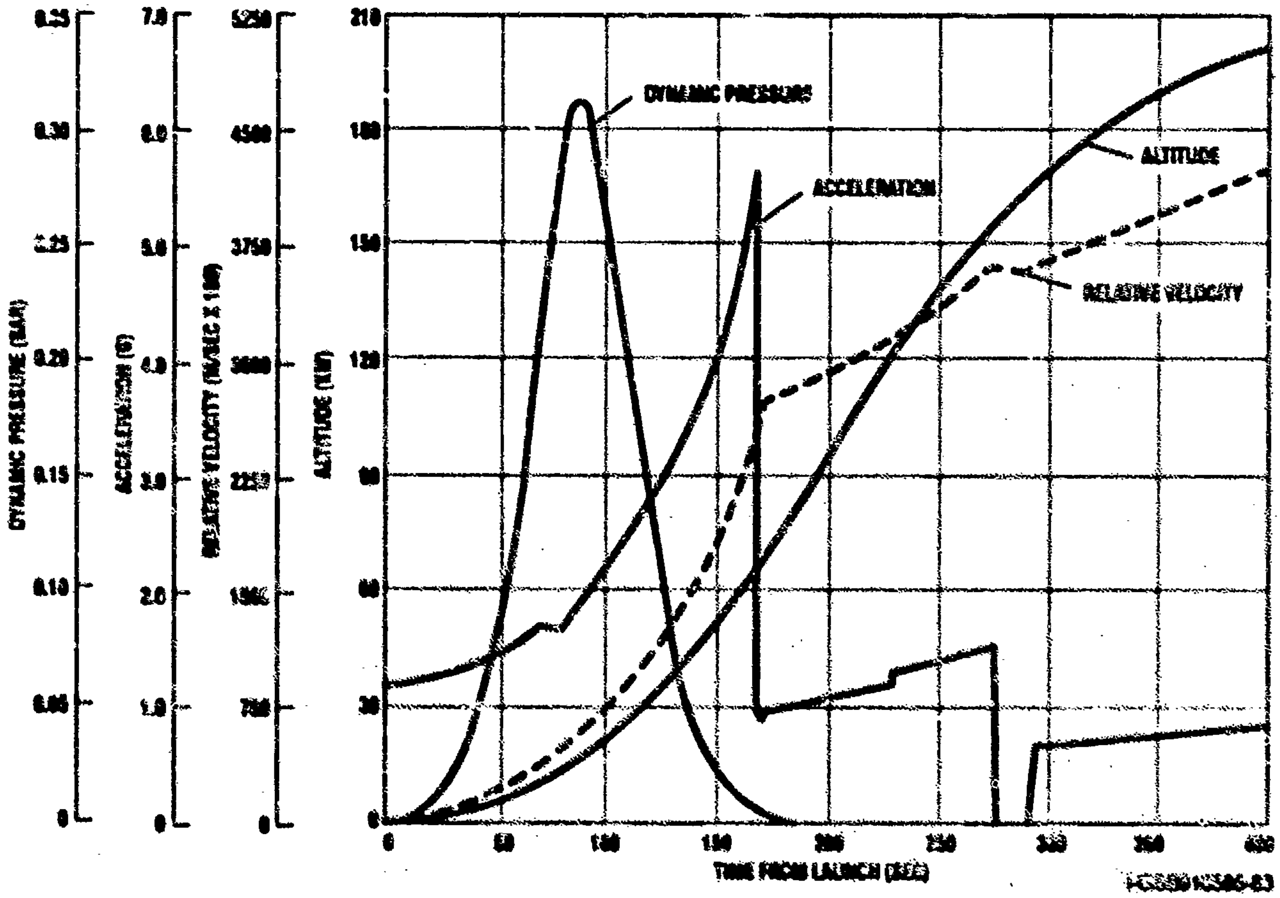

Figure 2-10. Atlas II nominal ascent data.

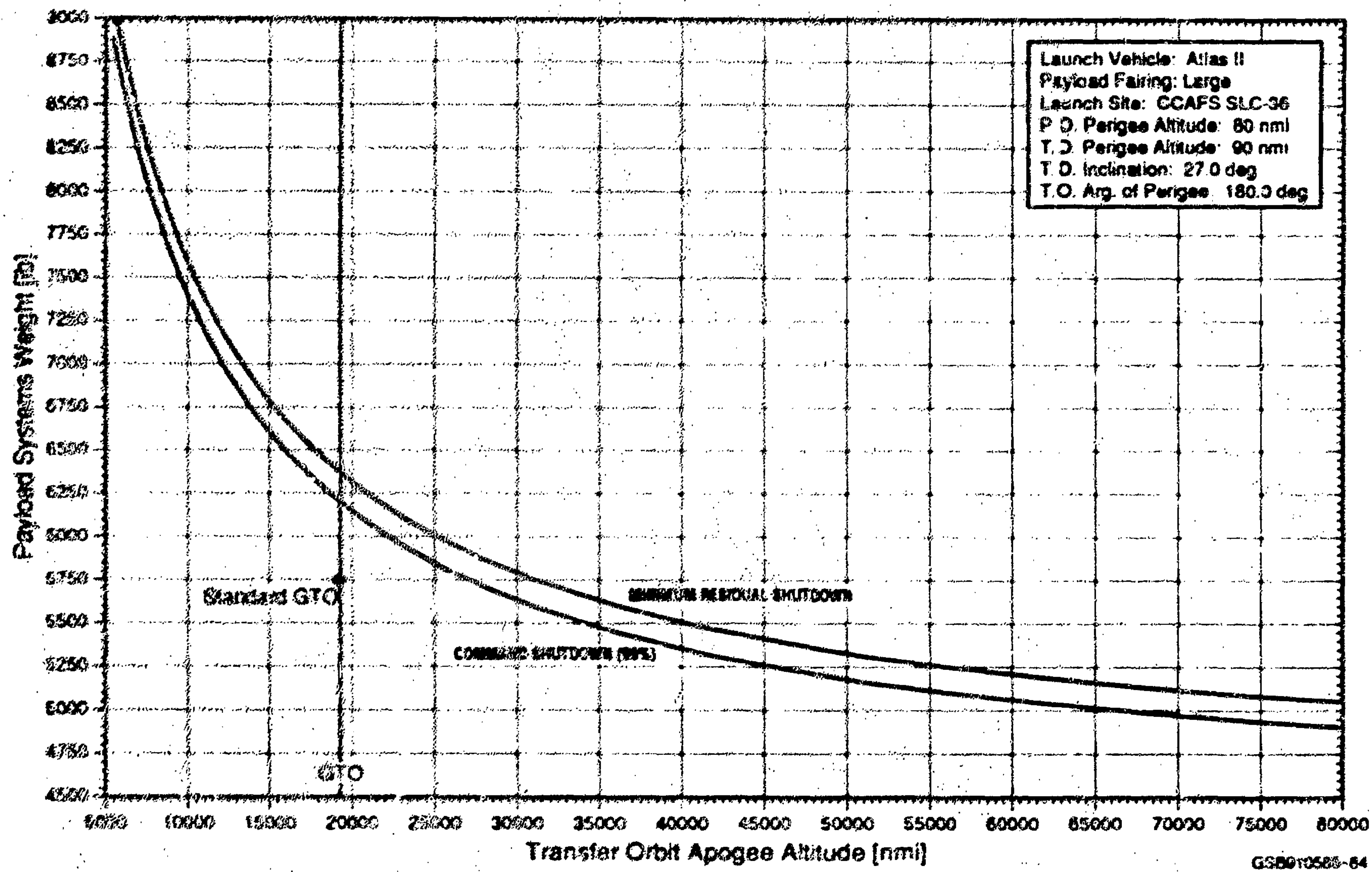

Figure 2-11a. Atlas II performance to elliptical transfer orbit

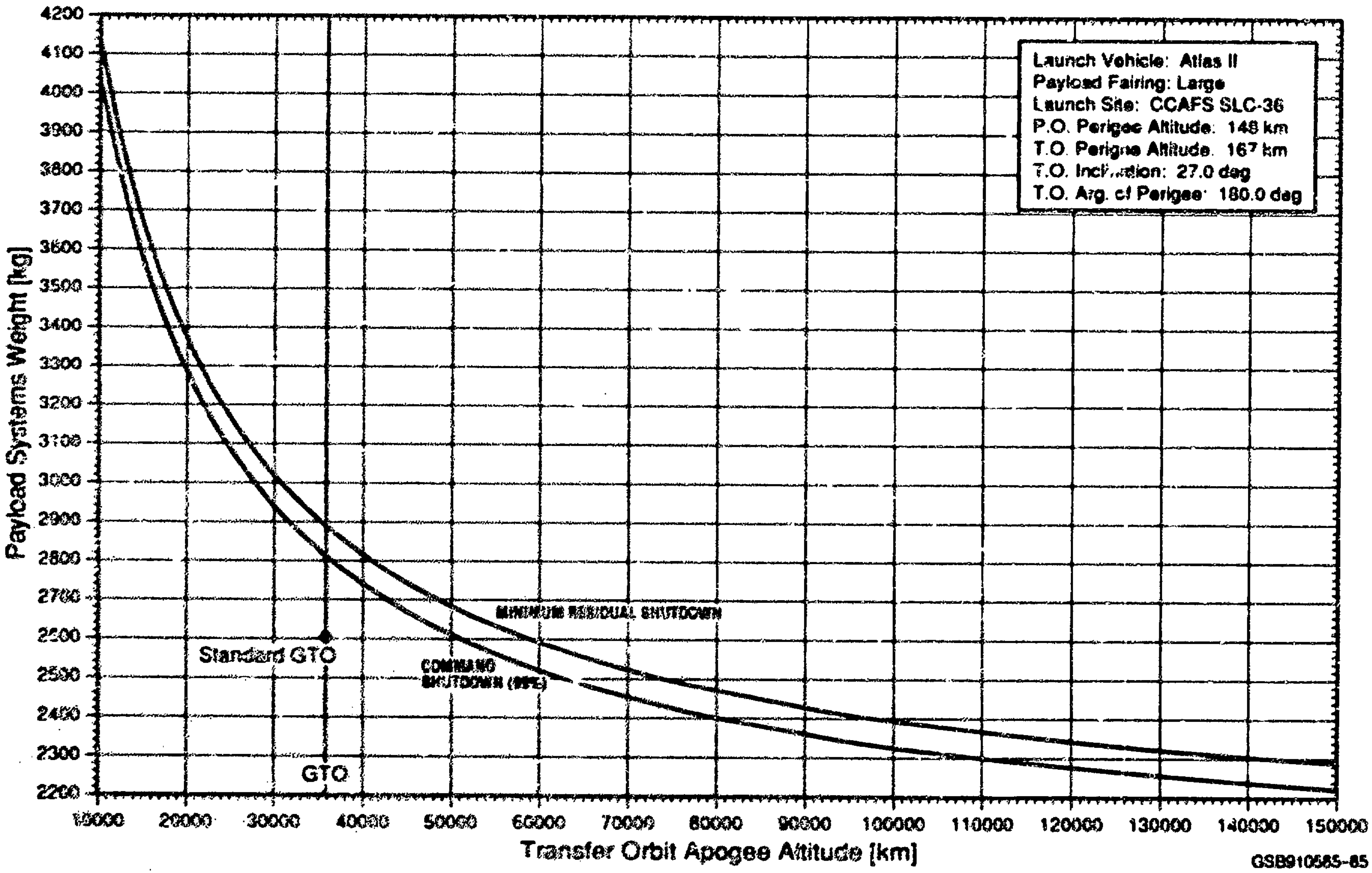

Figure 2-11b. Atlas II performance to elliptical transfer orbit (metric)

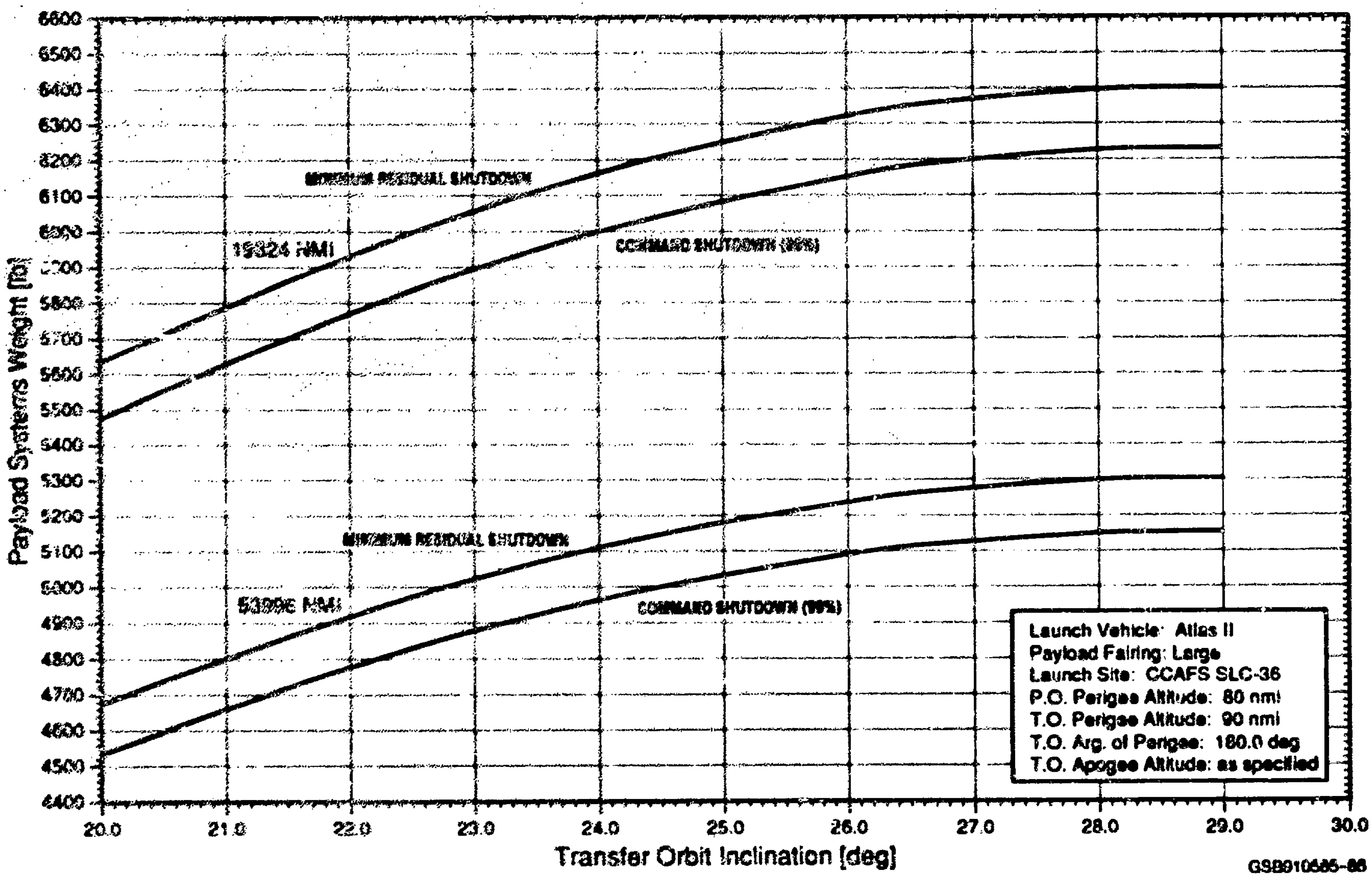

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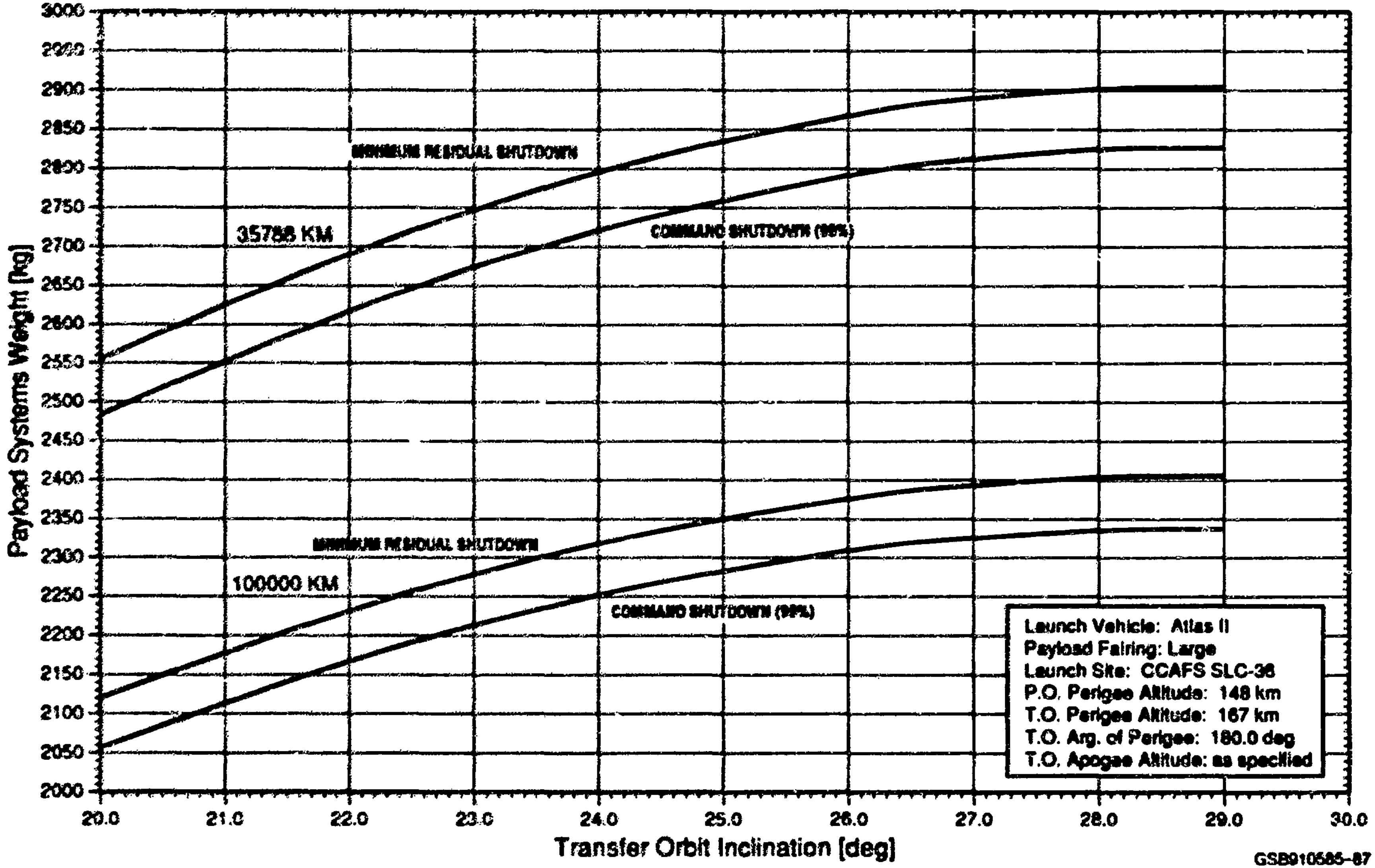
Figure 2-12a. Atlas II reduced inclination elliptical orbit performance.



gure 2-12b. *Atlas II reduced inclination elliptical orbit performance (metric).*

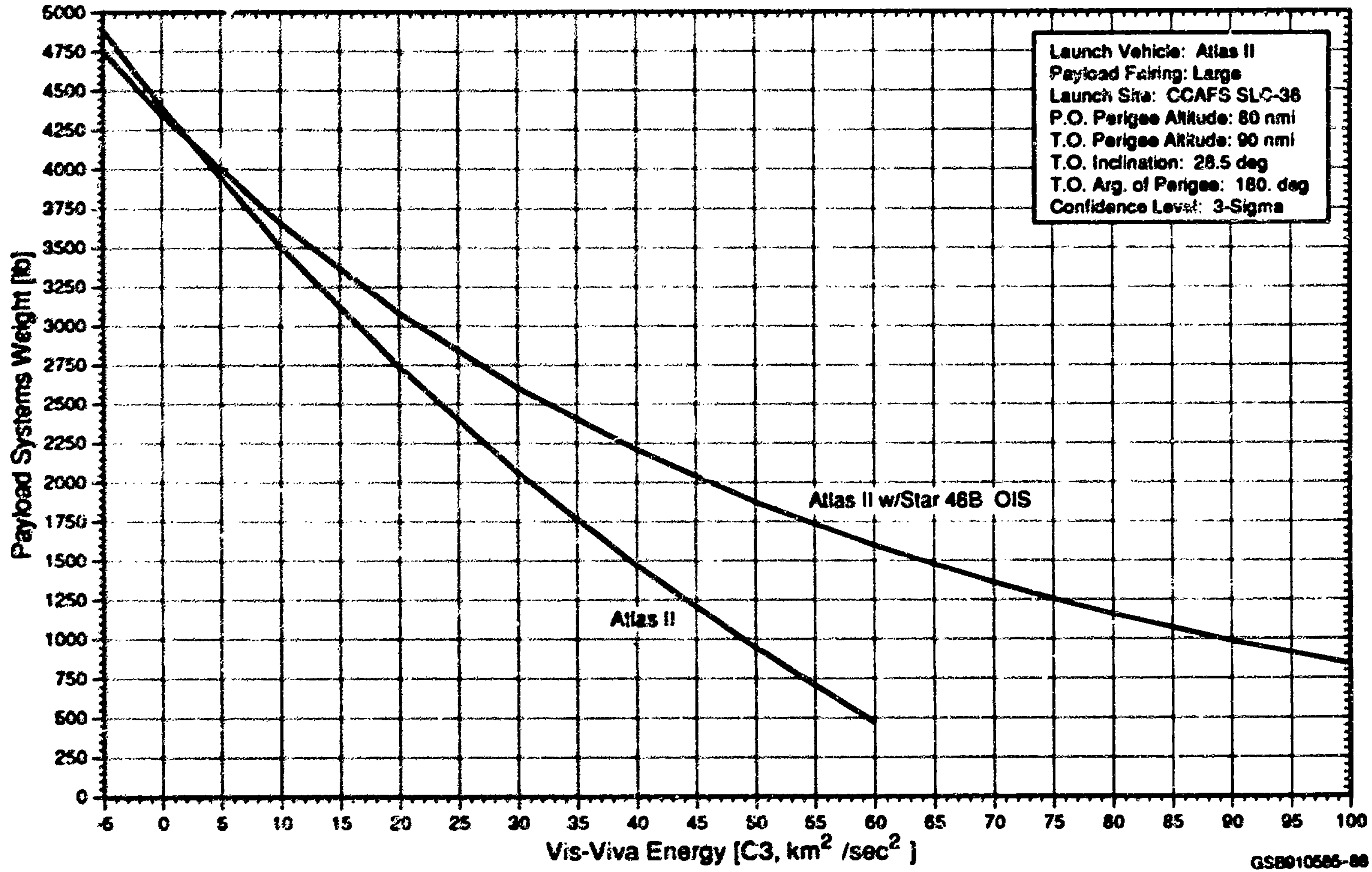


Figure 2-13a. Atlas II Earth-escape performance.

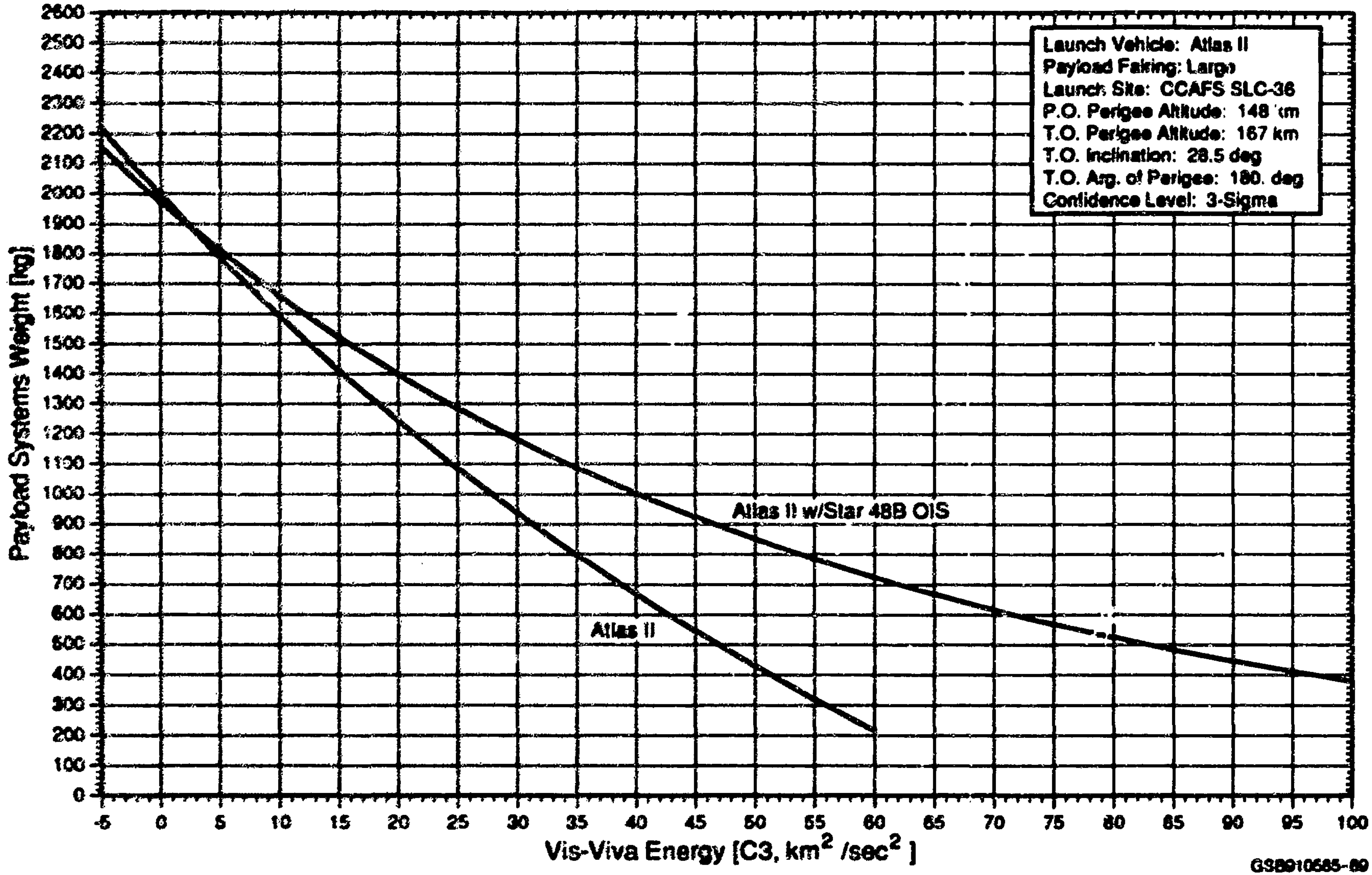


Figure 2-13b. Atlas II Earth-escape performance (metric).

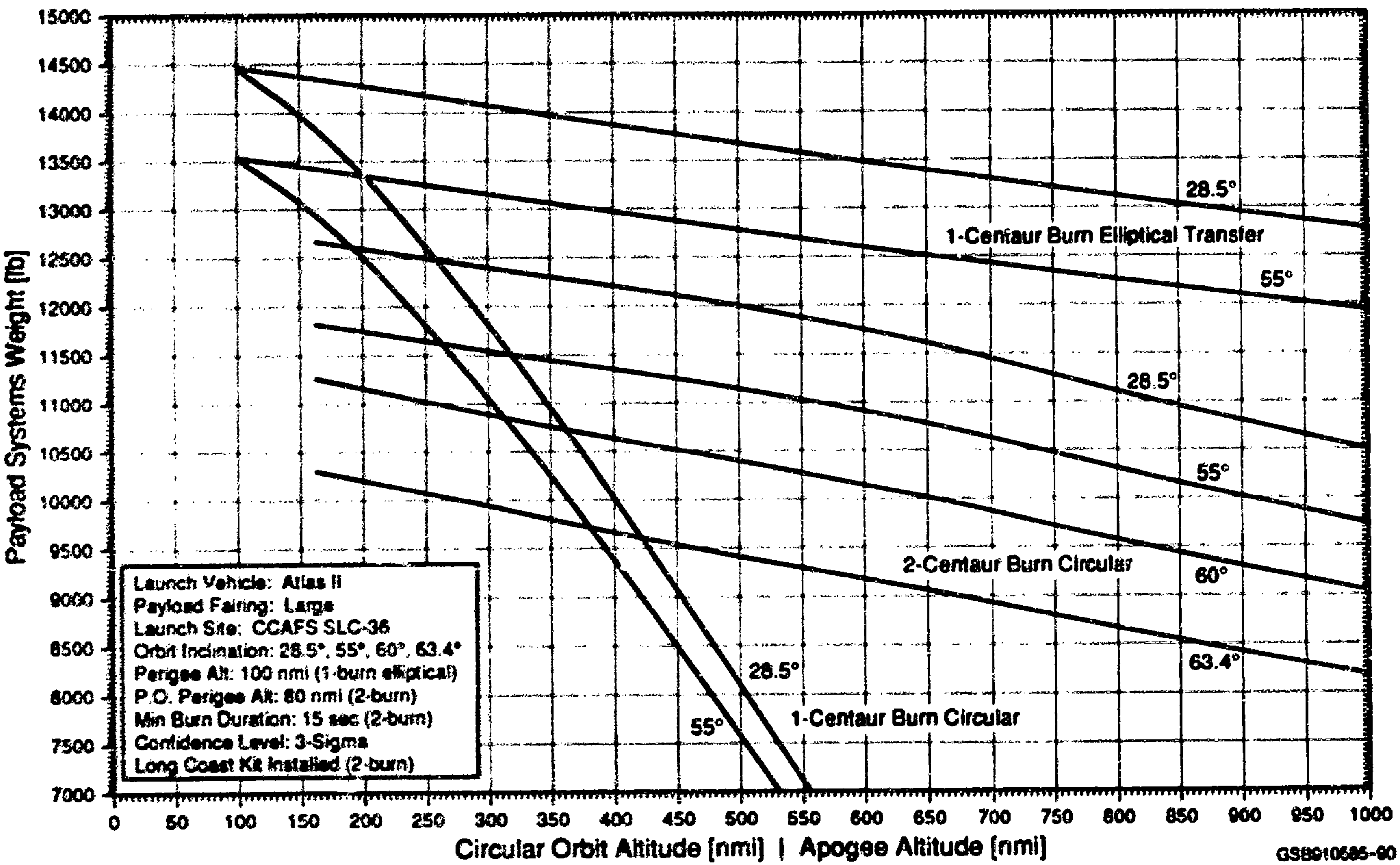


Figure 2-14a. Atlas II low Earth orbit performance.

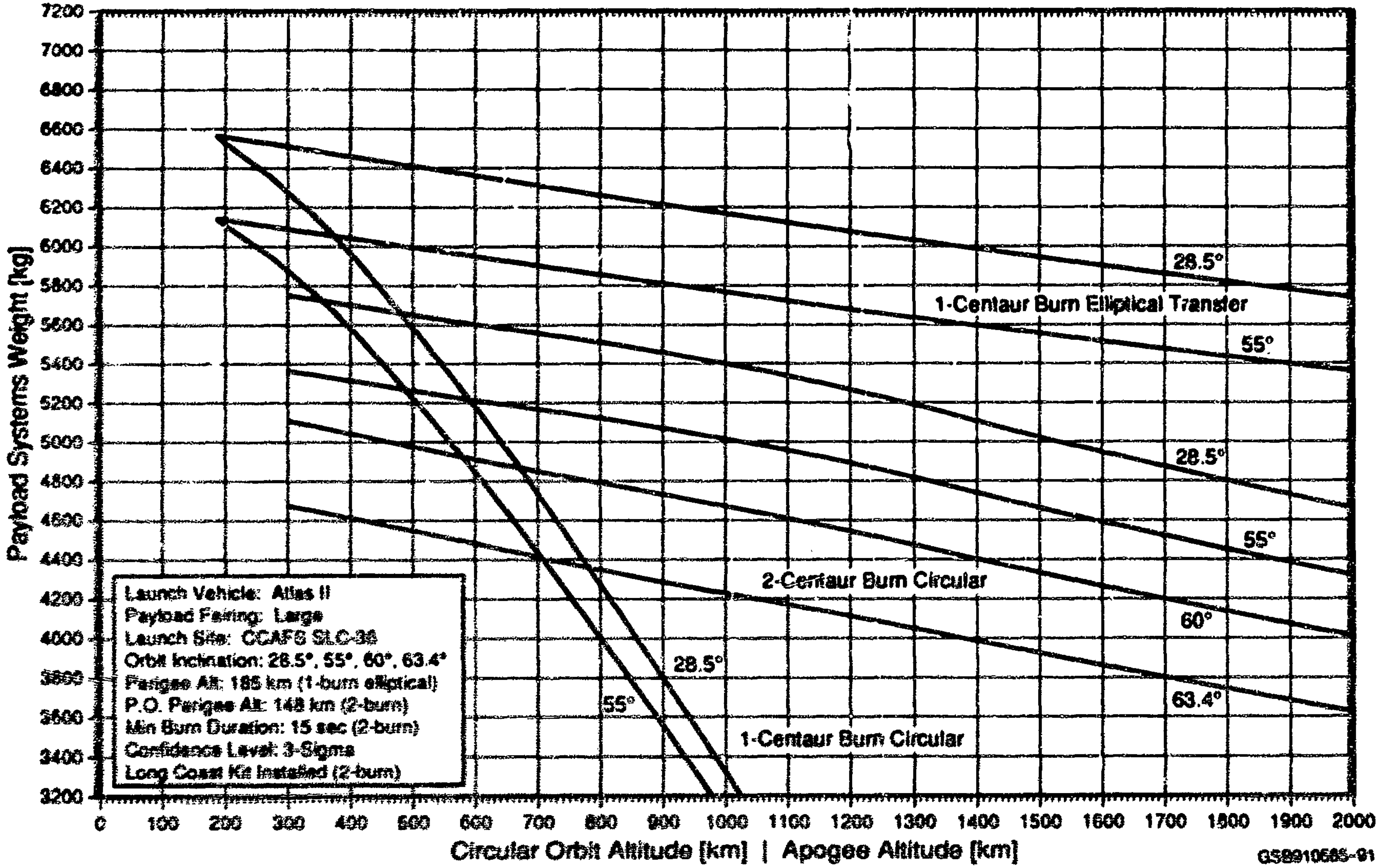


Figure 2-14b. Atlas II low Earth orbit performance (metric).

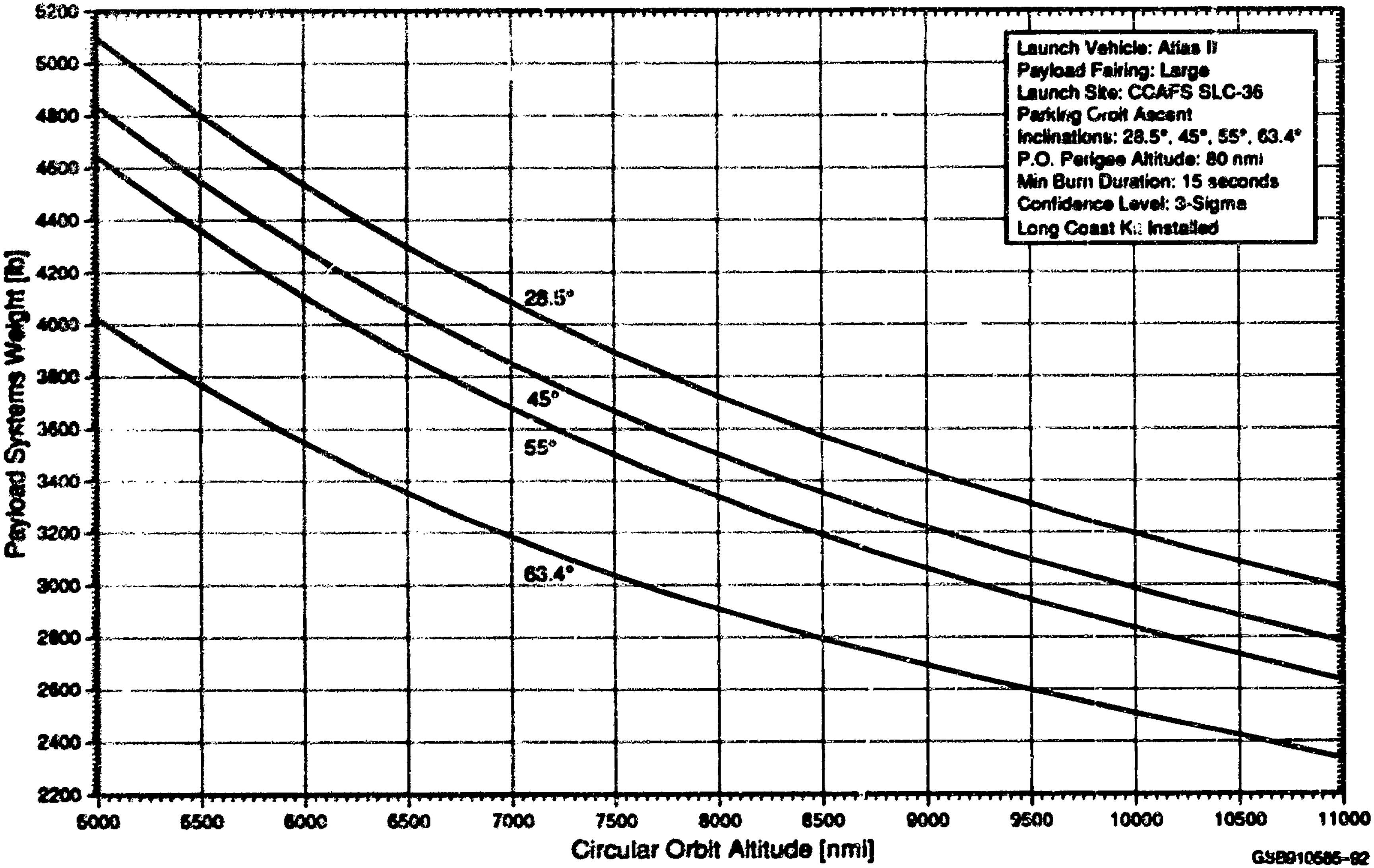


Figure 2-15a. Atlas II intermediate circular orbit performance.

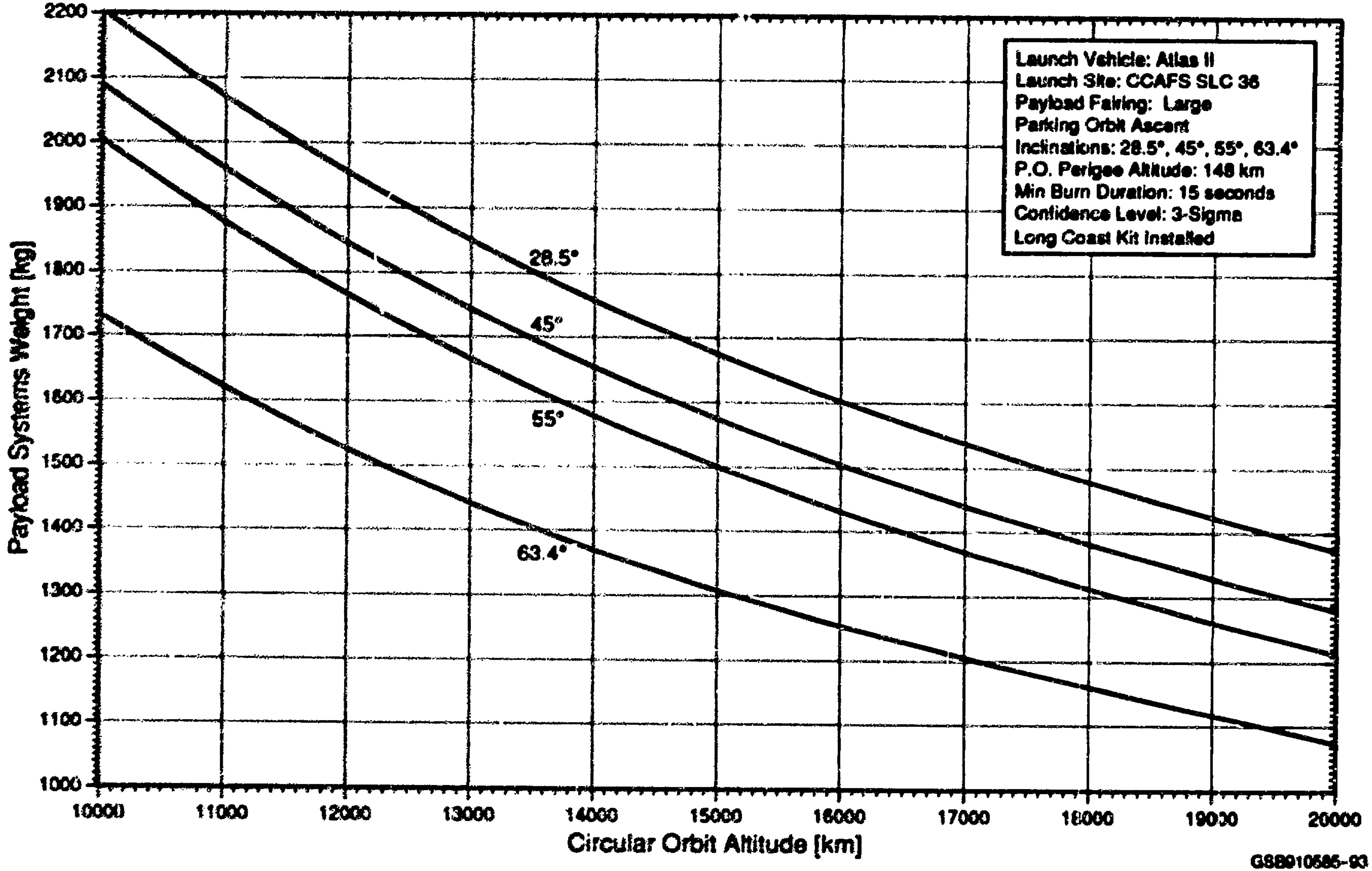


Figure 2-15b. Atlas II intermediate circular orbit performance (metric).

Table 2-11. Atlas II performance — PSW vs transfer orbit apogee altitude.

Apogee Altitude [nmi (km)]		Payload Systems Weight [lb (kg)]			
		MRS		99%	
80,994	(150000)	5,046	(2289)	4,900	(2223)
67,495	(125000)	5,139	(2331)	4,994	(2265)
53,996	(100000)	5,276	(2393)	5,127	(2325)
48,596	(90000)	5,353	(2428)	5,201	(2359)
43,197	(80000)	5,444	(2469)	5,291	(2400)
40,497	(75000)	5,498	(2494)	5,346	(2425)
37,797	(70000)	5,560	(2522)	5,407	(2453)
35,0[illegible]7	([illegible])	5,631	(2554)	5,474	(2483)
32,397	(60000)	5,712	(2591)	5,555	(2519)
31,048	(57500)	5,757	(2611)	5,599	(2539)
29,698	(55000)	5,805	(2633)	5,646	(2561)
28,348	(52500)	5,856	(2656)	5,698	(2585)
26,998	(50000)	5,915	(2683)	5,755	(2610)
25,648	(47500)	5,978	(2712)	5,815	(2637)
24,298	(45000)	6,046	(2743)	5,883	(2668)
22,948	(42[illegible]0)	6,122	(2777)	5,958	(2702)
21,598	(40000)	6,206	(2815)	6,041	(2740)
20,248	(37500)	6,300	(2857)	6,131	(2781)
19,324	(35788)	6,370	(2889)	6,200	(2812)
18,898	(35000)	6,404	(2905)	6,233	(2827)
[illegible]7,549	([illegible])	6,5[illegible]0	(2957)	6,349	(2880)
16,199	(30000)	6,653	(3018)	6,478	(2938)
14,849	(27500)	6,806	(3087)	6,629	(3007)
13,[illegible]	([illegible])	6,983	(3167)	6,802	(3085)
12,149	(22500)	7,185	(3259)	7,002	(3176)
10,799	(20000)	7,431	(3371)	7,244	(3286)
9,449	(17500)	7,722	(3503)	7,530	(3416)
8,099	(15000)	8,083	(3666)	7,881	(3575)
6,749	(12500)	8,532	(3870)	8,325	(3776)
5,400	(10000)	9,109	(4132)	8,893	(4034)

80 nmi (148 km) perigee parking orbit; 90 nmi (167 km) perigee transfer orbit; 27-degree inclination; 180-degree argument of perigee; large payload fairing

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ATLAS IIA PERFORMANCE

- Elliptical transfer orbit
- Reduced-inclination elliptical orbit
- Earth escape
- Low Earth orbit
- Intermediate circular orbit
- PSW versus transfer orbit apogee altitude

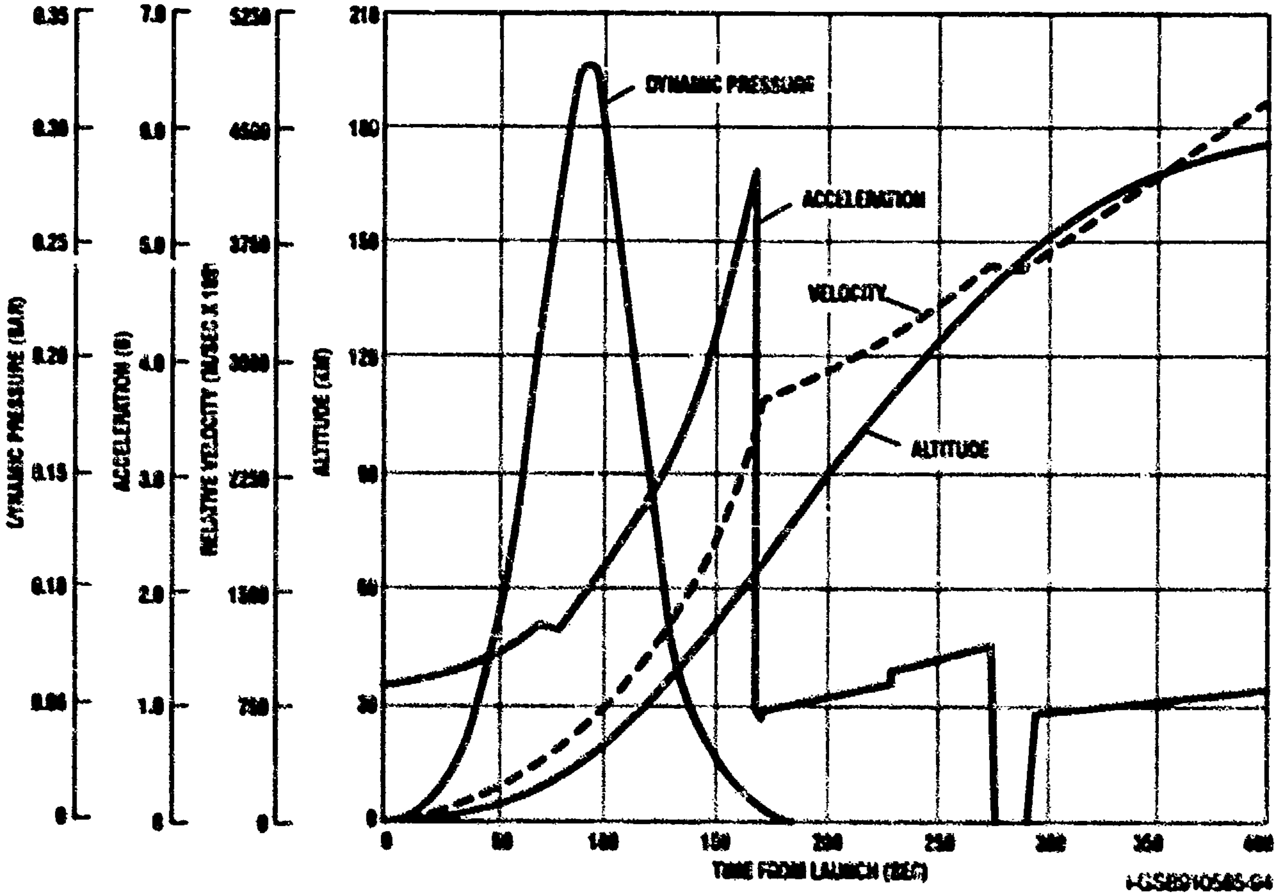

Figure 2-16. Atlas IIA nominal ascent data.

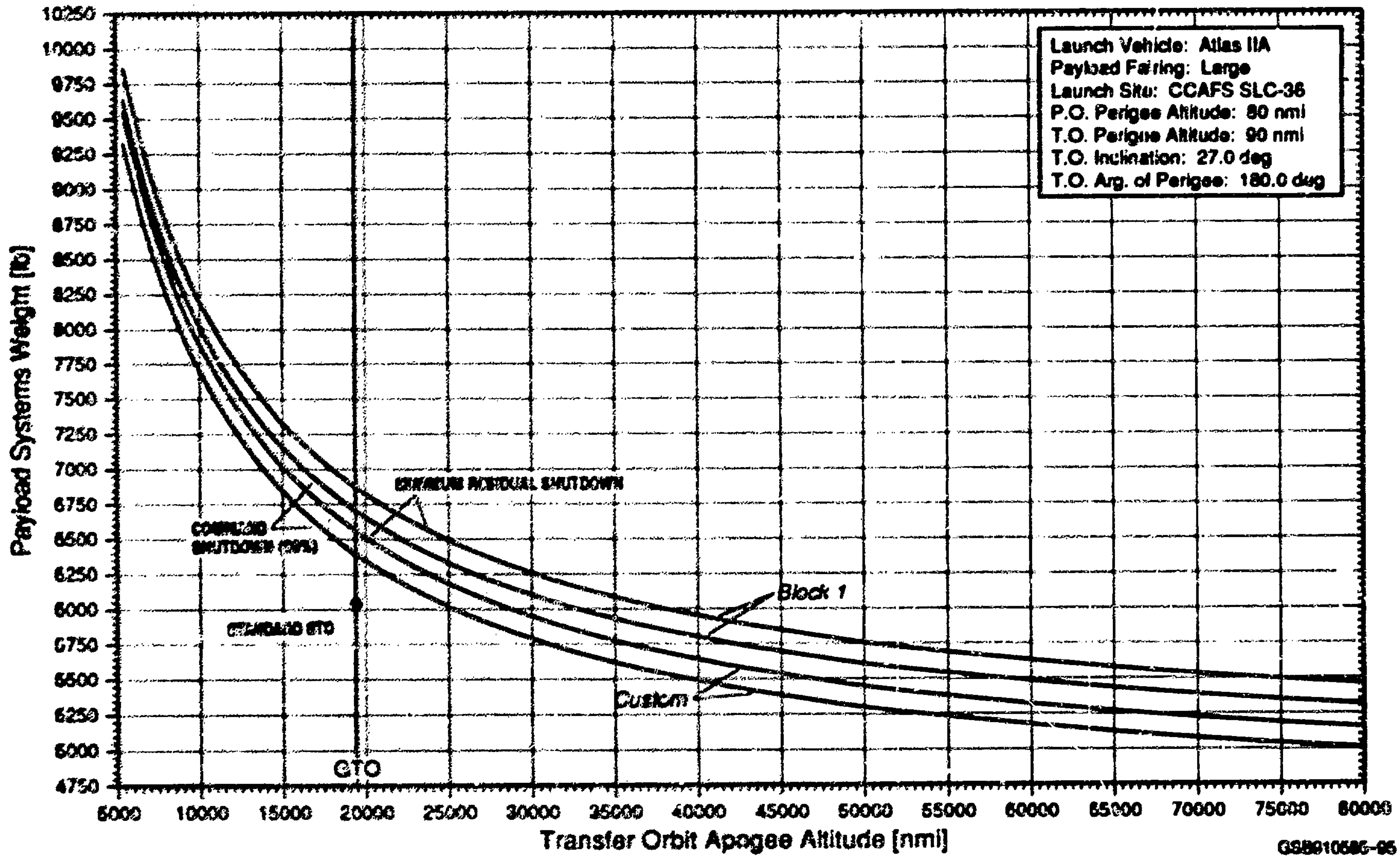

Figure 2-17a. Atlas IIA performance to elliptical transfer orbit.

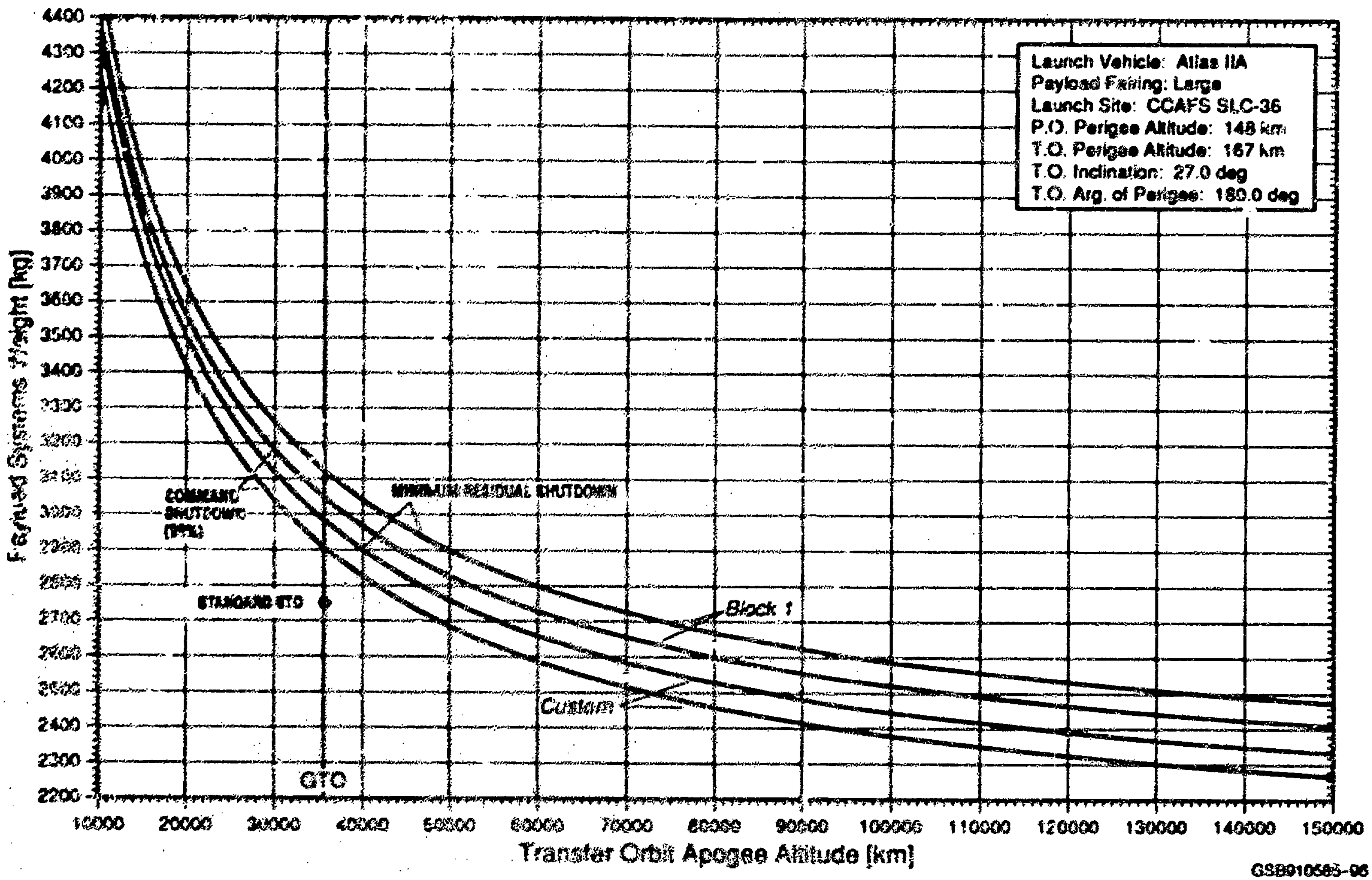

Figure 2-17b Atlas IIA performance to elliptical transfer orbit (metric).

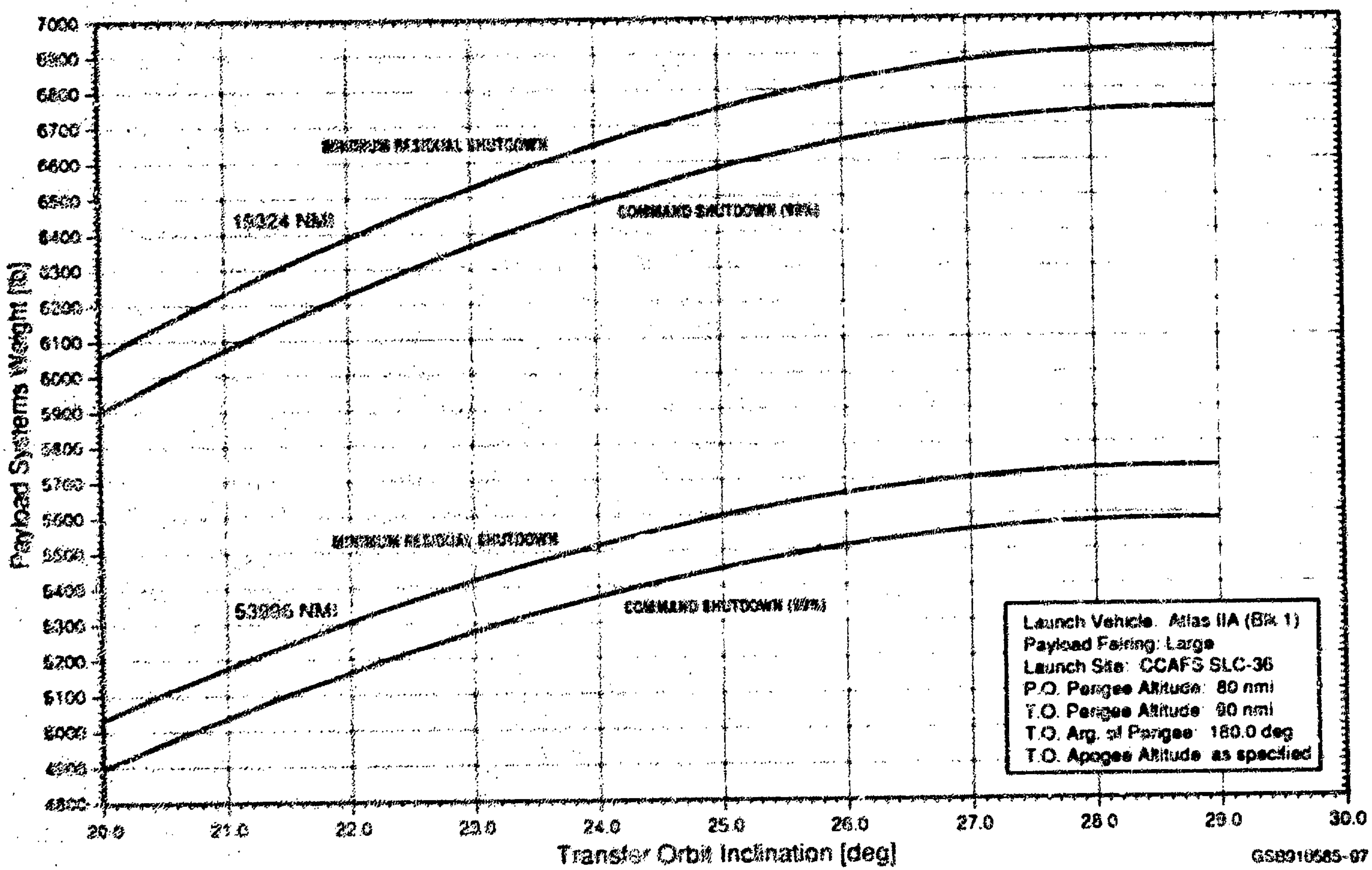

Figure 2-17a. Atlas IIA reduced inclination elliptical orbit performance

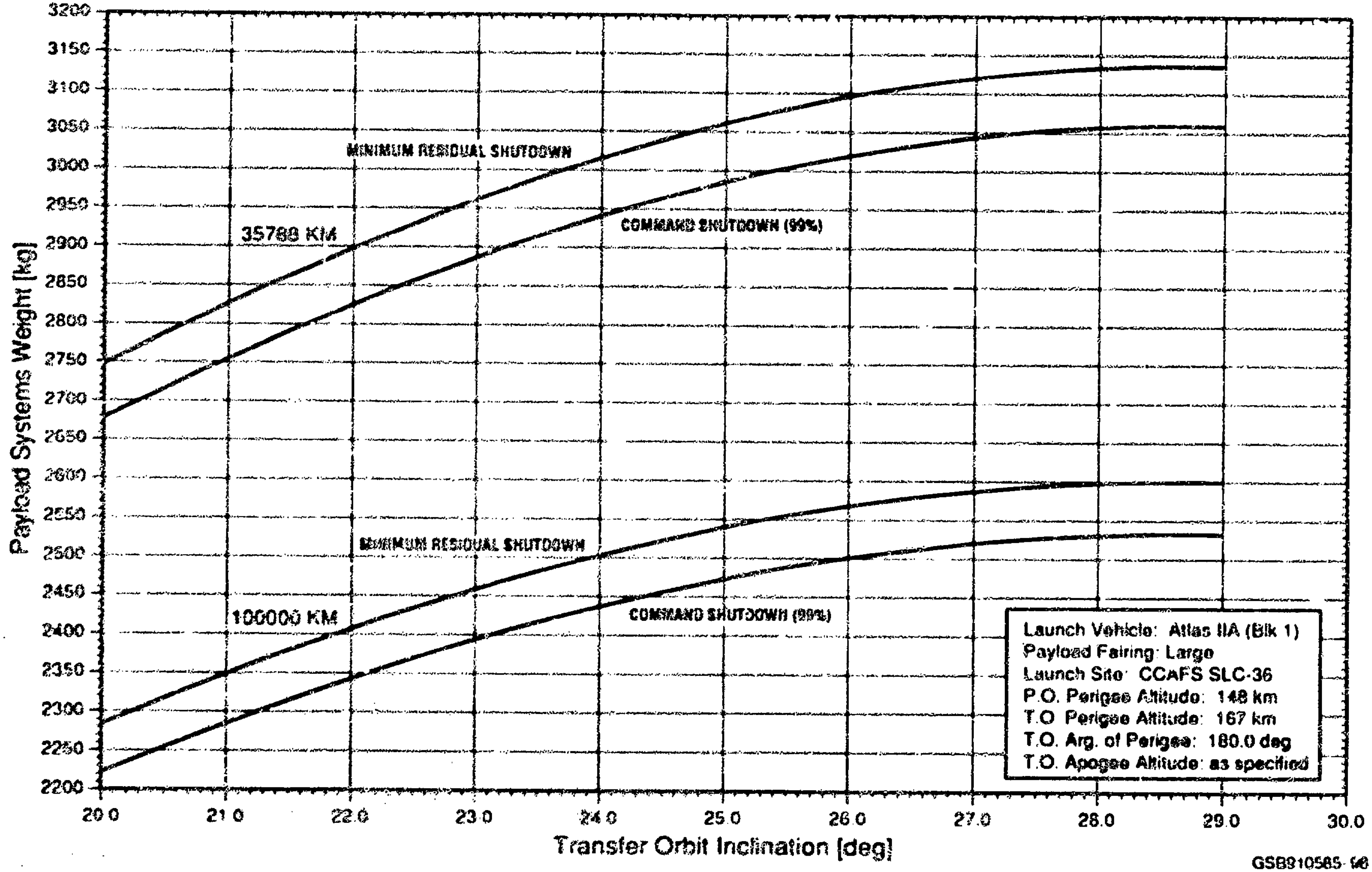

Figure 2-18b Atlas IIA reduced inclination elliptical orbit performance (metric).

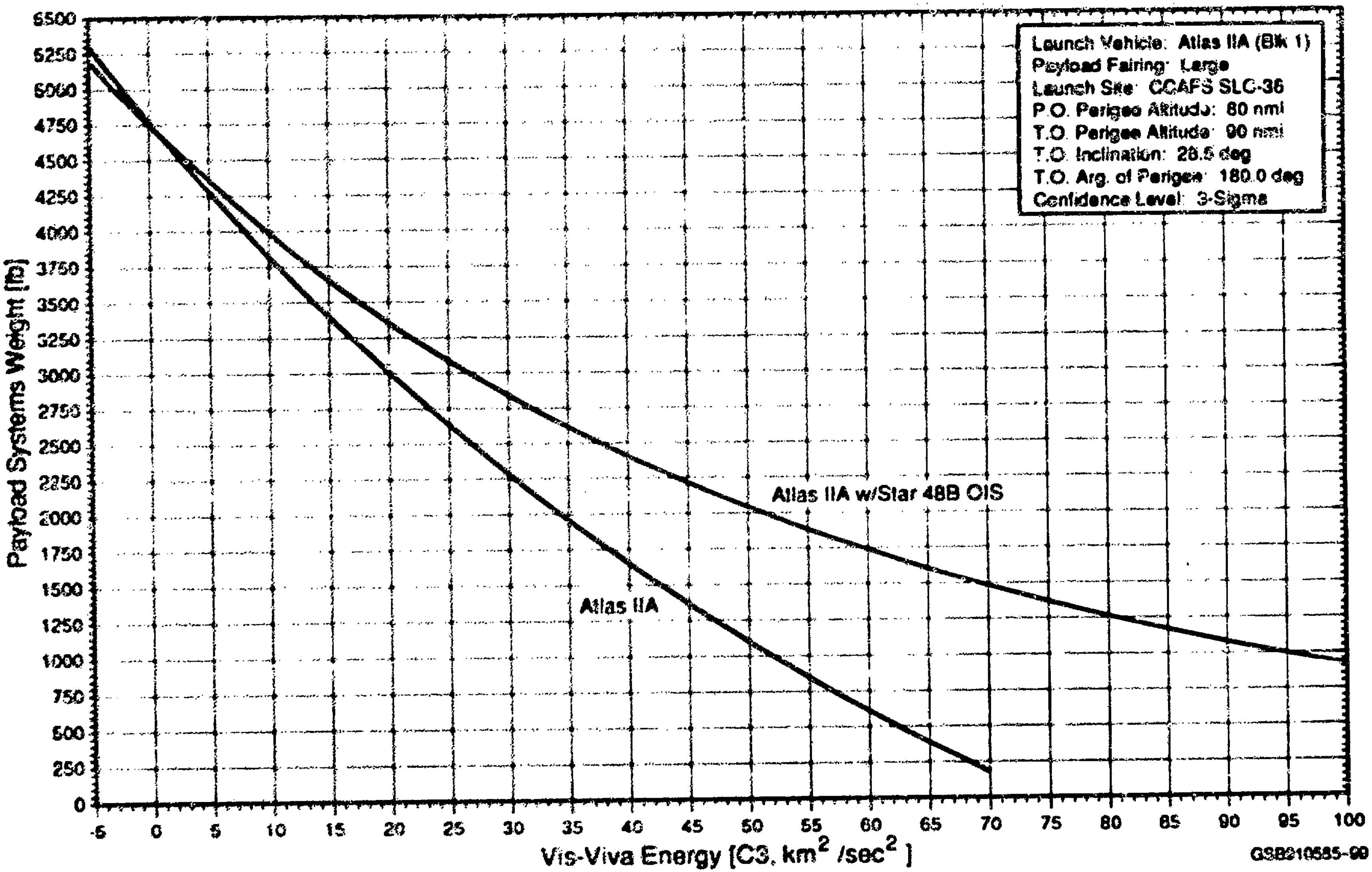

Figure 2-19a. Atlas IIA Earth-escape performance.

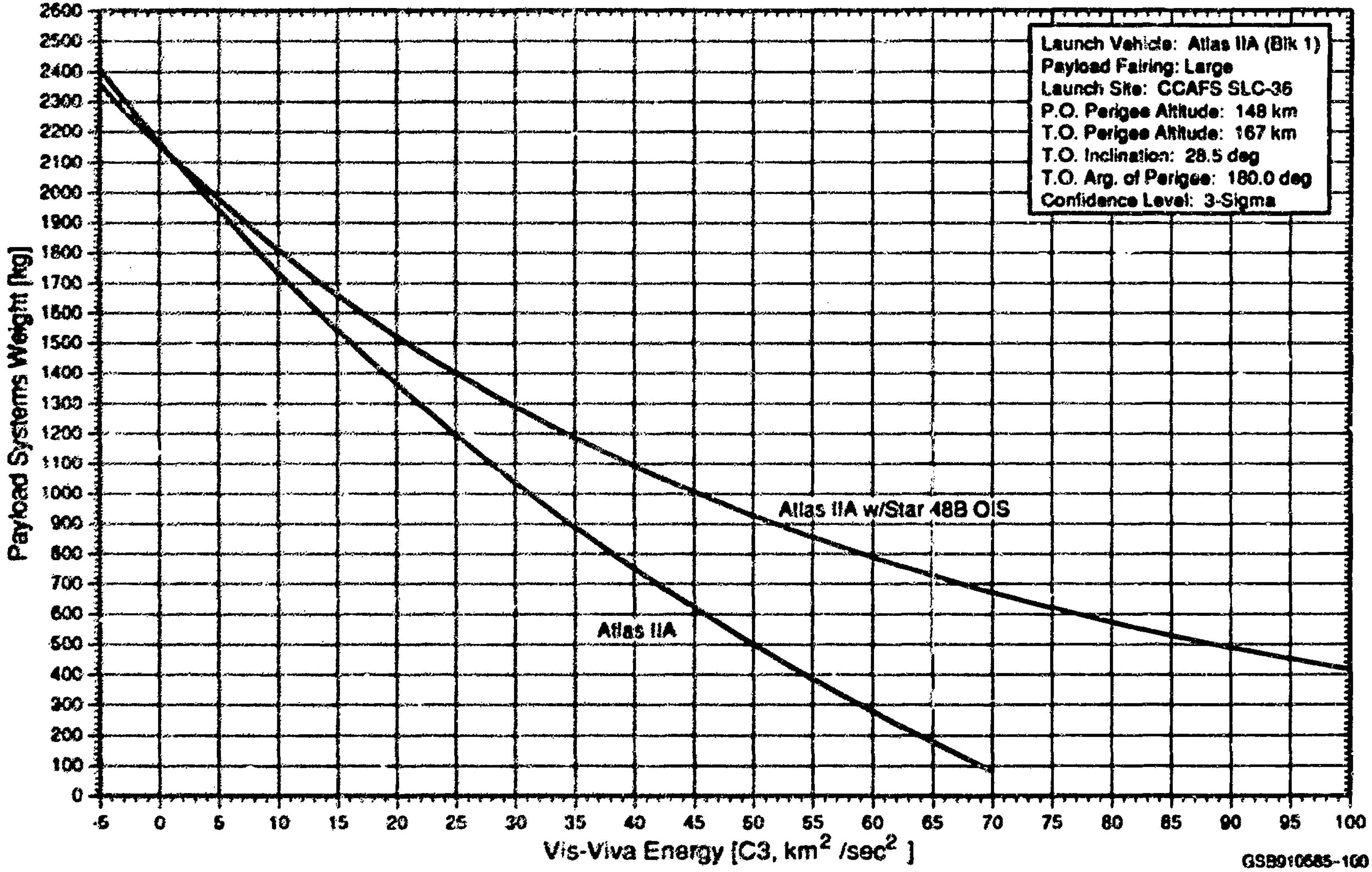

Figure 2-19b. Atlas IIA Earth-escape performance (metric).

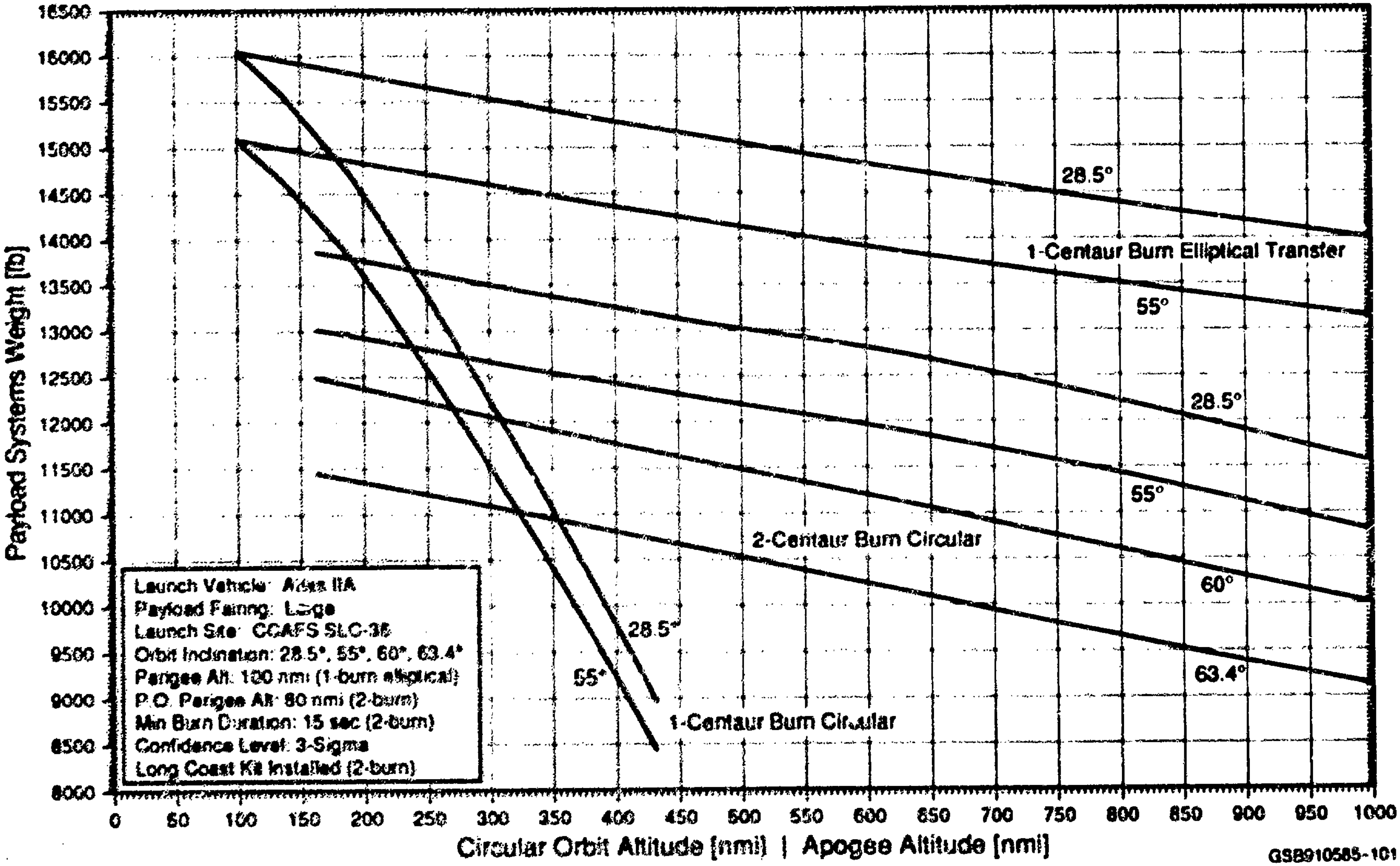

Figure 2-20a. Atlas IIA low Earth orbit performance.

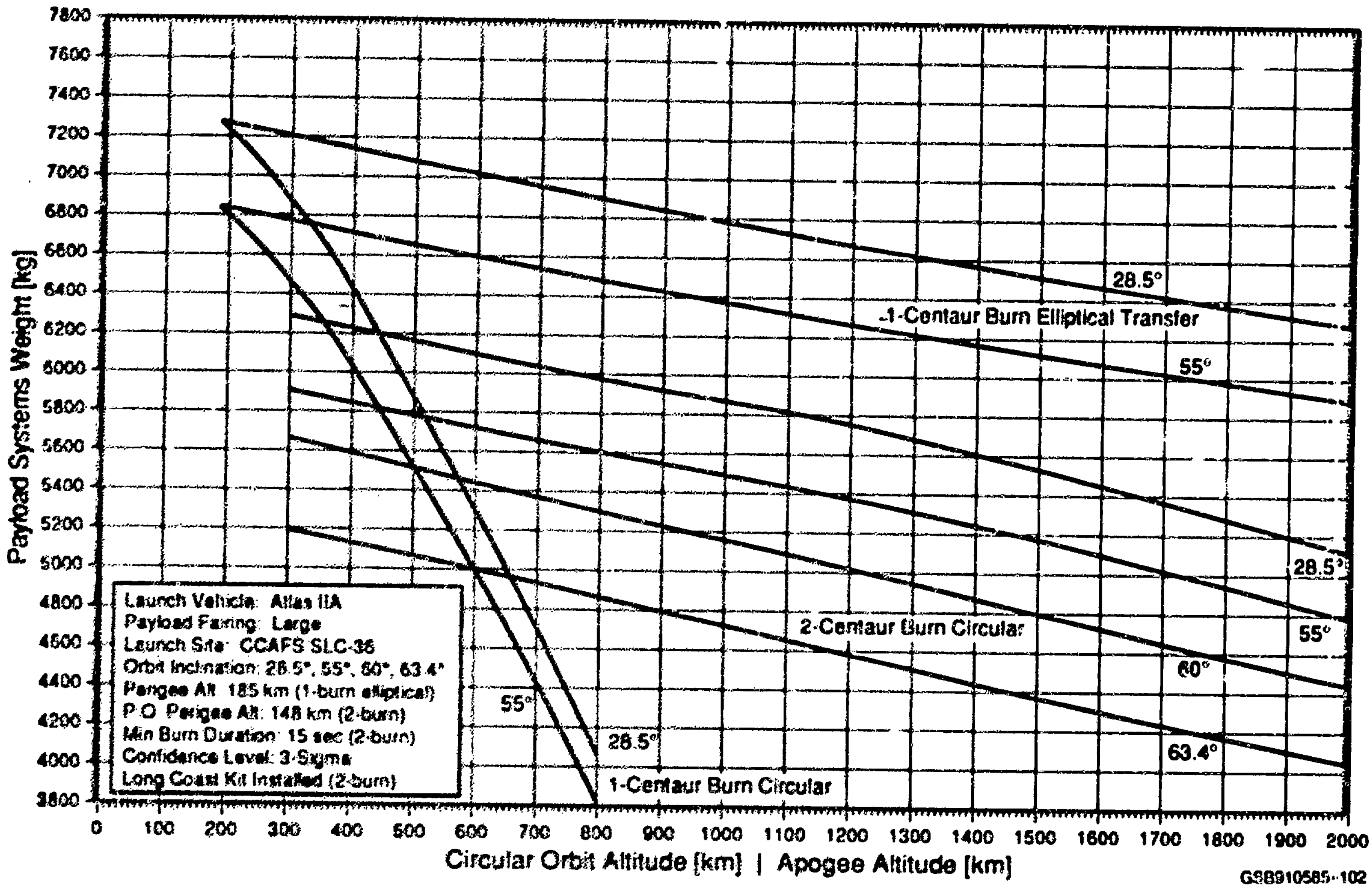

Figure 2-20b. Atlas IIA low Earth orbit performance (metric).

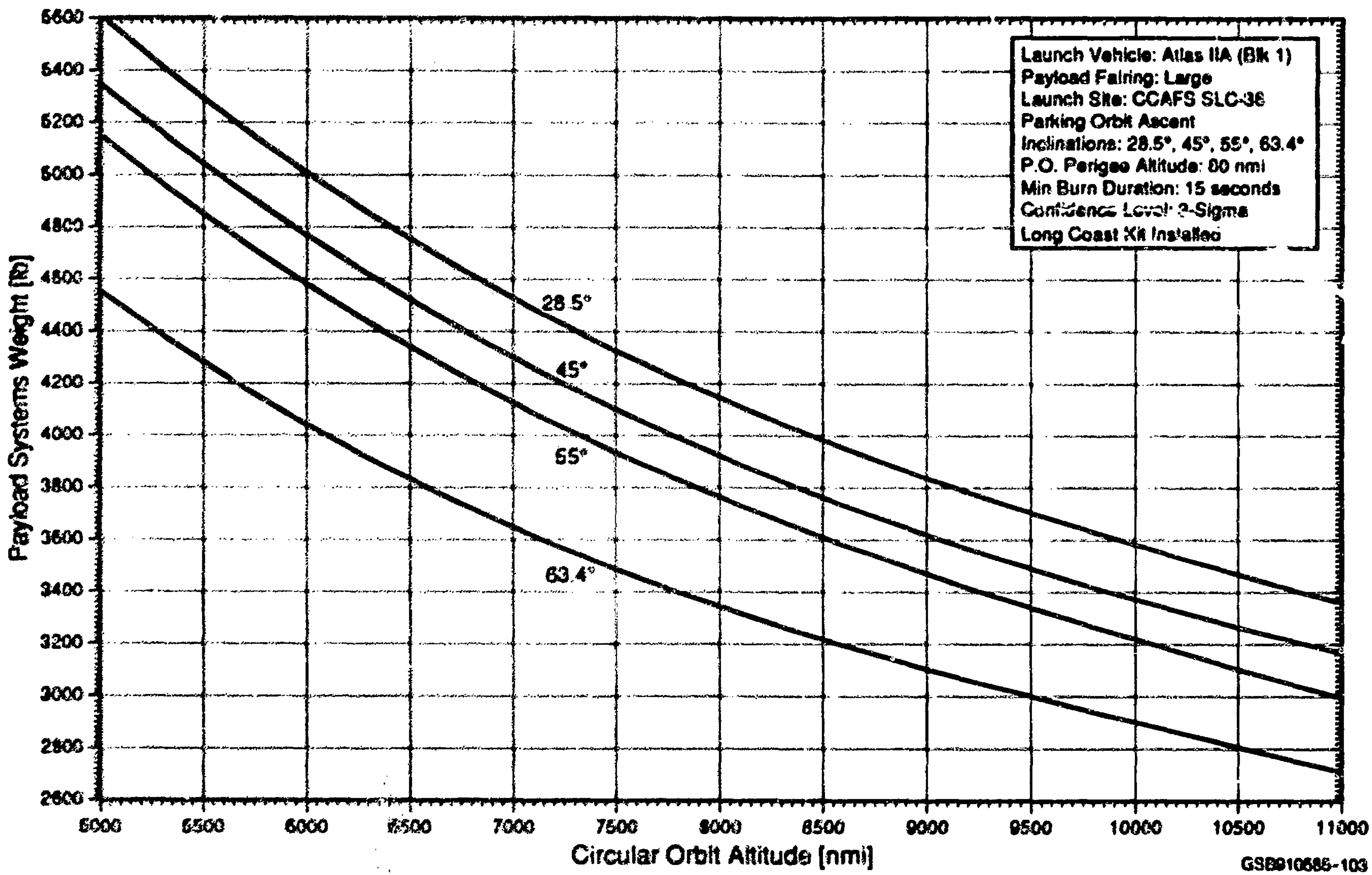

Figure 2-21a. Atlas IIA intermediate circular orbit performance.

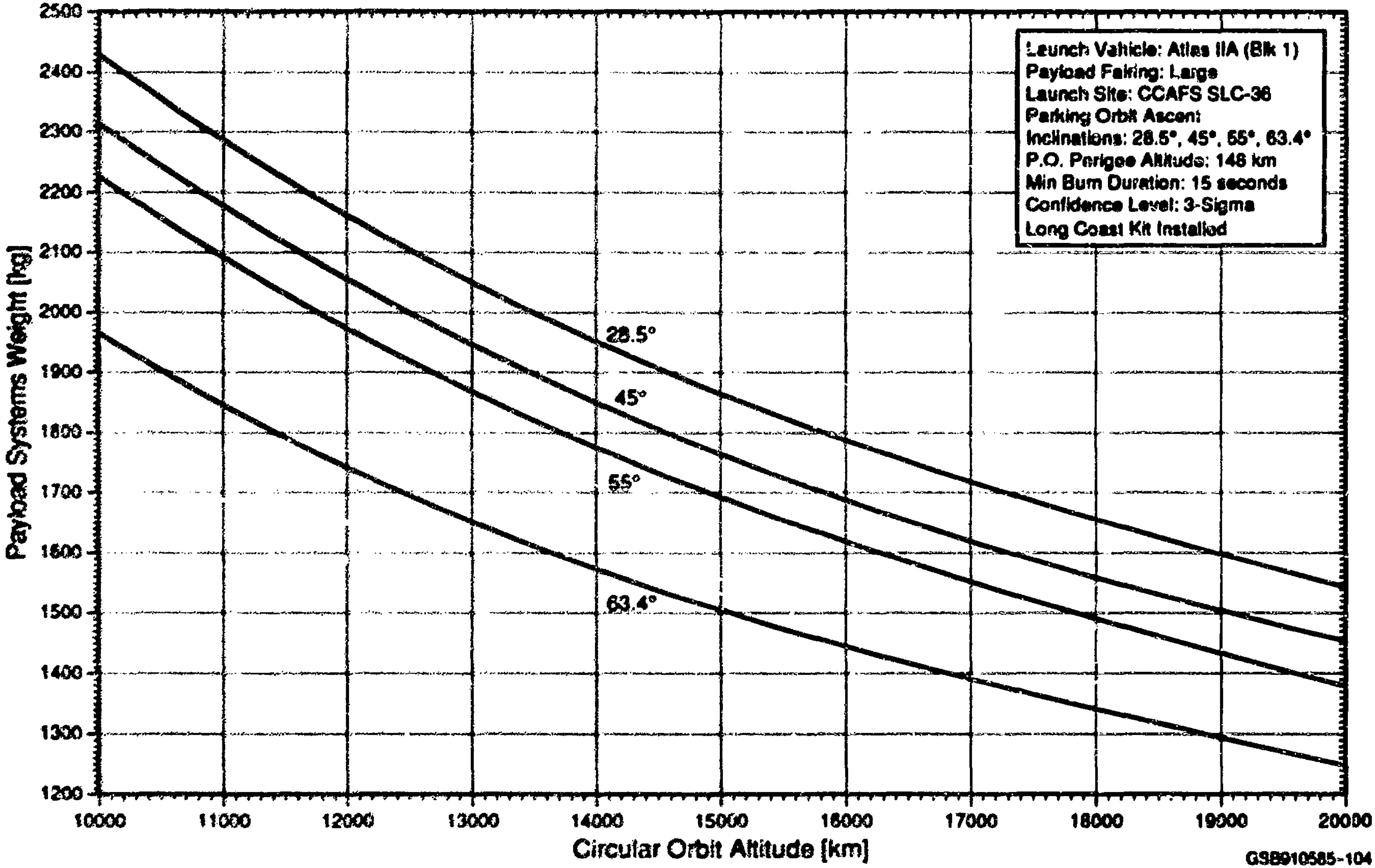

Figure 2-21b Atlas IIA intermediate circular orbit performance (metric).

Table 2-12. Atlas IIA (Block 1) performance — PSW vs transfer orbit apogee altitude.

Apogee Altitude [nmi (km)]		Payload Systems Weight [lb (kg)]			
		MRS		99%	
80,984	(150 000)	5,454	(2474)	5,310	(2409)
67,495	(125 000)	5,555	(2520)	5,409	(2453)
53,996	(100 000)	5,702	(2586)	5,555	(2520)
48,596	(90 000)	5,783	(2623)	5,634	(2555)
43,197	(80 000)	[illegible]	[illegible]	5,730	(2599)
40,497	(75 000)	5,942	(2695)	5,789	(2626)
37,797	(70 000)	6,007	(2725)	5,854	(2655)
35,097	(65 000)	6,083	(2759)	5,929	(2689)
32,397	(60 000)	6,170	(2799)	6,012	(2727)
31,048	(57 500)	6,219	(2821)	6,061	(2749)
29,698	(55 000)	6,272	(2845)	6,113	(2773)
28,348	(52 500)	6,327	(2870)	6,169	(2798)
26,998	(50 000)	6,389	(2898)	6,230	(2826)
25,648	(47 500)	6,457	(2929)	6,296	(2856)
24,298	(45 000)	6,532	(2963)	6,368	(2888)
22,948	(42 500)	6,613	(2999)	6,449	(2925)
21,598	(40 000)	6,703	(3040)	6,537	(2965)
20,248	(37 500)	6,804	(3086)	6,635	(3010)
19,324	(35 786)	6,880	(3121)	6,710	(3044)
18,898	(35 000)	6,916	(3137)	6,746	(3060)
17,549	(32 500)	7,043	(3195)	6,871	(3117)
16,199	(30 000)	7,188	(3260)	7,008	(3179)
14,849	(27 500)	7,345	(3332)	7,174	(3254)
13,499	(25 000)	7,543	(3421)	7,361	(3339)
12,149	(22 500)	7,766	(3523)	7,580	(3438)
10,799	(20 000)	8,030	(3642)	7,836	(3554)
9,449	(17 500)	8,340	(3783)	8,150	(3697)
8,099	(15 000)	8,732	(3961)	8,532	(3870)
6,749	(12 500)	9,226	(4185)	9,016	(4089)
5,400	(10 000)	9,862	(4473)	9,636	(4371)

80 nmi (148 km) perigee parking orbit; 90 nmi (167 km) perigee transfer orbit; 27-degree inclination; 180-degree argument of perigee; large payload fairing

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ATLAS IIAS PERFORMANCE

- Elliptical transfer orbit
- Reduced-inclination elliptical orbit
- Earth escape
- Low Earth orbit
- Intermediate circular orbit
- PSW versus transfer orbit apogee altitude

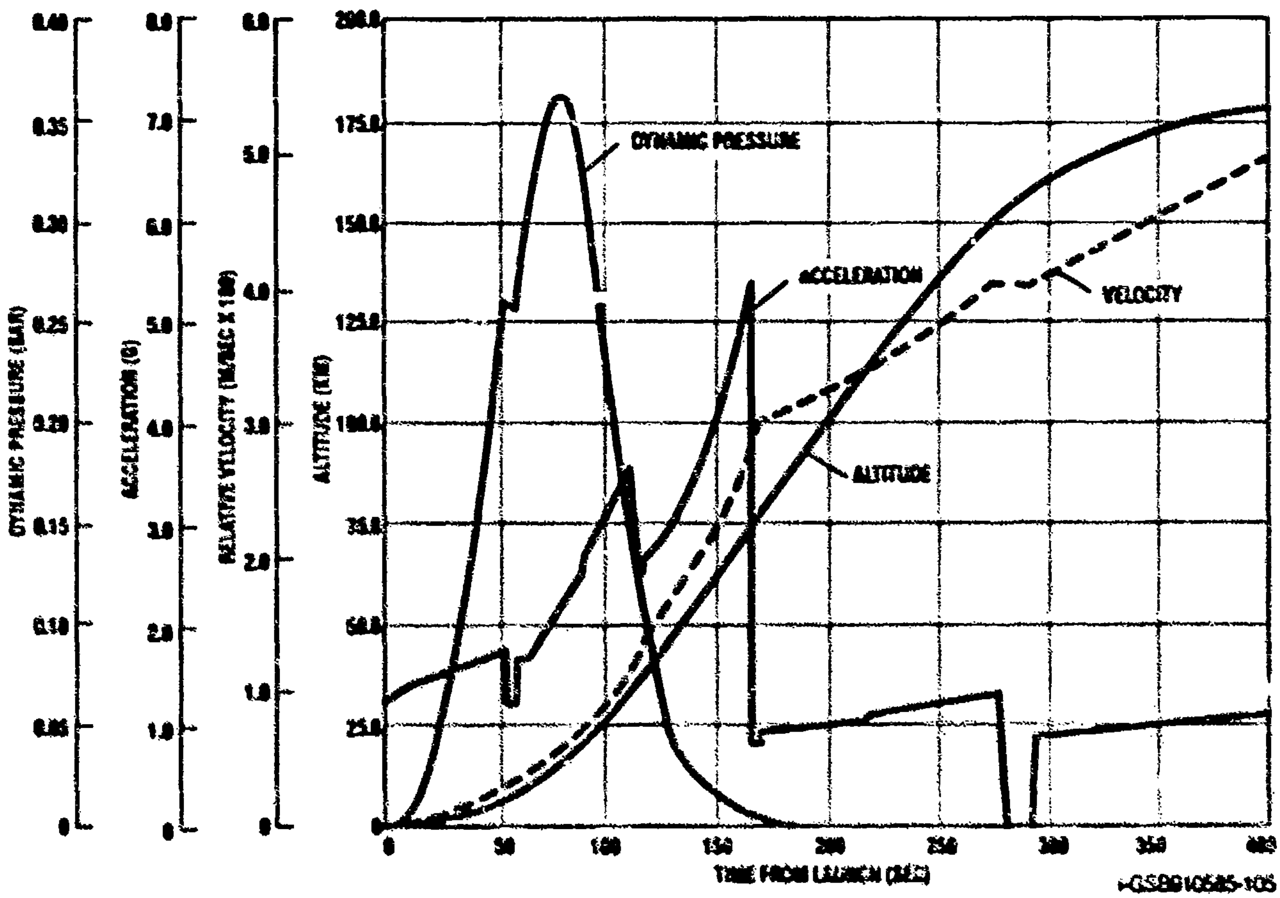

Figure 2-22. Atlas IIAS nominal ascent data.

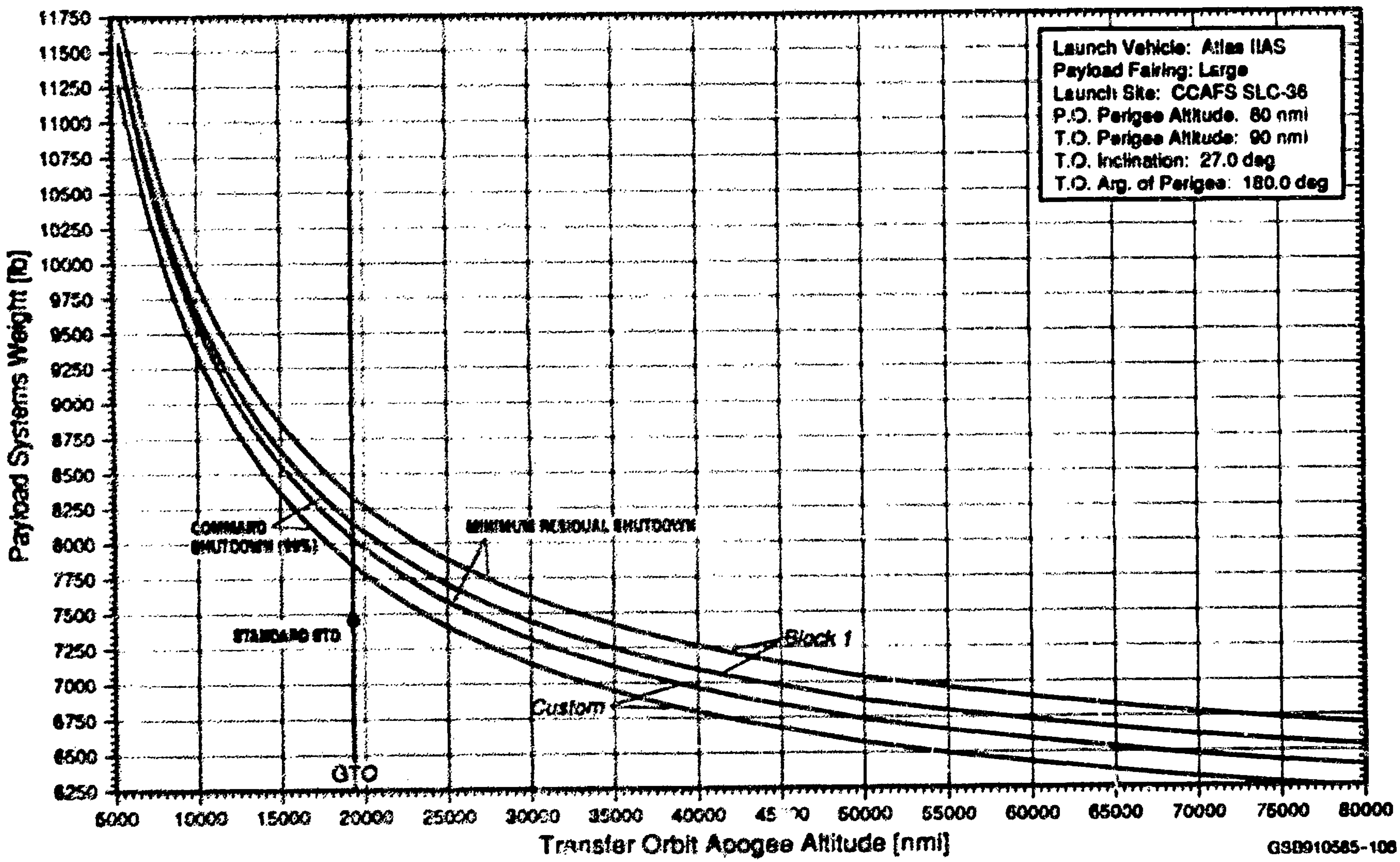

Figure 2-23a. Atlas IIAS performance to elliptical transfer orbit.

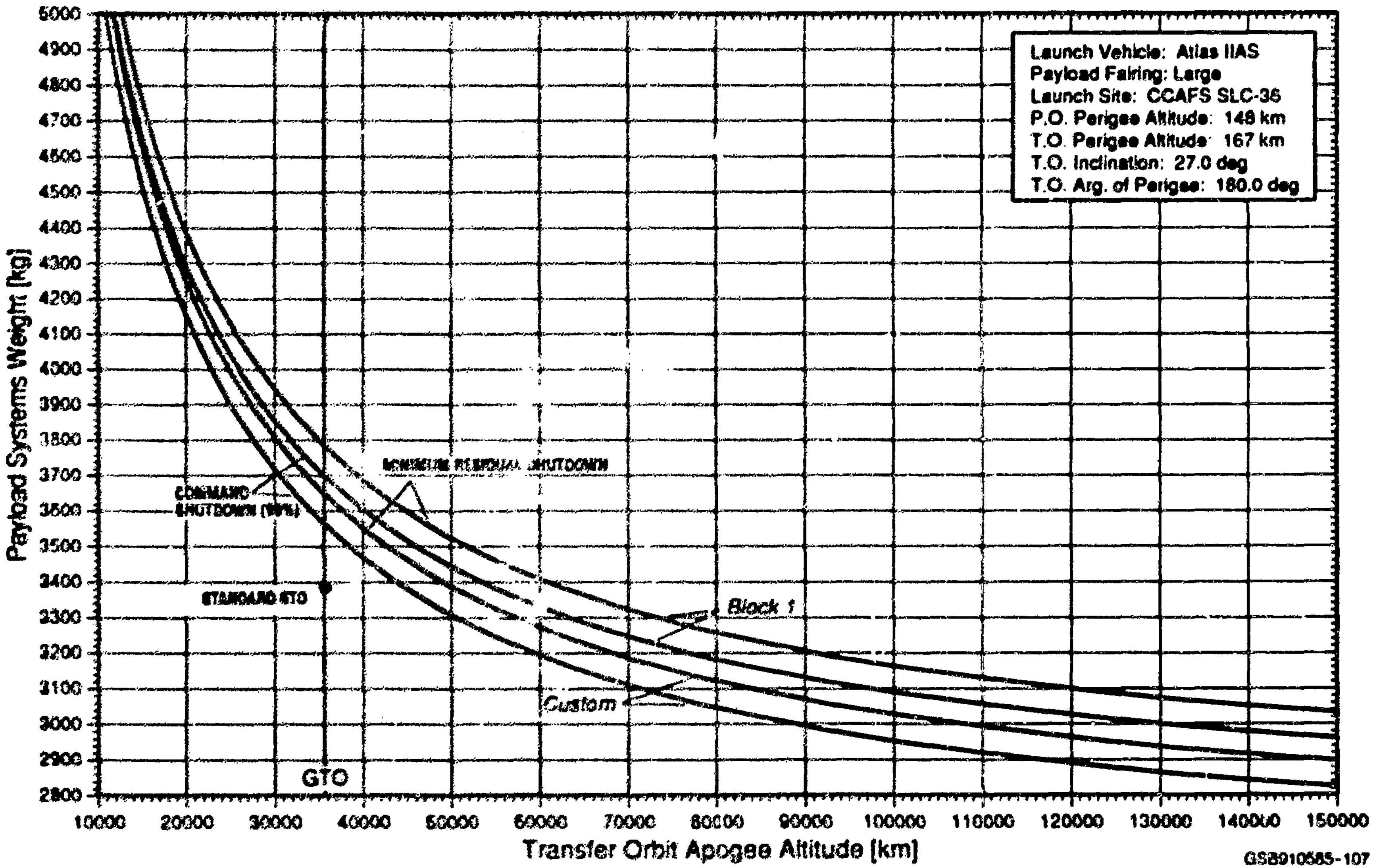

Figure 2-23b. Atlas IIAS performance to elliptical transfer orbit (metric)

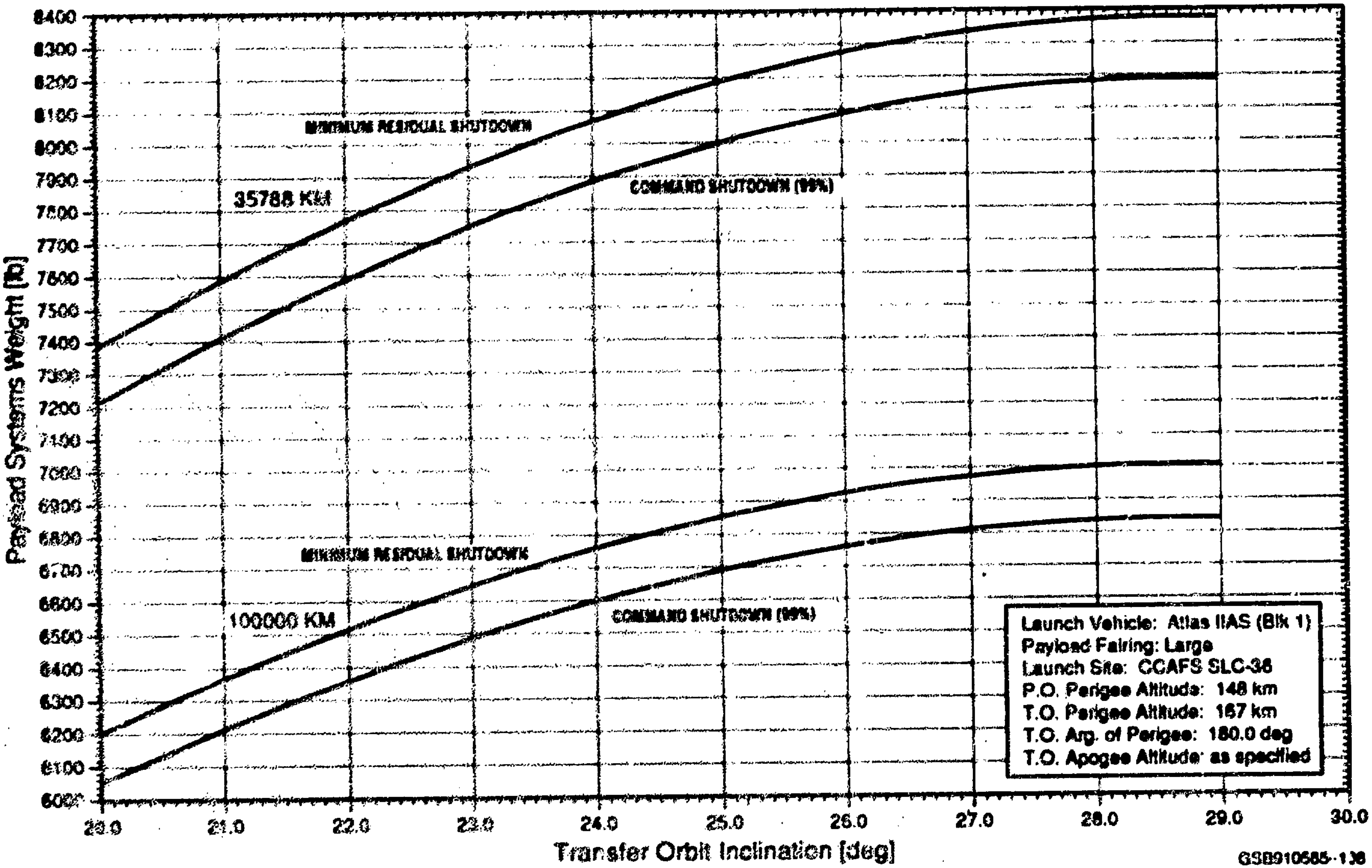

Figure 2-24a. Atlas IIAS reduced inclination elliptical orbit performance.

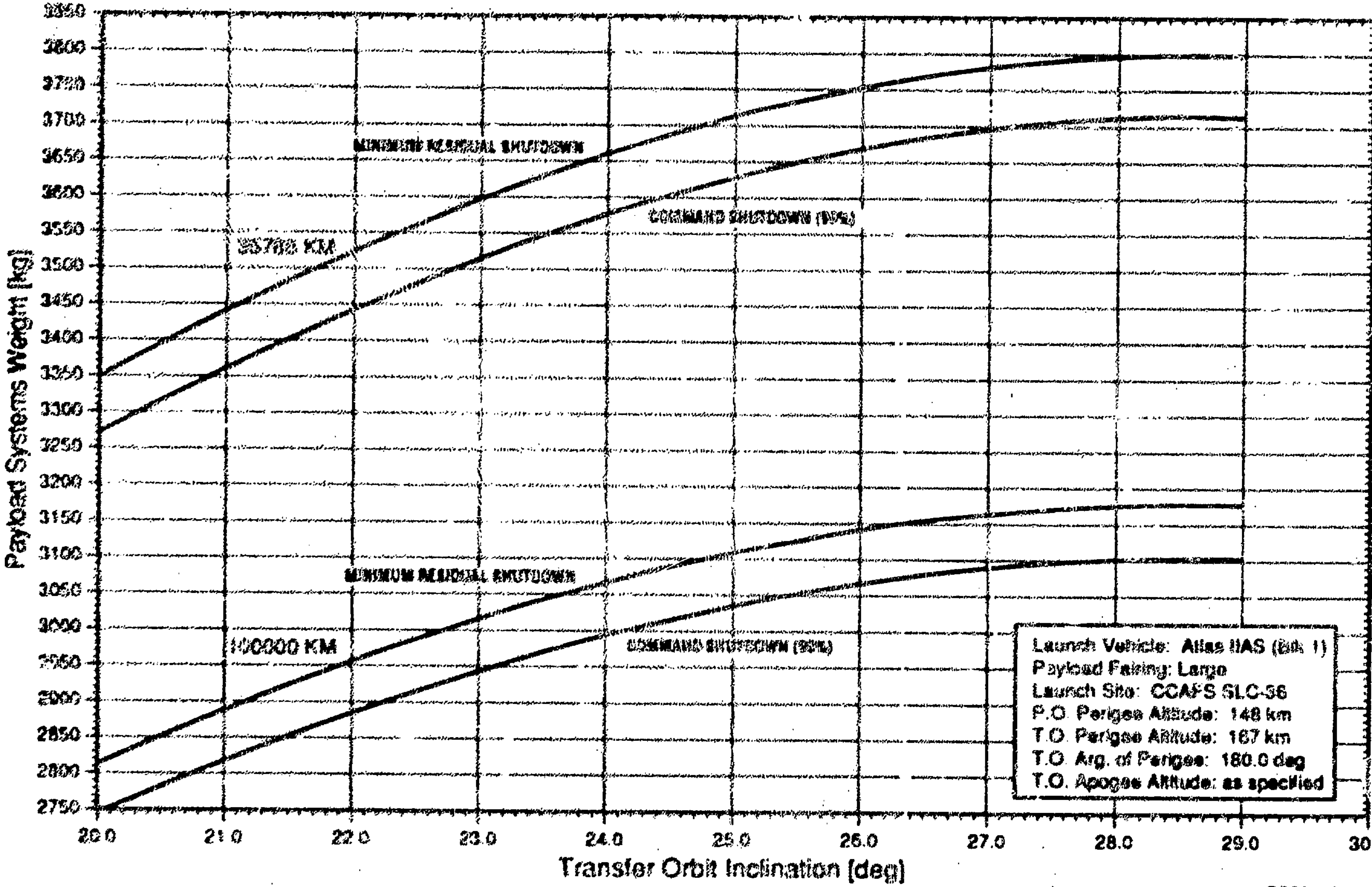

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Figure 2-24b. Atlas IIAS reduced inclination elliptical orbit performance (metric)

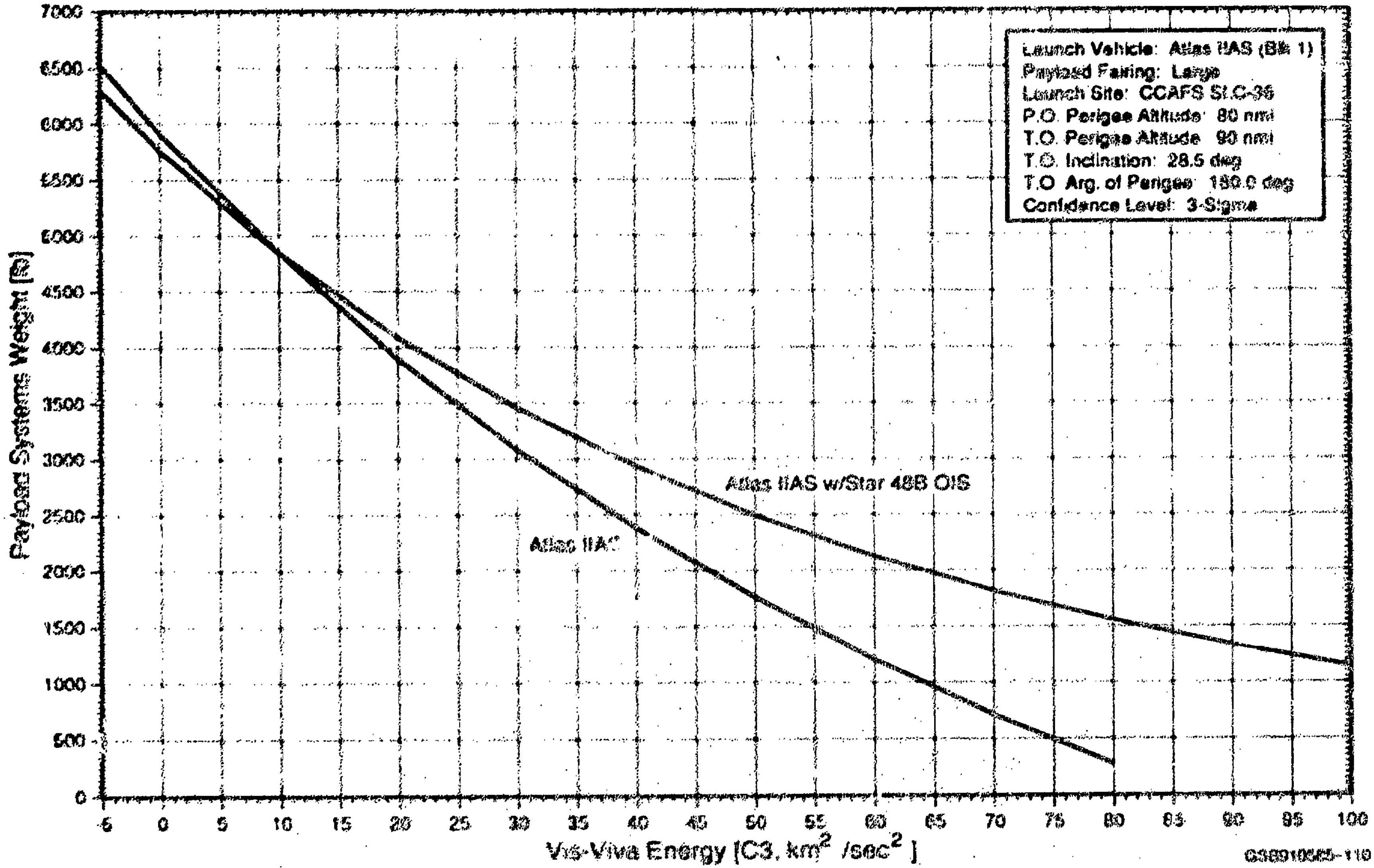

Figure 2-25a. Atlas IIAS Earth-escape performance.

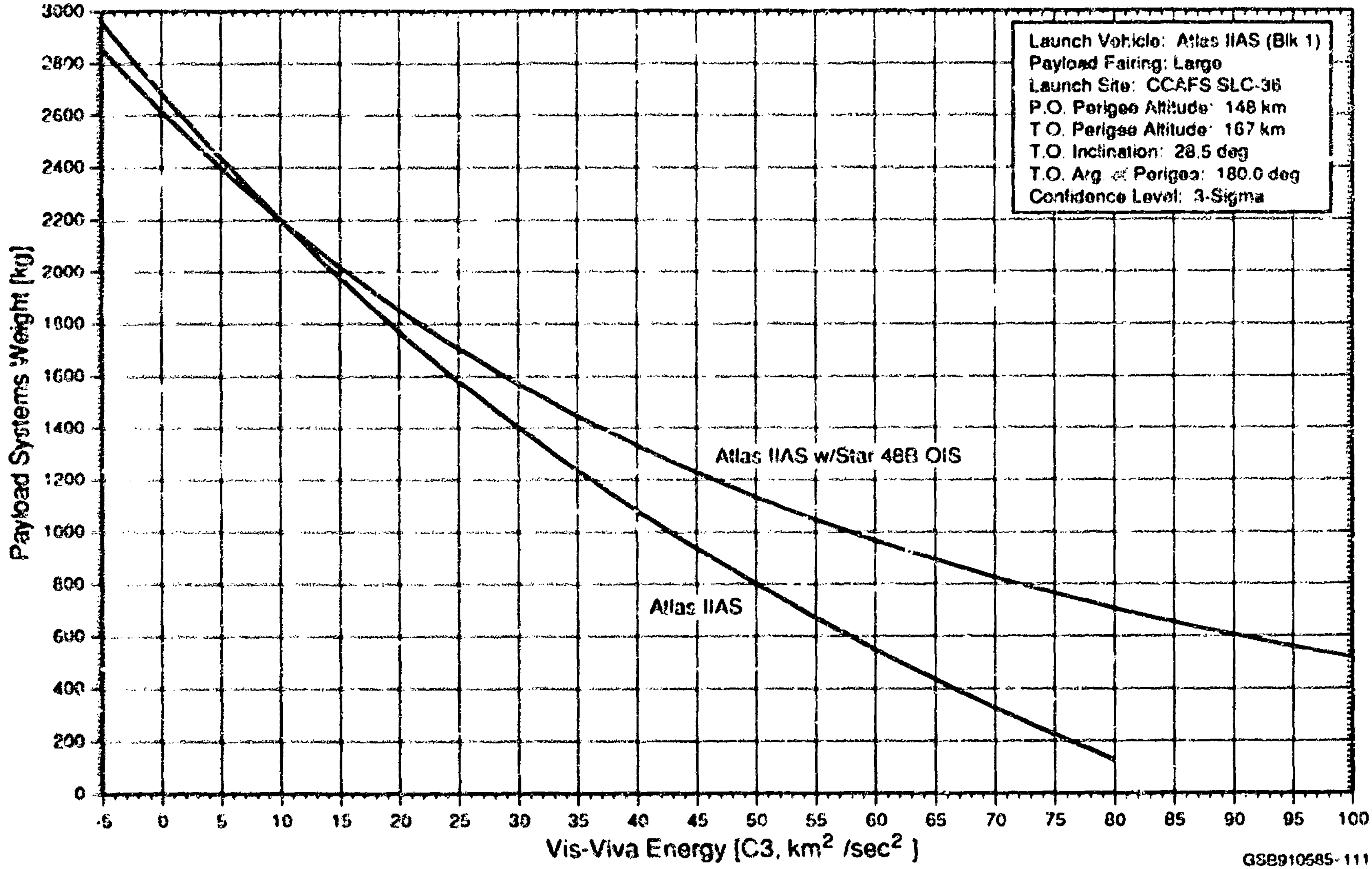

Figure 2-25b Atlas IIAS Earth-escape performance (metric)

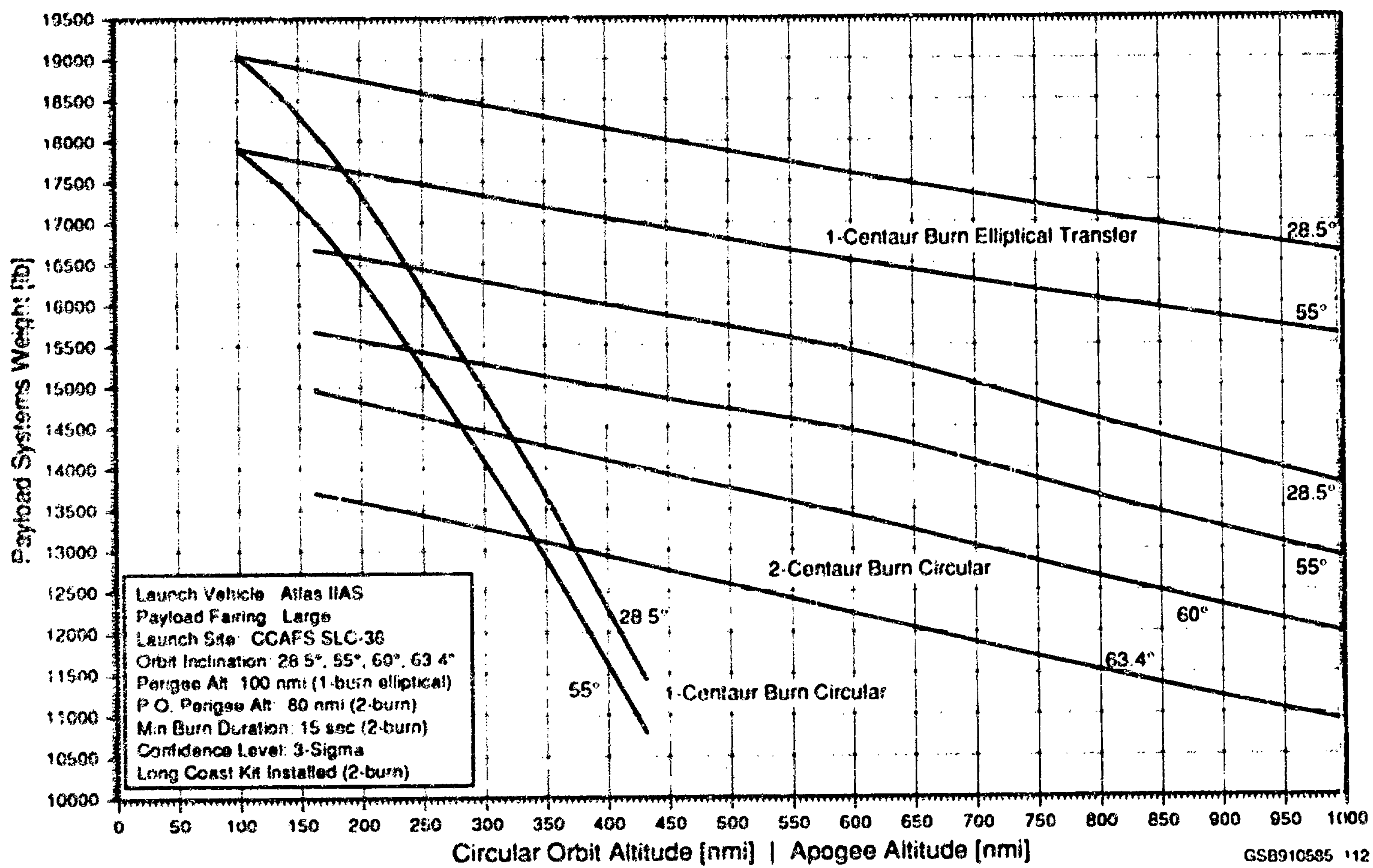

Figure 2-26a. Atlas IIAS low Earth orbit performance.

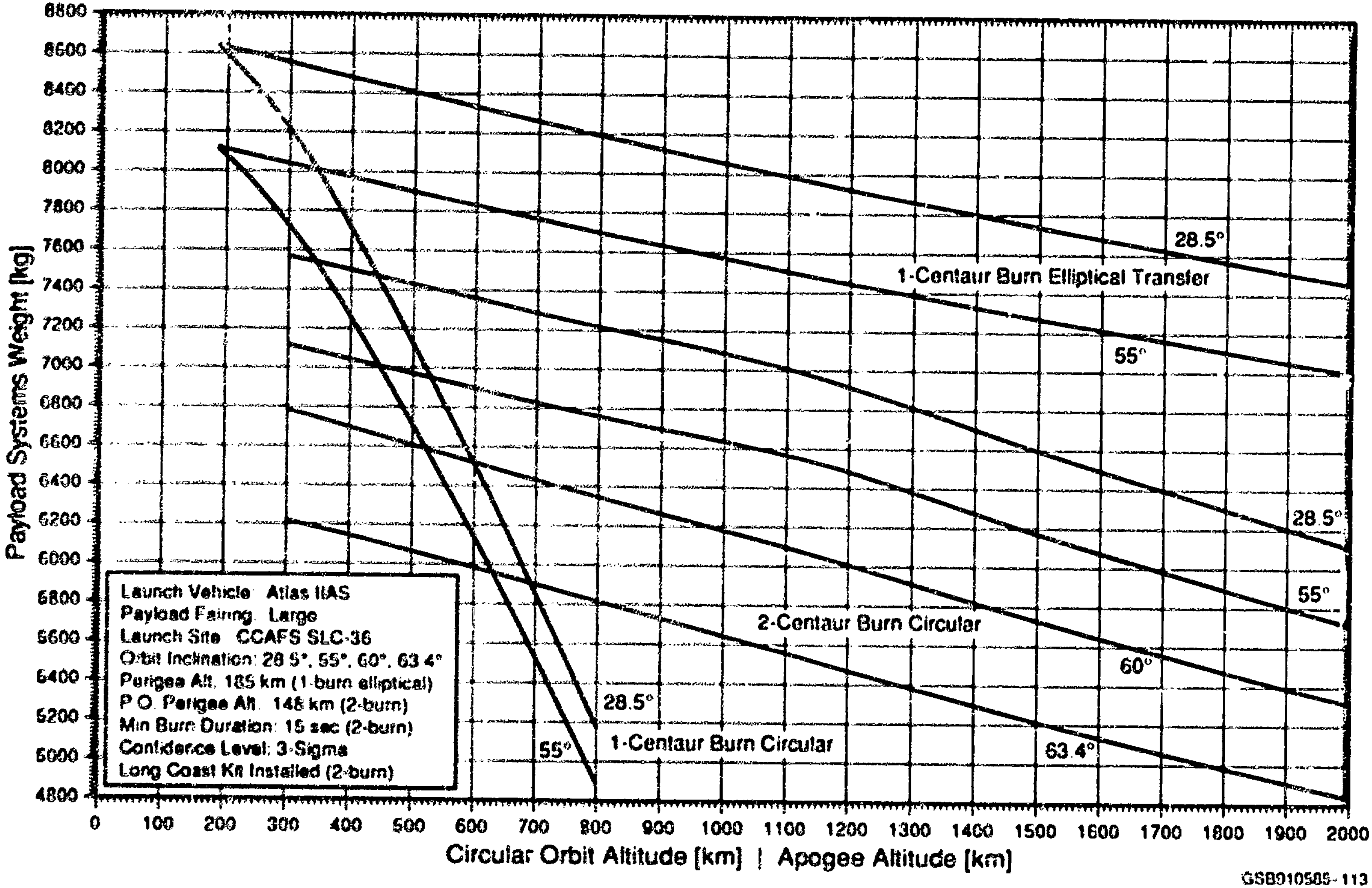

Figure 2-26b. Atlas IIAS low Earth orbit performance (metric).

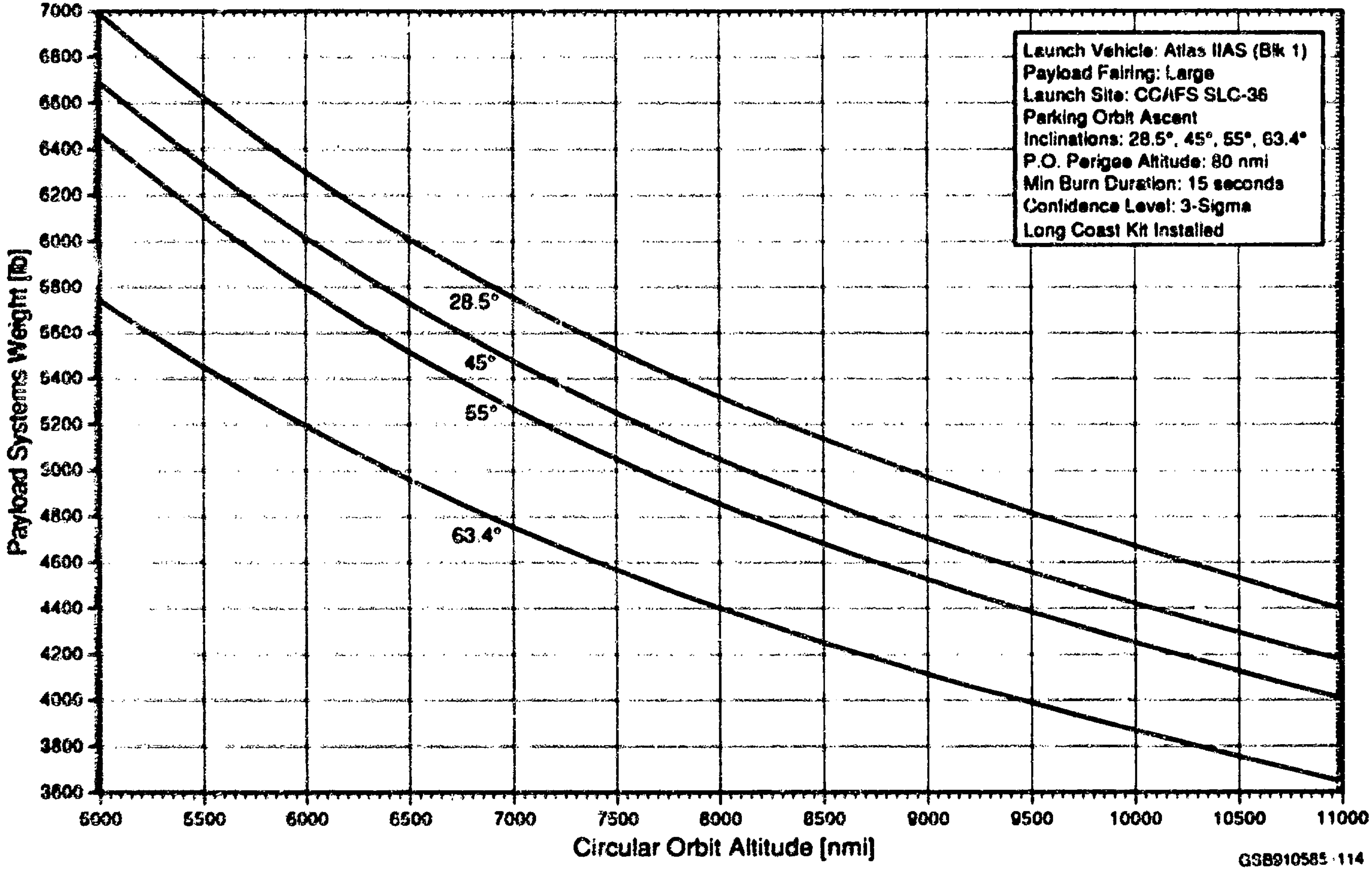

Figure 2-27a. Atlas IIAS intermediate circular orbit performance.

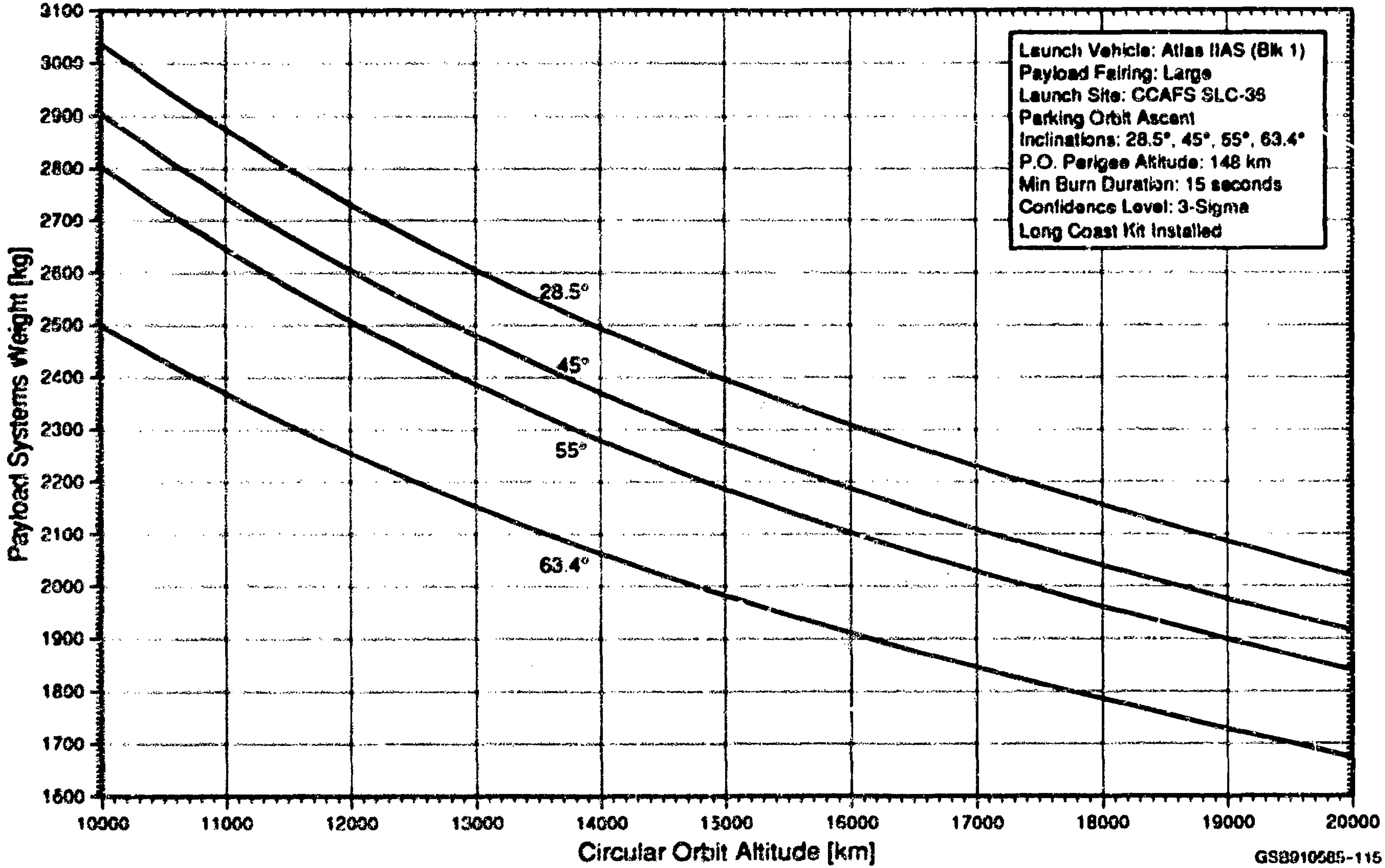

Figure 2-27b Atlas IIAS intermediate circular orbit performance (metric).

Table 2-13. Atlas IIAS (Block 1) performance — PSW vs transfer orbit apogee altitude.

Apogee Altitude [nmi (km)]		Payload Systems Weight [lb (kg)]			
		MRS		99%	
80,994	(150000)	6,686	(3033)	6,528	(2961)
67,495	(125000)	6,802	(3086)	6,642	(3013)
53,996	(100000)	6,974	(3163)	6,811	(3089)
48,596	(90000)	7,067	(3206)	6,902	(3131)
43,197	(80000)	7,181	(3257)	7,016	(3182)
40,497	(75000)	7,249	(3288)	7,082	(3212)
37,797	(70000)	7,326	(3323)	7,156	(3246)
35,097	(65000)	7,414	(3363)	7,243	(3285)
32,397	(60000)	7,514	(3408)	7,342	(3330)
31,048	(57500)	7,570	(3434)	7,397	(3355)
29,698	(55000)	7,631	(3462)	7,457	(3382)
28,348	(52500)	7,697	(3491)	7,521	(3412)
26,998	(50000)	7,769	(3524)	7,592	(3444)
25,648	(47500)	7,847	(3559)	7,669	(3479)
24,298	(45000)	7,933	(3598)	7,754	(3517)
22,948	(42500)	8,027	(3641)	7,846	(3559)
21,598	(40000)	8,132	(3689)	7,949	(3606)
20,248	(37500)	8,249	(3742)	8,063	(3657)
19,324	(35788)	8,337	(3781)	8,150	(3697)
18,898	(35000)	8,379	(3801)	8,192	(3716)
17,549	(32500)	8,529	(3869)	8,337	(3781)
16,199	(30000)	8,696	(3944)	8,501	(3856)
14,849	(27500)	8,888	(4031)	8,689	(3941)
13,499	(25000)	9,106	(4131)	8,907	(4040)
12,149	(22500)	9,369	(4250)	9,162	(4156)
10,799	(20000)	9,674	(4388)	9,464	(4293)
9,449	(17500)	10,047	(4557)	9,828	(4458)
8,099	(15000)	10,500	(4763)	10,276	(4661)
6,749	(12500)	11,073	(5023)	10,840	(4917)
5,400	(10000)	11,812	(5358)	11,566	(5246)

80 nmi (148 km) perigee parking orbit; 90 nmi (167 km) perigee transfer orbit; 27-degree inclination, 180-degree argument of perigee; large payload fairing

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3 • ENVIRONMENTS

This section describes the environments to which the spacecraft is exposed. Prelaunch environments are described in Section 3.1, flight environments are described in Section 3.2, and spacecraft tests requirements are described in Section 3.3.

3.1 PRELAUNCH ENVIRONMENTS

3.1.1 THERMAL — The spacecraft thermal environment is controlled during prelaunch activity, maintained during ground transport, and controlled after mate to the launch vehicle.

Environments in the spacecraft processing areas at Astrotech are controlled at 21–27°C (70–80°F) and 50 ± 5% relative humidity. Portable air conditioning units are available to further cool test equipment or spacecraft components as required.

During ground transport, the temperature within the payload fairing remains between 4 and 29°C (40 and 85°F), with positive pressure provided by a GN_2 purge. If required, an optional environmental control unit can maintain temperature between 15 and 25°C (50 and 77°F). In both cases, the relative humidity remains at or below 50%.

During hoisting operations the encapsulated spacecraft is purged with dry gaseous nitrogen with relative humidity at or below 50%.

Following spacecraft mate to Centaur, gas conditioning is provided to the payload fairing at the required temperature, humidity, and flowrate. Air with a maximum dew point of 40°F (4°C) is used until approximately two hours prior to launch, after which GN_2 with a maximum dew point of -35°F (-37°C) is used. Table 3-1 summarizes prelaunch gas conditioning temperature capabilities for the nominal configuration and for a fairing with a mission-peculiar thermal shield. The optional thermal shield allows greater control over payload fairing internal temperatures during prelaunch gas conditioning. The shield consists of a noncontaminating membrane attached to the inboard surfaces of the PLF frames as shown in Figure 3-1. Mission-peculiar arrangements for dedicated purges of specific components can be provided.

The gas to the payload area is supplied through a ground/airborne disconnect on the payload fairing and is controlled by prime and backup environmental control units. These units provide air or GN_2 conditioned to the following parameters (Figure 3-2):

- Cleanliness: Class 10,000 per FED-STD-209B
  Class 5,000 available on request

Table 3-1. Gas conditioning capabilities.

			Temperature Range Inside Payload Fairing*			
			11-ft PLF		14-ft PLF	
	Inlet Temp Capability	Flowrate Capability	Baseline	With Thermal Shield	Baseline	With Thermal Shield
Inside Tower	50-85°F (10-29°C)	80-160 lb/min (0.60-1.21 kg/sec)	54-75°F (12-24°C)	62-68°F (17-20°C)	51-76°F (11-26°C)	61-69°F (16-21°C)
After Tower Removal	50-85°F (10-29°C)	80-160 lb/min (0.60-1.21 kg/sec	47-83°F (8-28°C)	61-69°F (16-21°C)	40-87°F (4-31°C)	50-70°F (16-21°C)

* Temperature ranges are adjustable (within system capability) according to spacecraft requirements. Above ranges assume a 160 lb/min (1.21 kg/sec) flow rate.

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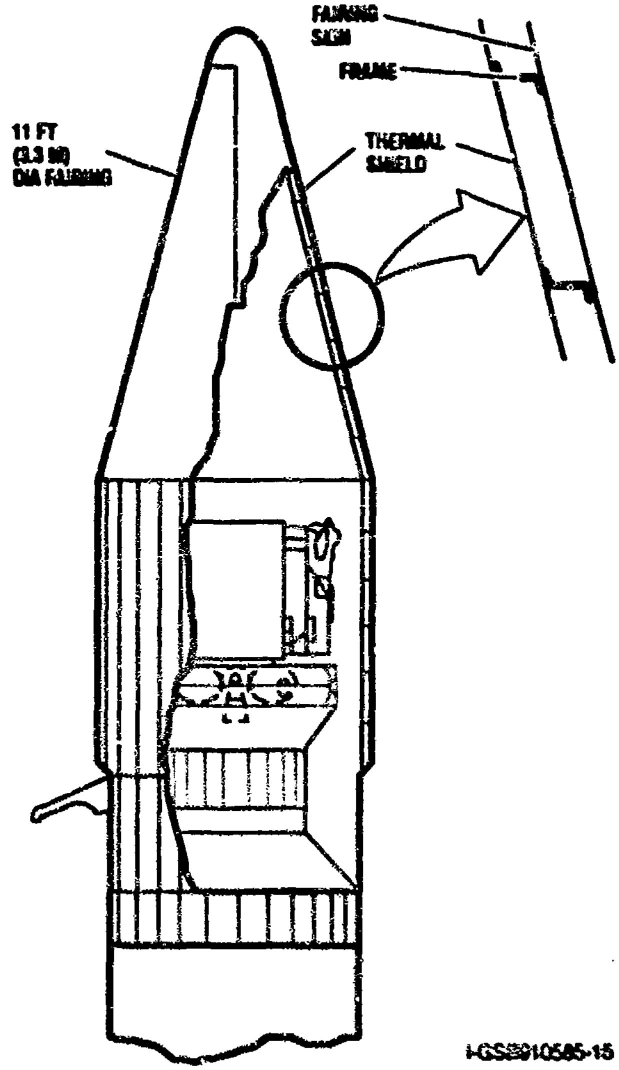

Figure 3-1. The thermal shield option.

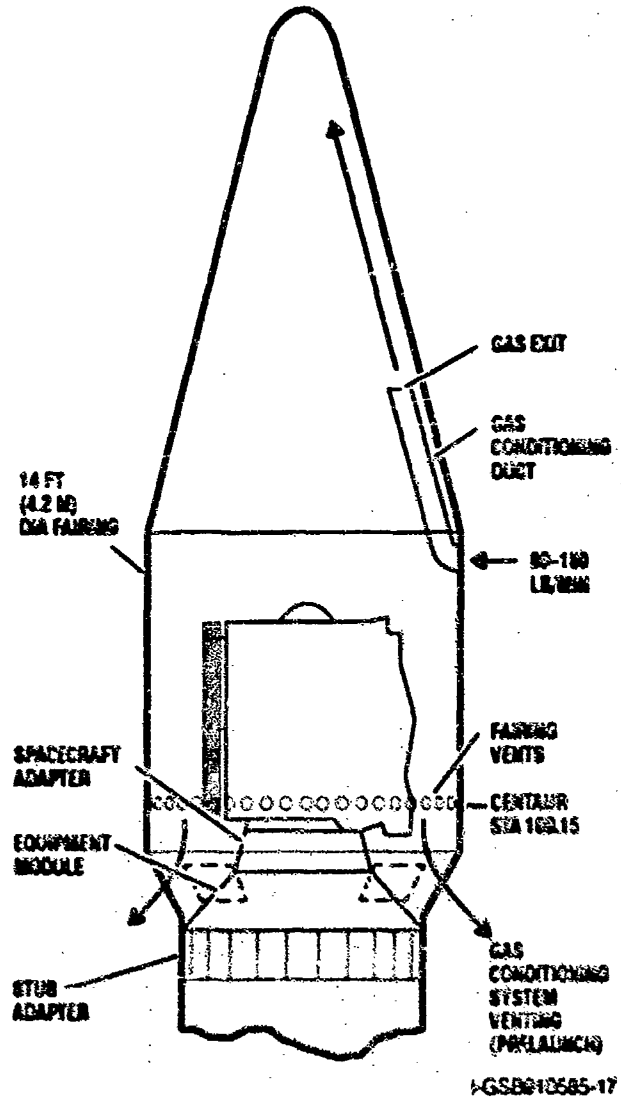

Figure 3-2. The payload fairing air conditioning system provides a controlled thermal environment during ground checkout and prelaunch activities.

- Inlet Temperature: Setpoint from 50 to 85°F (10 to 29°C)
- Inlet Temp Control: ± 3°F (2°C)
- Filtration: 0.3 microns
- Flow Rate: Adjustable to any set point between 80 and 160 lb/min (36.4 to 72.7 kg/min)
- Dewpoint (Max): 40°F (4.4°C) air; -35°F (-37.2°C) GN_2

Internal ducting directs the gas upward to provide uniform distribution and prevent direct impingement on the payload. The conditioning gas is vented to the atmosphere through one-way flapper doors in the aft end of the fairing. This baseline design ensures that average gas velocities across spacecraft components are less than 7 ft/sec (2 m/sec).

3.1.2 RADIATION AND ELECTROMAGNETIC — To ensure that electromagnetic compatibility (EMC) is achieved for each launch, the electromagnetic environment is thoroughly evaluated. The spacecraft customer will be required to provide all data necessary to support EMC analyses (see Appendix B tables) employed for this purpose.

3.1.2.1 Launch Vehicle-Generated Radio Environment — Launch vehicle intentional transmissions are limited to the S-band telemetry transmitters at 10.8 dBW and the C-band beacon transponder at 28.5 dBW (peak) or 26 dBm (average).

Figure 3-3 shows the absolute worst-case antenna radiation environment generated by the launch vehicle (LV). The curve is based on transmitter fundamental and "spurious output" requirements and assumes: (a) maximum transmit output power, (b) maximum antenna gain and minimum passive line loss (i.e., measured values at transmit frequency applied across the entire frequency spectrum), and (c) straight-line direct radiation. Actual levels encountered by the satellite (influenced by many factors) can only be less than the levels depicted on the figure. Initial reductions are provided to the user upon determination of which launch vehicle and payload adapter will be employed.

3.1.2.2 LV-Generated Electromagnetic Environment — The unintentional EM environments generated by the LV at the satellite location are depicted in Figures 3-4, 3-5, and 3-6. Actual levels to be encountered by the satellite will approach the typical levels depicted in the figures.

3.1.2.3 Launch Range Electromagnetic Environment — The EM environment of the launch range is based upon information contained in TOR-0084 (4338-42)-1 Reissue B. An EMC analysis will be performed to ensure electromagnetic compatibility of the spacecraft/launch vehicle with the range environment.

3.1.2.4 Spacecraft-Generated Environment Limitation — During ground and launch operation time frames through spacecraft separation, any spacecraft EMI radiated emissions (including antenna radiation) should not exceed the values depicted in Figure 3-7. LV/SC external interfaces (EMI-conducted emissions) must be addressed individually.

Each payload will be treated on a mission-peculiar basis. Assurance of the LV/SC EMC with respect to payload emissions will be a shared responsibility between General Dynamics and the individual spacecraft contractor.

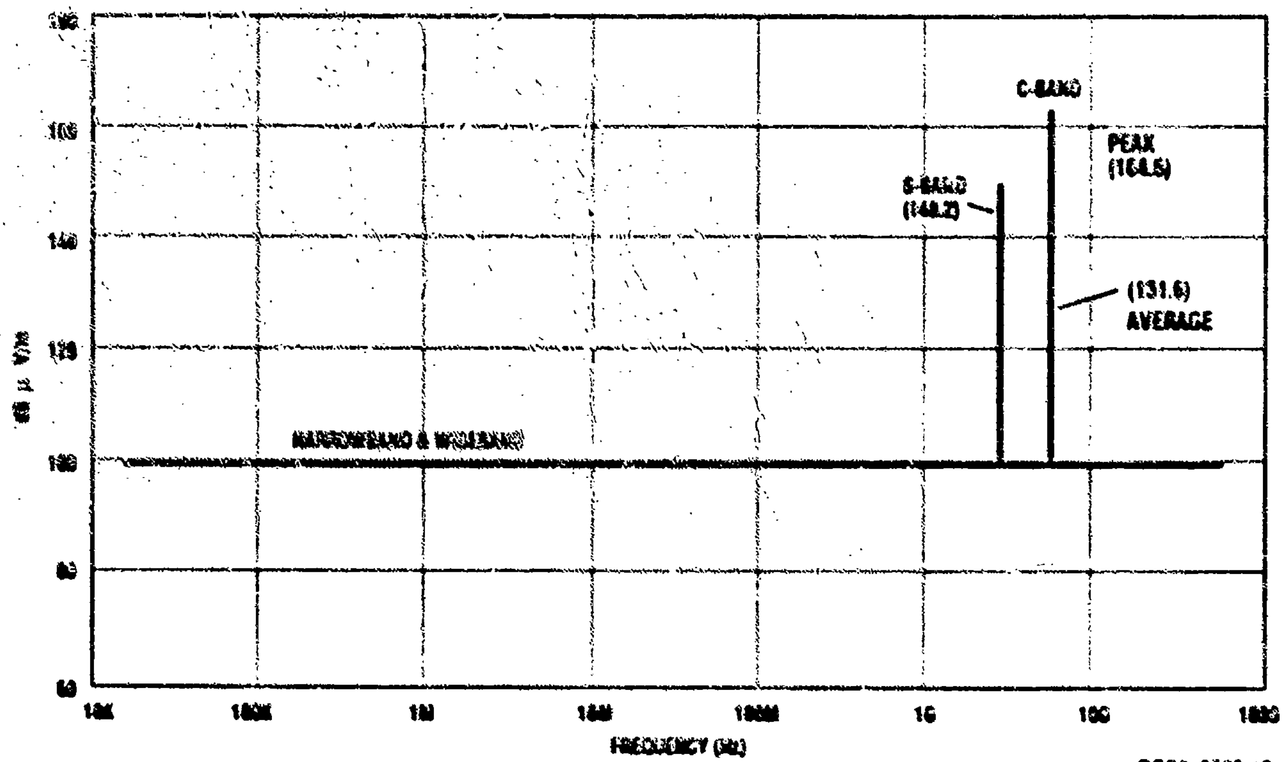

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Figure 3-3. Launch vehicle electric field radiation from antennas.

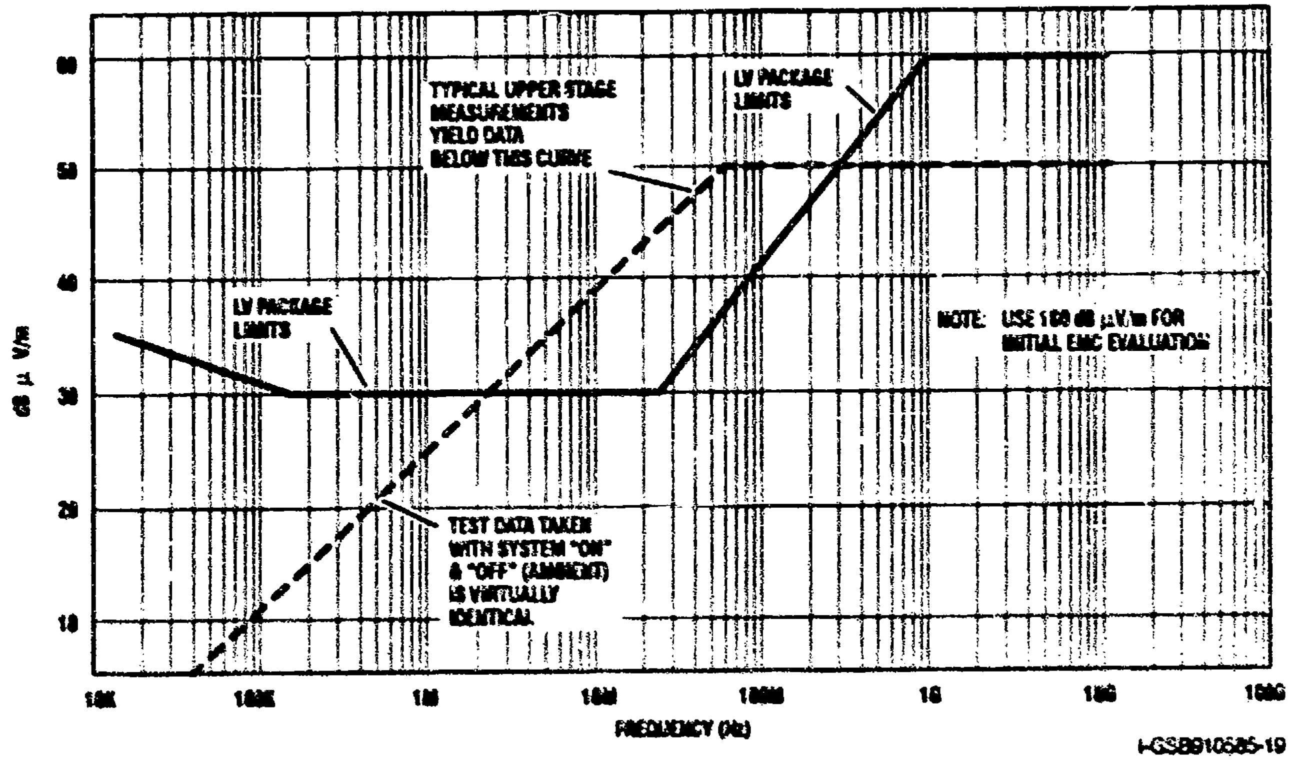

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Figure 3-4. Launch vehicle electric field radiation (narrow band).

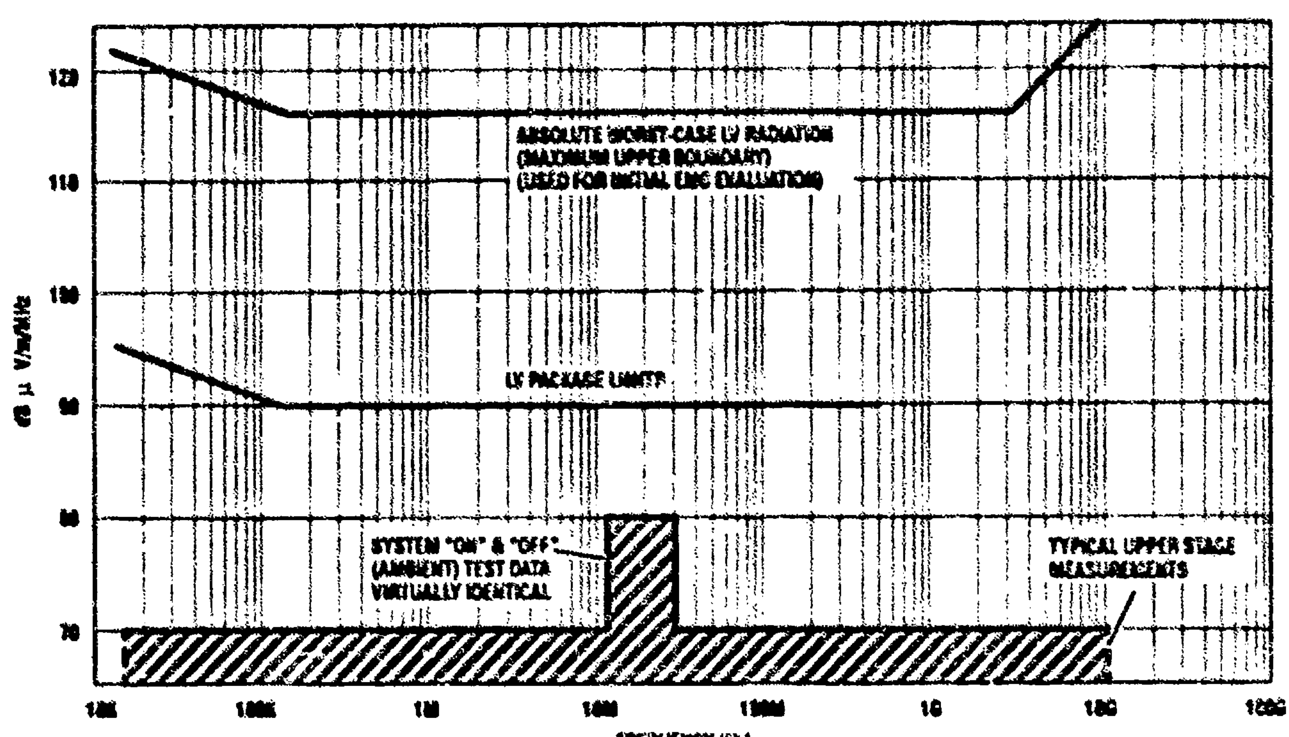

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Figure 3-5. Launch vehicle electric field radiation (wide band).

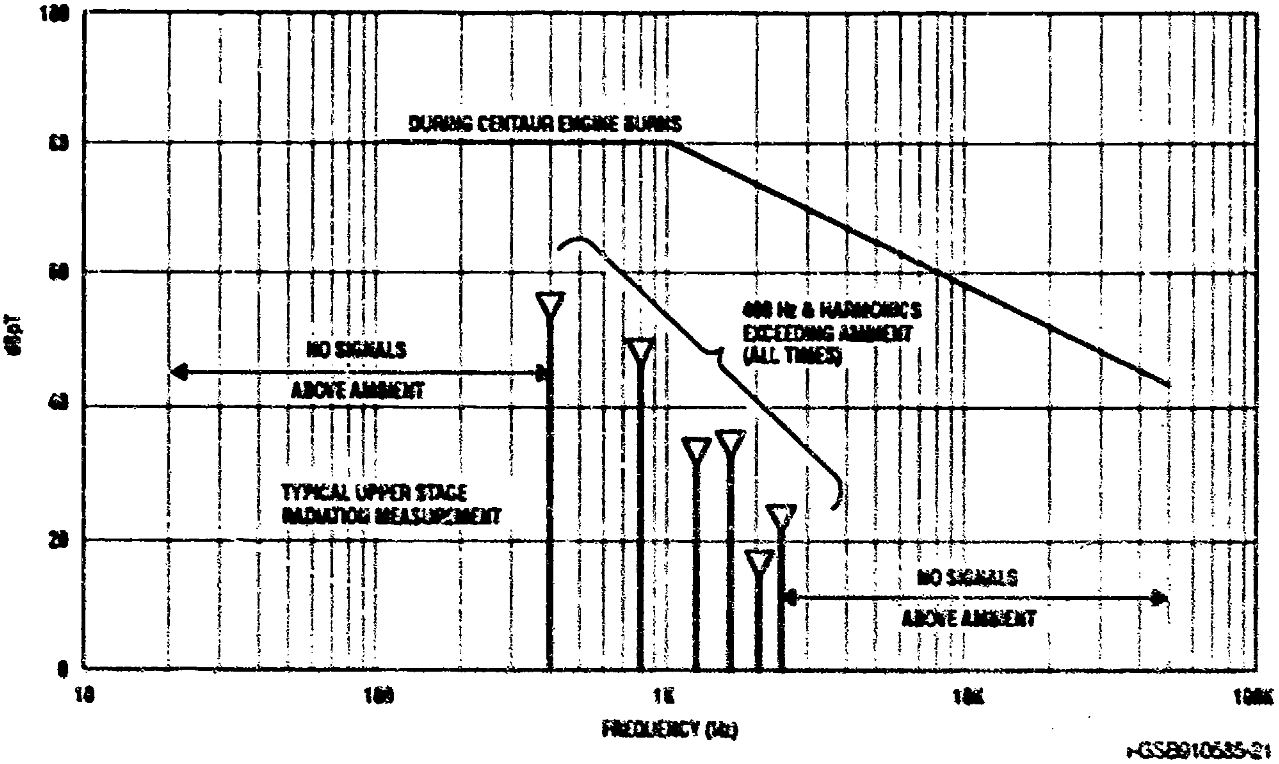

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Figure 3-6. Launch vehicle magnetic field spurious radiation (narrow band).

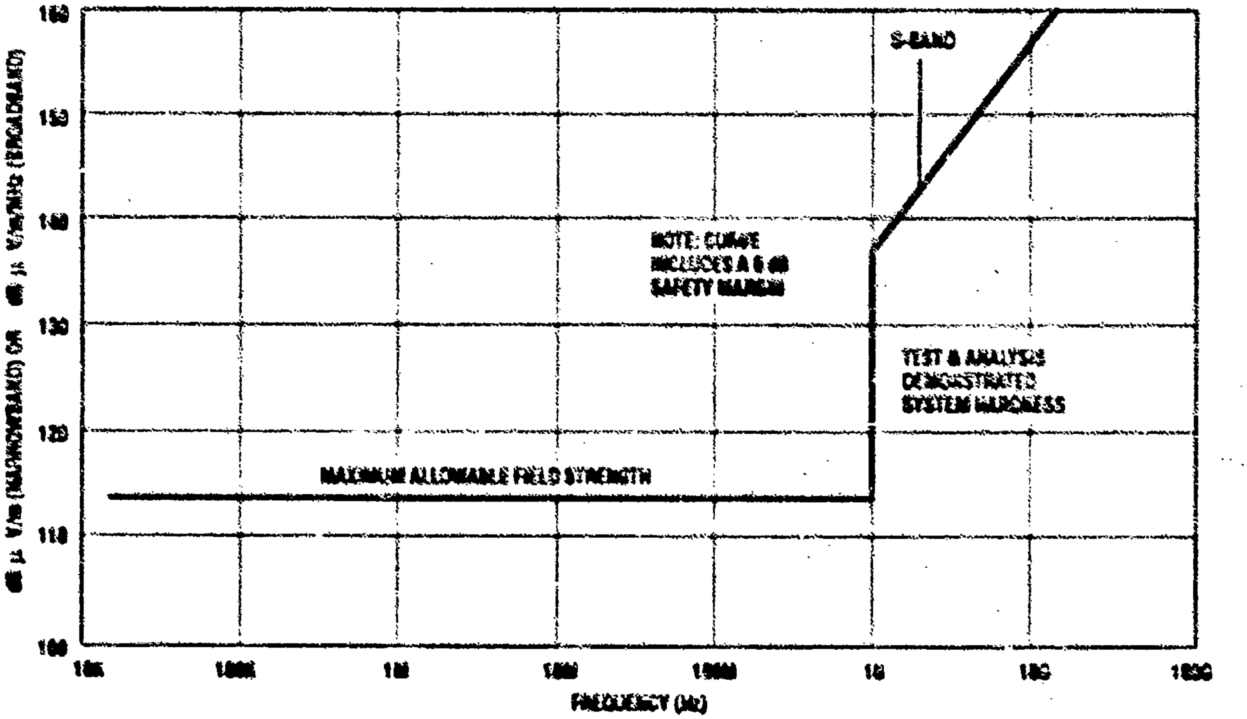

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Figure 3-7. Spacecraft electric field radiation impingement on LV.

3.1.3 CONTAMINATION AND CLEANLINESS

— Launch vehicle hardware that comes into contact with the payload environment has been designed and manufactured in accordance with strict contamination control guidelines. This hardware is defined as contamination-critical and includes the Centaur forward equipment area, the payload adapter, and the interior surface of the payload fairing. In addition, ground operations at the launch site have been designed to ensure a clean environment for the spacecraft. A comprehensive Contamination Control Plan has been written to identify these requirements and procedures. Some of the guidelines and practices employed in the plan are as follows.

1. Precautions are taken during manufacture, assembly, test, and shipment to prevent contamination accumulations on the contamination-critical launch vehicle surfaces.
2. Launch vehicle contamination-critical surfaces are cleaned using approved material and procedures. A final visual inspection for cleanliness conformance occurs just prior to encapsulation.
3. The encapsulation process is performed in a facility that is environmentally controlled to Class 100,000 conditions per FED-STD-209B. All handling equipment is cleanroom-compatible and will be cleaned and inspected before entering the facility. These environmentally controlled conditions are available for all remote encapsulation facilities (i.e., Astrotech).
4. Personnel controls are employed to limit access to the payload fairing to maintain spacecraft cleanliness. Contamination control training is provided to all launch vehicle personnel working in or around the encapsulated payload fairing. General Dynamics provides similar training to spacecraft personnel working on the spacecraft while on the launch tower to ensure that these individuals are familiar with the procedures.

3.1.3.1 Contamination Control Prior to Launch Site Delivery

Design and Assembly — Contamination control principles are employed in the design and manufacturing processes to limit the amount of contamination that can be expected from launch vehicle components. Interior surfaces include maintainability features to facilitate the removal of manufacturing contaminants. The Centaur vehicle is assembled in a Class 100,000 facility to ensure that the hardware surfaces, and in particular any entrapment areas, are maintained at an acceptable level of cleanliness prior to shipment to the launch site. Inspection points are provided to verify cleanliness throughout the assembly process.

Materials Selection — In general, materials are selected for use on contamination-critical hardware interior to the PLF that will not become a source of contamination to the spacecraft. Metallic or nonmetallic materials that are known to chip, flake, or peel, and cadmium plating, zinc plating, and unfused electrodeposited tin are prohibited from use. Corrosion-resistant materials are selected wherever possible and dissimilar materials are avoided or protected in accordance with MIL-STD-889B. Since most nonmetallic materials are known to exhibit some outgassing, these materials are evaluated against NASA SP-R-0022 criteria prior to selection.

3.1.3.2 Contamination Control Prior to Spacecraft Encapsulation

Cleanliness Levels — Contamination-critical hardware surfaces are cleaned and inspected to specific criteria. This checks for the absence of all particulate and molecular contaminants visible to the unaided eye at a distance of 6–18 inches (152–457 mm) with a minimum illumination of 100 foot-candles (1,076 lm/m^2). This criterion is Visibly

Clean Level 2. Hardware that is cleaned to this criterion at the assembly plant is protected to maintain this level of cleanliness through shipping and encapsulation.

Contingency cleaning may also be required to reattain this level of cleanliness if the hardware becomes contaminated. Contingency cleaning procedures outside of the encapsulation facility prior to encapsulation are subject to General Dynamics engineering approval. Cleaning of the launch vehicle hardware that is required in the vicinity of the spacecraft must also be approved by the cognizant spacecraft engineer.

Certain payloads may require that contamination-critical hardware surfaces be cleaned to a level of cleanliness other than Visibly Clean Level 2. Because additional cleaning and verification may be necessary, these requirements are implemented on a mission-peculiar basis.

Payload Fairing Cleaning Techniques — General Dynamics recognizes that effective cleaning of the large interior payload fairing surfaces depends on the implementation of well planned cleaning procedures. To achieve customer requirements, all cleaning procedures are verified by test and are reviewed and approved by Material and Processes Engineering. Final cleaning of the payload fairing is performed in a Class 100,000 facility just prior to encapsulation.

Cleanliness Verification — All contamination-critical hardware surfaces are visually inspected to verify the Visibly Clean Level 2 criteria described above. The additional verification techniques shown below can be provided on a mission-unique basis:

- Particulate Obscuration — Tape Lift Sampling
- Nonvolatile Residue (NVR) — Solvent Wipe Sampling
- Particulate/Molecular Fallout — Witness Plates

3.1.3.3 Contamination Control after Encapsulation

Contamination Diaphragm — After the two halves of the payload fairing are joined, the encapsulation is completed by closing the aft opening with a GSE reinforced plastic film diaphragm. The doughnut-shaped diaphragm stretches from the payload adapter to the aft end of the payload fairing cylinder and creates a protected environment for the spacecraft through mating to the Atlas/Centaur. After the spacecraft is mated, the diaphragm remains in place until final payload fairing closeout when all flight doors are installed. This assists in protecting the spacecraft from possible contamination during Centaur operations performed at the launch tower after mating.

Payload Fairing Purge — After encapsulation, the payload fairing environment is continuously purged with HEPA filtered gases (nitrogen or air) to ensure that the cleanliness of the environment does not exceed the requirements of Class 100,000 per FED-STD-209B. General Dynamics guarantees that the gas at the inlet to the payload fairing does not exceed Class 10,000.

Complex 36B ECA — The Complex 36B Service Tower access levels that are used to access the encapsulated payload fairing are designated as an Environmentally Controlled Area (ECA). These levels have been constructed so that they can be sealed off from the exterior environment. Upon sealing off these levels, a separate air conditioning system supplies HEPA filtered air to the environment. This provides for a controlled environment outside of the payload fairing when access doors are opened. Cleanroom garments can be provided to spacecraft personnel working on these levels to provide optimum control as dictated by spacecraft requirements.

3.2 LAUNCH AND FLIGHT ENVIRONMENTS

This section describes general environmental conditions that may be encountered by a spacecraft during launch and flight with the Atlas launch vehicle.

3.2.1 DESIGN LOAD FACTORS — Design load factors are provided in Tables 3-2 and 3-3 for use in preliminary design of primary structure and/or evaluating the suitability of the Atlas vehicle for an existing spacecraft. The load factors are intended for application at the center of gravity of the spacecraft to evaluate primary structure. The response of a spacecraft to launch vehicle transients will depend upon its mass properties, stiffness, and amount of axial-to-lateral coupling. The load factors given are intended to provide a conservative design envelope for a typical spacecraft in the 4,000 lb (1814 kg) to 8,000 lb (3628 kg) weight class with first lateral modes above 10 Hz and first axial mode above 15 Hz. Transient load analyses will be performed during the integration activity to provide the actual loads on the space vehicle for both primary and secondary structure. The load factors are separated into a quasi-steady-state and oscillatory dynamic. Total load factors in a direction are obtained by adding the steady-state and dynamic portion of the load factors.

Table 3-2. Spacecraft limit load factors for Atlas I, II, and IIA.

Load Condition	Direction	Steady-state (g)	Dynamic (g)
Launch	Axial	1.2	± 1.2
	Lateral	–	± 1.0
Flight winds	Axial	2.2	± 0.3
	Lateral	0.4	± 1.2
BECO/BPJ (max axial)	Axial	5.5	± 0.5
	Lateral	–	± 0.5
(max lateral)	Axial	2.5 - 1.0	± 1.0
	Lateral	–	± 2.0
SECO	Axial	2.0 - 0.0*	± 0.4
	Lateral	–	± 0.3
MECO (max axial)	Axial	4.0 - 0.0*	± 0.5
	Lateral	–	± 0.2
(max lateral)	Axial	0.0	± 2.0
	Lateral	–	± 0.6

* Decaying to zero

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Table 3-3. Spacecraft limit load factors for Atlas IIAS.

Load Condition	Direction	Steady-state (g)	Dynamic (g)
Launch	Axial	1.3	± 1.8
	Lateral	–	± 1.3
Flight winds	Axial	2.2	± 0.3
	Lateral	0.4	± 1.2
BECO/BPJ (max axial)	Axial	5.2	± 0.5
	Lateral	–	± 0.5
(max lateral)	Axial	2.5 - 1.0	± 1.0
	Lateral	–	± 2.0
SECO	Axial	2.0 - 0.0*	± 0.4
	Lateral	–	± 0.3
MECO (max axial)	Axial	4.0 - 0.0*	± 0.5
	Lateral	–	± 0.2
(max lateral)	Axial	0.0	± 2.0
	Lateral	–	± 0.6

Sign convention
- Longitudinal axis: + (Positive) = Compression; – (Negative) = Tension
- Pitch axis: ± May act in either direction
- Yaw axis: ± May act in either direction

Lateral and longitudinal loading may act simultaneously during any flight event.

Loading is induced through the cg of the spacecraft.

* Decaying to zero

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3.2.2 ACOUSTICS — The spacecraft is exposed to an acoustic environment throughout the boost phase of flight until the vehicle is out of the sensible atmosphere. Two portions of flight have significantly higher acoustic levels than the others. The highest acoustic level occurs for approximately 5 seconds during liftoff, when the acoustic energy of the engine exhaust is being reflected by the launch pad. The other significant level occurs for approximately 20 seconds during the transonic portion of flight and is due to transonic aerodynamic shock waves and a highly turbulent boundary layer. The acoustic level inside the payload fairing will vary slightly with different spacecraft. This is due to the acoustic absorption of each spacecraft depending on its size, shape, and surface material properties. Acoustic sound pressure levels for both the 14-foot and 11-foot payload fairings are provided in Figures 3-8a through 3-8c. The levels presented are for typical spacecraft of square cross sectional area with 50–60% fill of the fairing by cross sectional area. A mission-peculiar acoustic analysis is required for spacecraft with other fill factors to bound the acoustic environment. The spacecraft should be capable of functioning properly after one-minute exposure to this level. For the 14-foot payload fairing with acoustic blanket, special consideration should be given to components located within 30 in. (76 cm) of the payload fairing vents (the 11-foot payload fairing vents are fewer in number and located farther from the spacecraft envelope). Sound pressure levels for components located near the vents are given in Figure 3-8d.

An optional mission-peculiar acoustic blanket design is available (see Figure 3-9) to reduce high-frequency acoustic energy within the payload envelope. No reduction in clearance envelope results from inclusion of the blanket since the current design calls for placement between existing frames.

3.2.3 VIBRATION —The spacecraft is exposed to a vibration environment that may be divided into two general frequency ranges: 1) low-frequency quasi-sinusoidal vibration and 2) high-frequency broadband random vibration.

The low-frequency vibration tends to be the design driver for spacecraft structure. An envelope of the Atlas/Centaur flight measured low-frequency vibration under 100 Hz near the spacecraft interface is shown in Figure 3-10. The peak responses occur for a few cycles during transient events such as launch, gusts, BECO, jettison events, and MECO.

Coupled loads analyses show that interface accelerations are highly variable with the specific spacecraft dynamic characteristics. Thus, if the total spacecraft is tested with a sinusoidal base vibration, it is recommended that the input levels be tailored to the response levels consistent with the mission-

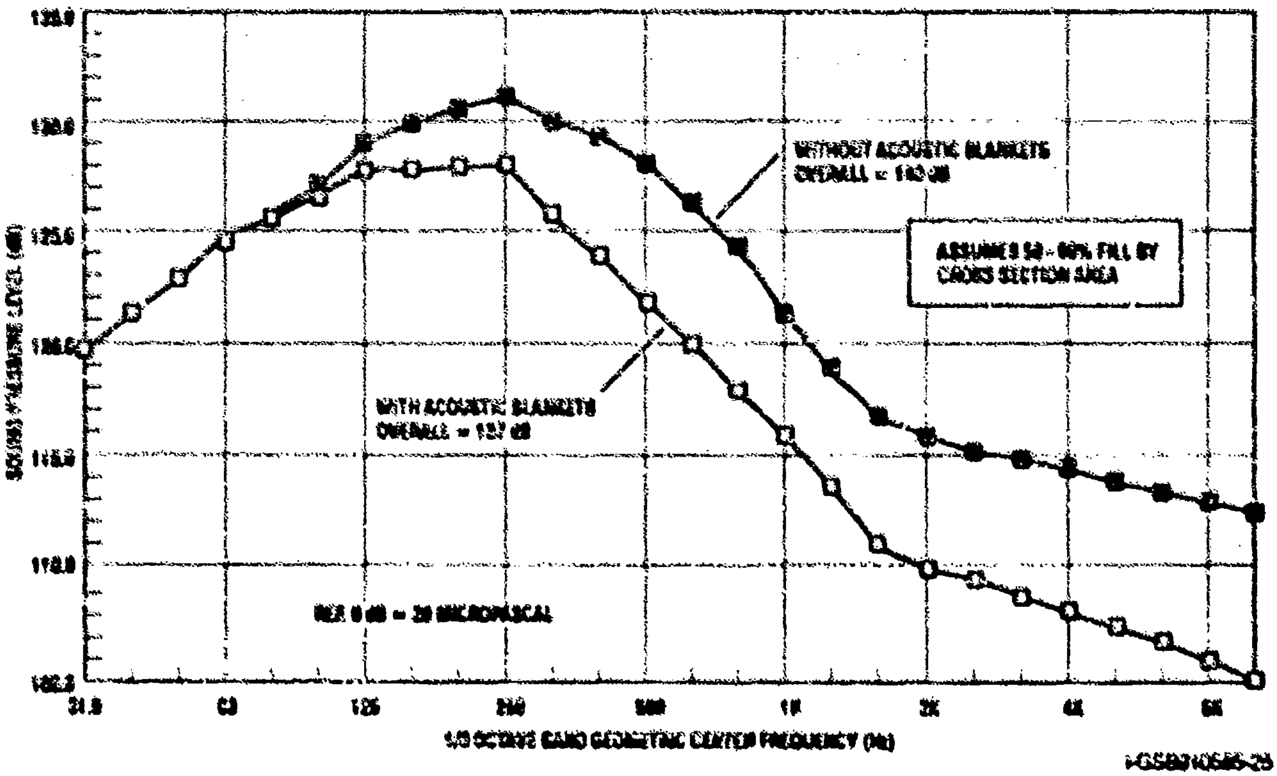

Figure 3-8a. Acoustic levels for Atlas I, II, and IIA with the 11-foot payload fairing.

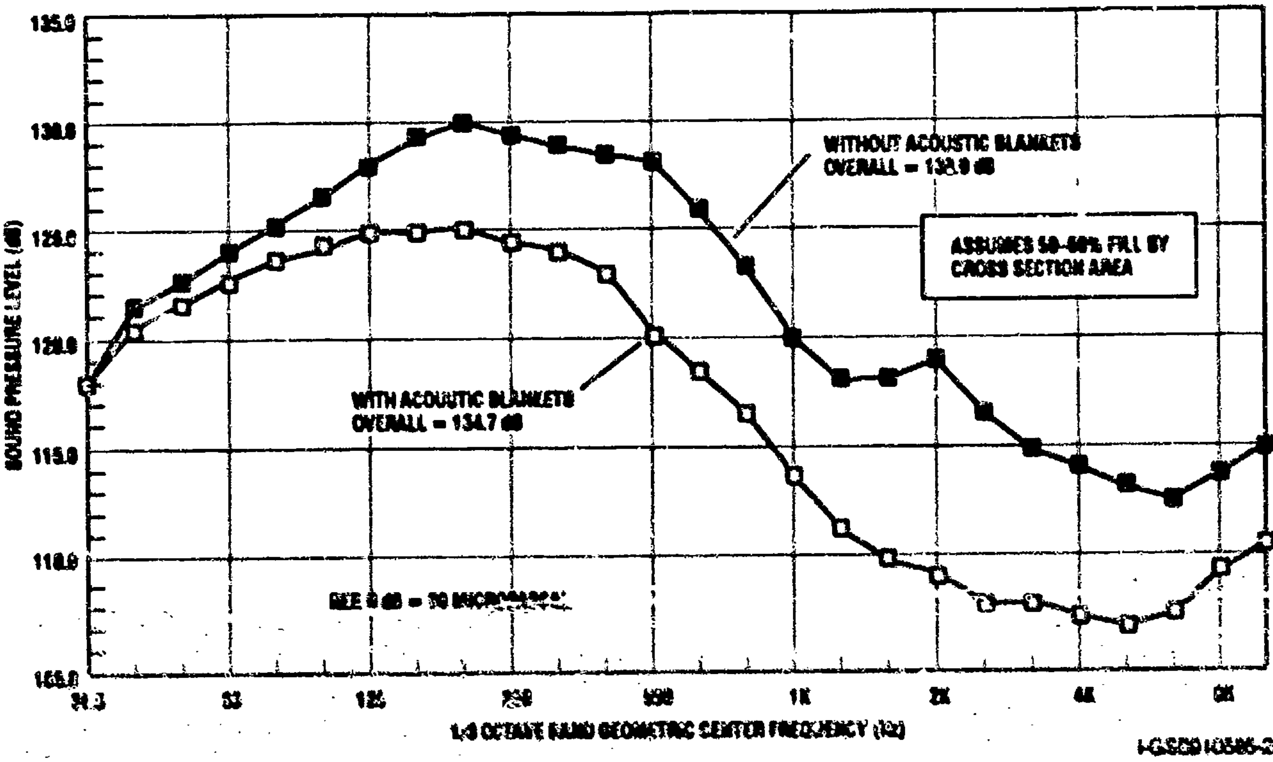

Figure 3-8b. Acoustic levels for Atlas I, II, and IIA with 14-foot payload fairing

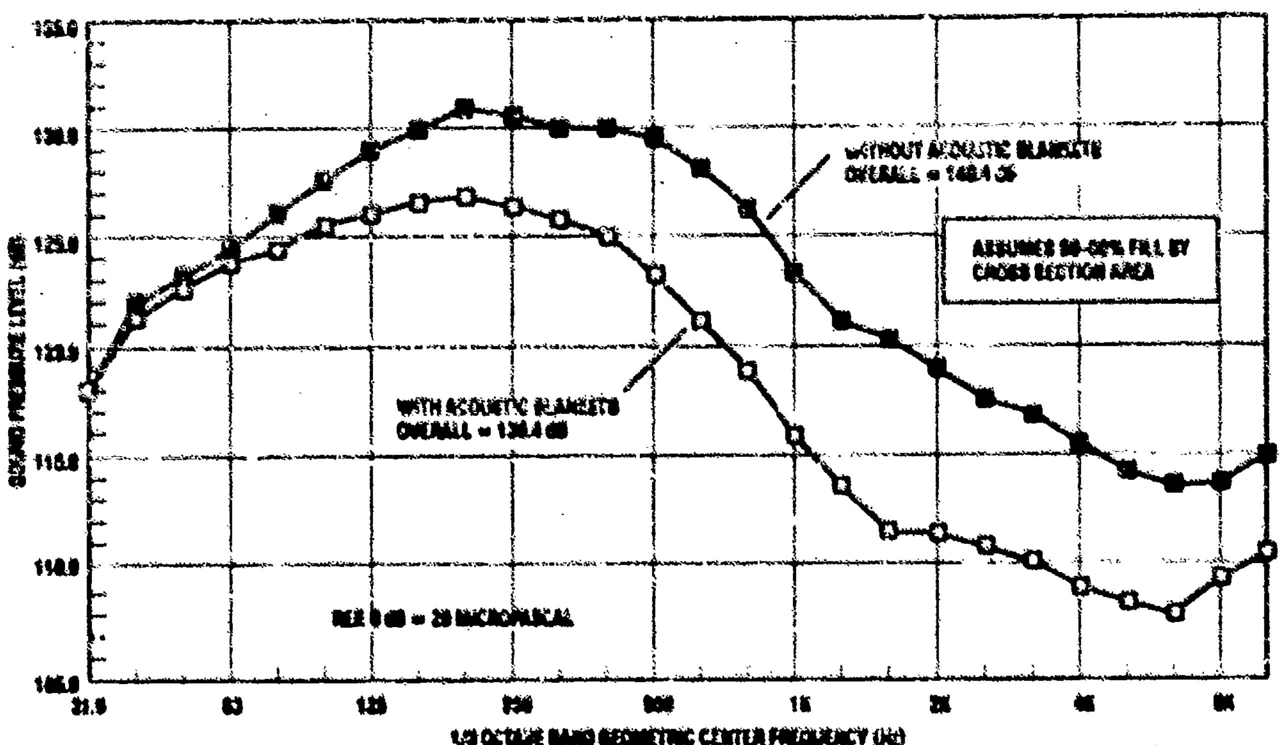

Figure 3-8c. Acoustic levels for Atlas IIAS with 14-foot payload fairing

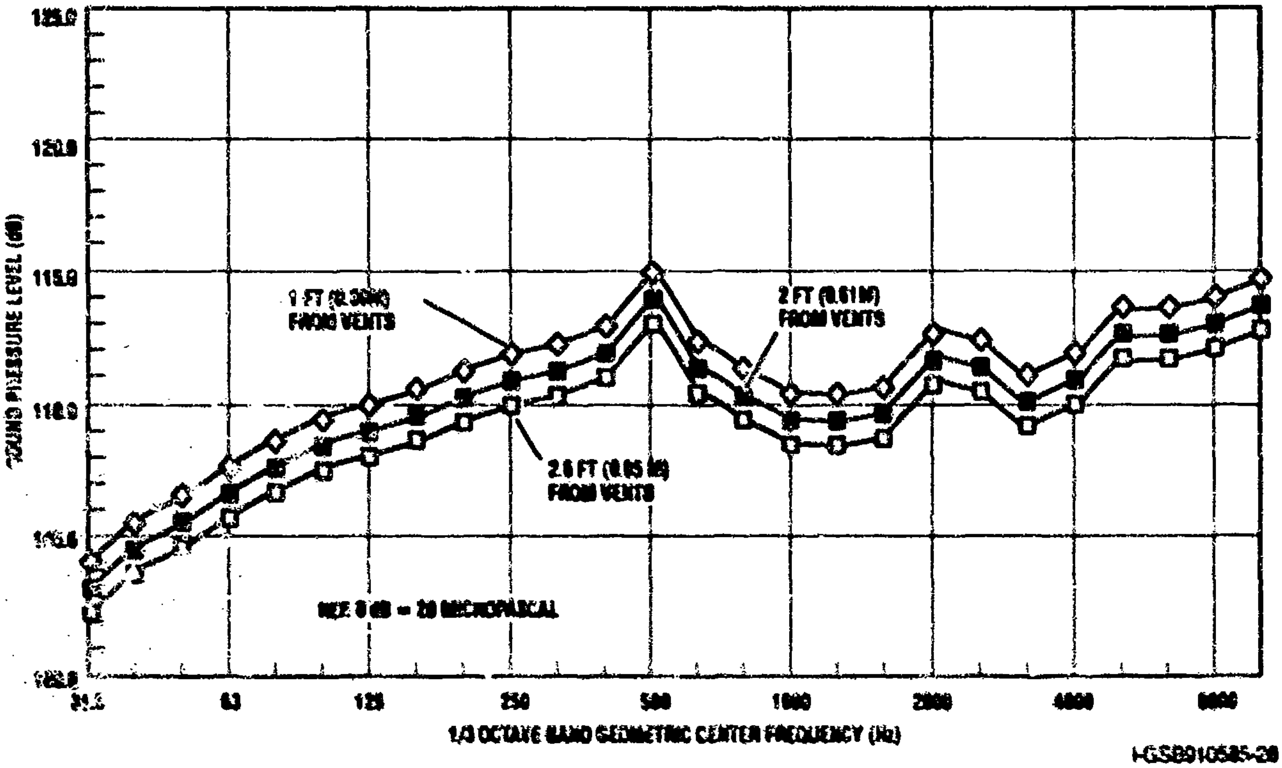

Figure 3-8. Acoustic levels near the vents with the 14-foot payload fairing.

peculiar coupled loads analysis and/or inflight responses of similar spacecraft.

The high-frequency random vibration that the spacecraft experiences is primarily due to the acoustic noise field, with a very small portion being mechanically transmitted from the engines. The acoustically excit-' random vibration environment tends to be the design driver for components and small structure supports. The high-frequency vibration level will vary from one location to another depending on physical properties of each area of the spacecraft. Since the vibration level at the payload interface depends on the adjacent structure above and below the interface, the exact interface level depends on the structural characteristics of the lower portion of the spacecraft, the particular PLA, and how the acoustic field is influenced by the particular spacecraft.

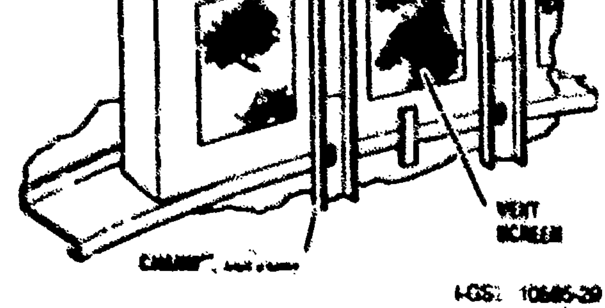

Figure 3-9. Acoustic blanket panels.

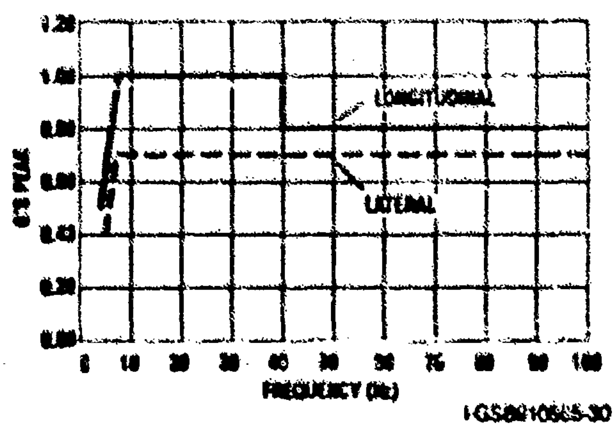

Figure 3-10. Low-frequency quasi-sinusoidal vibration levels.

An acoustic test of the spacecraft will more accurately simulate the high-frequency environment that it will undergo in flight than will a random vibration test. If the spacecraft is mounted to a test fixture that has structural characteristics similar to the PLA, then the vibration levels at the interface will be similar to the flight levels. It is not recommended to attach the spacecraft to a rigid fixture during the acoustic test since the interface vibration will be zero at the LV interface attach point, but at some distance away from the interface the vibration levels will be similar to flight levels.

3.2.4 SHOCK — There are four pyrotechnic shock events during flight on the Atlas I and three events on the Atlas II vehicles. These are insulation panel jettison (IPJ), payload fairing jettison (PFJ), Centaur separation from Atlas sustainer, and spacecraft separation. Since the system for Centaur separation from Atlas is located far from the spacecraft, the shock is highly attenuated by the time it reaches the spacecraft and does not produce a significant shock at the spacecraft interface. The separation devices for IPJ and PFJ are located closer to the spacecraft, and thus the shock at the SC interface is noticeable. The spacecraft separation device is at the SC/Centaur interface and produces the highest shock.

Figure 3-11 shows the expected shock levels for SC separation for a typical spacecraft at the spacecraft separation plane for the Type A, A1, B, B1, and D adapters. Based on separation shock tests, the shock levels for the Type B, B1, and D adapters have been raised from 50 to 100 and 150 g's at 100 Hz. For user-supplied adapters and separation systems, we recommend that the actual separation device be fired on a representative payload adapter and spacecraft to measure the actual level and/or qualify the spacecraft. Figure 3-12 shows the maximum acceptable shock level at the equipment module interface for a customer-provided separation system.

3.2.5 THERMAL

Within Fairing — The payload fairing protects the spacecraft during ascent to a nominal altitude of

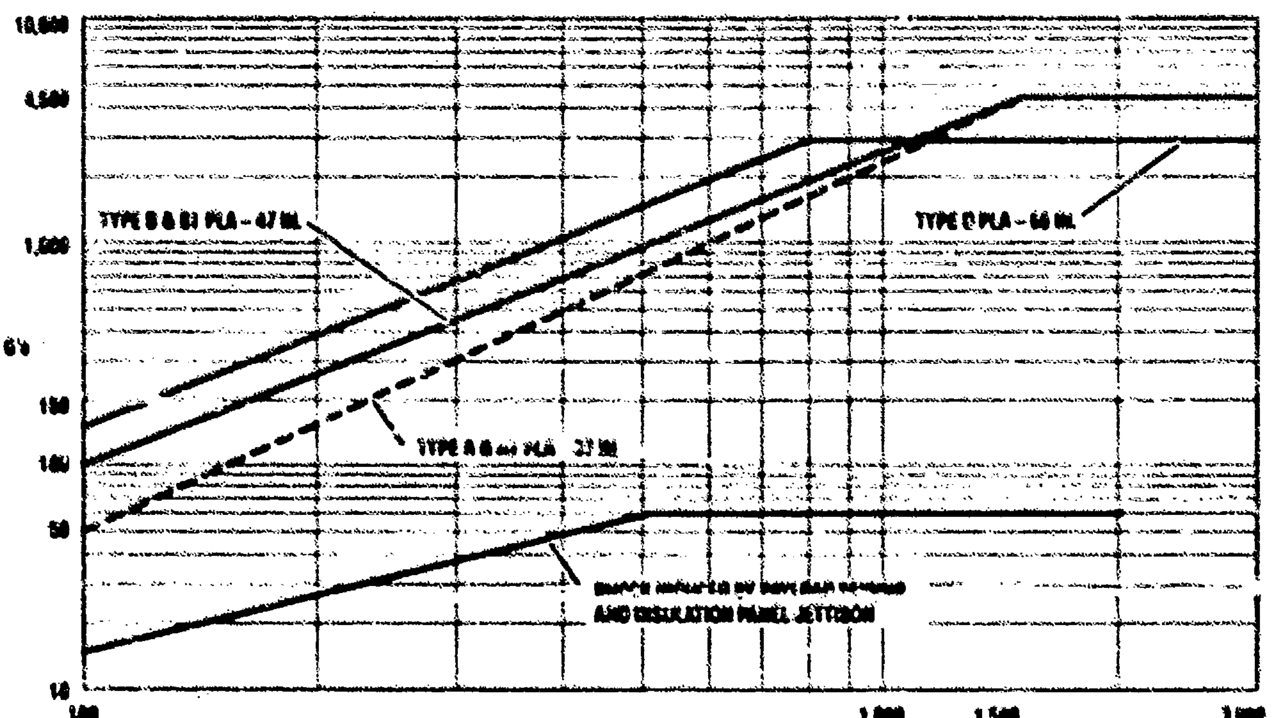

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Figure 3-11. Typical maximum Atlas shock levels — Type A, A1, B, B1, and D adapters

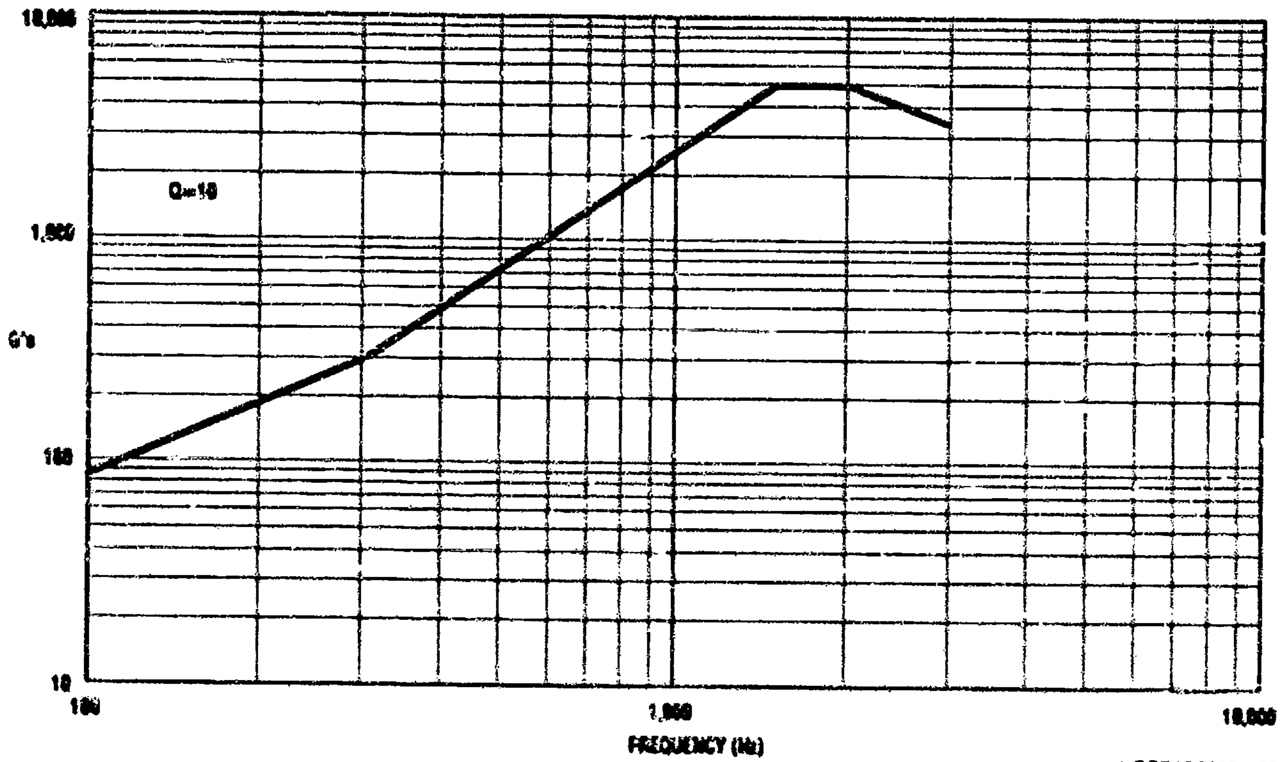

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Figure 3-12. Maximum allowable spacecraft-produced shock at equipment module interface.

approximately 370,000 ft (113 000 m). Aerodynamic heating on the fairing results in a time-dependent radiant heating environment around the spacecraft prior to fairing jettison. The fairing uses cork on the external conical surface to minimize fairing skin temperatures. The inner surfaces of the cone and cylinder have a low-emittance finish ($\epsilon < 0.1$), which minimizes heat transfer to the spacecraft. It should be noted that the payload fairing boattail and split barrel do not have a low-emittance coating and, therefore, have an emissivity of $\epsilon \leq 0.9$. The peak heat flux radiated by the cone and cylinder surfaces is less than 125 Btu/hr-ft^2 (400 W/m^2), and the peak temperatures remain below 400°F (190°C) at the warmest location.

After Fairing Jettison -- Fairing jettison occurs when the 3-sigma maximum free molecular heat flux decreases to 0.1 Btu/sec-ft^2 (1135 W/m^2). Jettison timing can be adjusted to meet specific mission requirements. A typical free molecular heating profile is shown in Figure 3-13. Since actual profiles are highly dependent on the trajectory flown, this data should not be used for design.

The spacecraft thermal environment following fairing jettison includes free molecular heating, solar heating, Earth albedo heating, and Earth thermal heating, plus radiation to the upper stage and to deep space. In addition, the spacecraft is conductively coupled to the forward end of the Centaur upper stage through the spacecraft adapter. Solar, albedo, and Earth thermal heating can all be controlled as required by the spacecraft by specification of launch times, vehicle orientation (including rolls), and proper mission design.

The Centaur itself nominally provides a benign thermal influence to the spacecraft, with radiation environments ranging from -50 to 125°F (-45 to 52°C) and interface temperatures ranging from 40°F to 120°F (4 to 49°C) at the forward end of the spacecraft adapter. Neither the upper stage main engine plumes nor reaction control system (RCS) engine plumes provide any significant heating to the space-

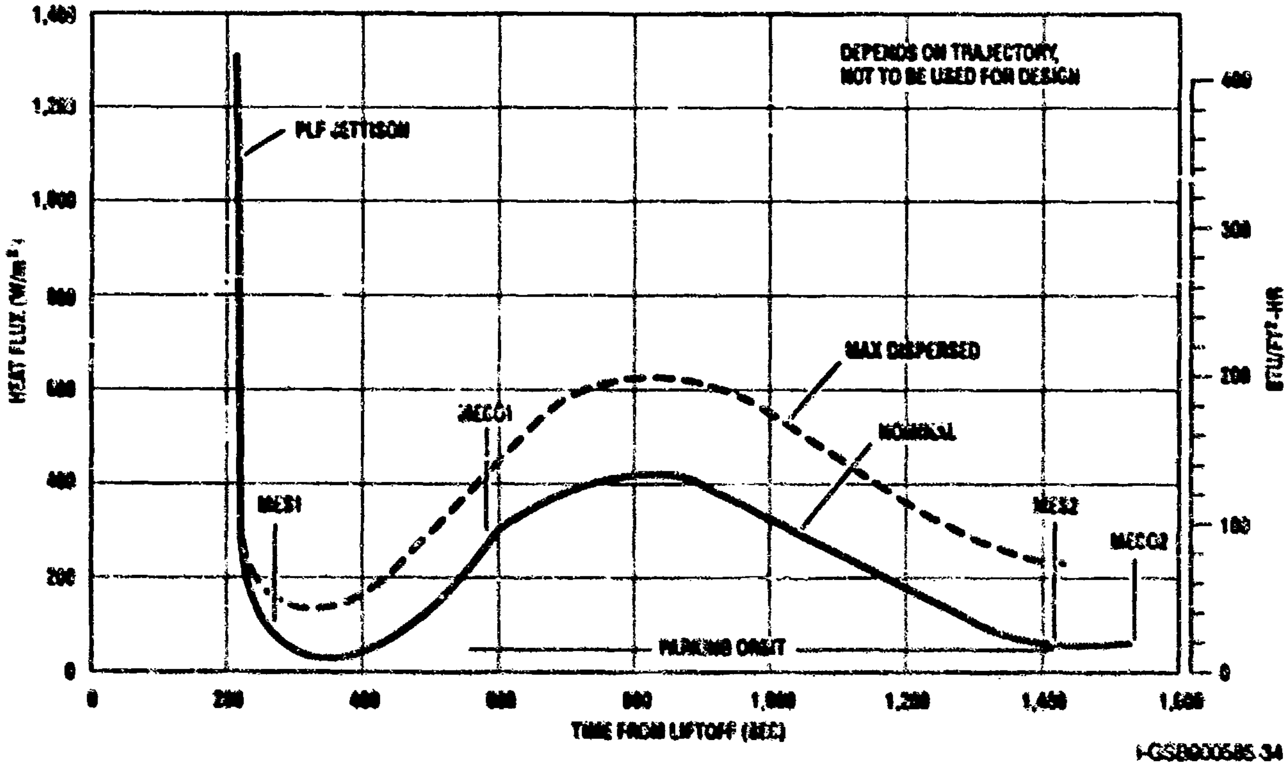

Figure 3-13. Typical free molecular heat flux profiles (example only: not to be used for design).

craft. The main engine plumes are nonluminous due to the high purity of the LH_2 and LO_2 reactants.

3.2.6 STATIC PRESSURE (PAYLOAD FAIRING VENTING) — The payload compartment is vented during boost flight through one-way vent doors. Payload compartment pressures and depressurization rates are a function of the fairing design and trajectory. The 14-ft fairing vent area was designed to have a depressurization rate of no more than 1.0 psi/sec (6.9 kPa/sec). Typical predicted pressure profiles and maximum depressurization rate profile for the 14-ft fairing are shown in Figures 3-14 and 3-15, respectively. As shown in Figure 3-15, the depressurization rate is typically less than 0.3 psi/sec (2.1 kPa/sec) except for a short period, when the launch vehicle approaches transonic speeds, during which it approaches 0.8 psi/sec (5.5 kPa/sec).

The 11-foot fairing vent area was designed for a depressurization rate of no more than 0.6 psi/sec (4.14 kPa/sec). Typical predicted pressure profiles and maximum depressurization rate profile for the 11-ft fairing are shown in Figures 3-16 and 3-17, respectively. As seen in Figure 3-17, the depressurization rate is typically less than 0.3 psi/sec (2.1 kPa/sec) except for the short transonic period, during which the depressurization rate approaches 0.6 psi/sec (4.14 kPa/sec).

The vent area of the launch vehicle payload adapters is designed assuming that the spacecraft does not vent an appreciable amount of internal volume through the payload adapter. If the user requires such venting, mission-peculiar modifications can be made.

3.2.7 CONTAMINATION CONTROL

3.2.7.1 Atlas Retrorockets — Approximately 270 seconds from launch, the Atlas sustainer is separated from Centaur. After separation, eight retrorockets near the aft end of the Atlas (Station 1133) are fired to ensure the expended Atlas stage moves away from the Centaur. These eight retrorockets use solid propellants, and exhaust products will consist of small solid particles and very low density gases.

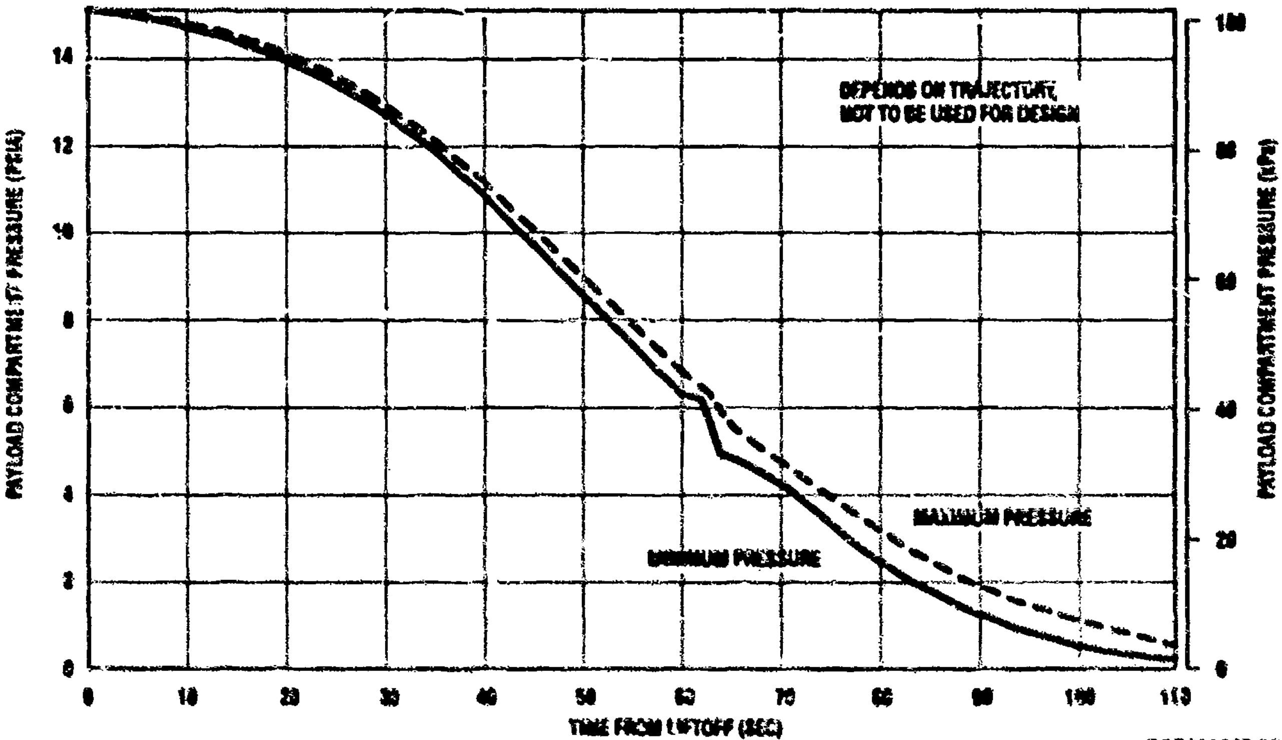

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Figure 3-14. Typical static pressure profiles inside 14-foot fairing

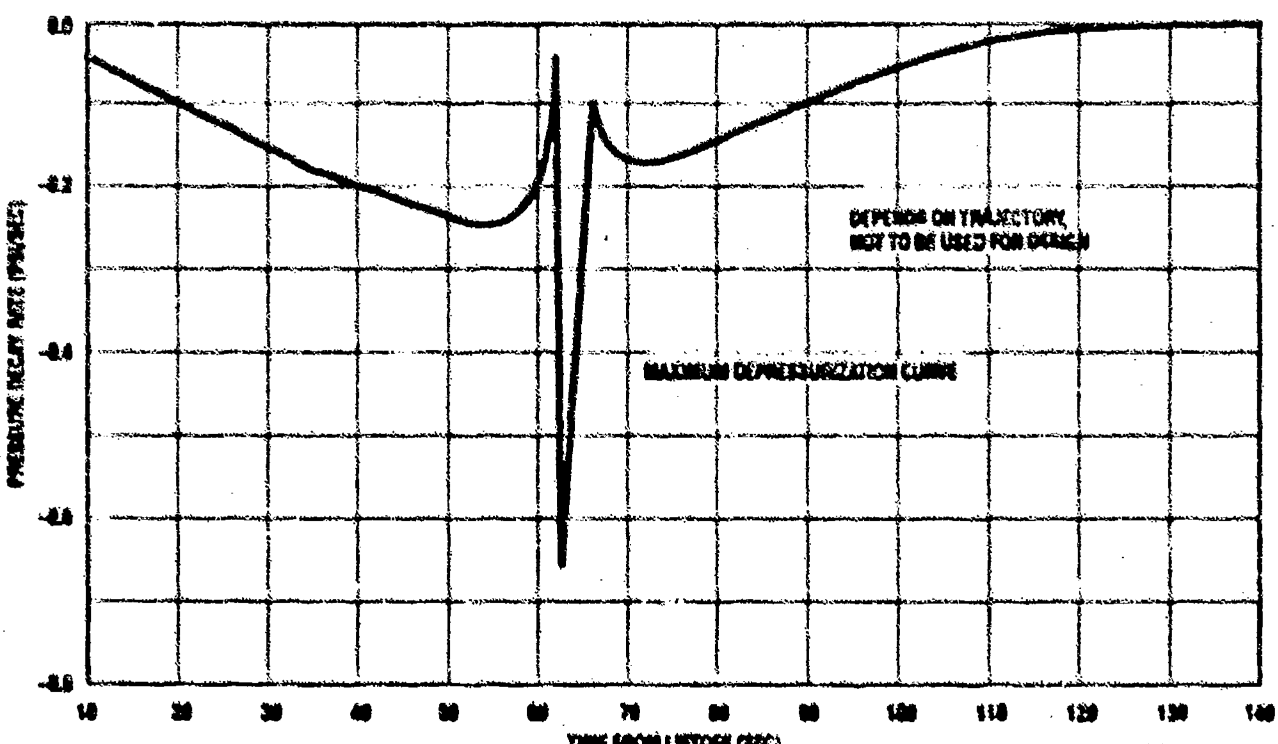

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Figure 3-15. Typical payload compartment pressure decay rate vs time from liftoff for the 14-ft payload fairing

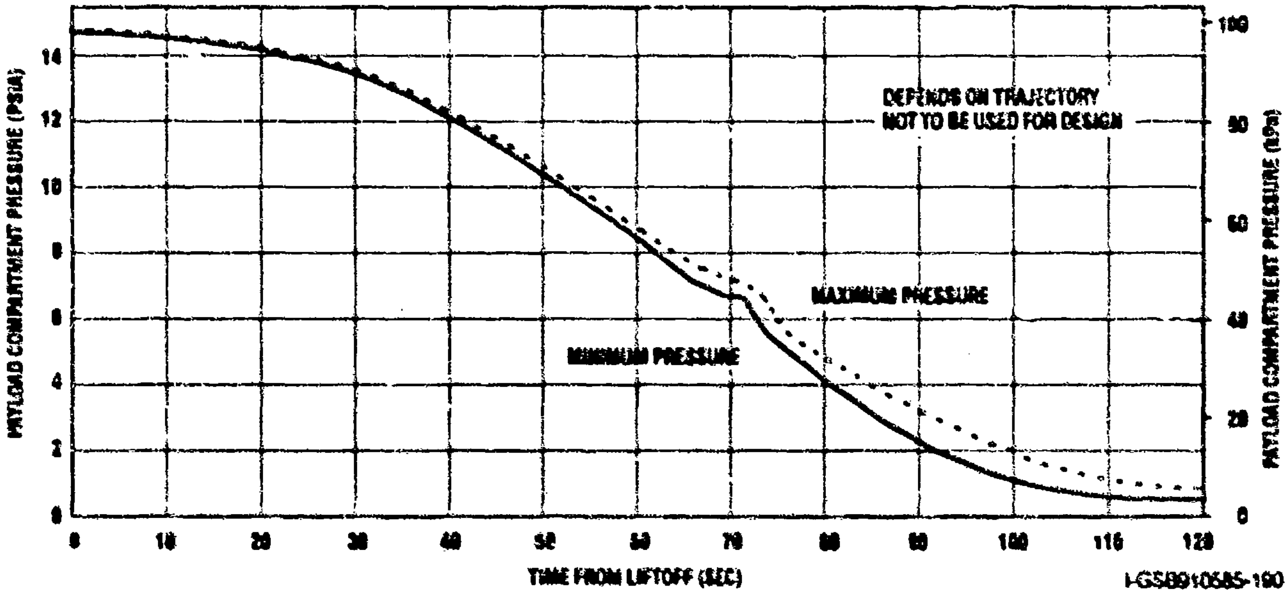

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Figure 3-16. Typical payload compartment pressure versus time of flight for the 11-ft payload fairing.

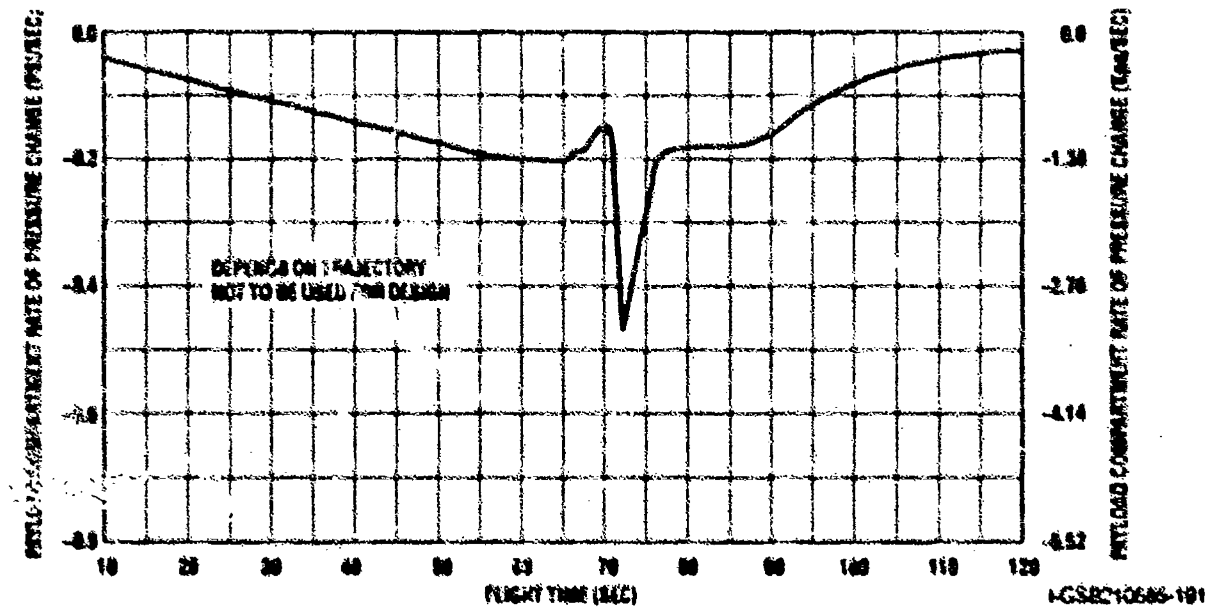

Figure 3-17. Typical payload compartment pressure decay rate versus time from liftoff for the 11-ft payload fairing.

The retrorocket nozzles are canted outboard 40 degrees from the thrust axis. This cant angle ensures that virtually no solid particles will impact the spacecraft. Exhaust gases that impinge upon the spacecraft are rarefied and should not be a contamination concern, as the spacecraft surfaces are still relatively warm from prelaunch payload compartment gas conditioning and the exhaust gases do not recondense.

3.2.7.2 Upper Stage Reaction Control System (RCS) — The upper stage RCS consists of twelve 6-lbf (27 N) hydrazine (N_2H_4) thrusters for settling, roll, and attitude control requirements. Four thrusters provide axial thrust, four provide pitch control, and four provide yaw control. They are located slightly inboard on the upper stage aft bulkhead.

Prior to upper stage/spacecraft separation, the spacecraft will not be exposed to RCS exhaust plumes. The RCS thruster's inboard location on the aft bulkhead precludes direct line of access between the spacecraft and thrusters.

After separation, some minor spacecraft impingement from thruster exhaust plumes may occur during the collision/contamination avoidance maneuver (CCAM). CCAM is designed to move the upper stage a safe distance from the spacecraft after separation.

A typical CCAM sequence is shown in Figure 3-18. This figure shows typical spacecraft motion after the separation event as longitudinal and lateral distance from the upper stage. Included are contour lines of constant flux density for the plumes of the aft-firing RCS settling motors during operation. The plumes indicate the relative rate of hydrazine exhaust product impingement on the spacecraft during the 2S-ON phase prior to blowdown and during hydrazine depletion. There is no impingement during the CCAM 4S-ON phase because the spacecraft is forward of the settling motors.

3.2.7.3 Upper Stage Main Engine Blowdown — As part of the CCAM, hydrogen and oxygen are expelled through the engine system to further increase upper stage/spacecraft separation distance. Hydrogen is expelled out the engine cooldown ducts, and oxygen is expelled out the main engine bells. The expelled products are hydrogen, oxygen, and trace amounts of helium, which are noncontaminating to the spacecraft. Figure 3-19 identifies typical main-

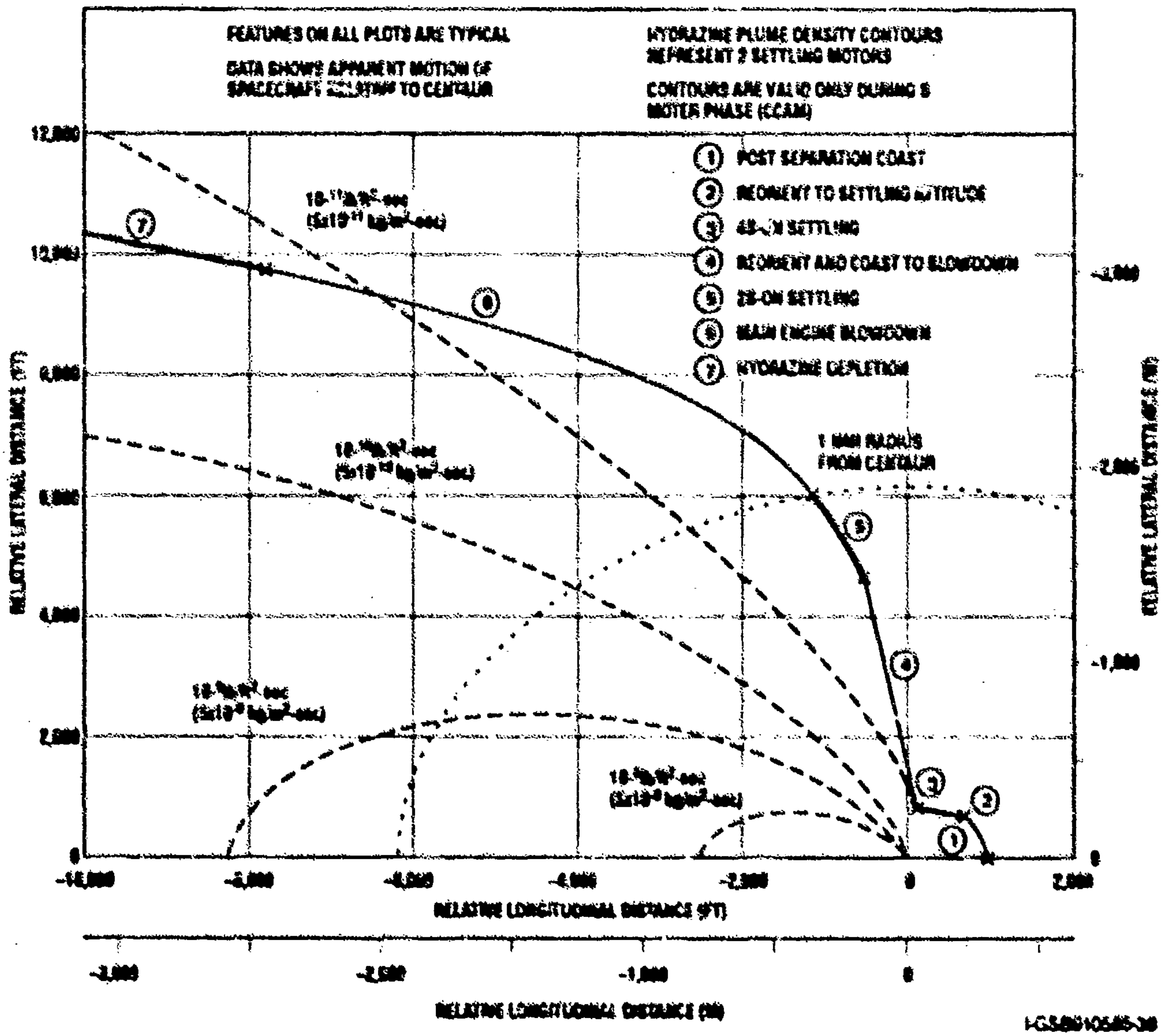

Figure 3-18. Typical spacecraft motion relative to Centaur upper stage.

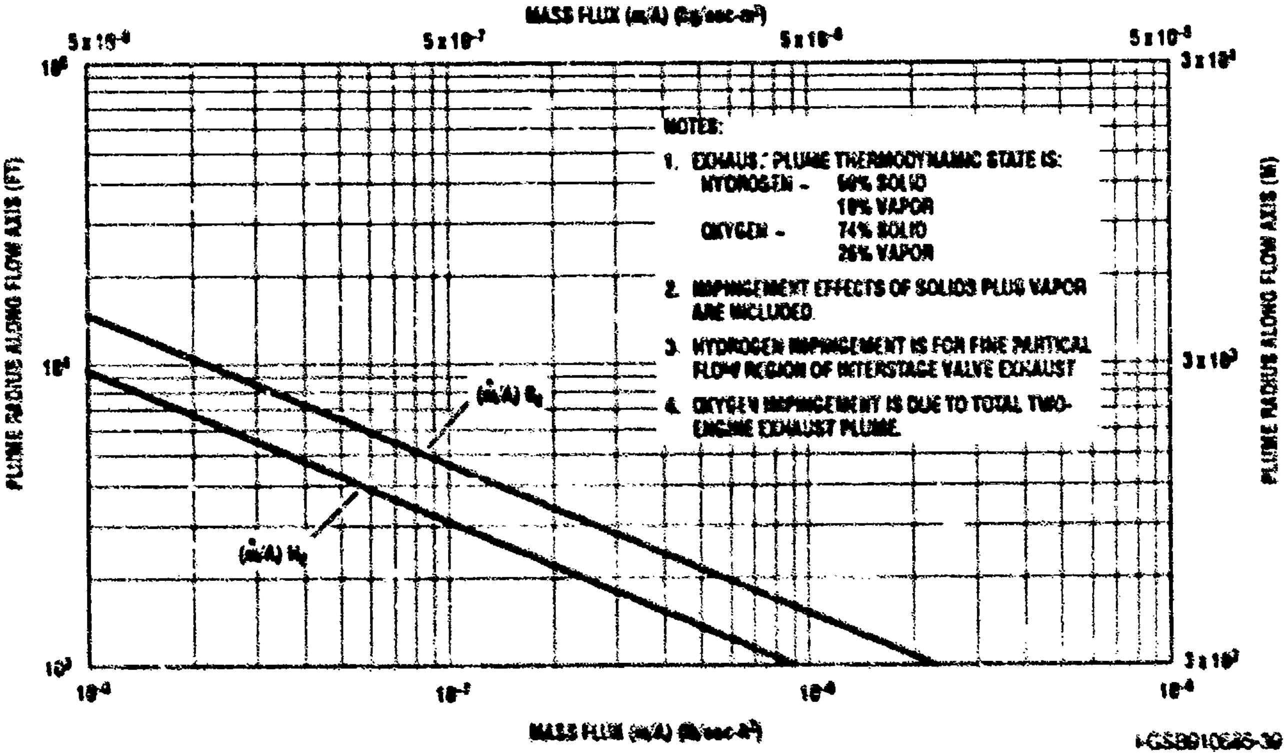

Figure 3-19. Typical spacecraft impingement fluxes during main engine blowdown.

engine blowdown exhaust product impingement rates on the spacecraft.

3.2.8 RADIATION AND ELECTROMAGNETIC COMPATIBILITY — The description of environments provided in Section 3.1.2 encompasses worst-case flight environments.

3.3 SPACECRAFT COMPATIBILITY TEST REQUIREMENTS

Spacecraft Structural Qualification and Acceptance Tests — General Dynamics requires that the spacecraft contractor provide the appropriate data to indicate compliance with the Atlas launch vehicle environments through their tests and analyses.

Spacecraft structural capability includes items such as shock, static load, modal survey, sine vibration, and acoustic tests and by the coupled loads analysis results. The coupled loads analysis is performed by General Dynamics using a customer-provided and approved math model of the spacecraft. Spacecraft test levels may be modified by coupled loads analysis results.

General Dynamics also suggests that the spacecraft contractor demonstrate the spacecraft capability to withstand the thermal and EMI/EMC environments.

Flight hardware fit checks are performed to verify the mating interfaces and envelopes. Table 3-4 identifies the suggested spacecraft qualification and acceptance tests to ensure adequate compliance with the Atlas environments. Specific test levels and margins, based on the flight environments provided, are provided early in the integration process.

Table 3-4 Spacecraft qualification and acceptance test requirement.

	ACOUSTIC	SHOCK	SINE VIBRATION	THERMAL	EMI/EMC	MODAL SURVEY	STATIC LOADS	FIT CHECK
QUALIFICATION	✓	✓	✓	✓	✓	✓	✓	
ACCEPTANCE	✓							✓

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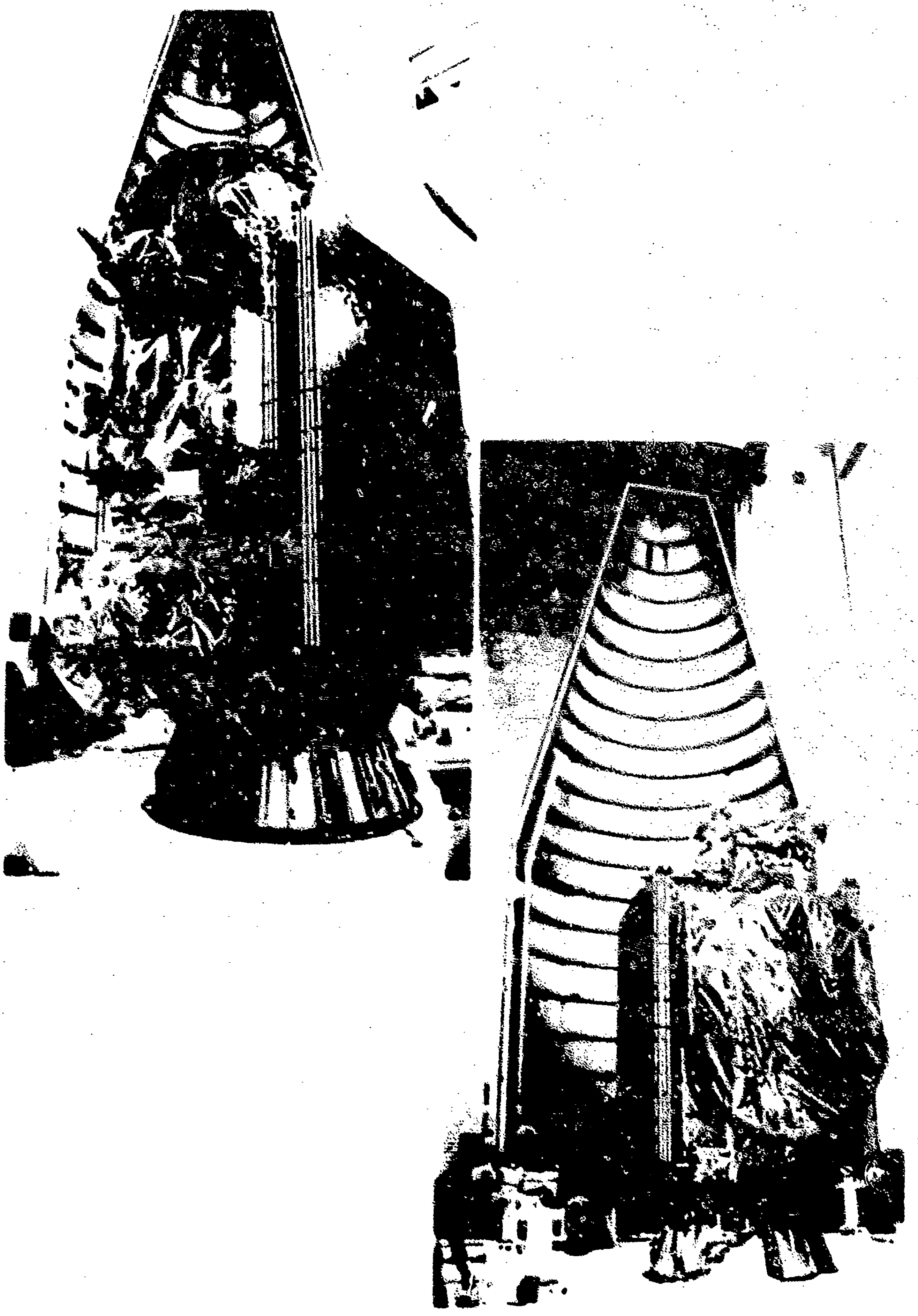


4 • SPACECRAFT INTERFACES

4.1 SPACECRAFT-TO-LAUNCH VEHICLE INTERFACES

All envelopes and adapter information given is to be used only as a guideline. Ultimate control of this information is through the mechanical interface control document. Slight modifications to the envelopes and adapters can be accommodated on a mission-peculiar/unique basis.

4.1.1 MECHANICAL INTERFACE—PAYLOAD FAIRINGS — The Atlas user has a choice between the large and medium payload fairing configurations.

4.1.1.1 Large Payload Fairing — The large payload fairing (LPF) is a two-half-shell structure, as shown in Figure 4-1. The structure consists of aluminum skin/stringer/frame construction with vertical, split-line longerons. The fairing cylindrical section is attached to the launch vehicle with the conical boattail section. The cylindrical section provides 13.8 feet (4.19 m) of length for the spacecraft. The cylindrical section is topped by a conical nose cone and a spherical cap. The 14.5-degree angle of the cone minimizes combined aerodynamic drag and weight losses.

The fairing and boattail provide a protective enclosure for the payload and Centaur equipment module packages during prelaunch and ascent. Portions of the external surfaces of the fairing are insulated with cork to limit temperatures to acceptable values. Noncontaminating thermal control coatings are used on internal surfaces to minimize incident heat fluxes to the spacecraft.

The fairing and boattail also provide mounting provisions for various other systems. Payload compartment cooling system provisions are contained in the cylindrical portion of the fairing. Electrical packages required for the fairing separation system are mounted on the internal surface of the boattail. Ducting for the upper stage hydrogen tank venting system and cooling ducts for the equipment module packages are also attached to the boattail. Four large doors on the boattail provide access to Centaur equipment module packages and the lower portion of the payload compartment.

The following mission-peculiar items can be provided: a thermal shield, acoustic blankets, spacecraft access doors, and an RF reradiating antenna.

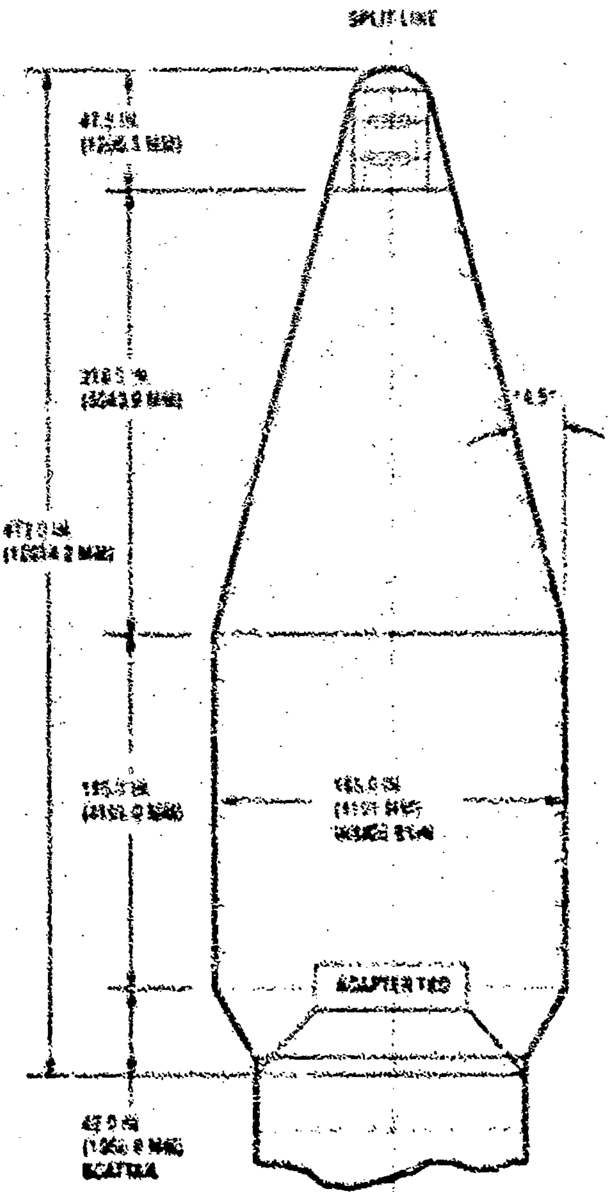


Figure 4-1. Large (14-foot) payload fairing configuration

The spacecraft thermal and acoustic environments can be tailored for each mission. Access doors in the fairing are provided as necessary to meet spacecraft requirements. RF reradiating antennas can be provided if required for RF access to the spacecraft.

4.1.1.2 Medium Payload Fairing — For payloads that do not require the volume of the large payload fairing, a 10.8-ft (3.3-m) diameter fairing is available (Figure 4-2).

The medium payload fairing (MPF) is also a two-half-shell structure consisting of skin/stringer/frame construction similar to the LPF. The fairing cylindrical section is attached to the launch vehicle with the split barrel. The cylindrical section provides 12.8 feet (3.9 m) of length for the spacecraft. The cylindrical section is topped by a conical nose cone and a spherical cap. Both the cone and cap are the same as the LPF.

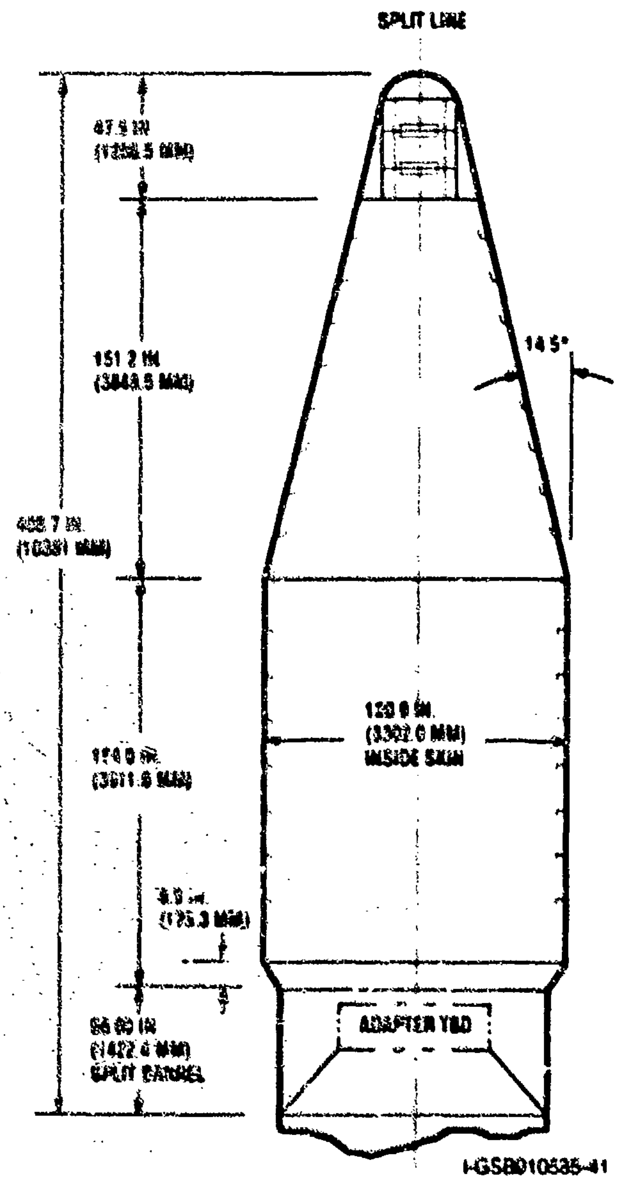


Figure 4-2. Medium (11-foot) payload fairing configuration

The MPF and the split barrel provide spacecraft protection (thermal, acoustic, electromagnetic, and environmental) similar to the LPF and boattail.

The MPF and split barrel include mounting provisions for equipment and systems similar to those in the LPF. Four large doors in the split barrel provide access to the Centaur equipment module packages and the lower portion of the payload compartment.

4.1.1.3 Spacecraft Static Envelopes — Figure 4-3 shows the usable spacecraft static volume provided by the large payload fairing. This envelope provides a usable diameter of 143.7 inches (3650 mm) in the cylindrical portion of the fairing. The aft portion of the envelope is reduced to allow for jettison clearances of the payload fairing hardware. The aft portion of the envelope around the spacecraft adapter varies for each user, depending on which spacecraft adapter configuration is used.

The usable static volume provided by the medium payload fairing envelope is shown in Figure 4-4. This provides a usable diameter of 115 inches (2921 mm) in the cylindrical portion of the fairing. The aft portion of the envelope is reduced to allow for jettison clearances of the fairing hardware. The aft portion of the envelope around the spacecraft adapter varies for each user, depending on which spacecraft adapter configuration is used.

Envelopes surrounding Type A, Type A1, Type B, Type B1, Type C, Type C1, and Type D adapters and the equipment module are shown in Figures 4-5 through 4-11. The spacecraft envelopes defined in these figures represent the maximum allowable spacecraft static dimensions (including manufacturing tolerances) relative to the spacecraft/attach fitting interface. Spacecraft dynamic deflections have been taken into account in arriving at these envelopes assuming that spacecraft primary structure first lateral modes are above 10 Hz and first axial modes are above 15 Hz. In addition to spacecraft dynamic deflections, these envelopes include allowances for payload fairing static and dynamic deflections, manufacturing tolerances, out-of-round conditions, and misalignments. With these assumptions, a minimum one-inch clearance between the spacecraft and the payload fairing is ensured. Reduced clearances may be permitted in the area of the spacecraft-to-launch vehicle interface and will be analyzed on a mission-peculiar basis.

Clearance layouts and analyses are performed for each spacecraft configuration and, if necessary, critical clearances are measured after the fairing is installed to ensure positive clearance during flight. To accomplish this, it is important for the spacecraft description to include an accurate physical location of all points on the spacecraft that are within 2 in. of the allowable envelope. The dimensions or spacecraft model must include the maximum manufacturing tolerances.

For spacecraft secondary structure (i.e., unsecured antennas, thermal shields, etc.) in the vicinity of the envelope, or for protrusions that may extend outside the envelopes shown, coordination with General Dynamics is required to define appropriate envelopes.

4.1.1.4 Spacecraft Accessibility — The four large doors in the aft (boattail or split barrel) portion of the payload fairings will provide primary access to the encapsulated spacecraft. These doors (Figure 4-12) provide an access opening approximately 30 inches (762 mm) wide by 26 inches (660 mm) tall (LPF boattail) and 40 inches (1016 mm) wide by 36 inches (914.4 mm) tall (MPF split barrel) and are located one per fairing quadrant. Work platforms can be inserted through these doors into the payload compartment to allow access to spacecraft hardware near the aft end of the payload compartment.

If additional access to the spacecraft is required, doors can be provided on a mission-peculiar basis on the cylindrical section of each payload fairing. Doors can be located in most areas of the fairing cylindrical sections except near the split lines and interface planes. Typical access doors are shown in Figure 4-12 for each fairing.

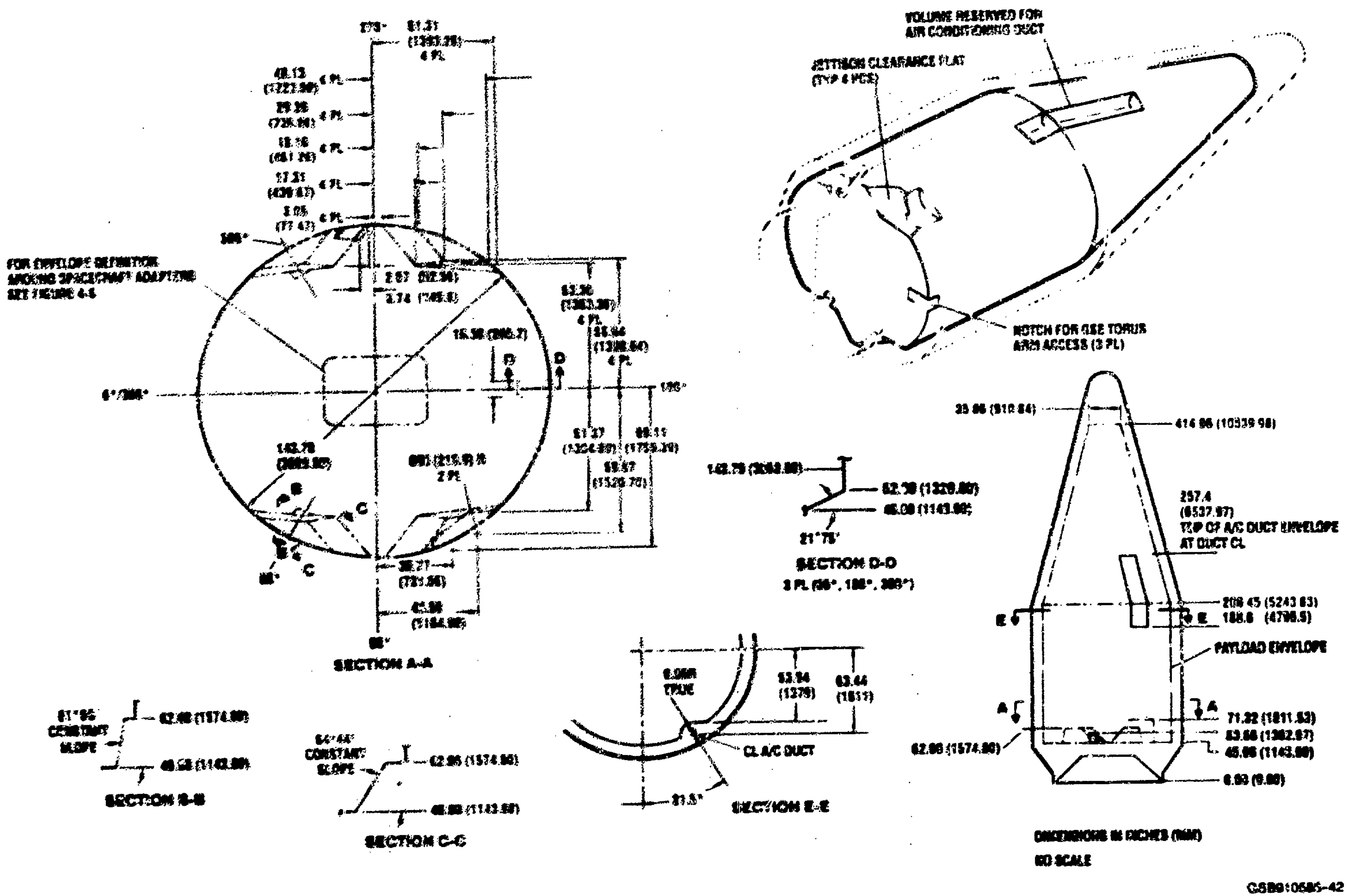
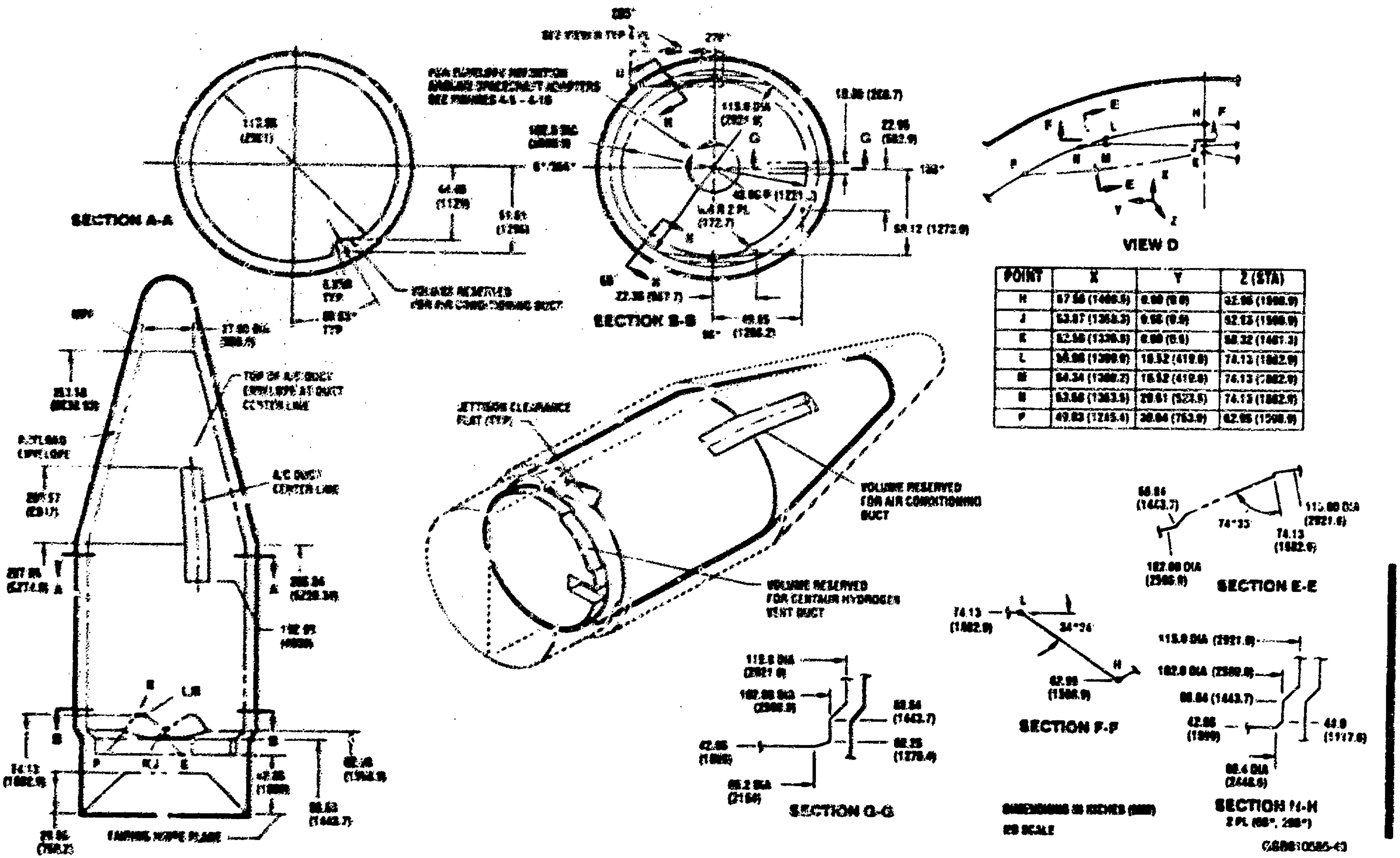


Figure 4-3. Large fairing payload envelope.



POINT	X	Y	Z (STA)
H	57.56 (1462.0)	0.00 (0.0)	62.95 (1598.9)
J	53.87 (1368.3)	0.00 (0.0)	62.13 (1598.9)
K	52.56 (1335.0)	0.00 (0.0)	58.32 (1481.3)
L	54.96 (1396.0)	16.52 (419.6)	74.13 (1882.9)
M	54.34 (1380.2)	16.52 (419.6)	74.13 (1882.9)
N	53.66 (1363.0)	20.61 (523.5)	74.13 (1882.9)
P	49.03 (1245.4)	30.04 (763.0)	62.95 (1598.9)

Figure 4-4. Medium fairing payload envelope.

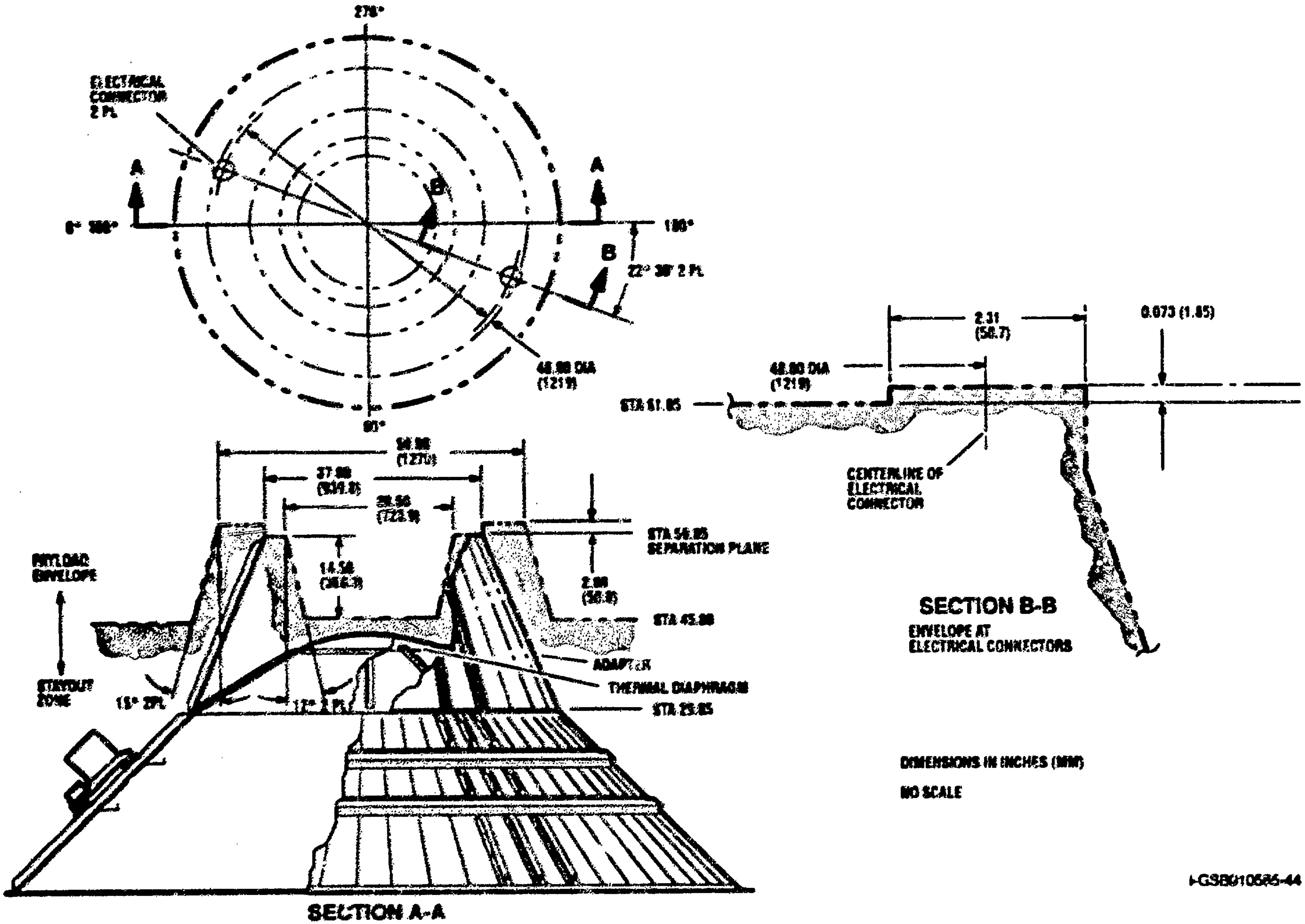


Figure 4-5 Type A payload adapter usable envelope

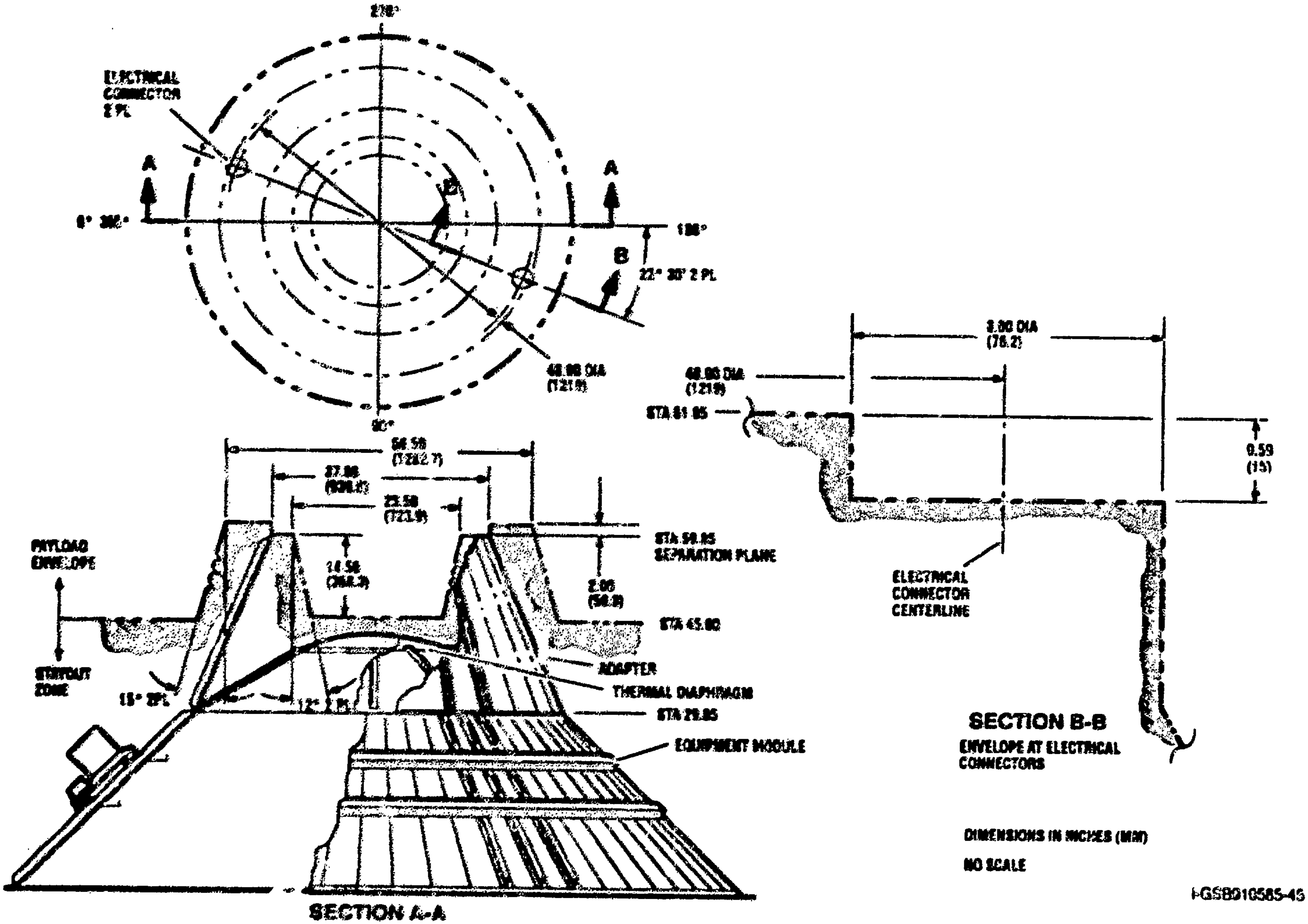


Figure 4-6. Type A1 payload adapter usable envelope.

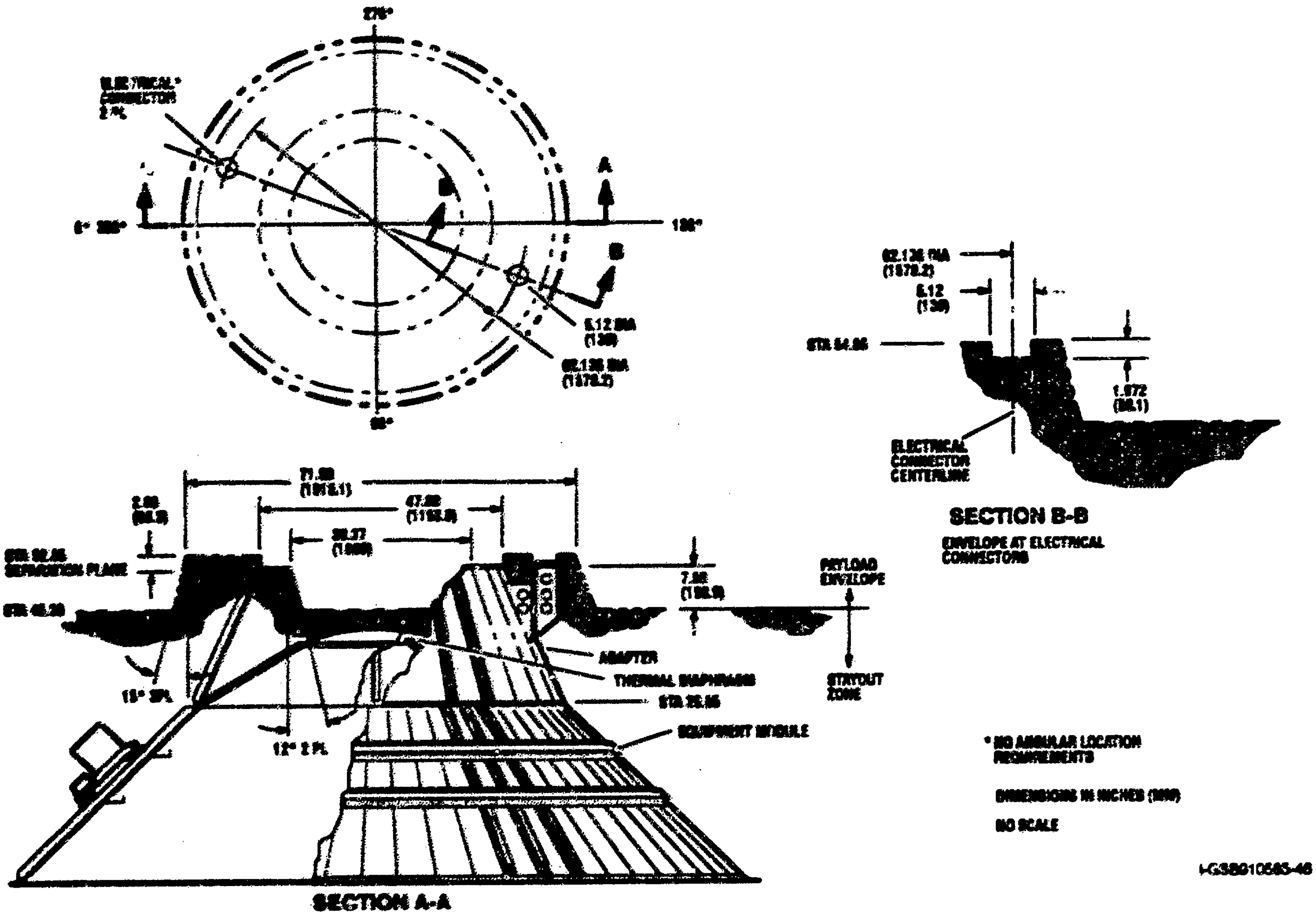


Figure 4-7. Type B payload adapter usable envelope.

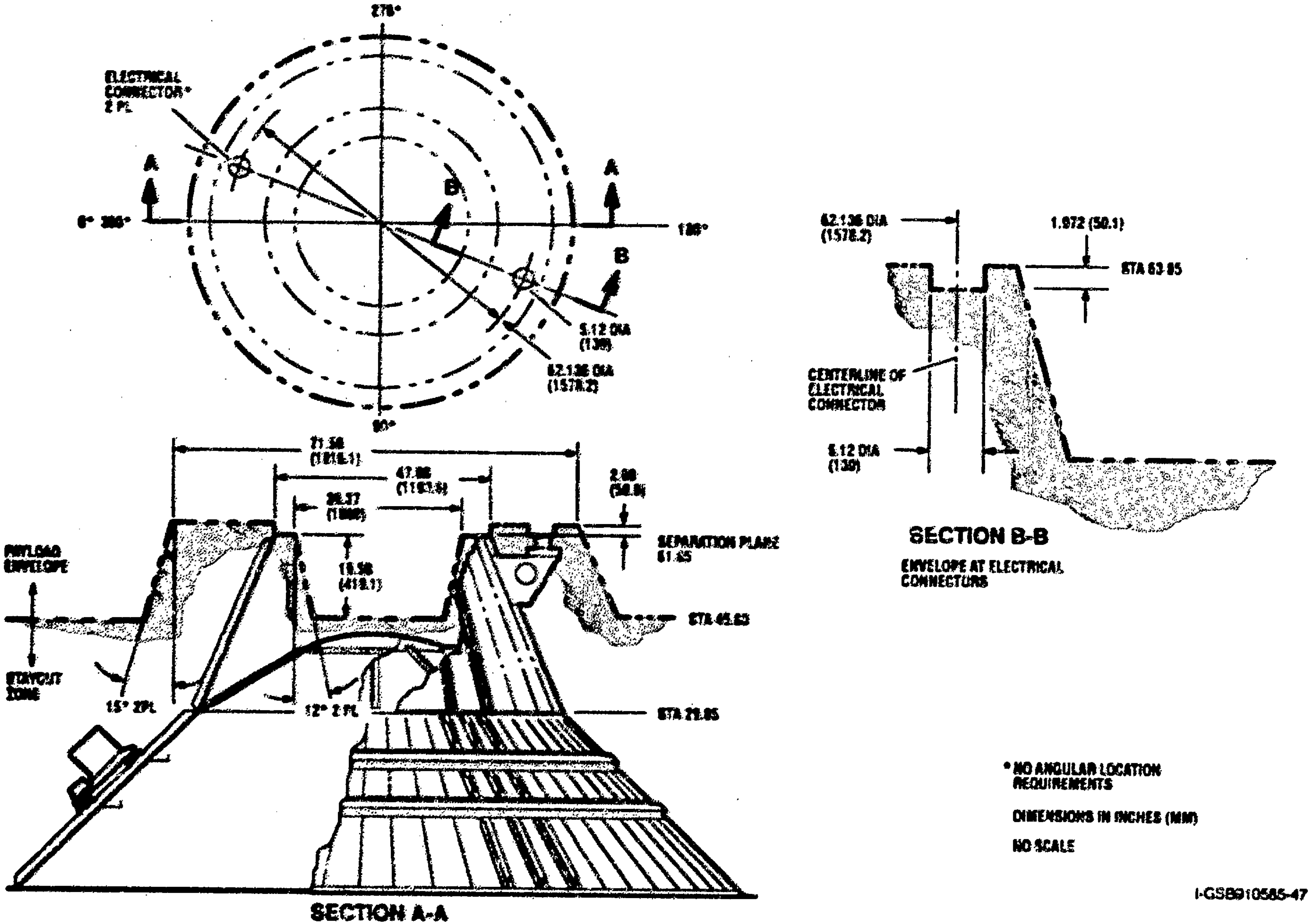


Figure 4-8. Type B1 payload adapter usable envelope.

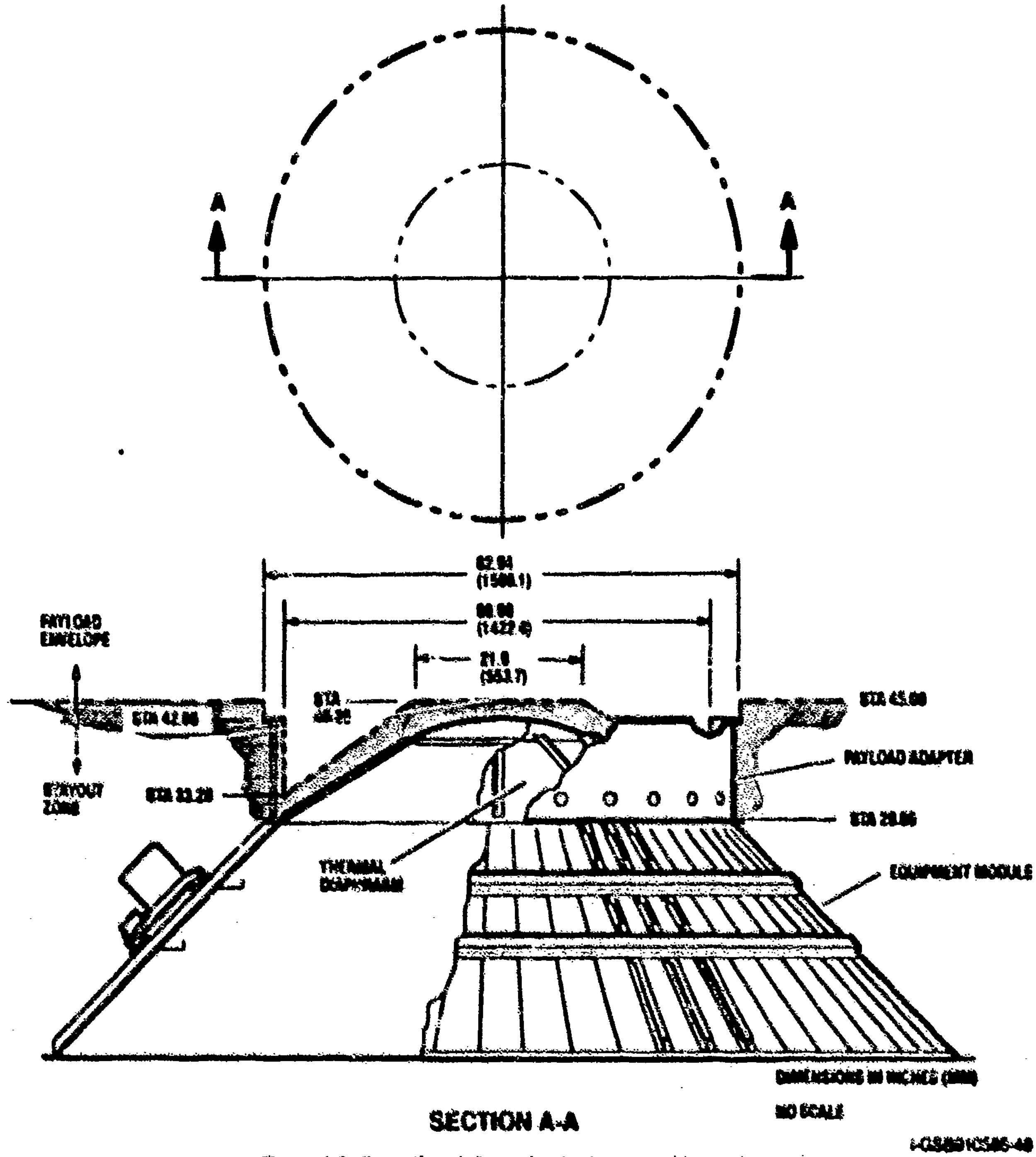


Figure 4-9. Types C and C1 payload adapter usable envelope.

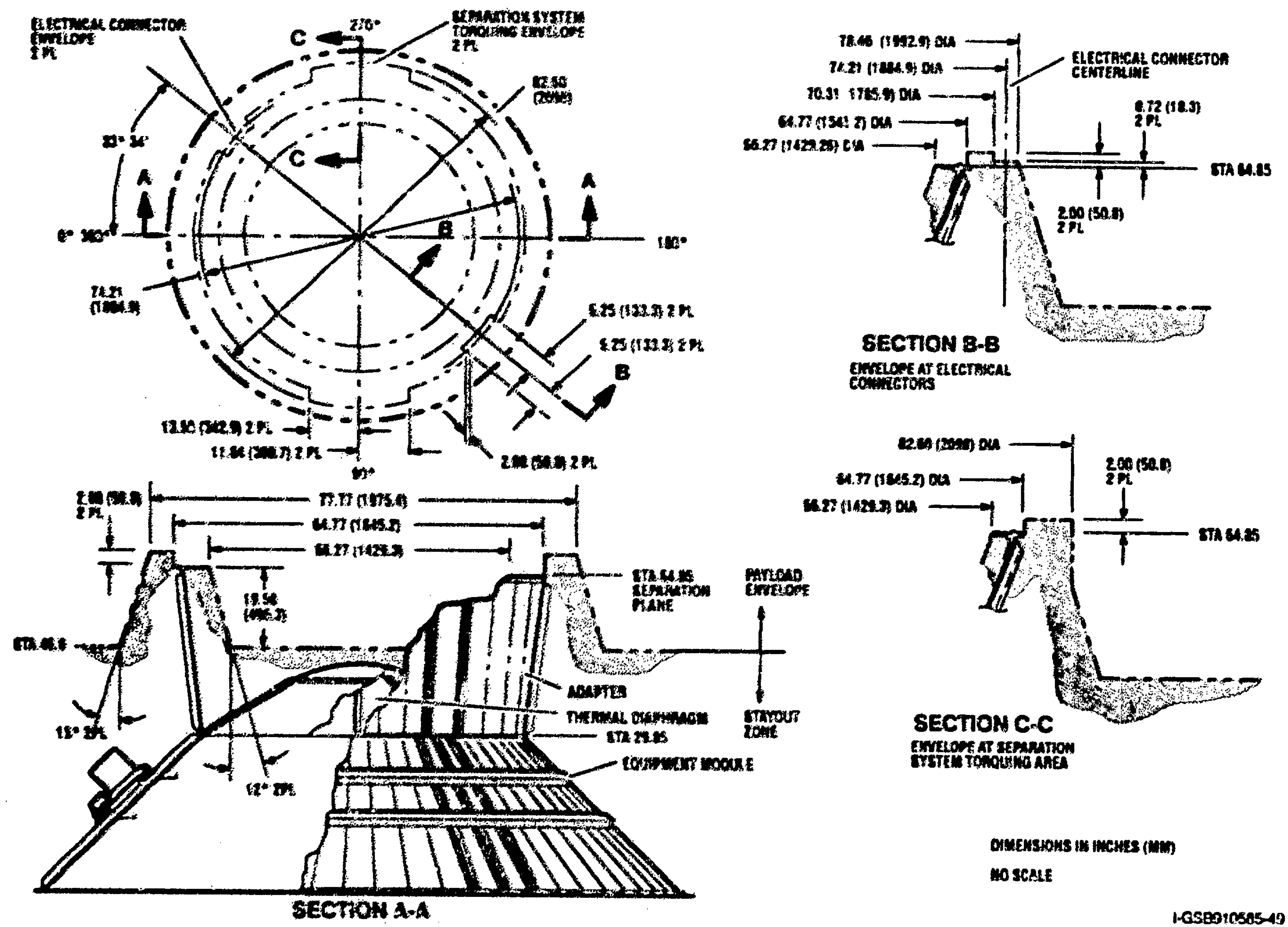


Figure 4-10. Type D payload adapter usable envelope.

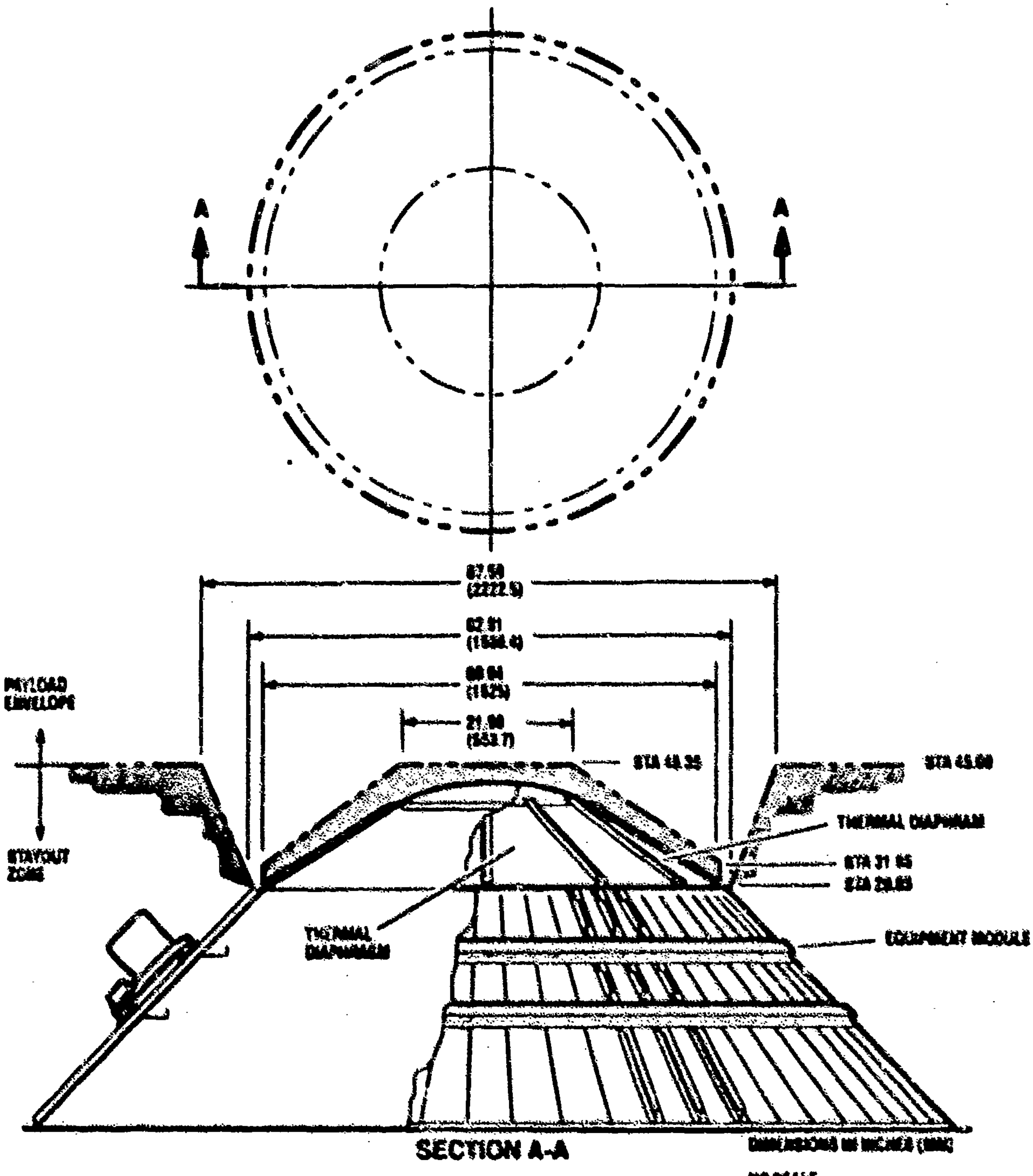


Figure 4-11 Equipment module usable envelope.

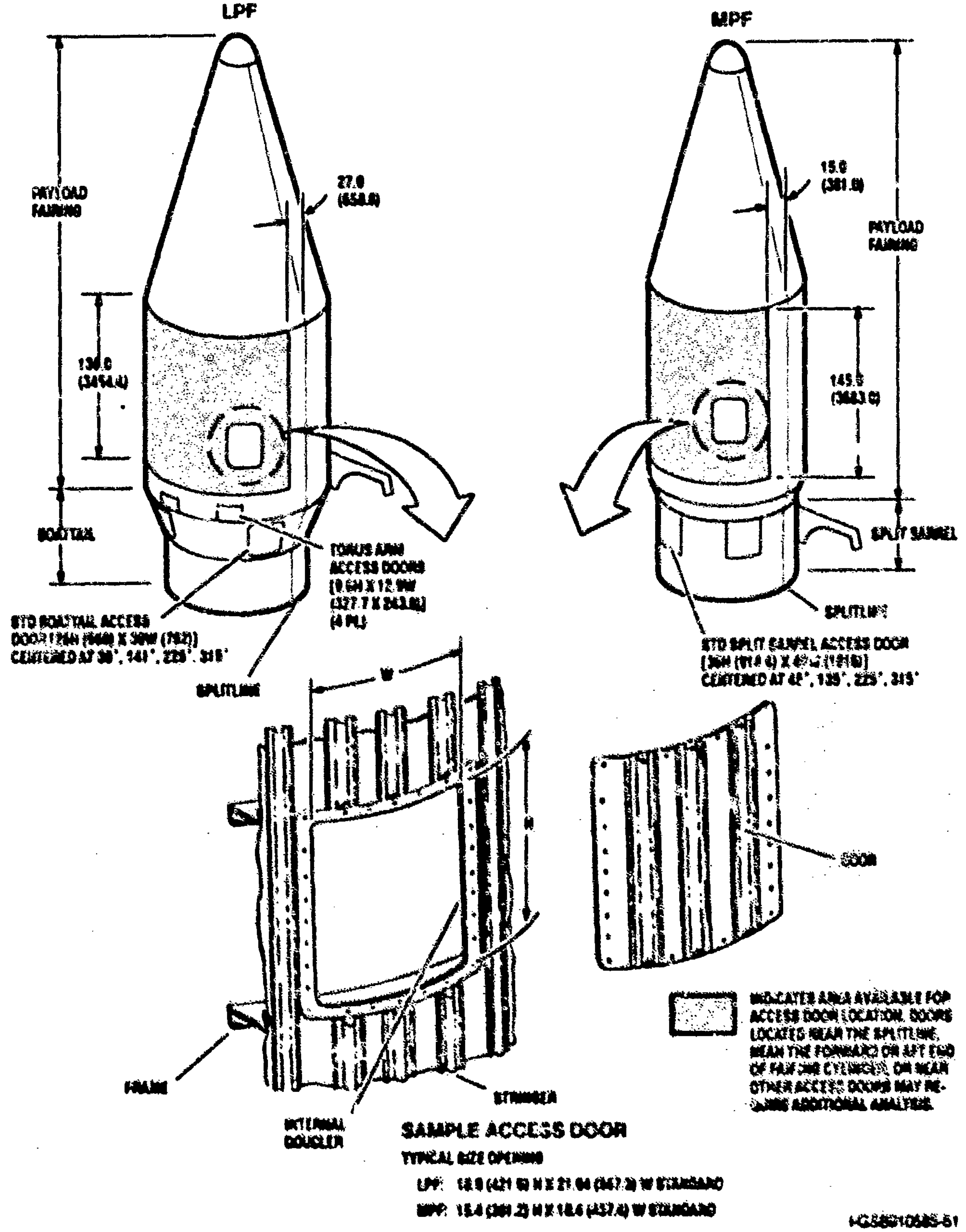


Figure 4-12. Fairing access doors allow access to the encapsulated spacecraft.

4.1.2 MECHANICAL INTERFACES — SPACECRAFT ADAPTERS — The spacecraft adapter interfaces typically consist of the basic spacecraft-to-launch vehicle attach ring with its securing provisions, the two spacecraft rise-off disconnects, and the separation springs. Electrical bonding is provided across all mechanical interface planes associated with these adapters. It should be noted that the clocking of the spacecraft adapter and separation system is not fixed but is determined based on spacecraft orientation. However, it is recommended that the separation system keyway on the Type A adapter be clocked in 6-deg increments relative to the launch vehicle axis.

4.1.2.1 Types of Adapters — The spacecraft adapter, equipment module, or other spacer adapters provide the mechanical interfaces between the spacecraft and launch vehicle. With the Type A, A1, B, B1, and D adapters, the launch vehicle provides the spacecraft separation system. For the user requiring an interface other than the above adapters, a bolted interface is provided by the equipment module and Type C and C1 adapters. If a customer-provided spacecraft adapter is used, it must provide interfaces for ground handling, encapsulation, and transportation equipment. In particular, there will need to be three torus arm fittings and an encapsulation diaphragm unless a GDSS intermediate adapter is employed. Figures 4-13a through 4-13g show the interfaces for the adapters. Figure 4-14 shows the interfaces for the equipment module. Type A adapter interfaces are compatible with the PAM-DII/937B adapters; Type A1 adapter interfaces are compatible with the PAM-D/937A; and Type B and B1 adapter interfaces are compatible with the 1194A adapter. The Type C1 adapter is a spacer adapter and provides the same bolt pattern as the equipment module. The spacecraft adapter also provides mounting for some of the mission-peculiar hardware. The spacecraft electrical umbilicals and range safety destruct unit both mount on the payload adapter. Type D adapter interfaces are compatible with the 1666A adapter. Alternate adapter designs can be developed on a mission-peculiar basis.

4.1.2.2 Interface Rings — The interface rings for the Type A, A1, B, B1, and D adapters are designed to provide for mounting of a V-band clamp separation system described below. The spacecraft ring interfaces with the spacecraft adapter ring and the V-band clamp holds the two together for the structural joint. Figures 4-15a through 4-15d show the interface ring requirements for the Type A, A1, B, B1, and D spacecraft adapters. The interface rings for the equipment module, Type C and C1 adapters, and other spacer adapters provide a bolt circle with which the spacecraft adapter will mate.

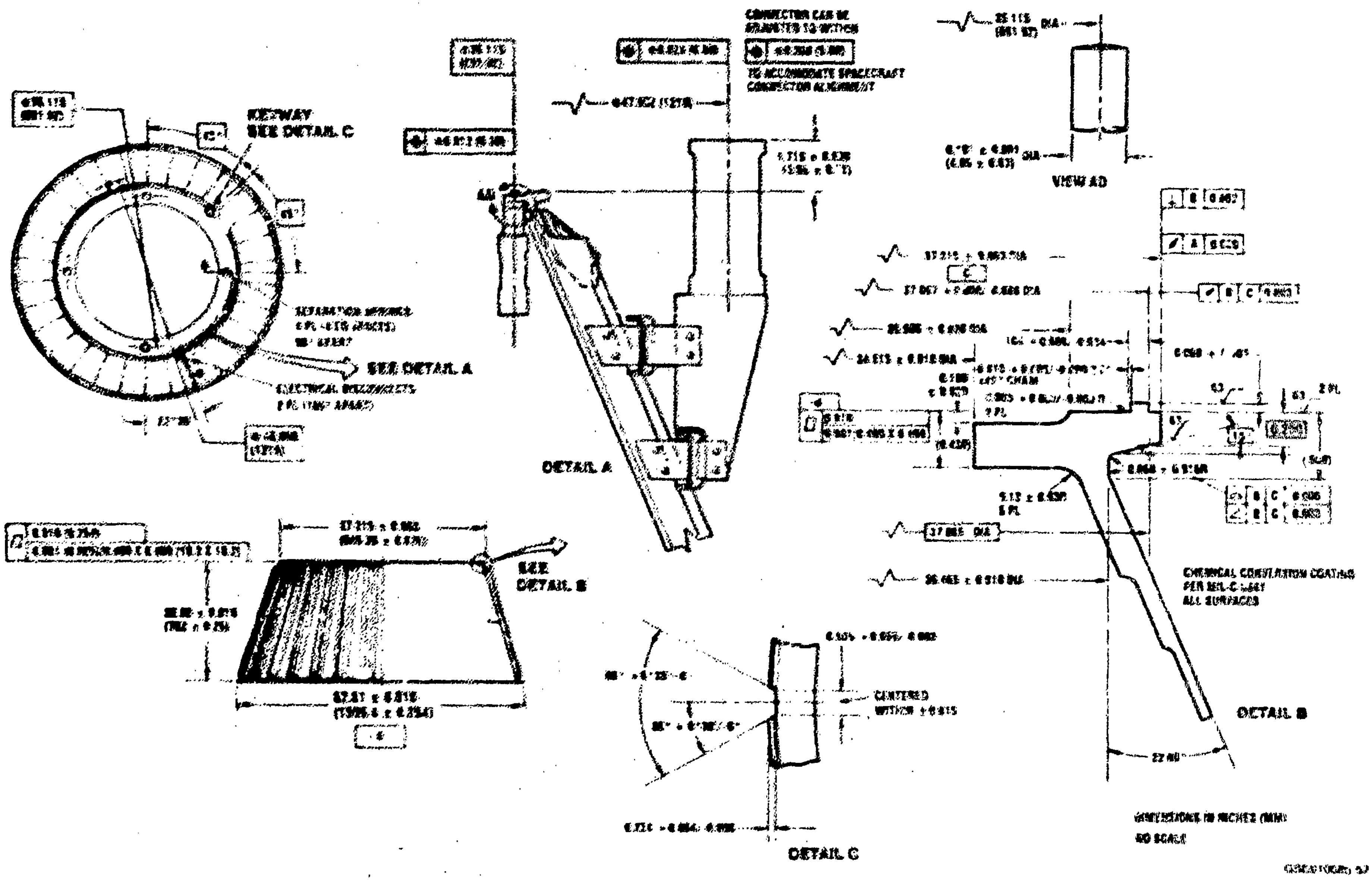


Figure 4-13a. Type A spacecraft adapter mechanical interfaces

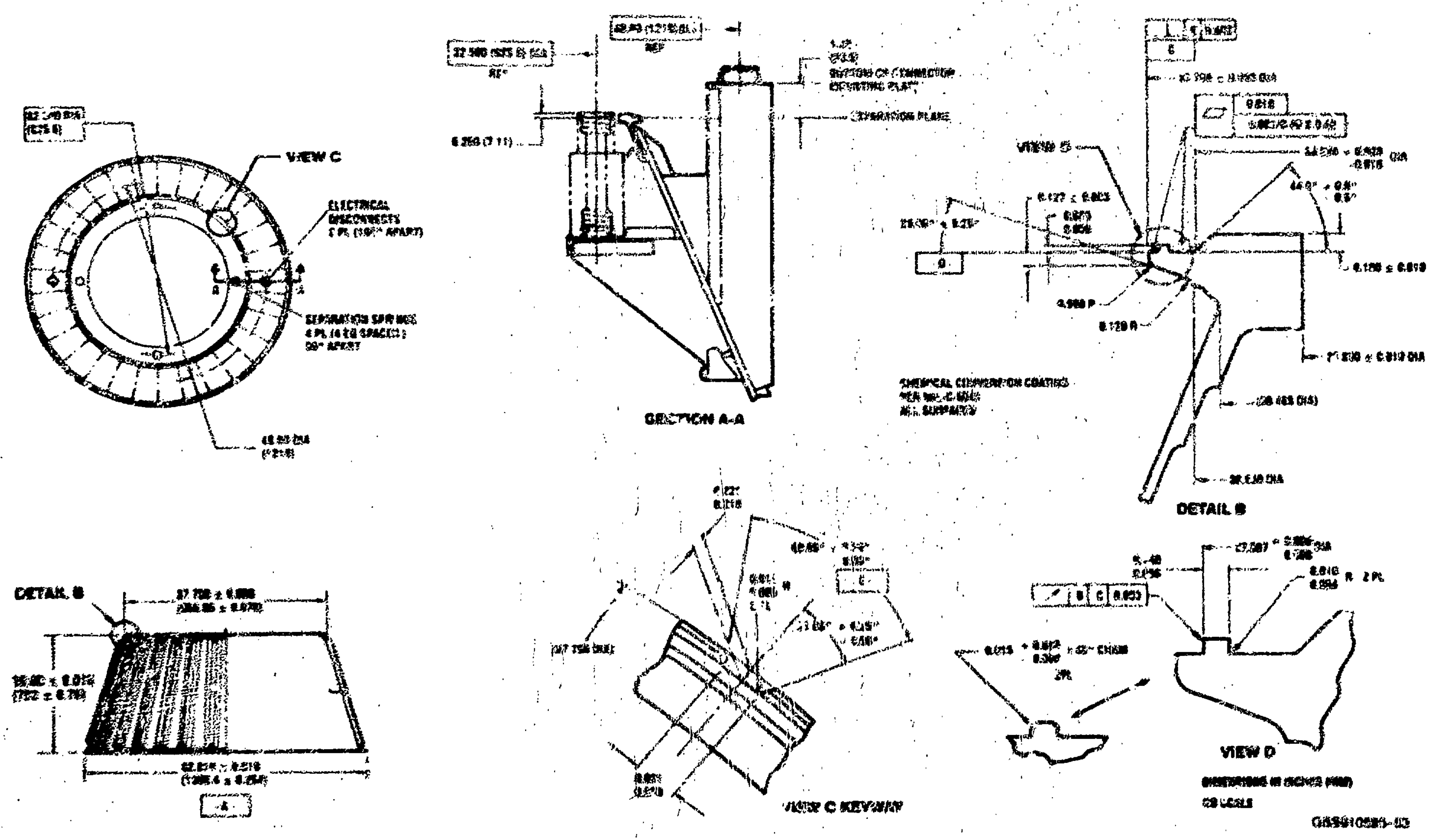


Figure 4-13b. Type A1 spacecraft adapter mechanical interfaces.

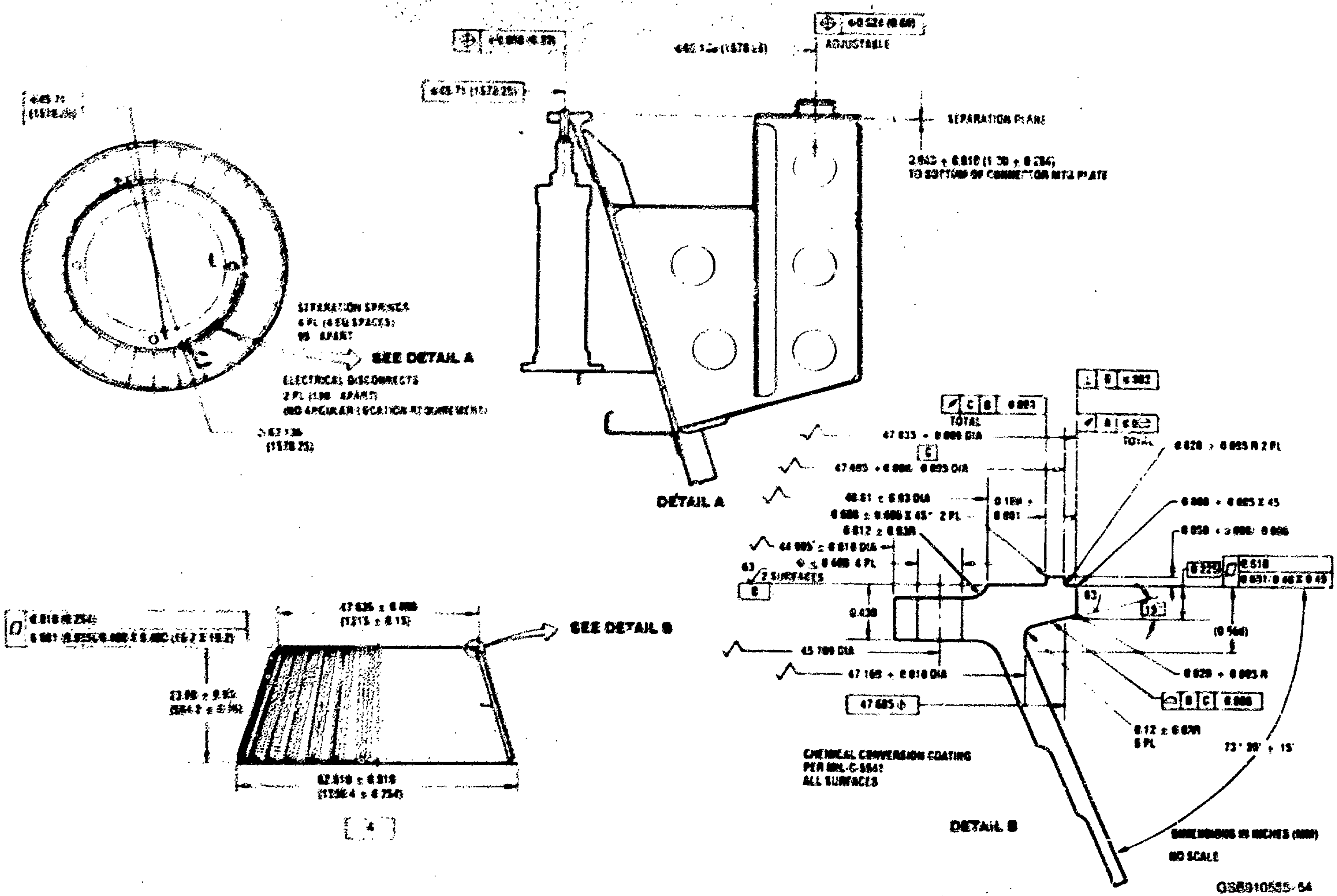


Figure 4-13c Type B spacecraft adapter mechanical interfaces.

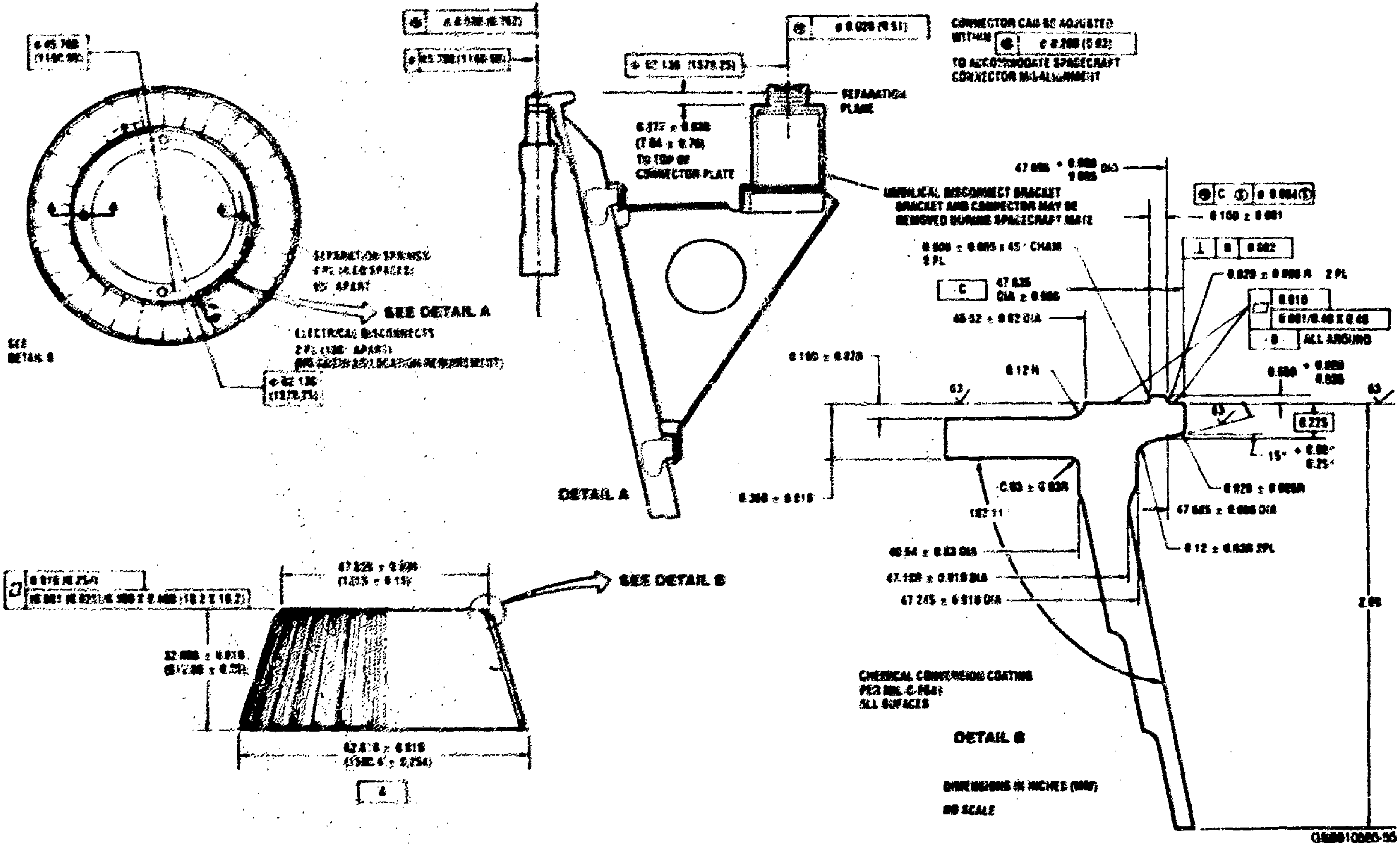


Figure 4-13d Type B1 spacecraft adapter mechanical interfaces.

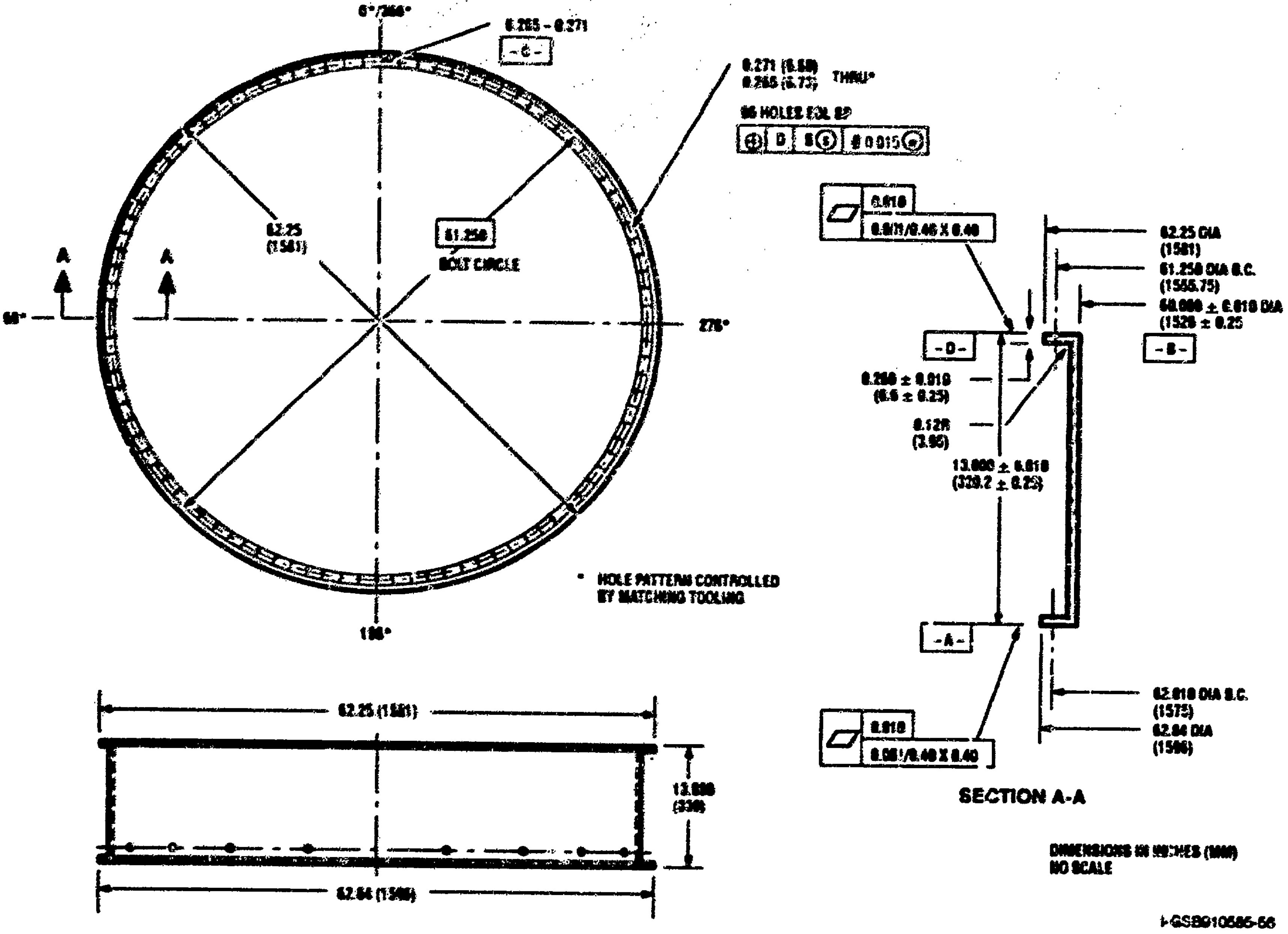


Figure 4-13e. Type C spacecraft adapter mechanical interfaces.

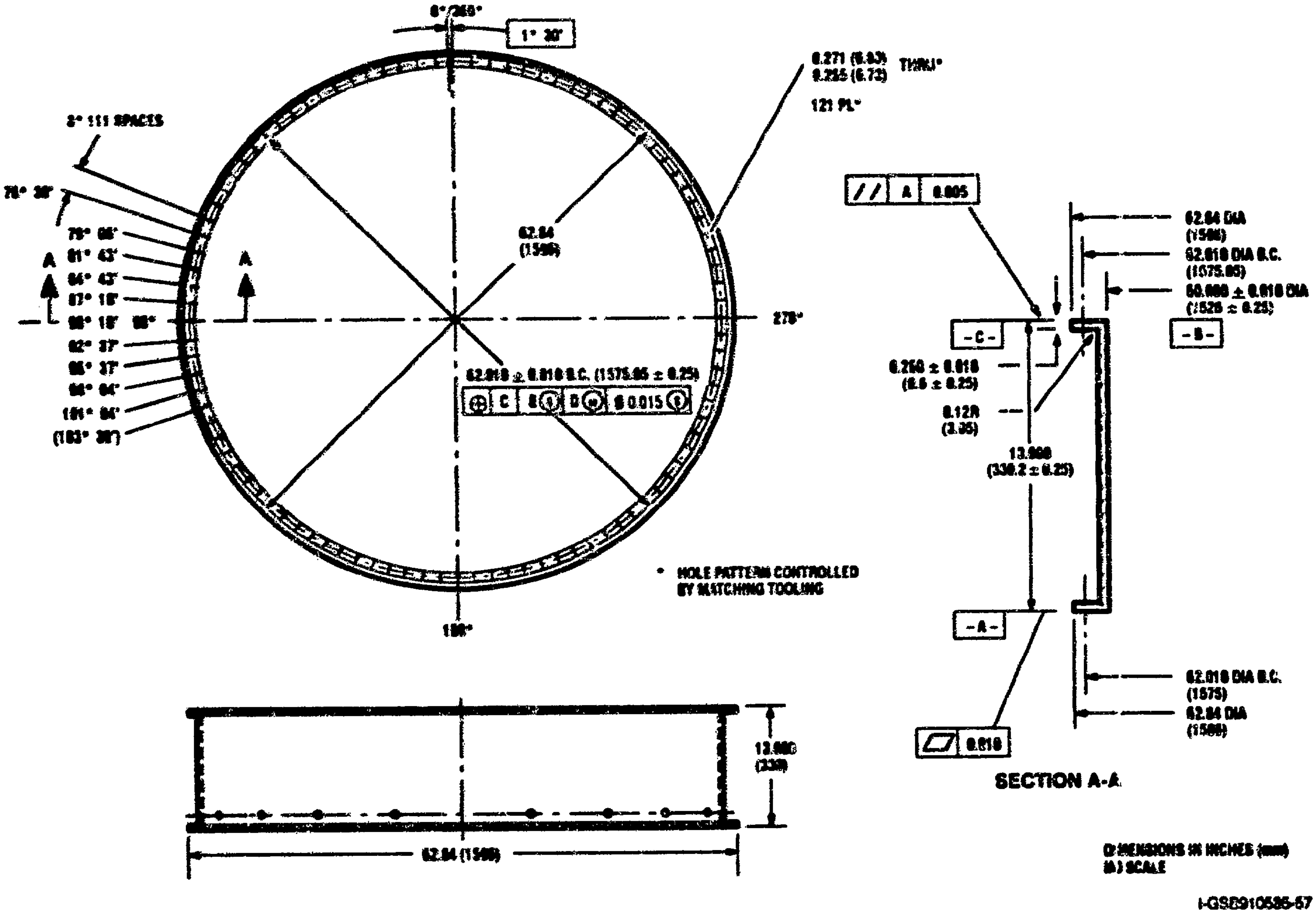


Figure 4-13f. Type C1 spacecraft adapter/spacer mechanical interfaces.

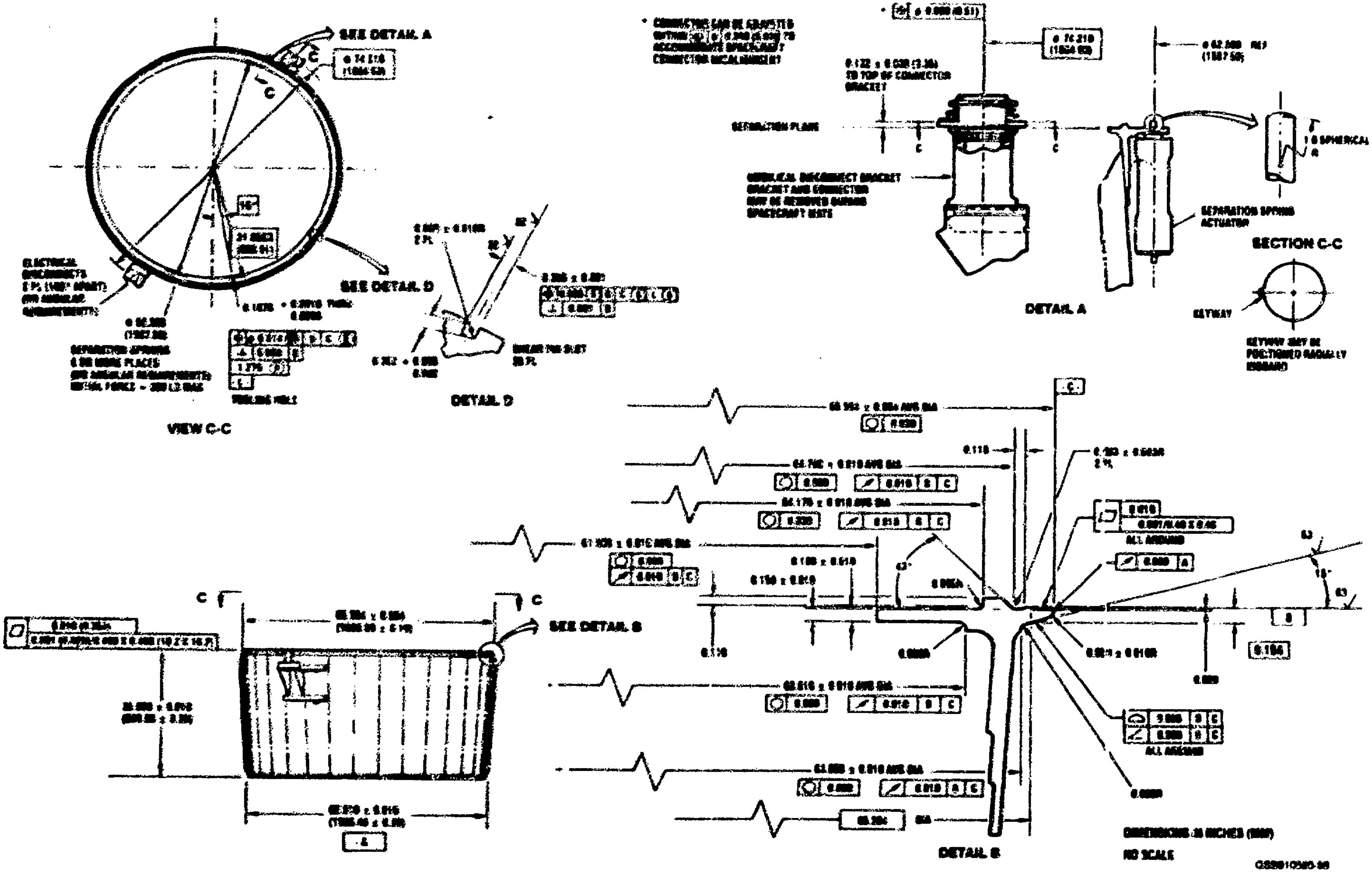


Figure 4-13g. Type D spacecraft adapter mechanical interfaces.

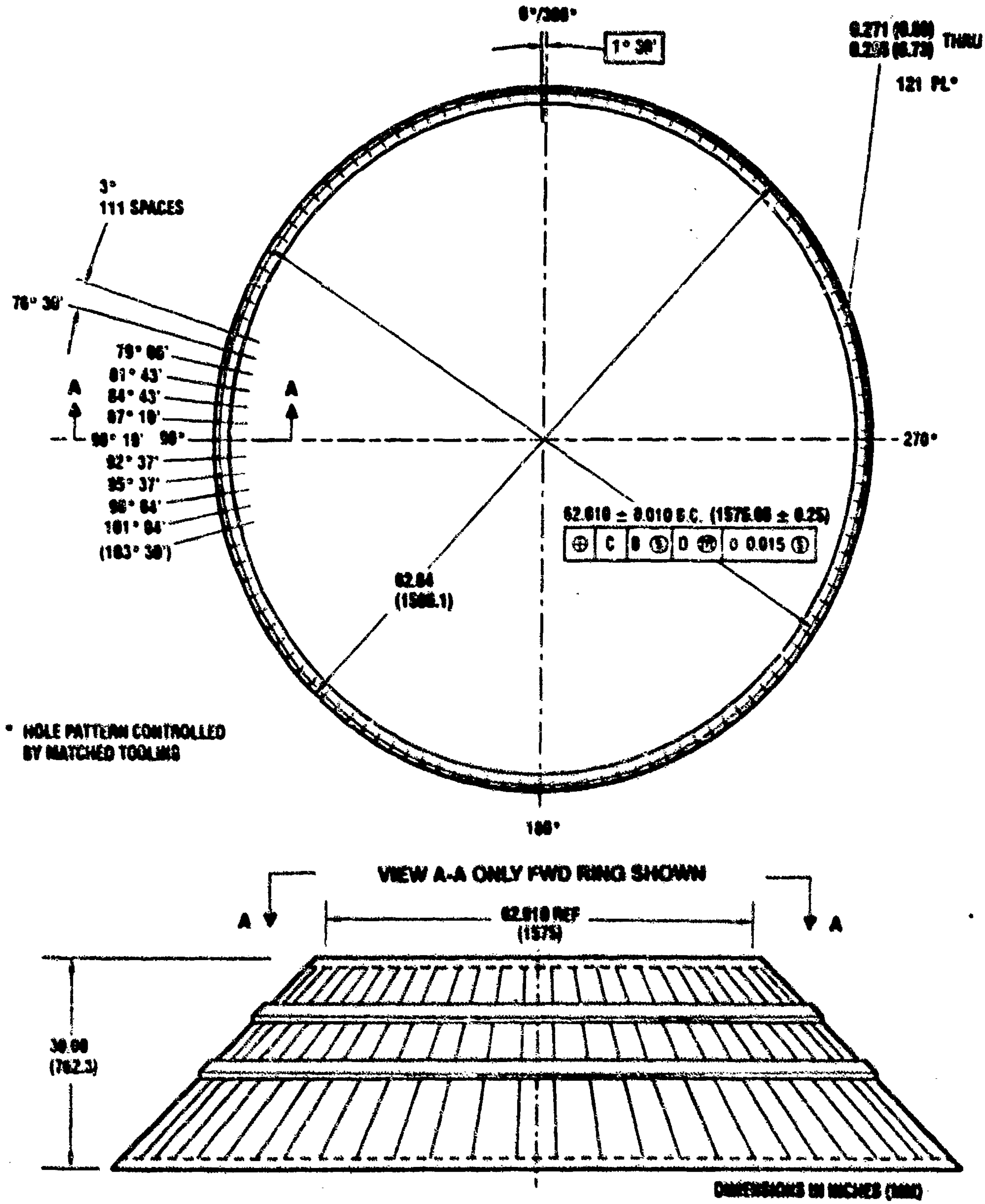


Figure 4-14. Equipment module mechanical interfaces.

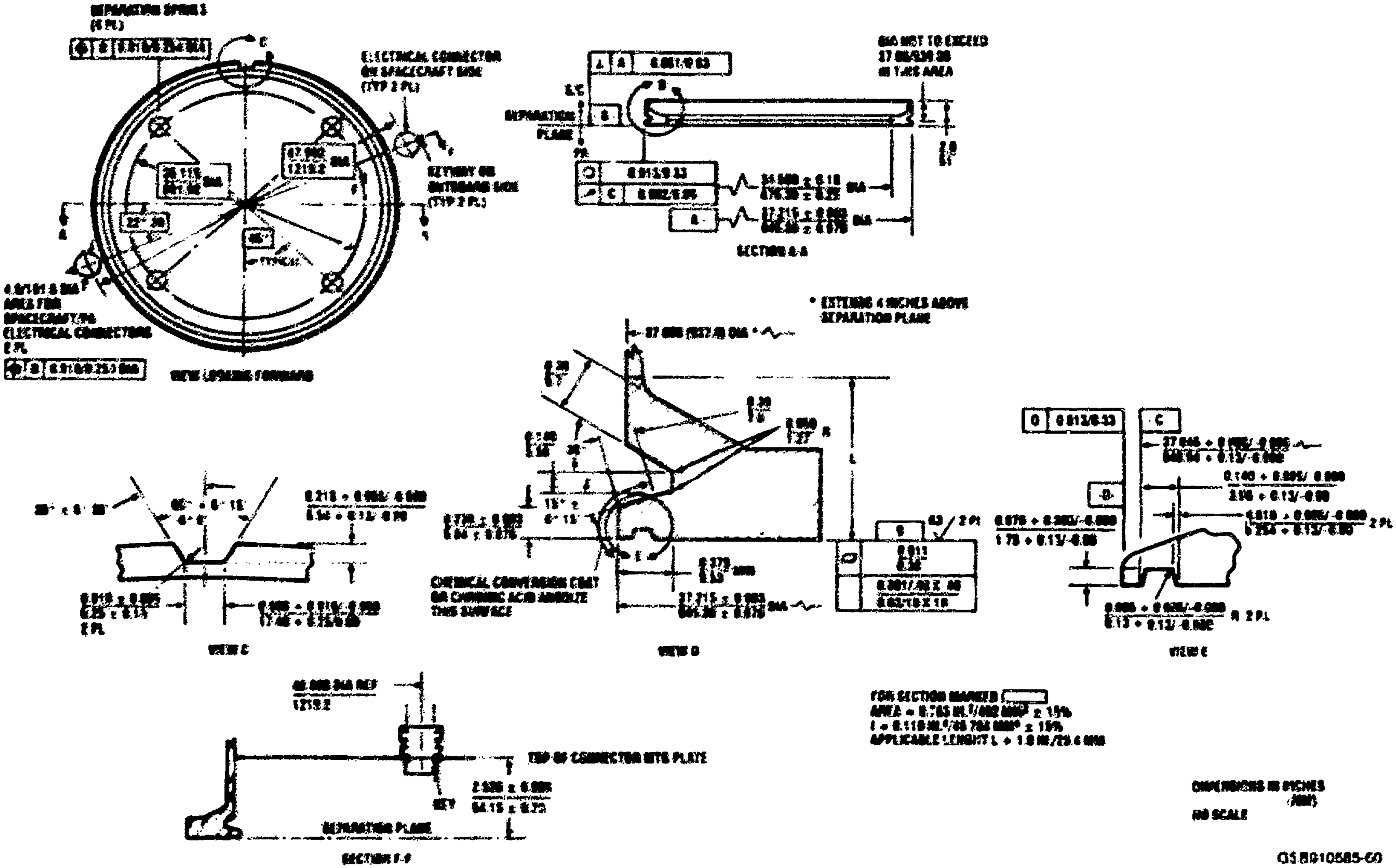


Figure 4-15a Type A adapter spacecraft interface requirements

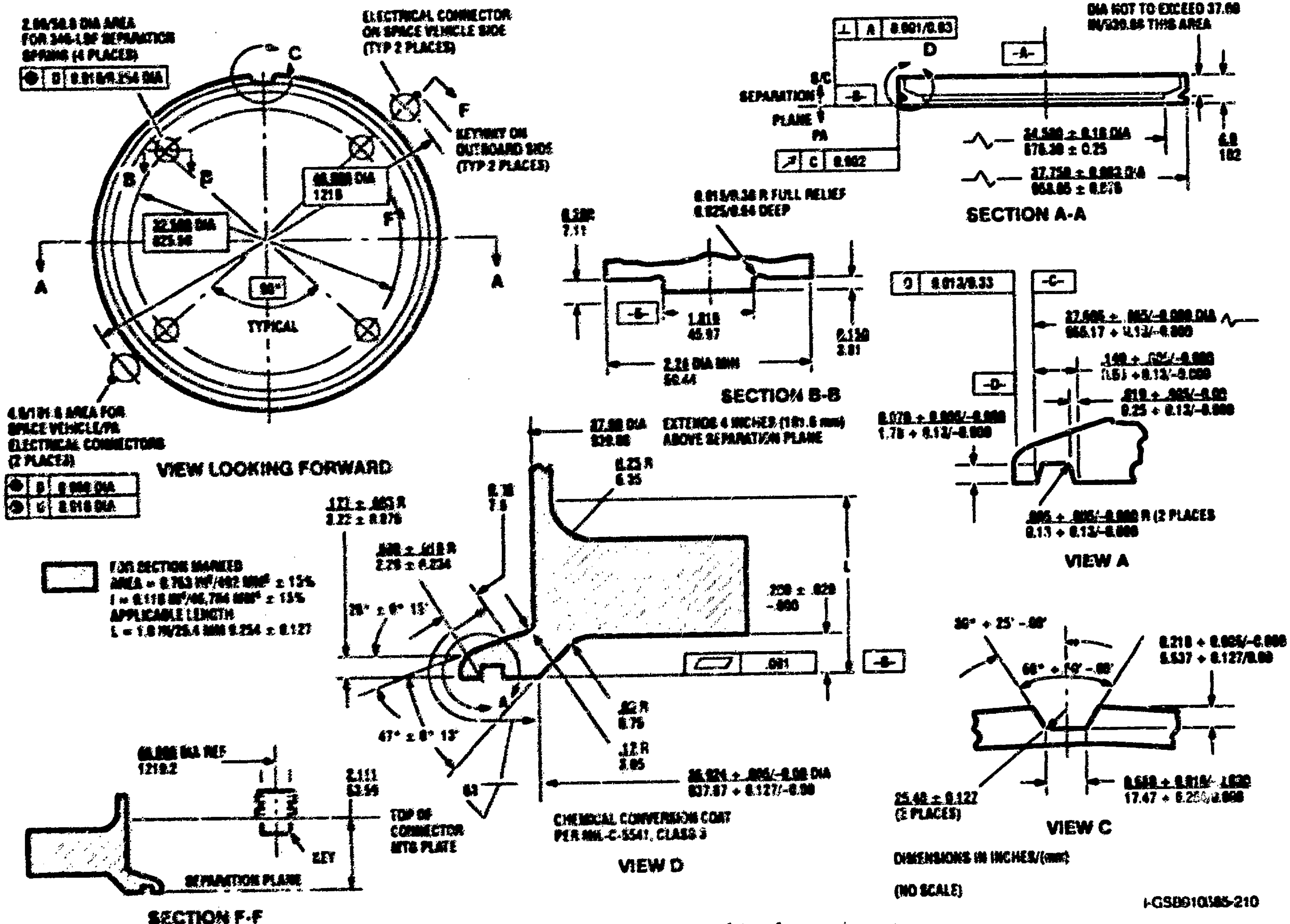


Figure 4-15b Type A1 adapter spacecraft interface requirements.

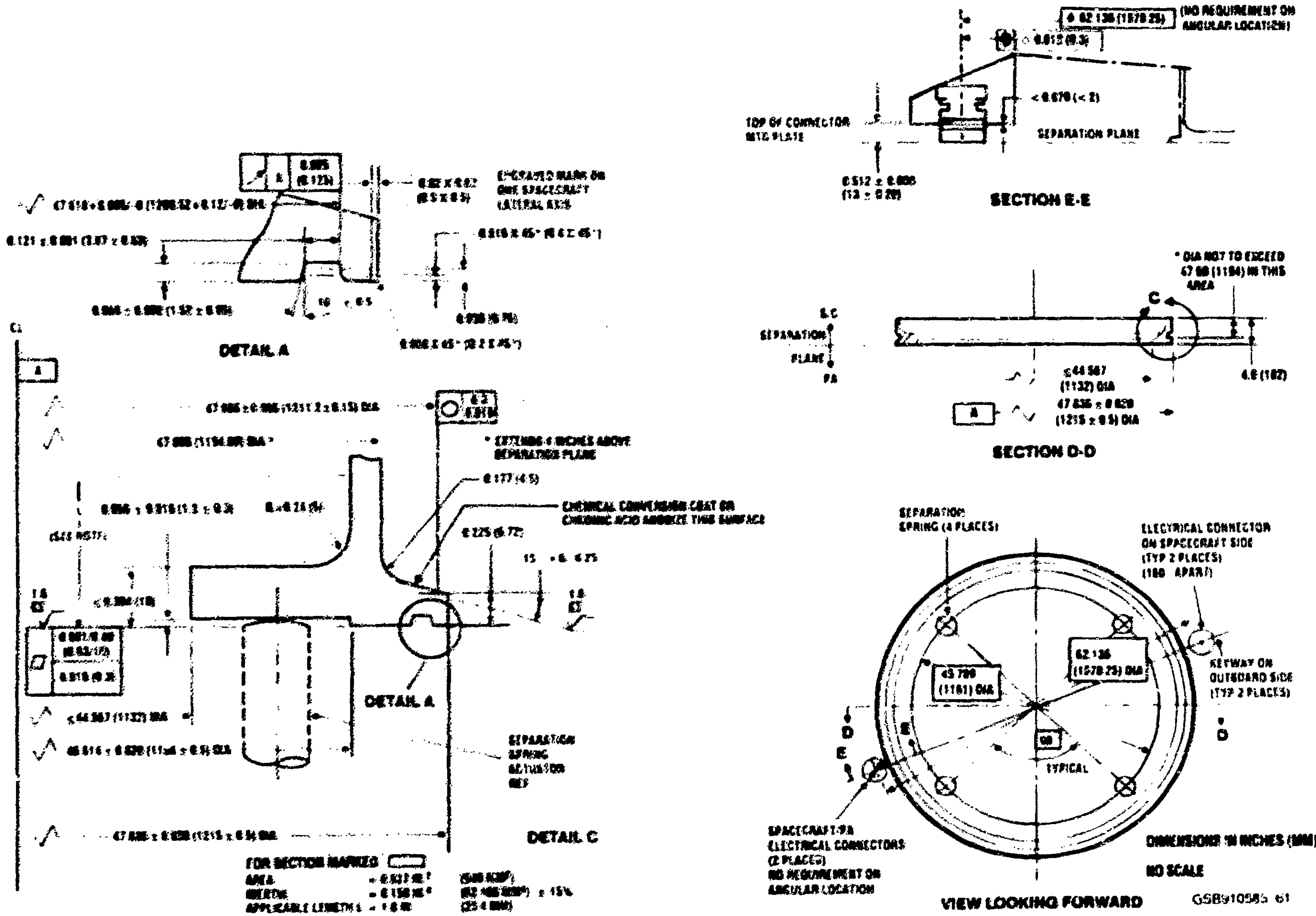


Figure 4-15c. Types B and B1 adapter spacecraft interface requirements.

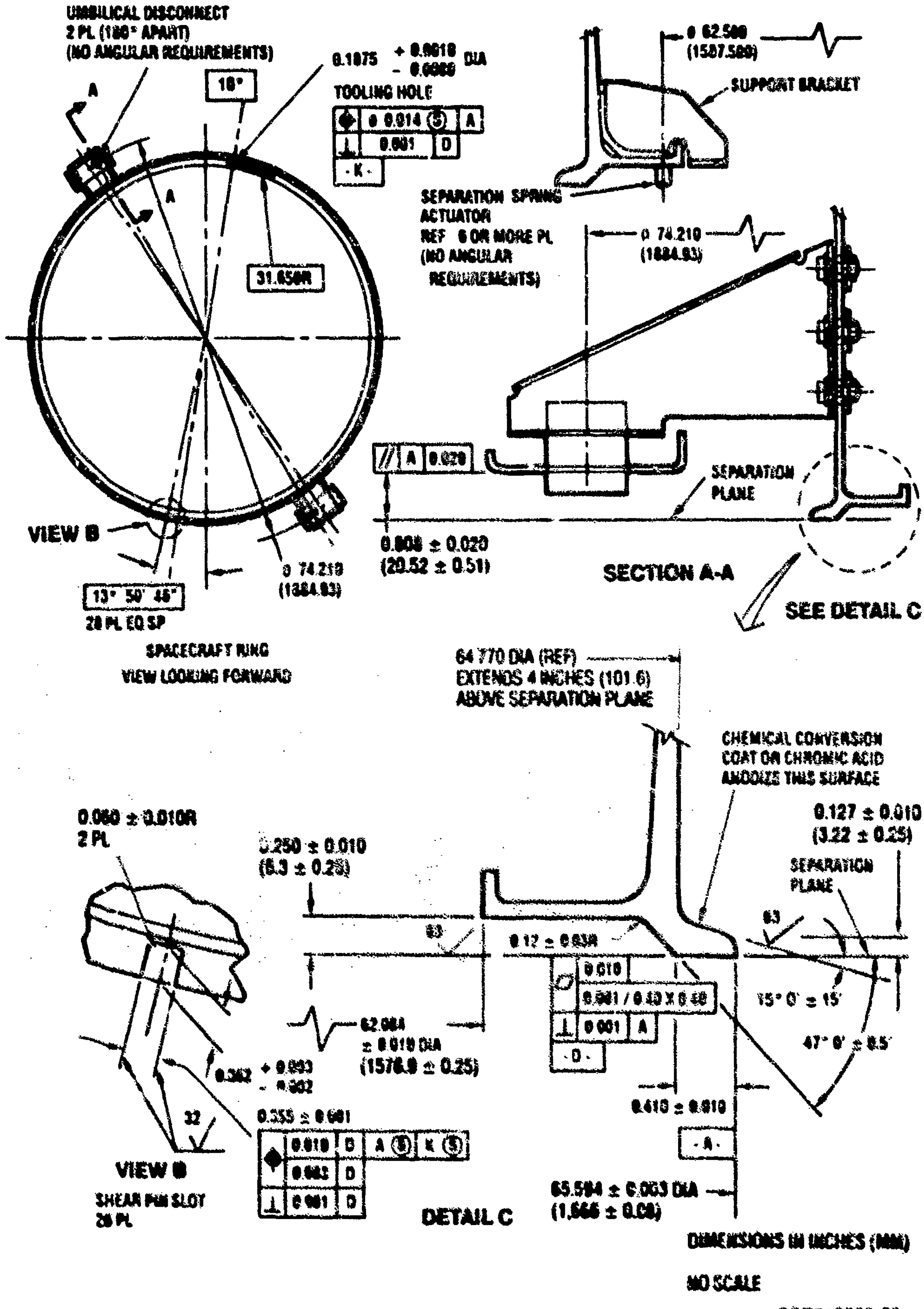


Figure 4-15a Type D adapter spacecraft interface requirements

Separation System — The separation systems provided with the standard Type A, A1, B, B1, and D adapters are similar. Each separation system consists of a clamp band set (Figure 4-16) and separation springs to give the necessary separation energy after the clamp band is released. The clamp band set consists of a clamp band for attaching the satellite to the adapter structure plus devices to extract, catch, and retain the clamp band on the adapter structure after separation. The separation spring assemblies are typically mounted inside the spacecraft adapter. The springs are integral with the spacecraft adapter and bear on supports fixed to the spacecraft rear frame. The springs are sized appropriately for each mission to provide the proper separation velocity between launch vehicle and spacecraft. The spacecraft adapter also provides mounting for some of the mission-peculiar hardware. The umbilicals for spacecraft electrical disconnects and the spacecraft range safety destruct unit both mount on the spacecraft adapter.

4.1.2.3 Spacecraft Adapter Structural Capabilities — The allowable spacecraft weights and longitudinal centers of gravity for the Type A, A1, B, B1, and D adapter/separation systems are shown in Figure 4-17. Note that the structural capability of the Type C and C1 adapters is limited by the structural capability of the equipment module. Figure 4-17 also shows equipment module load capability. This curve is used to assess the structural capability of the launch vehicle when user-supplied spacecraft adapters are used.

The allowable interface loads for the Type A, A1, B, B1, C, C1, and D payload adapters and equipment module are shown in Figures 4-18a through 4-18g.

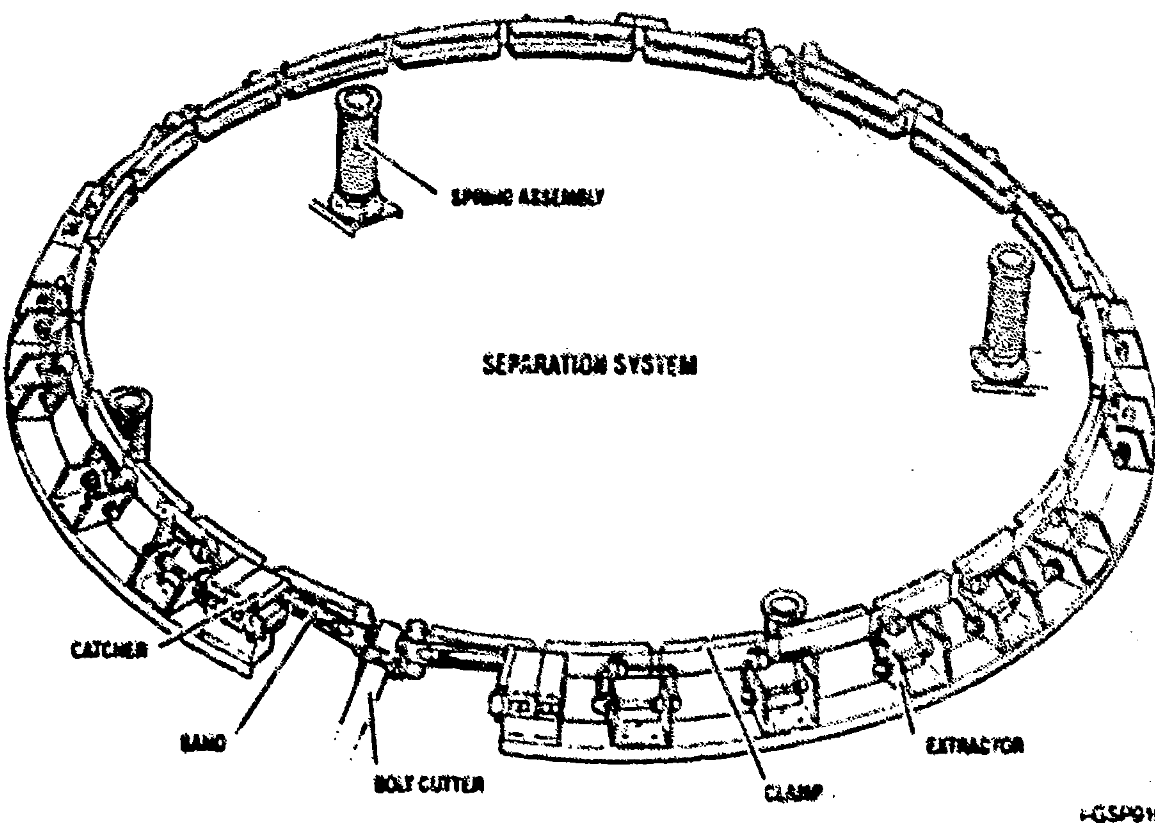


Figure 4-16. The separation system attaches the spacecraft to the payload adapter

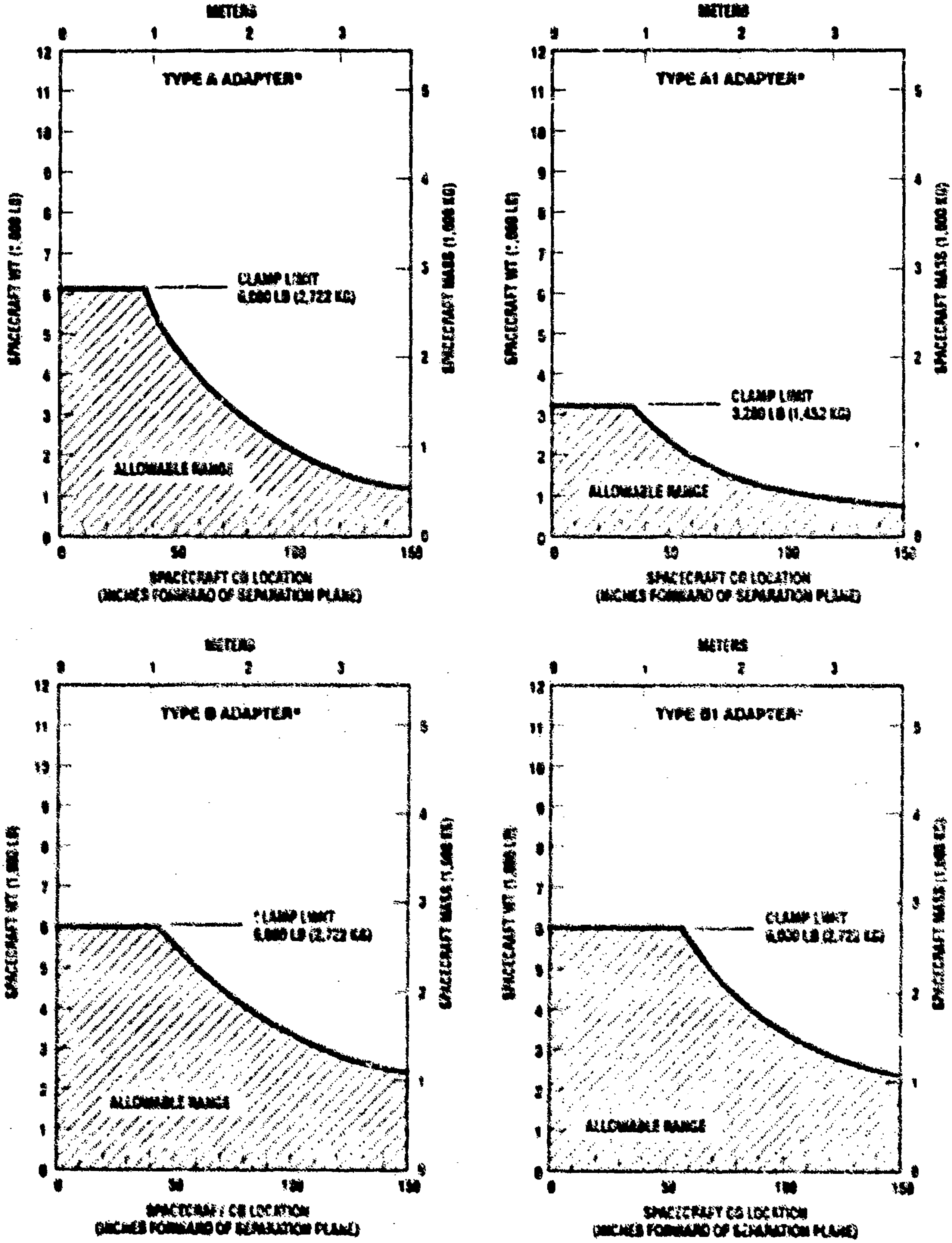


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Figure 4-17 Equipment module and space vehicle adapter/separation system structural capabilities

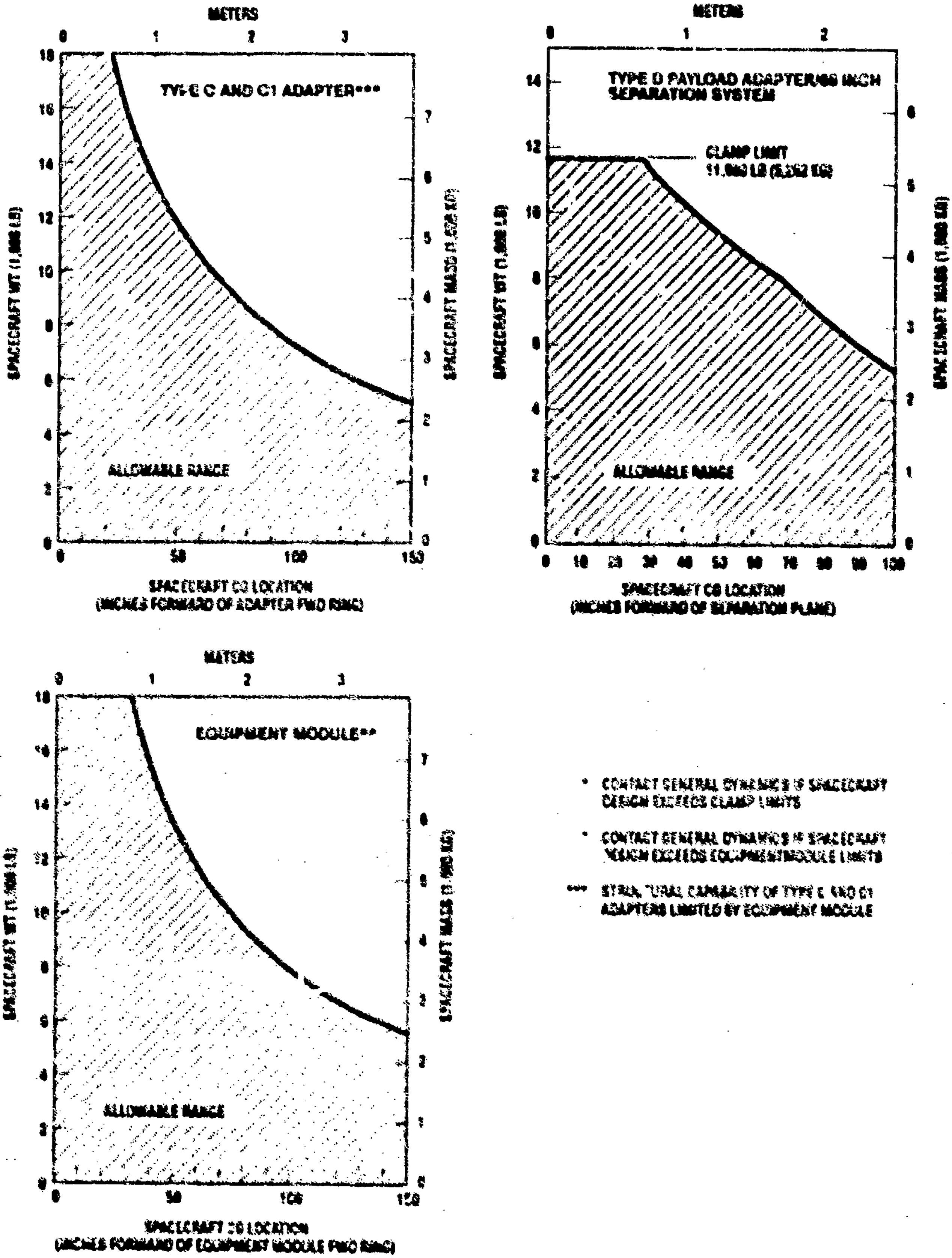

Figure 4-17 Equipment module and space vehicle adapter/separation system structural capabilities (continued)

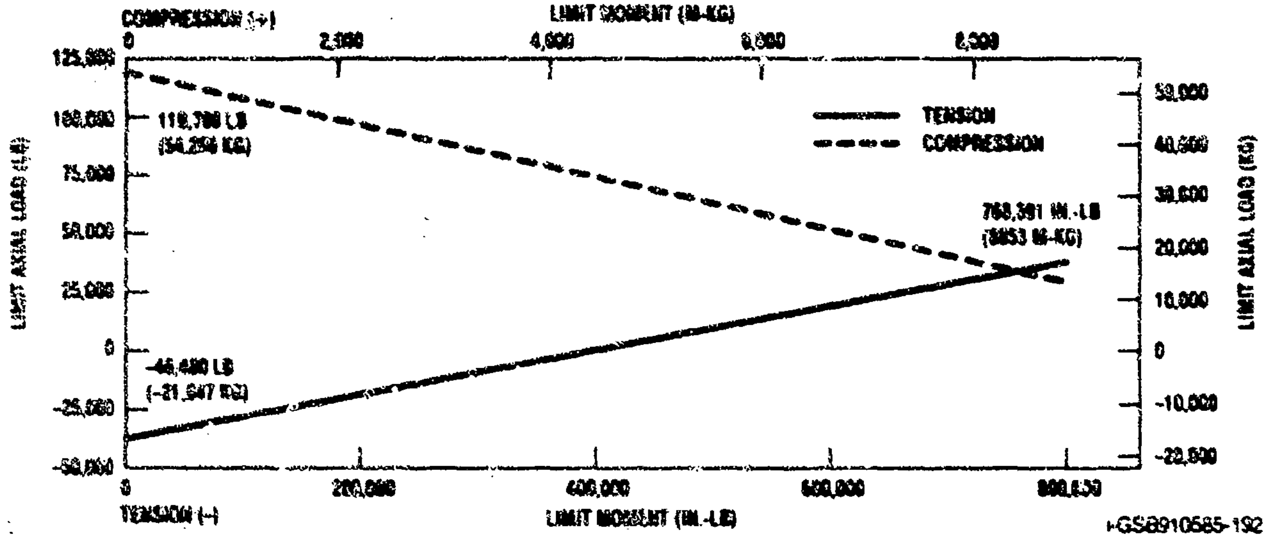

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Figure 4-18a. Type A payload adapter and separation system allowable interface loads.

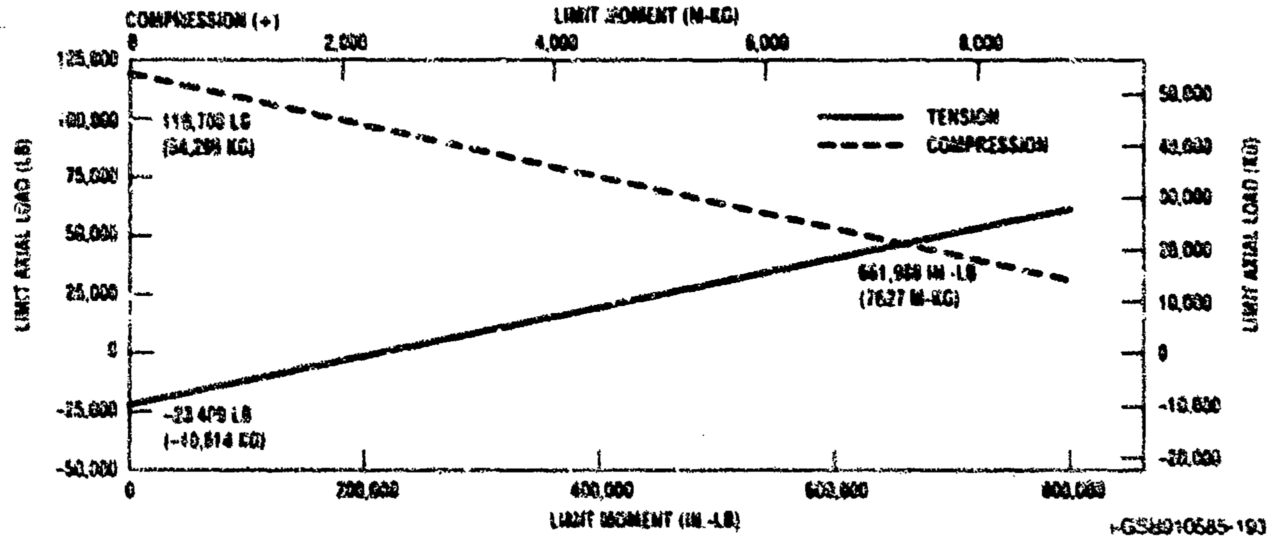

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Figure 4-18b. Type A1 payload adapter and separation system allowable interface loads

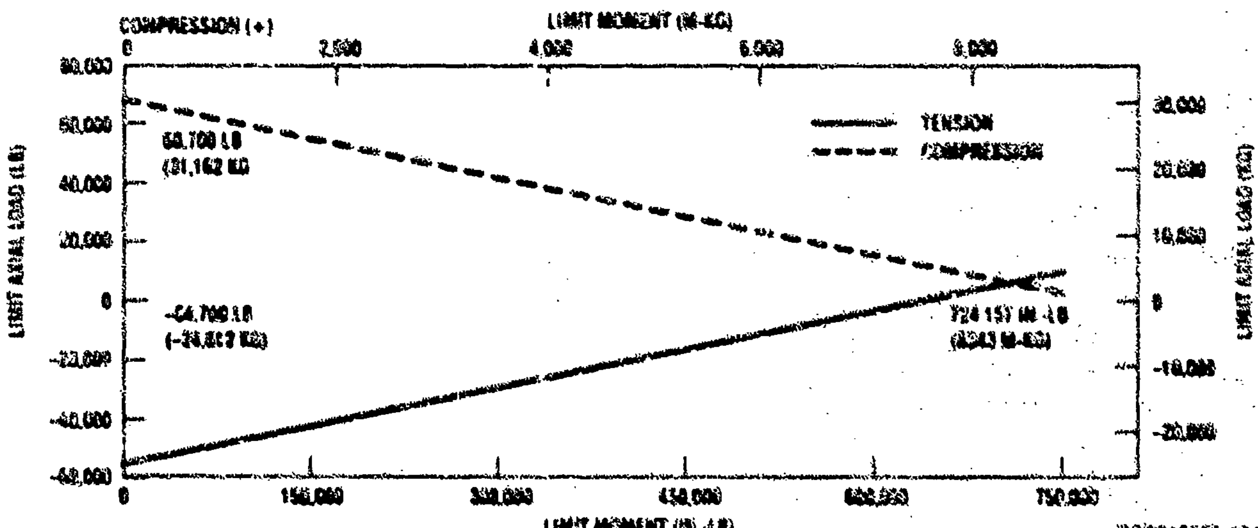

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Figure 4-18c. Type B payload adapter and separation system allowable interface loads

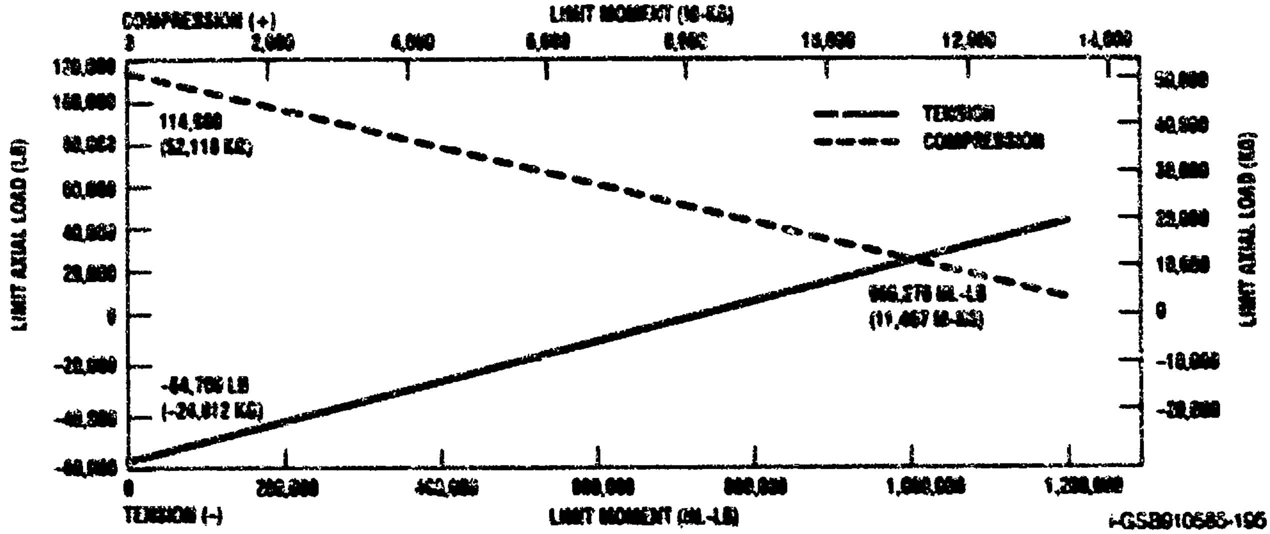

Figure 4-18d. Type B1 payload adapter and separation system allowable interface loads.

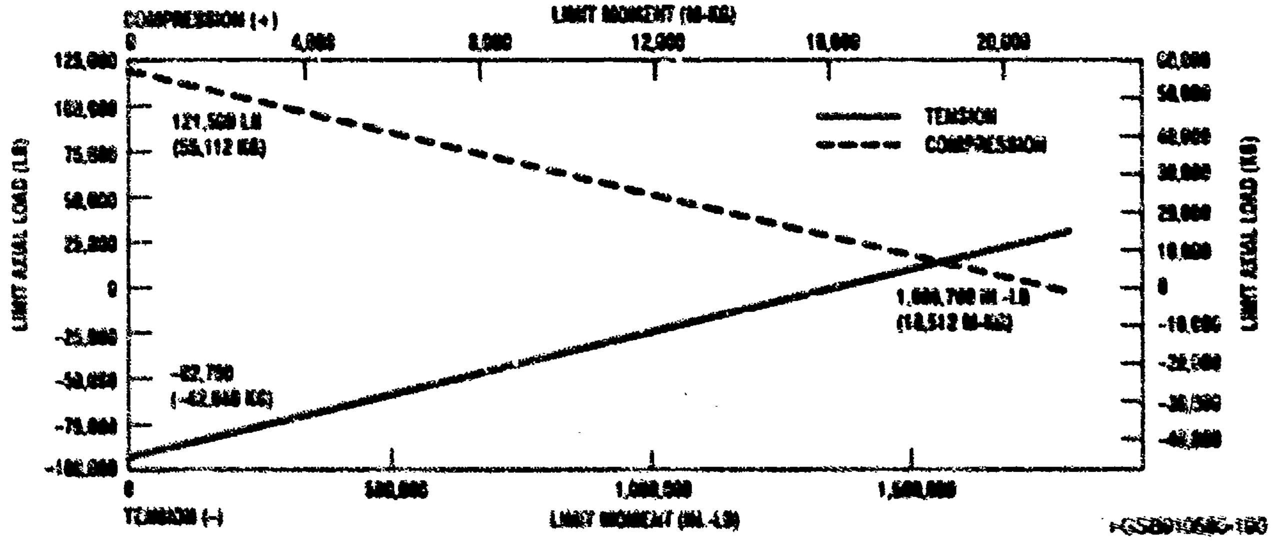

Figure 4-18e. Type C and C1 payload adapter allowable interface loads.

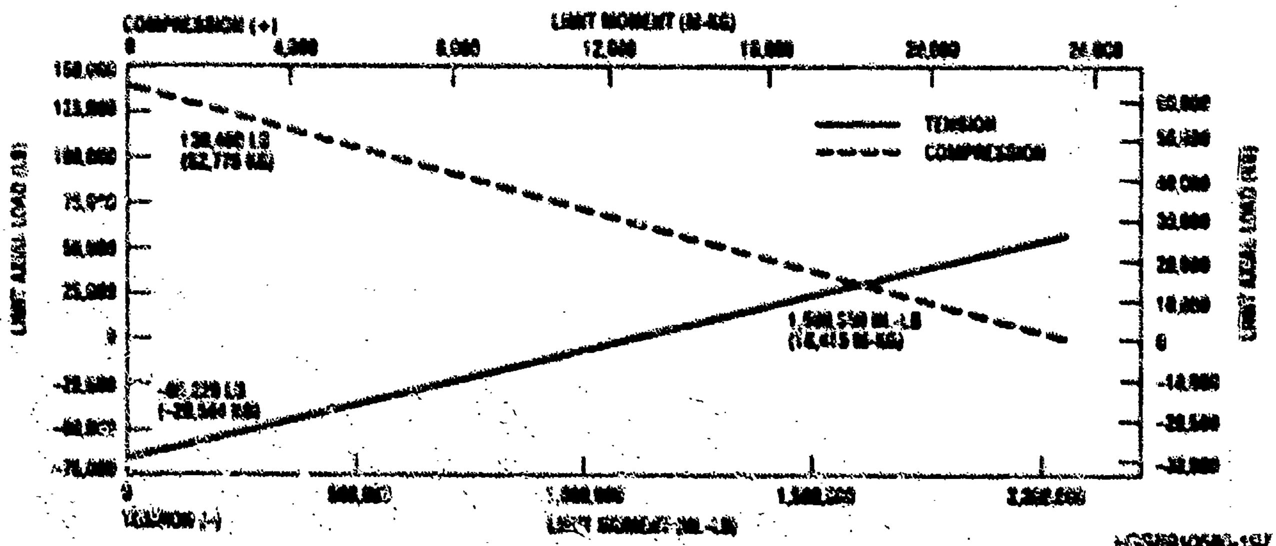

Figure 4-18f. Type D payload adapter and separation system allowable interface loads.

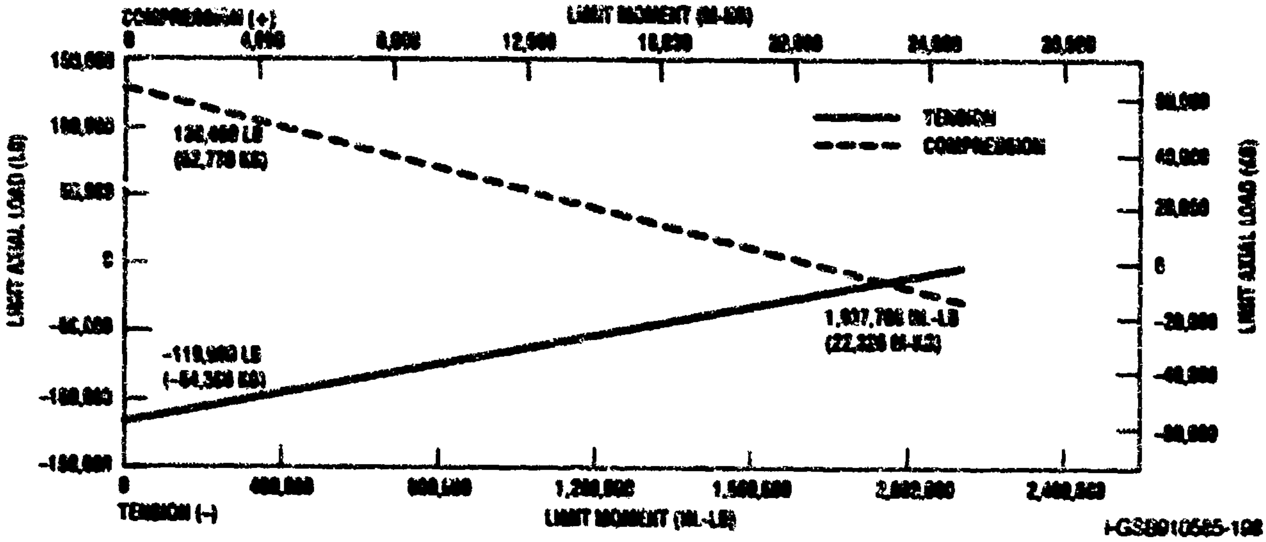

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Figure 4-18g. Equipment module allowable interface loads.

It should be noted that the spacecraft mass and center of gravity (cg) capabilities should be used only as a guideline in preliminary design. They are determined using generic spacecraft interface ring geometry as shown in Figures 4-15a through 4-15g, and quasi-static load factors shown in Tables 3-2 and 3-3. Actual spacecraft design allowables may vary depending upon interface ring stiffness and results of spacecraft mission-peculiar coupled loads analyses. Please contact General Dynamics for further discussion regarding spacecraft designs that exceed these generic allowables. (See Section 7 for additional equipment module capability options.)

4.1.3 ELECTRICAL INTERFACES — The spacecraft/launch vehicle electrical interfaces are shown in Figures 4-19 and 4-20. Typical standard interfaces include:

- A spacecraft-dedicated umbilical interface between the umbilical disconnect located on the

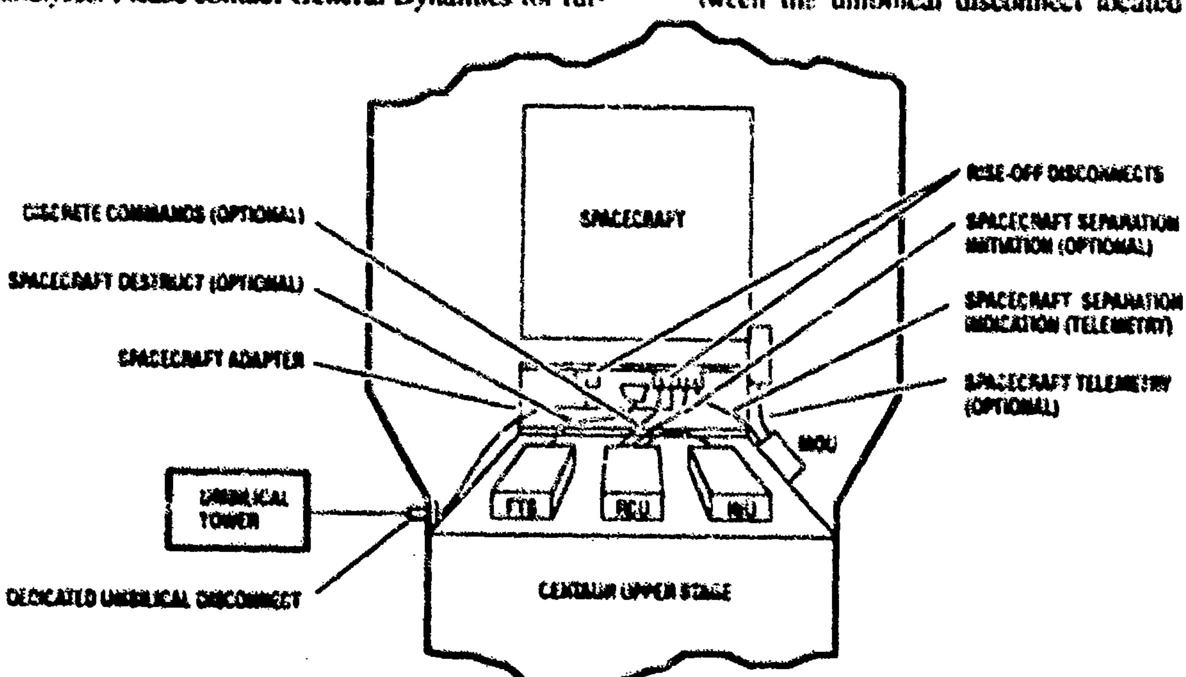

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Figure 4-19. Typical SC/LV electrical interface.

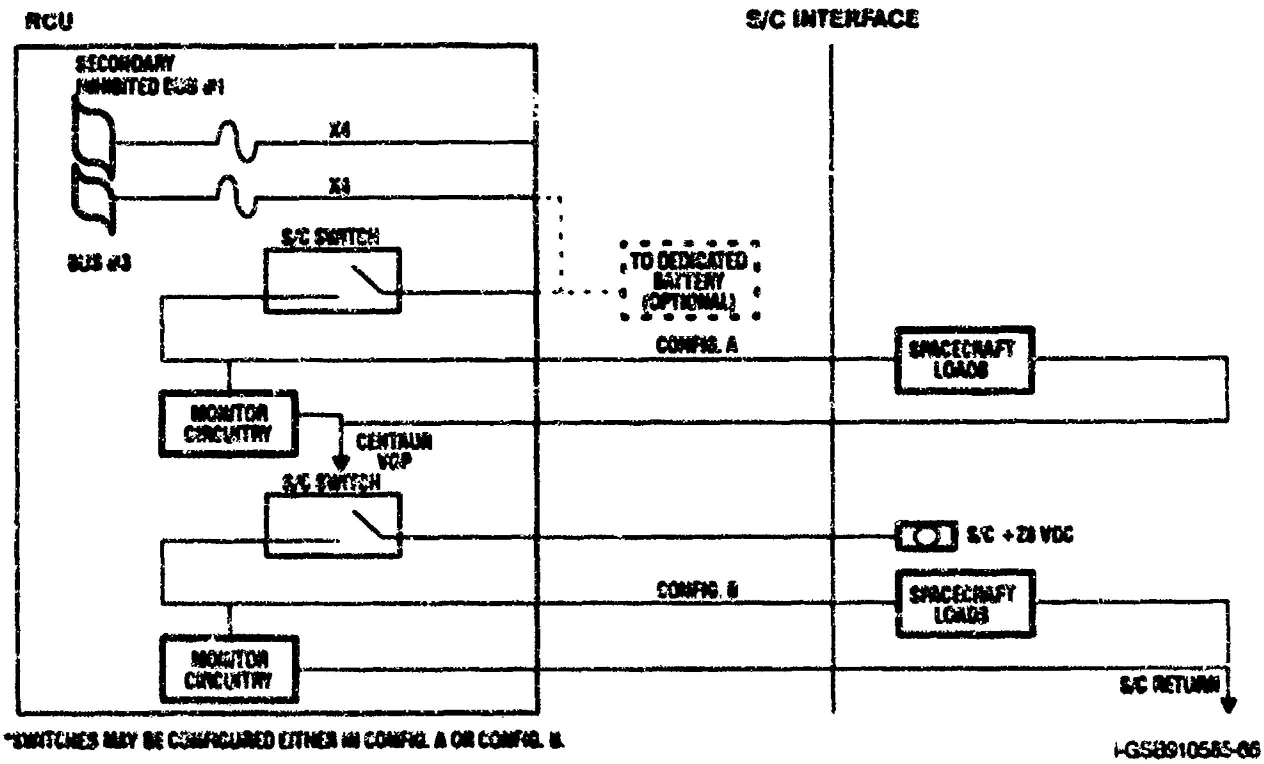


Figure 4-20. Atlas II, IIA, and IIAS spacecraft switch interface.

Centaur upper stage and rise-off disconnects at the spacecraft/launch vehicle interface

- Spacecraft/launch vehicle separation indicators located in the SC/LV rise-off disconnects to verify separation
- A spacecraft destruct interface activated by the Centaur range safety system
- Standard rise-off disconnects (MS 3446E 37-50P and MS3464 E37-50S) or other connectors from MIL-C-81703 that may be required by mission-peculiar changes. A unique keying arrangement for each connector is highly recommended.

The launch vehicle can also be configured to provide electrical interfaces for various mission-peculiar requirements. The complement of signals available is as follows: two separation commands, 16 or 10 (Atlas I) control commands that can be configured as 28 V discretes or switch closure functions, a standard umbilical with a mix of wire configurations, and an instrumentation interface, which contains two discrete inputs for detection of spacecraft separation, four analog inputs for general use, ten command feedback discretes, and two serial data interfaces for downlinking data from the spacecraft, if desired.

The paragraphs below describe the Atlas electrical interfaces in detail.

4.1.3.1 Umbilical Interface — A spacecraft-dedicated umbilical disconnect for on-pad operations is located on the Centaur forward umbilical panel. The umbilical interfaces with two SC-dedicated rise-off disconnects located on the spacecraft (see Figure 4-19). This umbilical interface provides signal paths between the SC and ground support equipment for SC system monitoring during prelaunch and launch countdown.

The umbilical disconnect separates at liftoff. The two spacecraft rise-off disconnects separate at SC/Centaur separation.

The SC/GSE umbilical contains the following complement of wires from SC to umbilical disconnect:

41 twisted shielded wire pairs — 20 AWG

6 twisted shielded wire triples — 20 AWG

4 twisted shielded controlled impedance wire pairs — 75 Ω ± 10%

Additional disconnects for signal interface between the SC and LV can be incorporated for mission-peculiar requirements. These disconnects also separate at SC/Centaur separation.

4.1.3.2 Electrical Disconnects — The standard adapters provide two 37-pin rise-off disconnects for the spacecraft interface. These rise-off disconnects typically provide a spacecraft-dedicated umbilical interface between the spacecraft and ground support equipment. General Dynamics can provide more disconnects on a mission-peculiar basis if the spacecraft requires them. Interface requirements for the disconnects are shown in Figures 4-15a, -15b, and -15c.

4.1.3.3 Spacecraft Separation System — The baseline separation system for SC/LV separation is a pyrotechnic V-type clamp band system.

The separation sequence is initiated by redundant commands from the upper stage guidance system. The upper stage typically controls the spacecraft pyrotechnically actuated separation system. Power for this is supplied from the main vehicle battery.

Positive spacecraft separation is detected via continuity loops installed in the spacecraft rise-off disconnects and wired to the Centaur instrumentation system. The separation event is then telemetered to the ground.

4.1.3.4 Control Command Interface — For the Atlas II, IIA, and IIAS vehicles, the remote control unit (RCU) provides as many as 16 control commands to the spacecraft. These commands are solid state switches configured as 28 Vdc commands or as switch closures (dry-loop commands).

Closure of the switches is controlled by the inertial navigation unit (INU). Parallel digital data from the INU is decoded in the RCU and the addressed relays are energized or de-energized under INU software control.

For the Atlas I vehicle, 10 command functions are provided by the sequence control unit (SCU). These functions are controlled by the INU and can be configured as 28 Vdc commands or as switch closures.

The basic switch configuration is shown in simplified form in Figure 4-20. The figure also shows a typical spacecraft interface schematic.

Command feedback provisions are also incorporated to ensure that control commands issued to the spacecraft are received through SC/LV rise-off disconnects. The SC is responsible for providing a feedback loop on the SC side of the interface.

4.1.3.5 Telemetry Interface — Two telemetry options for data transmission are available to the spacecraft.

- Incorporating an independent SC-based RF telemetry/command system. Use of this option will require modification to the metal nose fairing for implementation of a reradiating antenna system.
- Interfacing the spacecraft PCM and analog data with the upper stage master data unit (MDU). The MDU incorporates software programmable, integrated signal conditioning and multiplexing, and multiple-bit rates and PCM formats. The SC data is interleaved with the LV data and serially transmitted in a PCM bit stream.

A PCM hardwire link for SC and SC/GSE interface is available for system monitoring and verification.

4.1.3.6 Spacecraft Destruct Option — If required for range safety considerations, Atlas can provide a spacecraft destruct capability. A safe/arm initiator receives the destruct command from the Centaur Flight Termination System (FTS). The initiator ignites electrically initiated detonators, which set off a booster charge. The charge ignites a mild detonating fuse which, in turn, detonates a conically shaped explosive charge that perforates the spacecraft propulsion system.

4.2 SPACECRAFT-TO-GROUND EQUIPMENT INTERFACES

4.2.1 SPACECRAFT BLOCKHOUSE CONSOLE — Floor space is allocated on the operations level of Blockhouse 36 for installation of a spacecraft ground control console. This console is typically provided by the user, and interfaces with General Dynamics-provided control circuits through upper stage umbilicals to the spacecraft. The control circuits provided for spacecraft use are isolated physically and electrically from those of the launch vehicle to minimize EMI effects. Spacecraft that require a safe/arm function for apogee motors will also interface with the range-operated pad safety console. GD will provide cabling between the spacecraft blockhouse console and the pad safety console. The safe/arm command function for the spacecraft apogee motor must be inhibited by a switch contact in the pad safety console. Pad Safety will close this switch when pad evacuation has been verified.

4.2.2 POWER — Several types of electrical power are available at Complex 36 for spacecraft use. Commercial ac power is used for basic facility operation. Critical functions are connected to an uninterruptable power system (UPS). The dual-UPS consists of battery chargers, batteries, and a static inverter. The battery chargers are normally operated from the commercial system. However, one UPS may be operated on diesel generator power for major testing and launch. UPS power is available for spacecraft use in the blockhouse, launch service building, and umbilical tower.

Twenty-eight-volt dc power can be provided for spacecraft use in the blockhouse and the launch service building. The facility power supplies are operated on the UPS to provide reliable service.

4.2.3 LIQUIDS AND GASES

Gaseous Nitrogen (GN_2) — Three pressure levels of gaseous nitrogen are available on the service tower for spacecraft use. Nominal pressure settings are 2,000 psi (13 790 kN/m^2), 100 psi (689.5 kN/m^2), and approximately 10 psi (68.95 kN/m^2). The 10-psi system is used for purging electrical cabinets for safety and humidity control.

Gaseous Helium (GHe) — Gaseous helium at 2,200 psi (15 169 kN/m2) is available on the service tower.

Liquid Nitrogen (LN_2) — LN_2 is available at the Complex 36 storage facility. LN_2 is used primarily by the Atlas pneumatic and LN_2 loading systems. Small dewars can be filled at range facilities and brought to Complex 36 for spacecraft use.

4.2.4 PROPELLANT AND GAS SAMPLING — Liquids and gases provided for spacecraft use will be sampled and analyzed by the range propellant analysis laboratory. Gases, such as helium, nitrogen, and breathing air, and liquids such as hypergolic fuels and oxidizers, water, solvents, and hypergolic decontamination fluids may be analyzed to verify that they conform to the required specification.

4.2.5 WORK PLATFORMS — The Complex 36 service tower provides work decks approximately ten feet apart in the spacecraft area. Portable workstands will be provided to meet spacecraft mission requirements where the fixed work decks do not suffice. Access can be provided inside the encapsulated nose fairing. The access requirements will be developed during the planning stage of each mission.

4.3 SYSTEM AND RANGE SAFETY

4.3.1 REQUIREMENTS — Launch vehicle and spacecraft design and ground operations will be in accordance with the Eastern Space and Missile Center Regulation (ESMCR 127-1), Range Safety requirements, and Air Force regulations concerning Explosives Safety and Occupational Safety and Health. For spacecraft processing in Astrotech International Corporation facilities, compliance with their safety policy will be required. Should spacecraft processing be conducted in NASA facilities, compliance with ESMCR 127-1, as well as NASA safety regulations will be required.

Chapters 2 through 5 of ESMCR 127-1 identify spacecraft design and operational requirements that must be met to obtain range safety approval. For spacecraft that are designed to meet Space Transportation System criteria, compliance with ESMCR 127-1 should be specifically addressed in safety submittals.

General Dynamics System Safety engineers are available to evaluate, analyze, and provide guidance for spacecraft design to support range safety approval. CLS will provide a specific set of design and operational requirements that define spacecraft-peculiar Range Safety and spacecraft processing requirements. Should areas of noncompliance be determined, General Dynamics will evaluate and suggest when a waiver may be prudent while still meeting the intent of the safety requirement. For each program, a system safety manager will be designated, who acts as the spacecraft agent for all interface activities with the ESMC Range Safety Office.

4.3.1.1 Applicable Documents — The following documents are applicable to launch processing activities:

At CCAFS and/or KSC

ESMCR 127-1, Range Safety

AFR 127-100, Explosives Safety Standards

AFR 127-12, Air Force Occupational Safety, Fire Prevention and Health (AFOSH) Program

MIL-STD-1522A, Standard General Requirements for Safe Design and Operation of Pressurized Missile and Space Systems

MIL-STD-1576, Electroexplosive Subsystem Safety Requirements and Test Methods for Space Systems

GDSS Launch Site Safety Manual

At Astrotech

Astrotech Safety Policy

4.3.1.2 Application -- System and Range Safety requirements will be analyzed and controls established for both design and operational/test procedures that are determined to be hazardous to the launch vehicle system or personnel. For launch operations from Cape Canaveral Air Force Station, spacecraft must meet the range safety requirements contained in ESMCR 127-1.

The following areas will be evaluated for compliance:

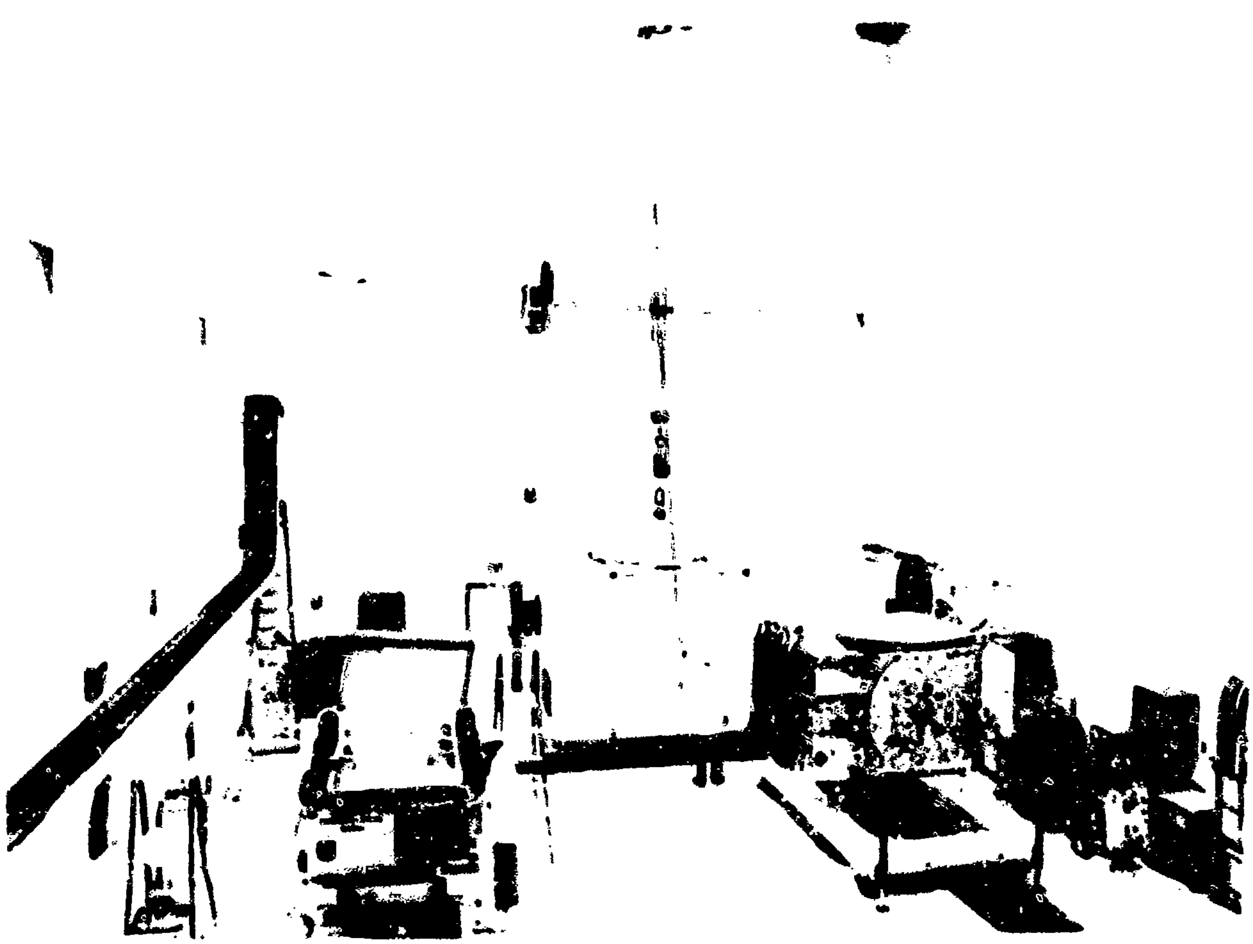
- Propellants and propulsion systems
- Pressurized systems
- Ordnance systems
- Electrical/electronic equipment
- Ground support equipment
- Non-ionizing radiation
- Ionizing radiation sources
- Acoustic (noise) criteria

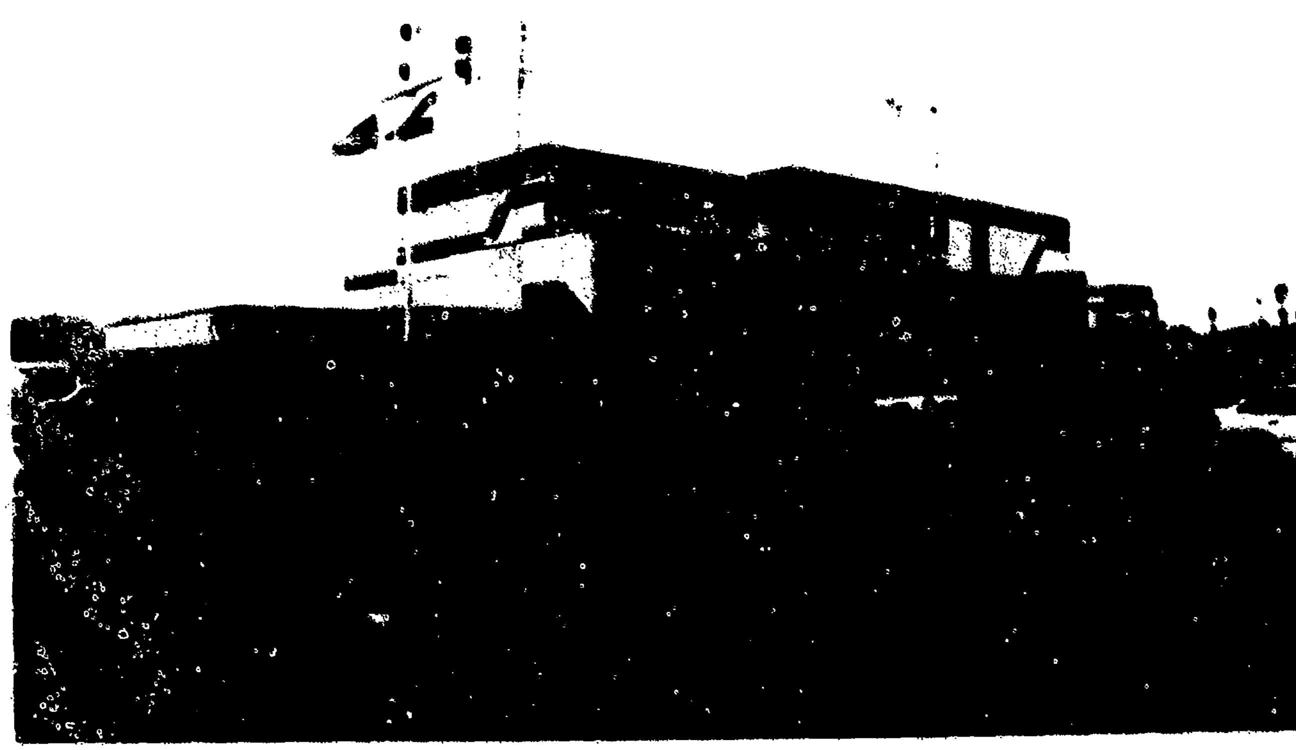
- Hazardous materials

4.3.1.3 Safety Data Submittals — The customer and spacecraft supplier shall perform system safety hazard analyses for the spacecraft and provide these analyses in a Missile System Prelaunch Safety Package (MSPSP).

This data package will describe, in detail, all potentially hazardous subsystems of the spacecraft design. The subsystems designated as hazardous are: propulsion, pressure, electrical, ordnance, radiation sources, and hazardous materials. This document summarizes the system safety analyses for the spacecraft and includes analyses of the spacecraft to launch vehicle interface. The MSPSP also includes information concerning all hazardous and nonhazardous procedures accomplished, as well as all ground support equipment used during payload processing. Safety Working Groups and Technical Interchange Meetings will be held to ensure exchange of the safety data necessary to verify compliance with range safety requirements.

Flight Analysis Data Package — This data package, prepared by GDSS, will contain general spacecraft performance, flight, and trajectory (land overflight) information.





5 • SPACECRAFT AND LAUNCH FACILITIES

General Dynamics has formal agreements with U.S. Air Force 45th Space Wing (45 SPW) and the National Aeronautics and Space Administration's Kennedy Space Center (KSC) for the use of payload and launch vehicle processing facilities and Complex 36A and 36B at Cape Canaveral Air Force Station (CCAFS) (see Figure 5-1). We also have agreement on the range services and equipment to be provided.

A customer mission support facility is provided on Cape Road at CCAFS. Conference rooms and management offices are available for use during the launch campaign. This facility is conveniently located between SLC-36 and Astrotech (see Figures 5-1 and 5-2).

The information that follows is a summary of the launch service capabilities that General Dynamics provides. Additional details are available in the Atlas Launch Services Facilities Guide.

5.1 SPACECRAFT FACILITIES

The commercial payload processing facility owned and operated by Astrotech Space Operations, L.P. is the primary facility for processing commercial spacecraft. This facility contains separate nonhazardous and hazardous processing buildings, storage buildings, and offices. The facilities and floor plans are described in the following sections. Astrotech complies fully with all applicable federal, state, regional, and local statutes, ordinances, rules, and regulations relating to safety and environmental requirements.

Should the Astrotech facility not adequately satisfy commercial spacecraft requirements, government facilities, as provided for in General Dynamics/NASA and General Dynamics/USAF agreements, are available. These facilities, too, are described in the following sections.

5.1.1 PAYLOAD PROCESSING FACILITIES

Astrotech Building 1 (Payload Processing Building) — Astrotech Building 1, with its high bay expansion, is considered the primary payload processing facility. With overall dimensions of approximately 200 ft (61.0 m) by 125 ft (38.1 m) and a height of 49 ft (14.9 m), its major features are:

- Airlock
- High bays (three identical and one expansion)
- Control rooms (two per high bay)
- Office complex, administrative area, communications mezzanine, and support areas.

The Building 1 floor plan is depicted in Figures 5-3 and 5-4. The expansion adds a high bay complex adjacent to the south wall with a passage between the new high bay and existing airlock. Table 5-1 lists the details of room dimensions, cleanliness, and crane capabilities of this facility.

Building AE — Building AE is a spacecraft and missions operations facility originally constructed by the Air Force. With overall dimensions of approximately 120 ft (36.6 m) by 320 ft (97.5 m), it features:

- Spacecraft checkout areas (high and low bays)
- Mission Director's Center and VIP Observation Room (see Figure 5-5)
- Telemetry Ground Station and Laboratory (including Astrotech's communications link with the range)

Building AE is located in the CCAFS Industrial Area on Hanger Road. Table 5-2 provides a detailed description of Building AE facilities.

Building AM — Building AM is a specially designed two-story facility constructed by NASA for spacecraft processing and checkout. With dimensions of approximately 103 ft (31.4 m) by 195 ft (59.4 m), it features dual spacecraft checkout areas. Building

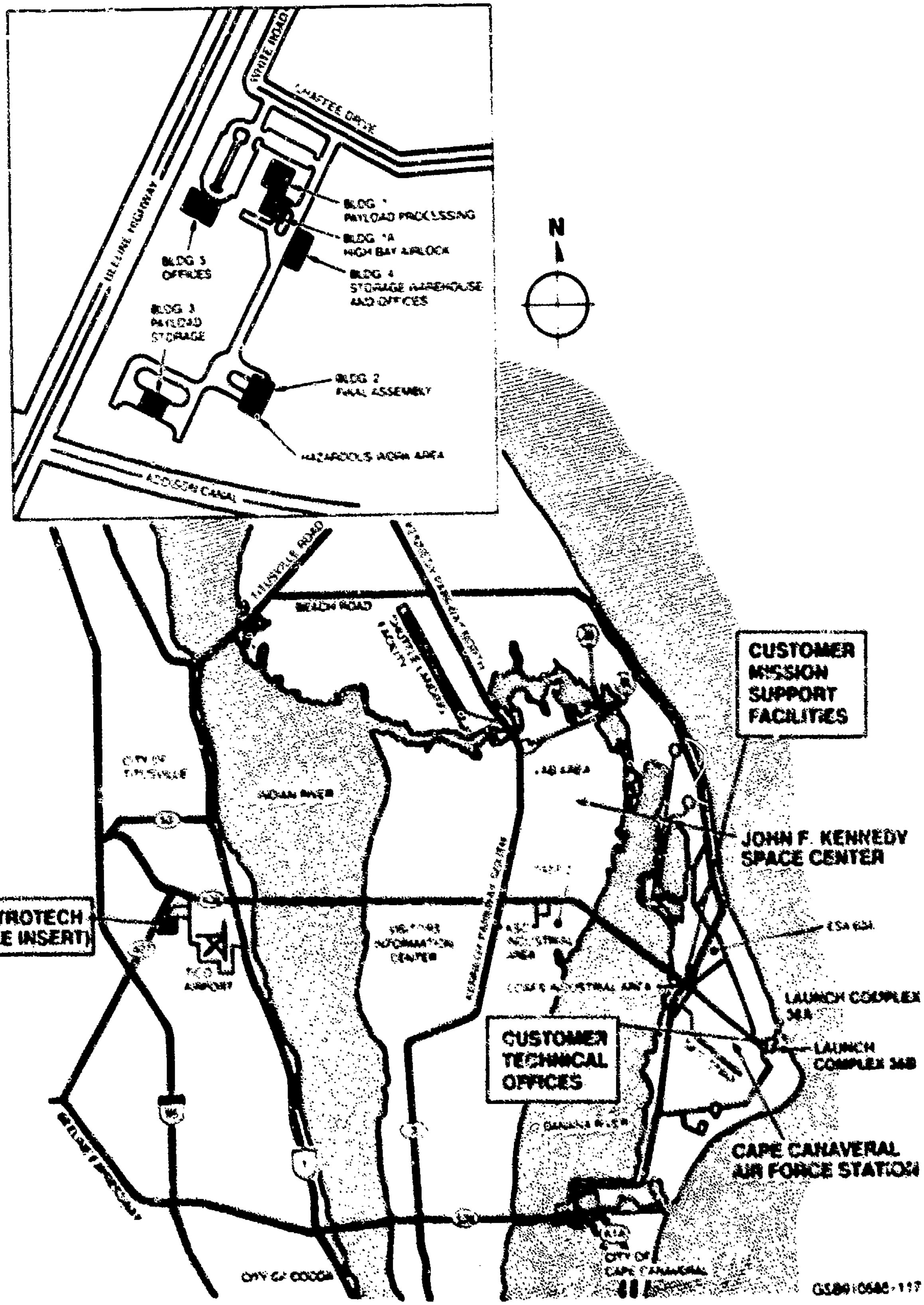


Figure 5-1. Payload processing and launch facilities.

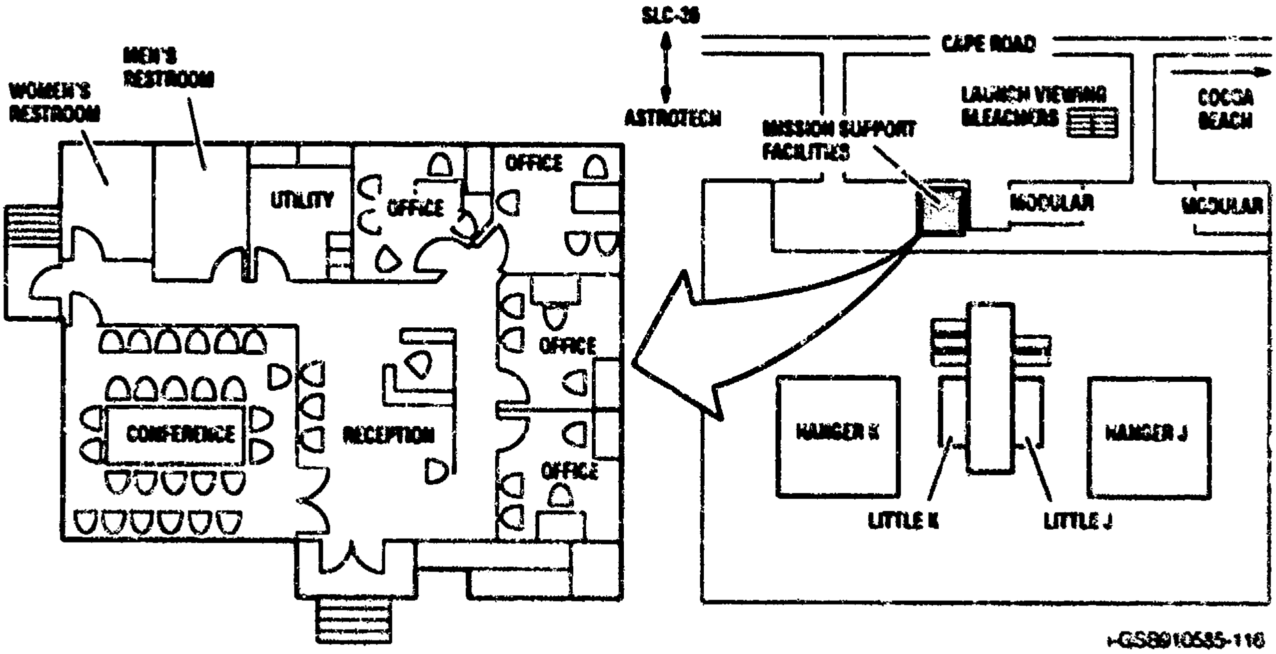


Figure 5-2 CCAFS customer support facility.

AM is located in the CCAFS Industrial Area on Hanger Road. Table 5-3 provides a detailed description of Building AM facilities.

Building AO — Building AO is a specially designed two-story facility constructed by NASA for spacecraft processing and checkout. With dimensions of approximately 180 ft (54.8 m) by 185 ft (56.4 m), it had been used primarily by JPL for interplanetary and lunar spacecraft processing. Building AO is located in the CCAFS Industrial Area on Hanger Road. Table 5-4 provides a detailed description of Building AO facilities.

Table 5-1 Astrotech's Building 1 features four high bay areas

Building 1 High Bays (2): Class 100,000 Clean Room		
Temperature	23.8 ± 2.8°C	75 ± 5°F
Relative humidity	50 ± 5%	
Usable floor space	18 x 12.19 m	60 x 40 ft
Floor area	223.1 m²	2 400 ft²
Clear height	13.1 m	43 ft
Crane type (each bay)	Bridge	
Crane capacity	9,072 kg	10 ton
Crane hook height	11.3 m	37.1 ft
Door size	6.1 x 7 (h) m	20 x 23 (h) ft
High Bay Control Rooms (2/bay)		
Size	9.14 x 4.27 m	30 x 14 ft
Area	39.0 m²	420 ft²
Ceiling height	2.67 m	8.75 ft
Door size	2.44 x 2.44 m	8 x 8 ft
Bay window size	1.22 x 2.44 m	4 x 8 ft
Temperature	23.8 ± 2.8°C	75 ± 5°F
Airlock: Class 100,000 Clean Room		
Temperature	23.8 ± 2.8°C	75 ± 5°F
Relative humidity	50 ± 5%	
Usable floor space	36 x 9.14 m	120 x 30 ft
Clear ceiling height	7.32 m	24 ft
Door size	6.1 x 7 m	20 x 23 ft
Offices (second floor)		
Floor size (1 room)	12.8 x 7.3 m	42 x 24 ft
Floor area	93.44 m	1 008 ft
Floor size (2 room)	12.5 x 7.3 m	41 x 24 ft
Floor area	92.25 m	984 ft
Acoustic ceiling height	2.43 m	8 ft
Bay window size	1.22 x 1.22 m	4 x 4 ft
Bldg 1A High Bay: 100,000 Clean Room		
Temperature	23.8 ± 2.8°C	75 ± 5°F
Relative humidity	50 ± 5%	
Usable floor space	15.5 x 38.1 m	51 x 125 ft
Floor area	592.2 m²	6 375 ft²
Clear height	18.3m	60 ft
Crane type (each bay)	Bridge	
Crane capacity	27 216 kg	30 ton
Crane hook height	16.8 m	55 ft
Door size	6.1 x 15.2 (h) m	20 x 50 (h) ft
Bldg 1A Bay Airlock		
Temperature	23.8 ± 2.8°C	75 ± 5°F
Relative humidity	50 ± 5%	
Usable floor space	12.2 x 15.2 m	40 x 50 ft
Clear ceiling height	18.3 m	60 ft
Door size	6.1 x 15.2 (h) m	20 x 50 (h) ft
Bldg 1A Bay Control Rooms (2)		
Size	12.2 x 13.1 m	40 x 43 ft
Area	159.8 m²	1 720 ft²
Temperature	23.8 ± 2.8°C	75 ± 5°F

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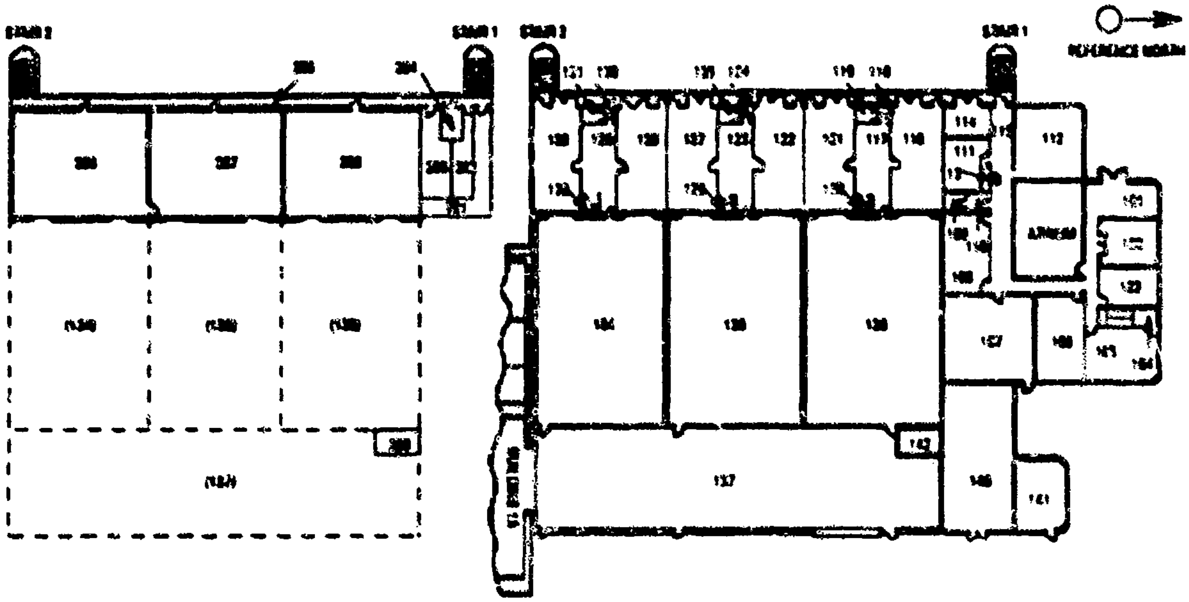

Rm	Function	Rm	Function	Rm	Function
108	Conference	121	Control Room A2	133	Control Room C2
109	Women's Restroom	122	Control Room B1	134	High-bay C
110	Women's Lounge	123	Change Room B	135	High-bay B
111	Men's Restroom	124	Vestibule B	136	High-bay A
112	Break/Lunch	125	Storage B	137	Common Airlock
114	Machine Shop	126	Restroom B	202	Women's Restroom
115	Corridor	127	Control Room B2	203	Men's Restroom
116	Control Room A1	128	Control Room C1	205	Corridor
117	Change Room A	129	Change Room C	206	Office Area C
118	Vestibule A	130	Vestibule C	207	Office Area B
119	Storage A	131	Storage C	208	Office Area A
120	Restroom A	132	Restroom C		

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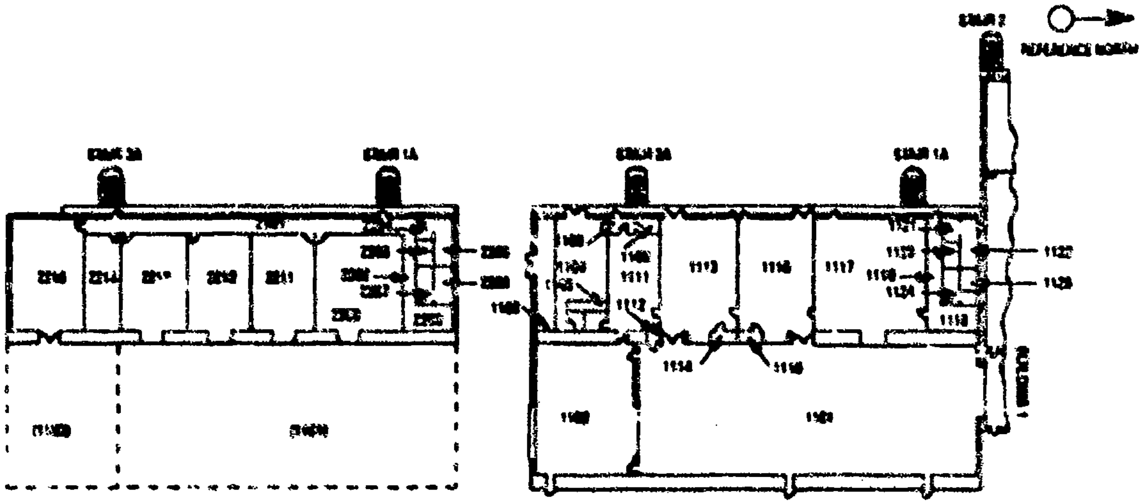
Figure 5-3 Astrotech Building 1 detailed floor plan



Rm	Function	Rm	Function	Rm	Function
1101	Large High-bay D	1117	Office Area D1	2205	Men's Restroom
1102	Large Airlock	1118	Break Room	2207	Women's Washroom
1104	Conference D1	1119	Corridor	2208	Women's Restroom
1105	Closet	1121	Men's Washroom	2209	Office Area D2
1106	Restroom	1122	Men's Restroom	2211	Office Area D3
1107	Restroom	1124	Women's Washroom	2212	Office Area D4
1108	Vestibule	1125	Women's Restroom	2213	Office Area D5
1111	Change Room D	2201	Corridor	2214	Conference D2
1112	Air Shower	2202	Corridor	2215	Office Area D6
1113	Control Room D2	2203	Break Room		
1115	Control Room D1	2204	Men's Washroom		

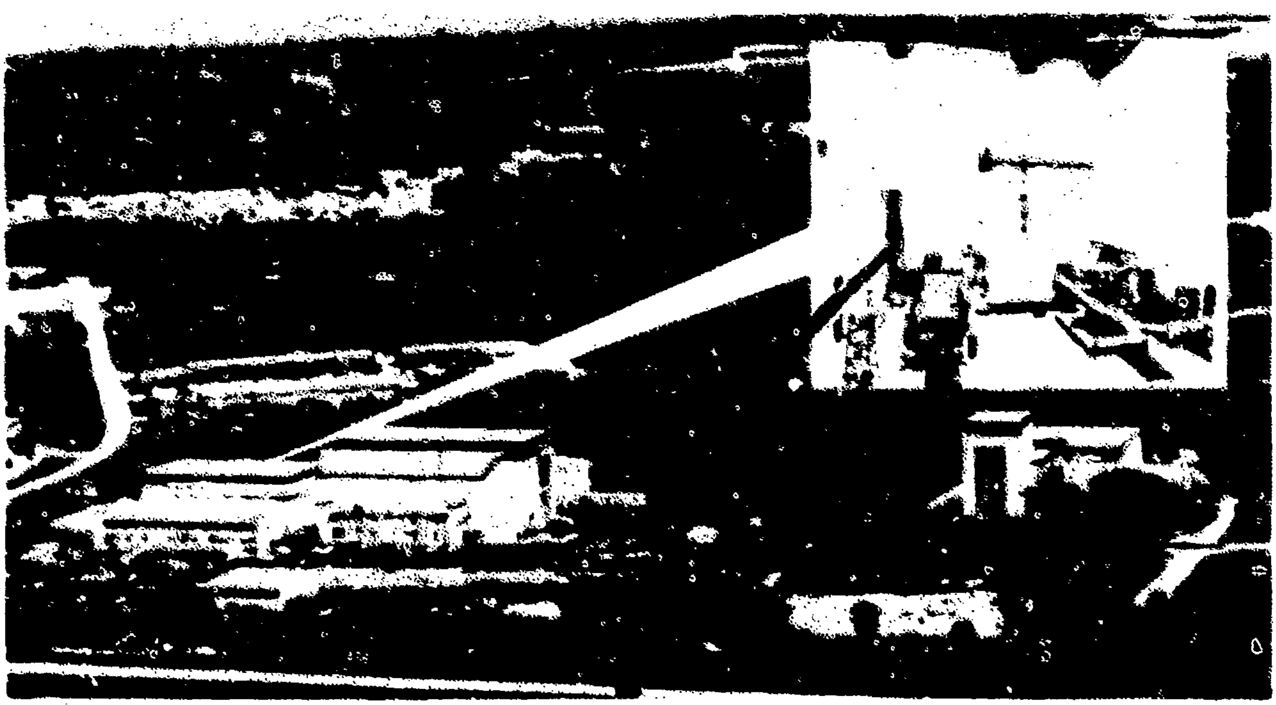


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Figure 5-4. Astrotech Building 1A detailed floor plan — first floor

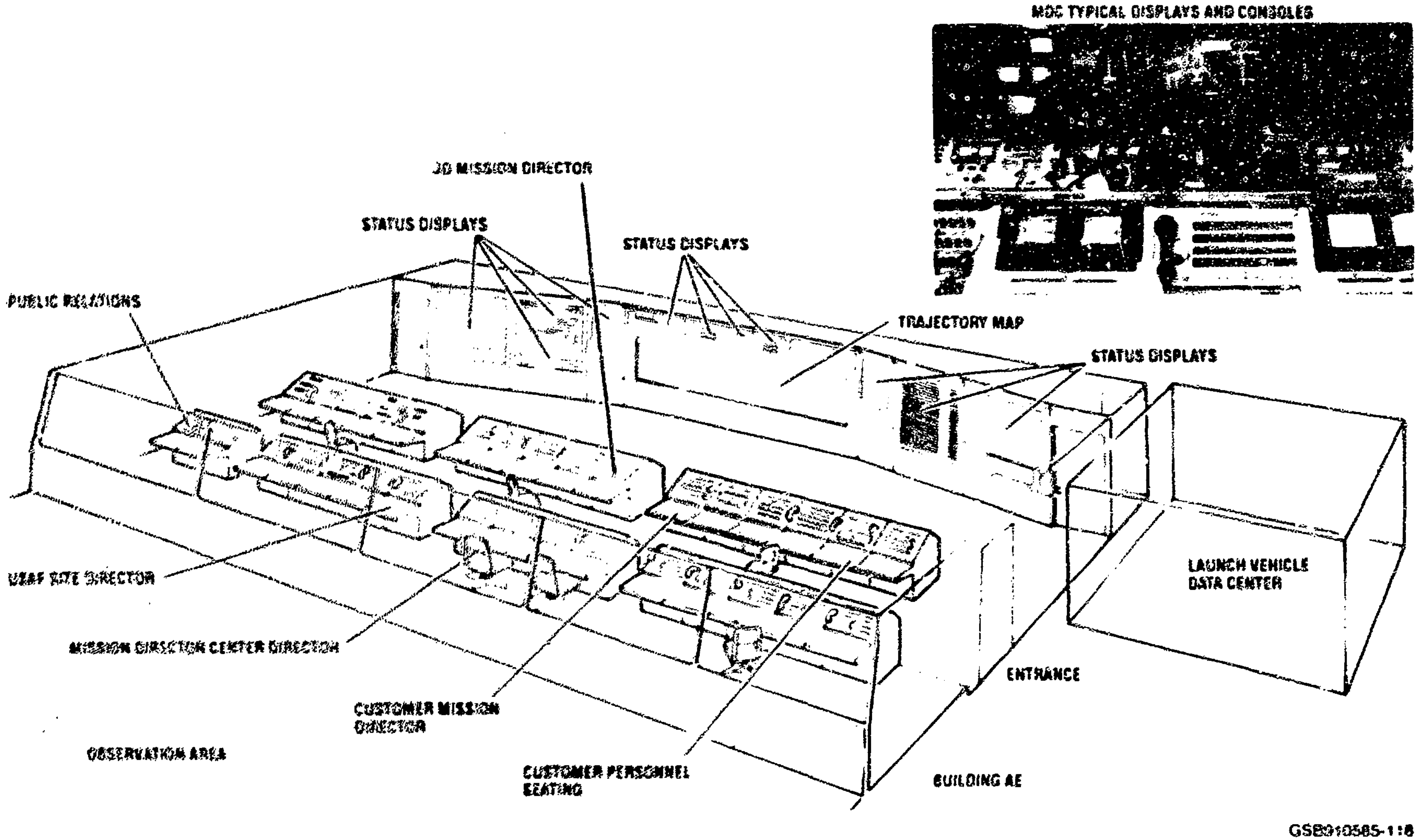

Figure 5-5 Mission Director's Center

Table 5-2. Building AE provides adequate space for smaller spacecraft processing activities.

Spacecraft Laboratory		
Low bay floor area	108 m^2	1,161 ft^2
Height	6.1 m	20 ft
Door size	2.4 X 3 m	8 X 10 ft
Crane type	Bridge	
Crane capacity	1,814 kg	2 ton
Crane hook height	11.3 m	37 ft
High Bay Clean Room (Four Rooms)		
Test Area		
Floor space	14.6 X 19.2 m	48 X 63 ft
Floor area	280.32 m^2	3,024 ft^2
Room height	12.8 m	42 ft
Crane type	Bridge	NOT clean room compatible
Crane capacity	2,722 kg	3 ton
Crane hook height	10.3 m	33.75 ft
Crane type	Monorail	
Crane capacity	1,814 kg	2 ton
Crane hook height	11.3 m	37 ft
Airlock		
Floor space	5.5 X 14.6 m	18 X 48 ft
Floor area	80.3 m^2	864 ft^2
Room height	12.8 m	42 ft
Crane type	Monorail	
Crane capacity	5,443 kg	6 ton
Crane hook height	10.3 m	33.75 ft
Airlock Outside Door		
Width	4.9 m	16 ft
Height	11.1 m	36.5 ft
Test and Storage Area Rooms		
South-side room		
Floor space	9.4 X 12.2 m	31 X 40 ft
Floor area	114.68 m^2	1,240 ft^2
Room height	3 m	10 ft
South door	3 X 3 m	10 X 10 ft
North door	2.5 X 2 (h) m	8.5 X 7 (h) ft

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5.1.2 HAZARDOUS PROCESSING FACILITIES

Astrotech Building 2 (Hazardous Processing Facility) — Astrotech's Building 2, with its high bay expansion, is considered the primary hazardous processing facility. With overall dimensions of approximately 159 ft (48.5 m) by 112 ft (34.1 m) and a height of 49 ft (14.9 m), the major features are:

- Airlock
- Spacecraft processing high bay and operations room
- Spin high bay and control room
- PAM processing high bay and control room
- Expansion high bay/airlock
- Support areas

The Building 2 floor plan is depicted in Figure 5-6. Each of the high bays is built to explosion-proof or equivalent standards to support operations involving liquid propellant transfer, solid propellant motor preparations, and ordnance in-

Table 5-3. Building AM is a good alternate to the primary payload processing facility.

High Bay		
Bay dimensions	21 X 38 m	48 X 63 ft
Total area	795 m^2	8,750 ft^2
Crane type	Bridge	
Crane capacity	4,536 kg	5 ton
Crane hook height	10.7 m	35 ft
North Area		
Floor dimensions	10.7 X 21 m	35 X 70 ft
Total area	224.7 m^2	2,450 ft^2
Clear height	10.7 m	35 ft
Ceiling height	12.8 m	42 ft
Floor space	4.6 X 10.3 m	15 X 34 ft
South Area		
Floor dimensions	14.6 X 15.2 m	48 X 50 ft
Total area	221.9 m^2	2,400 ft^2
Clear height	10.7 m	35 ft
Ceiling height	12.8 m	42 ft
Floor space	4.6 X 6.1 (h) m	15 X 20 (h) ft
Clean Room Complex (consists of a clean room, airlock, equipment receiving room, personnel change room & anteroom)		
Clean Room		
Dimensions	4.4 X 5 m	14.3 X 16.5 ft
Clear ceiling height	3 m	10 ft

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Table 5-4. Building AO provides the largest high bay floor area.

High Bay Class 100,000 Clean Room		
Usable floor space	14 X 53.3 m	46 X 175 ft
Floor area	747.8 m^2	8,050 ft^2
Clear height	14 m	46 ft
Crane type	Bridge	
Crane capacity	9,072 kg	10 ton
Crane hook height	13.7 m	45 ft
Airlock		
Floor space	7.9 X 8.8 m	26 X 29 ft
Clear height	14.6 m	48 ft
Door size (2)	7.6 X 12.2 (h) m	25 X 40 ft
Crane type & number	Bridge (2)	
Crane capacity	9,072 kg	10 ton
Crane hook height	13.7 m	45 ft
Class 100 laminar flow clean room (second floor) No air shower nor airlock — "portable" used		
Dimensions	3.7 X 3.8 m	12 X 12.5 ft
Area	14.06 m^2	151.2 ft^2
Height	2.6 m	8.5 ft
Second Floor Freight Elevator		
Capacity	1816 kg	2 ton
Size	2.4 X 2.4 X 2.4 m	8 X 8 X 8 ft
Clear ceiling height	3 m	10 ft

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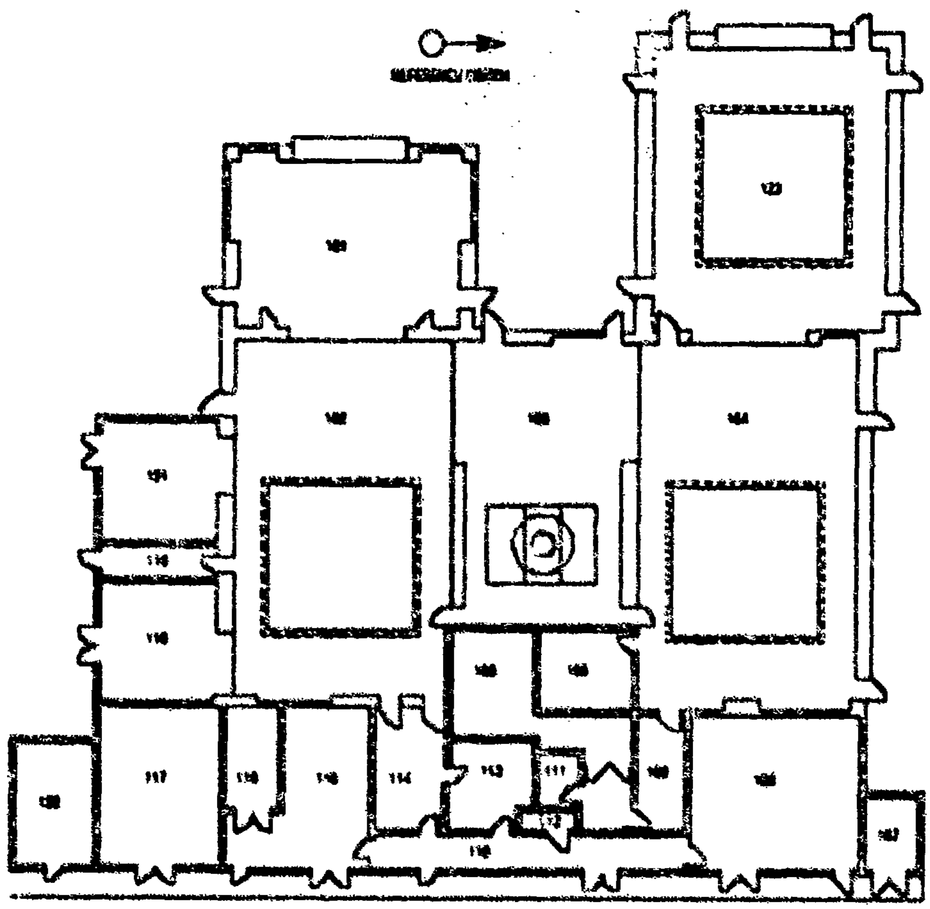
stallation. Table 5-5 lists the details of room dimensions, cleanliness, and crane capabilities of this facility.

Explosive Safe Area 60A (ESA 60A) — ESA 60A is located north of the CCAFS Industrial Area on Titan III Road. Designed and built by NASA, ESA 60A consists of four separate buildings:

- Sterilization and Assembly Building (5,660 square feet)
- Propellant Laboratory (1,100 square feet)
- Instrumentation Laboratory (1,350 square feet)
- Ground Support Equipment (GSE) Building (1,400 square feet)

The Sterilization and Assembly Building and 1,000 square feet of the Propellant Laboratory are Class 100,000 clean room environments. All test areas within ESA 60A are either reinforced concrete or earth revetments to meet blast requirements. Table 5-6 provides the details of this facility.



Rm	Function	Rm	Function	Rm	Function
101	South Airlock	109	North Change Room	115	South Control Room
102	South High-bay	110	Corridor	116	Balance Control Room
103	Center High-bay	111	Women's Restroom	118	Corridor
104	North High-bay	112	Janitor	119	Oxidizer Cart Room
105	Office	113	Men's Restroom	121	Fuel Cart Room
106	North Control Room	114	South Change Room	123	North Airlock High-bay

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Figure 5-6 Astrotech's Building 2 provides excellent hazardous operations facilities for liquid and solid propellants

Table 5-5. Astrotech's Building 2 features three blast-proof processing rooms.

Spacecraft Processing High Bay: Class 100,000 Clean Room		
Temperature	23.8 ± 2.8°C	75 ± 5°F
Relative humidity	50 ± 5%	
Usable floor space	18.27 x 11.58 m	60 x 37 ft
Floor area	211.6 m^2	2,220 ft^2
Clear height	13.11 m	43 ft
Crane type (each bay)	Bridge	
Crane capacity	9,072 kg	10 ton
Crane hook height	11.33 m	37.1 ft
Filtration	Class 100,000	
Floor trench (fueling)	7.62 x 7.62 m	25 x 25 ft
Spacecraft High Bay Control Room:		
Temperature	21.1 to 25.6°C	70 to 75°F
Size	7.62 x 7.62 m	25 x 25 ft
Area	58.1 m^2	625 ft^2
Ceiling	2.84 m	9.33 ft
Bay window	0.71 x 0.71 m	2.33 x 2.33 ft
Spin Balance High Bay: Class 100,000 Clean Room		
Temperature	23.8 ± 2.8°C	75 ± 5°F
Relative humidity	50 ± 5%	
Usable floor space	14.63 x 8.23 m	48 x 27 ft
Floor area	122.1 m^2	1,296 ft^2
Clear height	13.11 m	43 ft
Crane type (each bay)	Bridge	
Crane capacity	9,072 kg	10 ton
Crane hook height	8.23 m	27.0 ft
Filtration	Class 100,000	
Spin room doors	6.10 x 12.20 m	25 x 40 ft
Spin Balance High Bay Control Room:		
Temperature	23.8 ± 2.8°C	75 ± 5°F
Usable floor space	3 x 4.6 m	10 x 15 ft
Floor area	13.93 x 4.6 m^2	150 ft^2
Clear height	2.84 m	9.33 ft
Door size (2)	1.8 x 2 m	6 x 6.66 (h) ft
PAM Processing High Bay: Class 100,000 Clean Room		
Temperature	23.8 ± 2.8°C	75 ± 5°F
Relative humidity	50 ± 5%	
Usable floor space	18.29 x 11.27 m	60 x 37 ft
Floor area	206.1 m^2	2,220 ft^2
Clear height	13.11 m	43 ft
Crane type (each bay)	Bridge	
Crane capacity	9,072 kg	10 ton
Crane hook height	11.33 m	37.1 ft
Floor trench (fueling)	7.62 x 7.62 m	25 x 25 ft
PAM High Bay Control Room:		
Temperature	23.8 ± 2.8°C	75 ± 5°F
Size	9.14 x 7.62 m	30 x 25 ft
Area	69.6 m^2	750 ft^2
Ceiling	2.8 m	9.33 ft
Airlock: Class 100,000 Clean Room		
Temperature	23.8 ± 2.8°C	75 ± 5°F
Relative humidity	50 ± 5%	
Usable floor space	11.6 x 8.8 m	38 x 29 ft
Floor area	102.4 m^2	1,102 ft^2
Clear height	13.11 m	43 ft
Fuel and Oxidizer Storage Rooms		
Usable floor space	6.1 x 6.1 m	20 x 20 ft
Floor area	37.21 m^2	400 ft^2
Clear height	2.84 m	9.33 ft
Door size (to bay)	3.05 x 2.74 (h) m	10 x 9 (h) ft
Door size (exterior)	1.83 x 2.03 (h) m	6 x 6.66 (h) ft
Expansion High Bay Area: Class 100,000 Clean Room		
Temperature	23.8 ± 2.8°C	75 ± 5°F
Relative humidity	50 ± 5%	
Usable floor space	12.2 x 15.2 m	40 x 50 ft
Floor area	185.8 m^2	2,000 ft^2
Clear height	19.8 m	65 ft
Crane type (each bay)	Bridge	
Crane capacity	27,216 kg	30 ton
Crane hook height	16.8 m	55 ft
Floor trench (fueling)	7.62 x 7.62 m	25 x 25 ft
Door Size	6.1 x 15.2 (h) m	20 x 50 (h) ft
Spacecraft Control Room		
	TBS	TBS

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Spacecraft Assembly and Encapsulation Facility No. 2 (SAEF 2) — SAEF 2 is a NASA facility located southeast of the KSC Industrial Area. It features:

- High bay
- Low bays (two)
- Test cell
- Control rooms (two)

Details of the SAEF 2 facility are in Table 5-7.

Payload Hazardous Servicing Facility (PHSF) — The PHSF is a new NASA facility located southeast of the KSC Industrial Area (adjacent to SAEF2). Additional features of the PHSF Service Building are described in Table 5-8.

5.1.3 SPACECRAFT SUPPORT FACILITIES

Astrotech Building 3 (Payload Storage Building) — Astrotech's Building 3 is a thermally controlled storage area for short-term storage in conjunction with payload processing activities or long-term satellite storage. Storage bay and door dimensions and thermal control ranges are identified in Table 5-9.

Astrotech Building 4 (Warehouse Storage Building) — Astrotech's Building 4 is a storage area without environmental controls and is suitable for storage of shipping containers and mechanical GSE. Table 5-10 details this facility's dimensions.

Astrotech Building 5 (Customer Office Building) — Astrotech's Building 5 provides 3,600 ft^2 (334.4 m^2) of office space divided into 17 individual offices with a reception area sufficient to accommodate up to three secretaries.

5.1.4 SPACECRAFT SERVICES

ELECTRICAL POWER AND LIGHTING — The Astrotech facility is served by 480 Vac/three-phase commercial 60 Hz electrical power that can be redis-

Table 5-6. ESA 60A provides diverse services for hazardous processing operations.

Spacecraft Assembly Building		
High Bays (2)		
Floor dimensions	11.6 X 17 m	38 X 56 ft
Floor area	197.7 m²	2,126 ft²
Ceiling height	11.9 m	39 ft
Door dimensions	5.5 X 10.7 (h) m	18 X 35 (h) ft
Airlock		
Usable floor space	7.3 X 14.6 m	24 X 48 ft
Crane type	Bridge	
Crane capacity	4536 kg	5 ton
Crane hook height	10.4 m	34 ft
Door dimensions	7.3 X 10.7 m	24 X 35 ft
Environmental Controls		
Filtration	Class 100,000	
Temperature	20 to 35 ± 1.7°C	68 to 95 ± 3°F
Relative humidity	50 ± 5%	
Propellant Laboratory: Single high bay with earth revetments on the north & south walls. A dynamic balance table capable of 0 to 300 rpm and weights to 5445 kg (12,000 lb) is located in the high bay.		
Usable floor space	9.1 X 11 m	30 X 36 ft
Floor area	100.1 m²	1,080 ft²
Crane type	Bridge	
Crane capacity	9072 kg	10 ton
Crane hook height	9.4 m	31 ft
Equipment door	6 X 9.1 (h) m	20 X 30 (h) ft
Environmental Controls		
Filtration	Class 100,000	
Temperature	23 ± 1.7°C	73 ± 3°F
Relative humidity	50 ± 5%	
Instrumentation Laboratory & Storage Building: The instrumentation Building contains a cleaning laboratory, shop & instrumentation laboratory		
Storage floor space	9.1 X 15.2 m	30 X 50 ft
Storage area door	3 X 6 m	10 X 20 ft
Crane type	Monorail	
Crane capacity	5443 kg	6 ton
Crane hook height	10.3 m	33.75 ft

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tributed as 480 Vac/three-phase/30A, 120/180 Vac/three-phase/60A, or 120 Vac/single-phase, 20A power to any location in Buildings 1 and 2. Commercial power is backed up by a diesel generator during critical testing and launch periods. Astrotech can provide 35 kV of 50 Hz power, which is also backed up by a diesel generator.

The high bays and airlocks in Buildings 1 and 2 are lighted by 400-watt metal halide lamps to maintain 100 foot-candles illumination. Control rooms, offices, and conference areas have 35-watt fluorescent lamps to maintain 70 foot-candles of illumination.

COMMUNICATIONS

- **Telephone and Facsimile** — Astrotech provides all telephone equipment, local telephone service,

Table 5-7. SAEF 2 is the larger of the alternate hazardous processing facilities.

Control Rooms (2)		
Floor space	9.14 X 10.97 m	30 X 36 ft
Floor area	100.3 m²	1,080 ft²
Raised flooring	0.31 m	1 ft
Ceiling height	2.44 m	8 ft
High Bay		
Floor size	14.94 X 30.18 m	49 X 99 ft
Floor area	450.9 m²	4,851 ft²
Clear ceiling height	22.56 m	74 ft
Filtration	Class 100,000	
Crane type (each bay)	Bridge	
Crane capacity	9072 kg	10 ton
Crane hook height	19.8 m	65.0 ft
Low Bays (2)		
Floor size (No. 1)	5.79 X 21.95 m	19 X 72 ft
Floor area (No. 1)	127.1 m²	1,368 ft²
Clear height (No. 1)	7.62 m	25 ft
Floor size (No. 2)	5.79 X 8.23 m	19 X 27 ft
Floor area (No. 2)	47.7 m²	513 ft²
Clear height (No. 2)	13.26 m	44 ft
Test Cell		
Floor size	11.28 X 11.28 m	37 X 37 ft
Floor area	127.2 m²	1,369 ft²
Clear ceiling height	15.85 m	52 ft
Door size	6.7 X 12.2 m	22 X 40 (h) ft

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Table 5-8. The PHSF Service Building is a new NASA facility for hazardous processing.

Service bay		
Floor space	18.4 X 32.6m	60 X 107 ft
Ceiling height	28.9m	94 ft 10 in
Door dimensions	10.8 X 22.9m	35 X 75 ft
Crane capacity	45,400 kg	50 ton
Hook height	25.5m	83 ft 6 in
Airlock		
Floor space	15.3 X 25.9m	50 X 80 ft
Ceiling height	27.4m	89 ft 10 in
Door dimensions	10.8 X 22.9m	35 X 75 ft
Crane capacity	13,600 kg	15 ton
Hook height	22.9m	75 ft
Equipment airlock		
Usable space	4.4 X 8.0m	14 X 26 ft
Ceiling height	3.2m	10 ft 4 in
Door dimensions	3.0 X 3.0m	10 X 10 ft
Environmental controls		
Filtration	Class 100,000 (planned)	
Air change rate	Four per hour minimum	
Temperature	21.7 ± 3.3C	71 ± 6F
Relative humidity	50%	

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and long distance access. A Group 3 facsimile machine is available and commercial telex service can be arranged.

- **Intercommunication Systems** — Astrotech provides a minimum of three channels of voice communications among all work areas. The facility is connected to the NASA/USAF Operational In-

Table 5-9 Astrotech's Payload Storage Building is used for short-term hardware storage.

Building 3: Thermally controlled (21.1 to 25.6°C; 70 to 78 °F) storage facility		
Floor space (6 rooms)	7.62 X (6.71 or 7.32) m	25 X (22 or 24) ft
Floor area (6 rooms)	51.1 or 55.8 m²	550 or 600 ft²
Clear height	8.53 m	28 ft
Door size (4 rooms)	6.1 X 7.62 (h) m	20 X 25 (h) ft
Door size (2 rooms)	5.49 X 7.62 (h) m	18 X 25 (h) ft

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Table 5-10. Astrotech's Warehouse Storage Building is suitable for storage of items not requiring climate controls.

Building 4: Storage without environmental control		
Floor space	15.24 X 38.1 m	50 X 125 ft
Floor area	580.6 m²	6,250 ft²
Door size	5.49 X 7.92 m	18 X 26 ft

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tercommunications System and Transistorized Operational Paging System (TOPS) to provide multiple-channel voice communications between the Astrotech facility and selected locations at Cape Canaveral Air Force Station.

- **Closed-Circuit Television (CCTV)** — CCTV cameras are located in the high bays of Building 2 and can be placed in the high bays of Building 1, as required, to permit viewing operations in these areas. CCTV can be distributed within the Astrotech facility to any location desired. In addition, Astrotech has the capability to transmit and receive a single channel of CCTV to and from KSC/CCAFS via a dedicated microwave link.
- **Command and Data Links** — Astrotech provides both wideband and narrowband data transmission capability via a dedicated microwave link and the KSC/CCAFS cable transmission system to all locations served by the KSC/CCAFS network. If a spacecraft requires a hardline transmission capability, the spacecraft is responsible for providing correct signal characteristics to interface to the KSC/CCAFS cable transmission system.

Astrotech provides antennas for direct S-band, C-band, and Ku-band air links from the Astrotech facility to Launch Complex 36, and antennas for C-band and Ku-band air links between Astrotech Buildings 1 and 2.

OTHER SERVICES

- **Temperature/Humidity Control** — The environment of all Astrotech high bays and airlocks is maintained at a temperature of 24 ± 2.8°C (75 ± 5°F) and a relative humidity of 50 ± 5%. The environment of all other areas is maintained by conditioned air at a temperature between 21 and 25°C (70 to 78°F) and a comfortable humidity.
- **Compressed Air** — Regulated compressed air at 125 psi is available in Buildings 1 and 2.
- **Security and Emergency Support** — Perimeter security is provided 24 hours a day. Access to the Astrotech facility is via the main gate, where a guard is posted during working hours to control access. Internal security is provided by cypher locks on all doors leading into payload processing areas. Emergency medical support is provided by Brevard County and emergency fire support by the City of Titusville. In case of an accident, personnel will be transported to Jess Parish Hospital in Titusville. Both medical and fire personnel have been trained by NASA.

5.2 SPACECRAFT INSTRUMENTATION SUPPORT FACILITIES

The CCAFS area facilities described in this section can be used for spacecraft checkout as limited by compatibility to the spacecraft systems. Special arrangements and funding are required to utilize these assets.

5.2.1 TEL 4 TELEMETRY STATION — The ESMC operates an S-band telemetry receiving, recording, and real-time relay system on Merritt Island. This system is used for prelaunch checkout of launch vehicles and spacecraft. A typical ground checkout configuration would include a reradiating antenna at the PPF, HPF, or launch pad directed toward the Tel 4 antenna. The telemetry data can be recorded on magnetic tape or routed by hardline data circuits to the spacecraft ground station for analysis. Tel 4 also acts as the primary terminal for telemetry data transmitted from the ESMC downrange stations.

5.2.2 GSFC GSTDN/TDRSS MILA STATION— The Goddard Space Flight Center (GSFC) station is also located on Merritt Island and is the ESMC launch area station for NASA's Ground Spaceflight Tracking and Data Network (GSTDN). Included are satellite ground terminals providing access to worldwide communications. Circuits from MILA to HPF, PPF, and Complex 36 are available to support checkout and network testing during prelaunch operations as well as spacecraft telemetry downlinking during launch and orbital operations.

The MILA station can also support ground testing with Tracking and Data Relay Satellite System (TDRSS)-compatible spacecraft to include TDRSS links to White Sands, New Mexico. Special arrangements and documentation are required for TDRSS testing. The GSTDN is scheduled for phaseout when the TDRSS system becomes fully operational.

5.2.3 JPL MIL-71 STATION — This station is colocated at MILA on Merritt Island and is an element of the Jet Propulsion Laboratory (JPL) Deep Space Network (DSN). This station can be configured for ground tests similar to Tel 4. In addition, data from spacecraft that are compatible with the DSN can be relayed to the JPL in Pasadena, California.

5.3 SPACE LAUNCH COMPLEX 36 CONFIGURATION

The Atlas launch facility is Space Launch Complex 36 (Figure 5-7), located at CCAFS. The major facilities include the mobile service tower (MST), umbilical tower (UT), and the blockhouse. See Figure 5-8 for a plan view of Space Launch Complex 36.

5.3.1 MOBILE SERVICE TOWER (MST) — The MST (Figure 5-9) is an open steel structure with an interior enclosure that contains retractable vehicle servicing and checkout levels/platforms. The tower contains an electric, trolley-mounted 10-ton (9072 kg) overhead bridge crane used to hoist spacecraft, fairings, and the upper stage vehicle into position. Two elevators serve all MST levels. The entire MST assembly is on a rail system, which allows it to be moved from the launcher platform for launch.

RF cabling and reradiating antennas can be made available on the service tower for spacecraft use.

5.3.2 UMBILICAL TOWER — The umbilical tower (UT) (see Figure 5-10) is a fixed structural steel tower extending above the launch pad. Retractable service booms are attached to the UT. The booms provide electrical power, instrumentation, propellants, pneumatics, and conditioned air or GN_2 to the vehicle and spacecraft. These systems also provide quick-disconnect mechanisms at the respective vehicle interface and permit boom retraction at vehicle launch. A payload umbilical junction box is provided to interconnect the spacecraft to the electrical ground support equipment. Limited space is available within this junction box to install spacecraft-unique electrical ground support equipment.

5.3.3 LAUNCH PAD GROUND SYSTEM ELEMENTS — The launch complex is serviced by gaseous nitrogen (GN_2), gaseous helium (GHe), and propellant storage facilities within the complex area. Environmental Control Systems (ECS) exist for both the launch vehicle and the spacecraft. Detailed descriptions of the capabilities of these systems to provide for spacecraft activity is defined in Section 4.2, Spacecraft to Ground Equipment Interfaces.

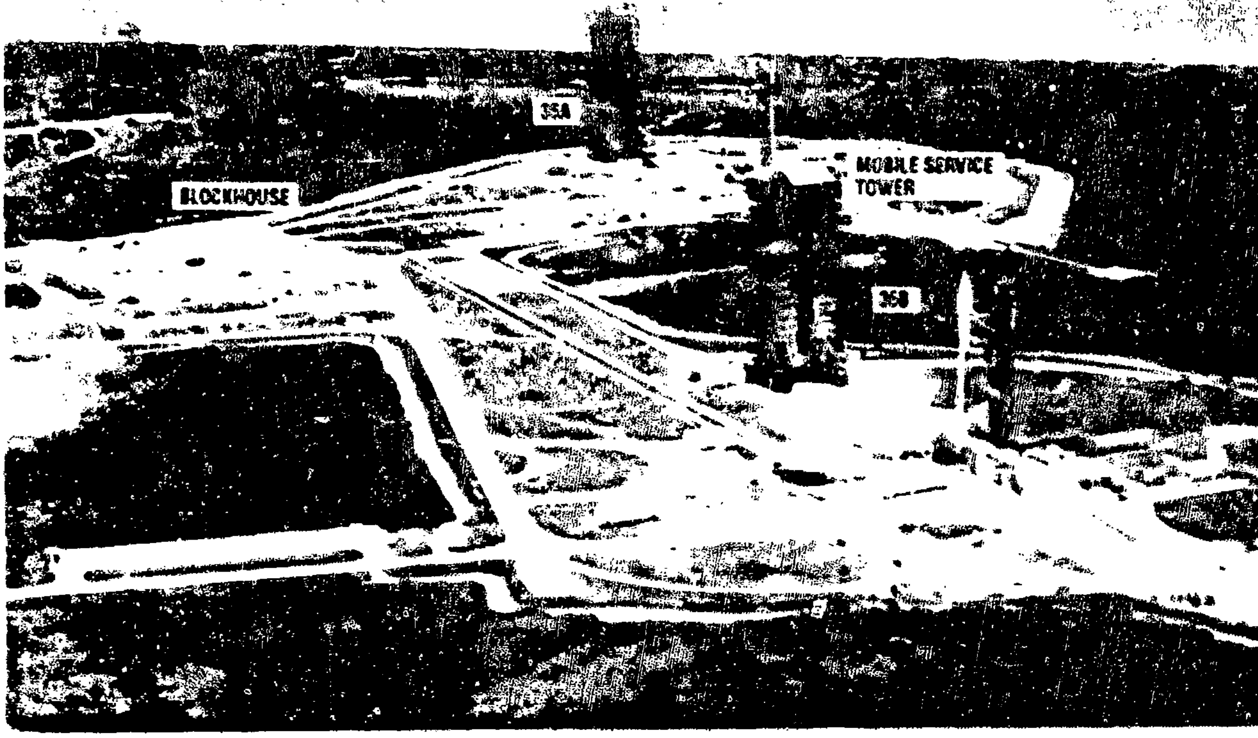

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Figure 5-7 Space Launch Complex SLC-36

5.3.4 BLOCKHOUSE — The blockhouse (Figures 5-11 and 5-12) serves as the operations and communications center for the launch complex. It contains all necessary control and monitoring equipment. The launch control, electrical, landline instrumentation, and ground computer systems are the major systems in this facility.

The launch control provides consoles and cabling for control of the launch complex systems. The landline instrumentation system (coupled with the closed-circuit TV system) monitors and records safety and performance data during test and launch operations. The ground computer system consists of redundant computer-controlled launch sets (CCLS) and a telemetry ground station. The CCLS provides control and monitoring of the vehicle guidance, navigation, and control systems and monitors vehicle instrumentation for potential anomalies during test and launch operations.

5.3.5 LAUNCH SERVICE BUILDING — The launch service building (LSB) provides the means for erecting the launch vehicle, interconnecting electrical wiring between the umbilical tower and the blockhouse, and locating spacecraft remote GSE; and it provides the launching platform. The LSB is a two-story structure as shown in Figure 5-13.

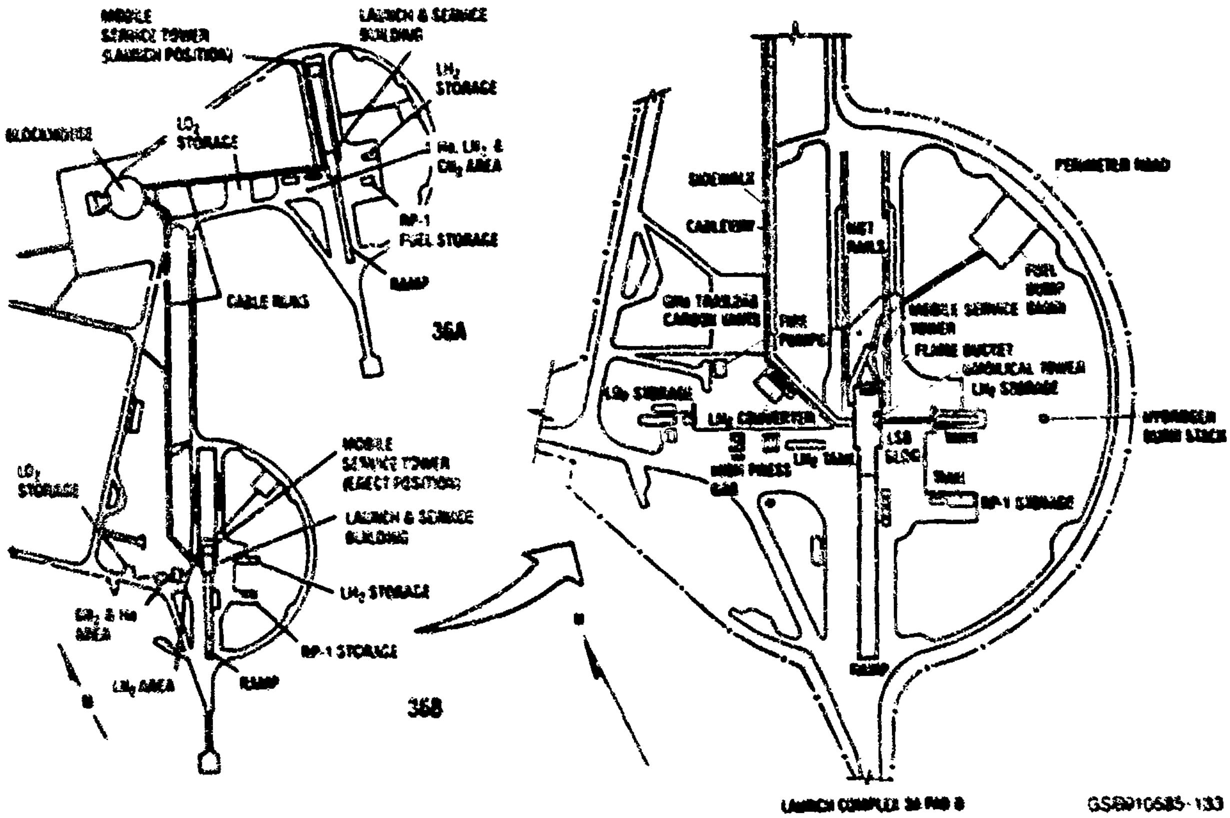

Figure 5-8. Space Launch Complex SLC-36B at CCAFS

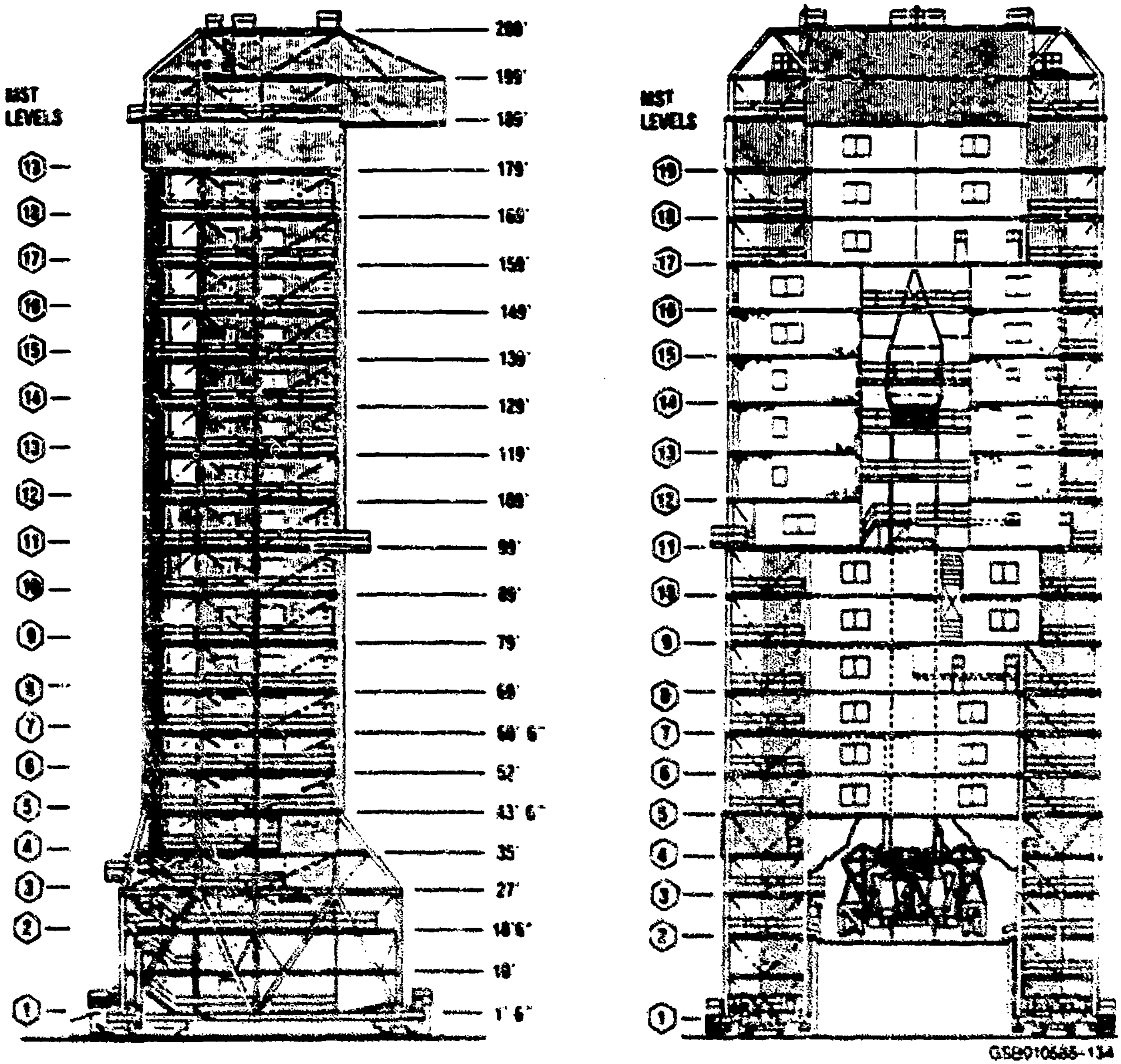

Figure 5-9 The SLC-36B MST provides access and environmental protection to the launch vehicle and spacecraft during prelaunch processing

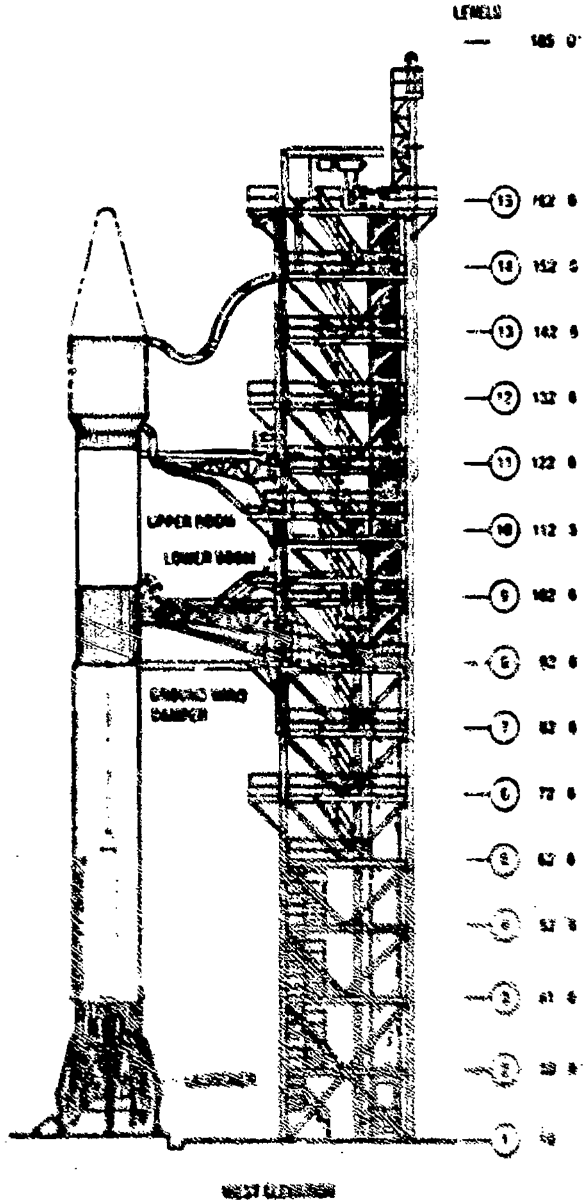

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Figure 5-10 The SLC-36B umbilical tower retractable service arms allow for quick disconnect and boom retraction at vehicle launch.

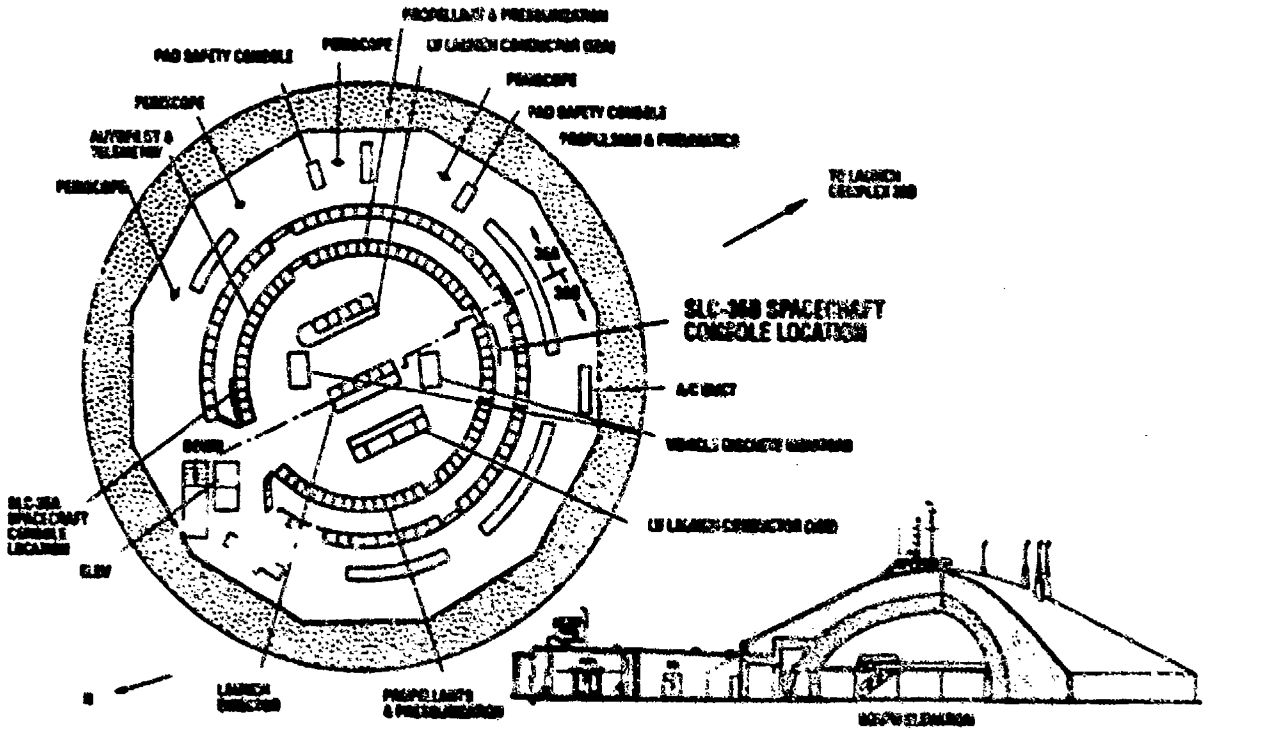

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Figure 5-11. Test and launch operations are controlled and monitored from the blockhouse.

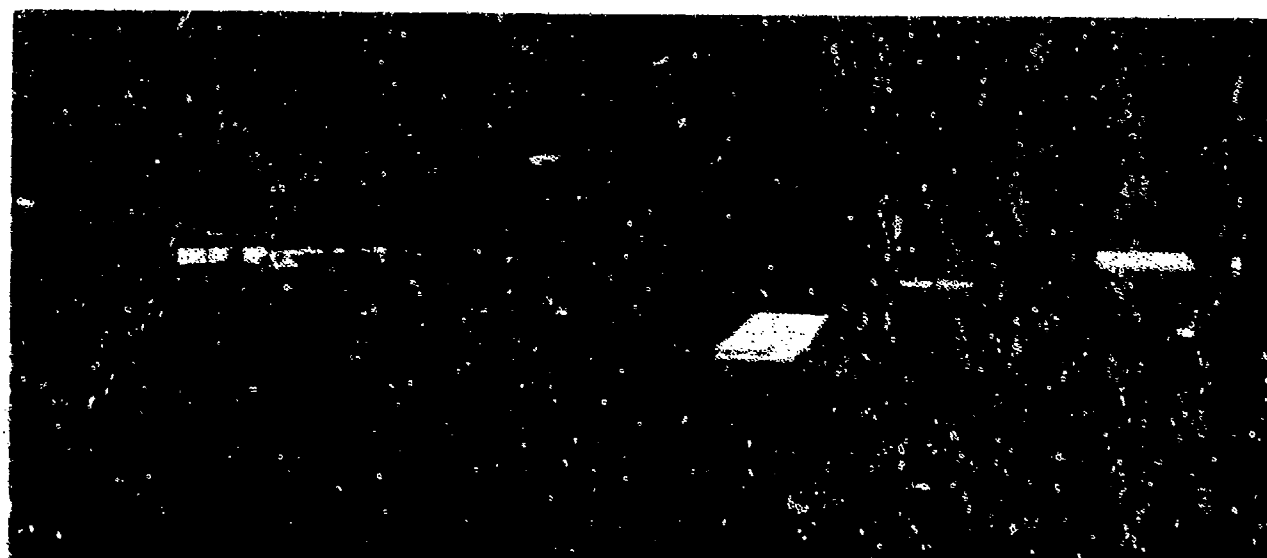

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Figure 5-12. Aerial view of common SLC-36 blockhouse.

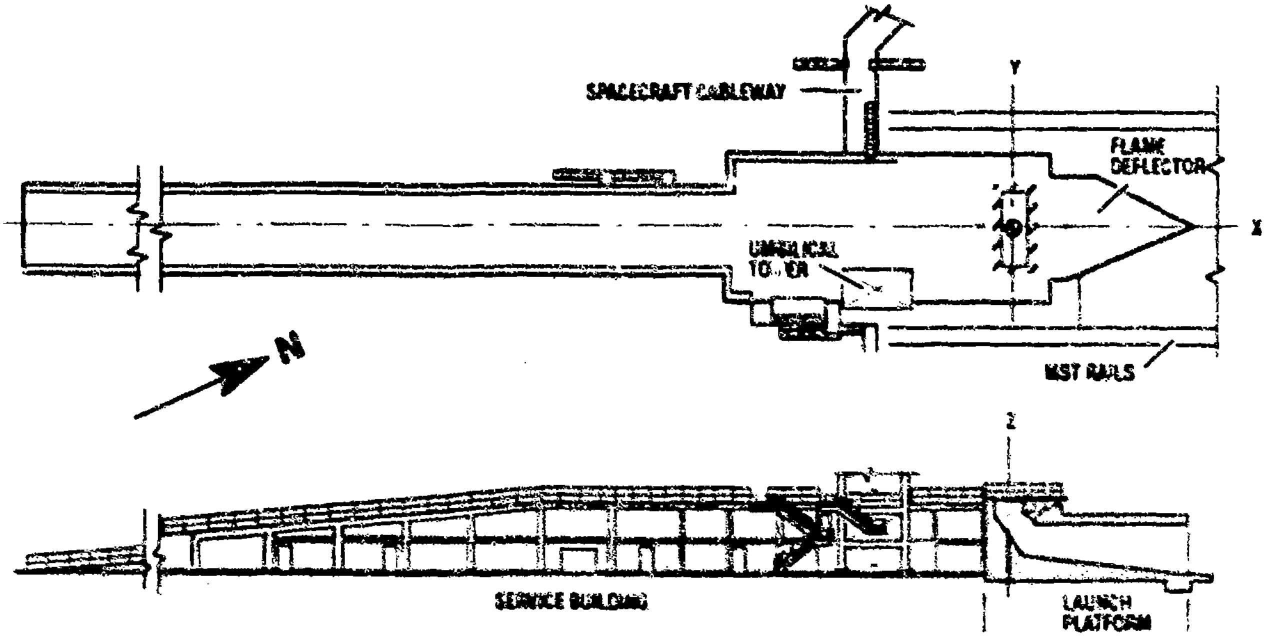

Figure 5-13 SLC-36B launch and service building

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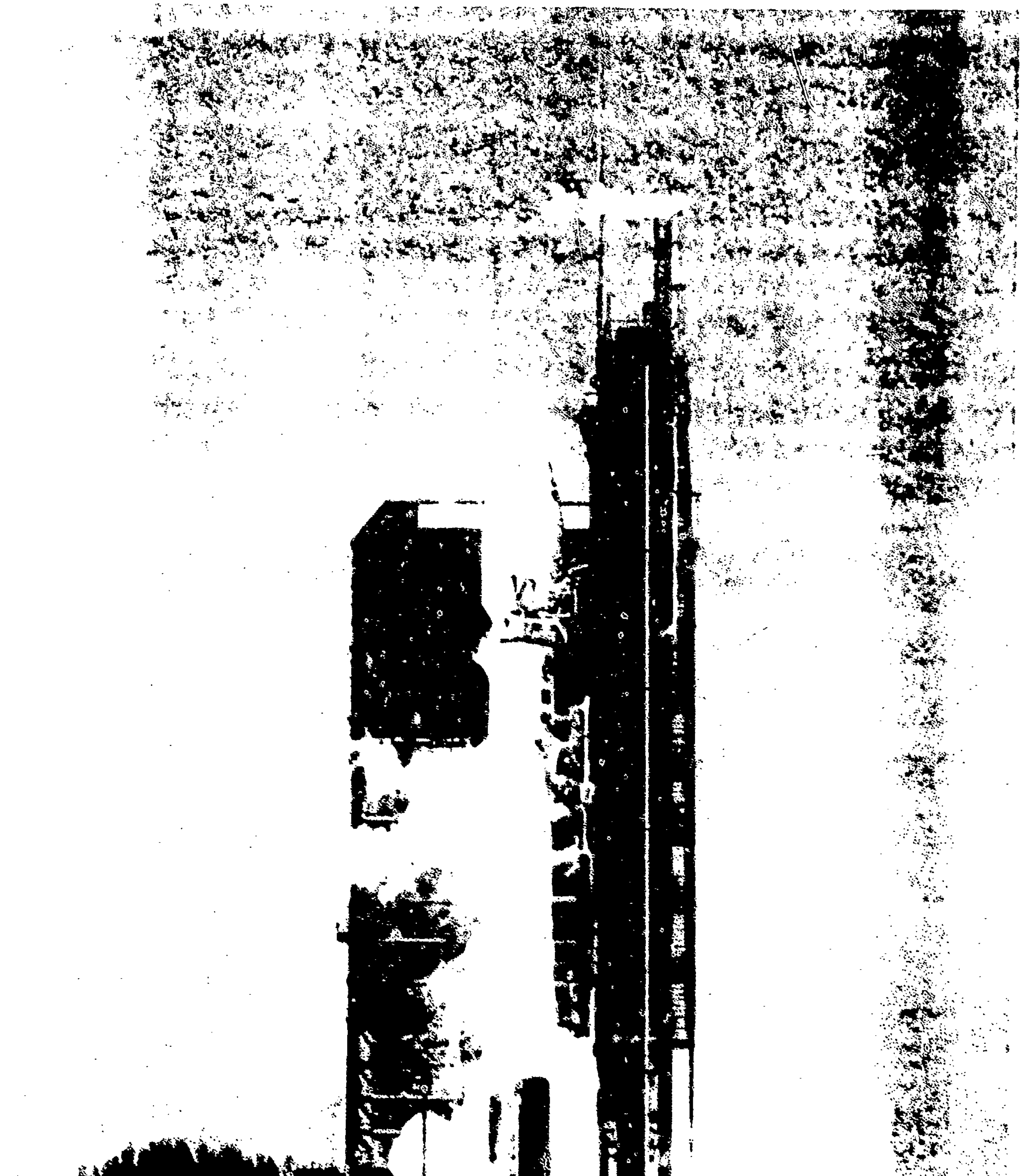

6 • MISSION MANAGEMENT AND LAUNCH OPERATIONS SERVICES

6.1 SPACECRAFT/LAUNCH VEHICLE INTEGRATION

Clear communication between the spacecraft and launch vehicle contractors is vital to mission success. Procedures and interfaces have been established to delineate areas of responsibility and authority.

6.1.1 LAUNCH VEHICLE RESPONSIBILITIES — General Dynamics is responsible for Atlas design, integration, checkout, and launch. This work is done primarily at the General Dynamics Space Systems Division's Kearny Mesa Plant in San Diego, shown in Figure 6-1. Major subcontractors are Pratt & Whitney (upper stage main engines), Honeywell (inertial navigation unit), and Rockwell International—Rocketdyne (Atlas engines). As the spacecraft-to-launch vehicle integrating contractor, General Dynamics is responsible for payload integration (i.e., electrical, mechanical, environmental, and electromagnetic compatibility), guidance system integration, mission analysis, software design, range safety documentation/support, launch site processing and coordination etc. General Dynamics produces all launch vehicle-related software for Atlas launches and is responsible for launch vehicle ascent trajectory, data acquisition, performance analysis, targeting, guidance analysis, and range safety analysis.

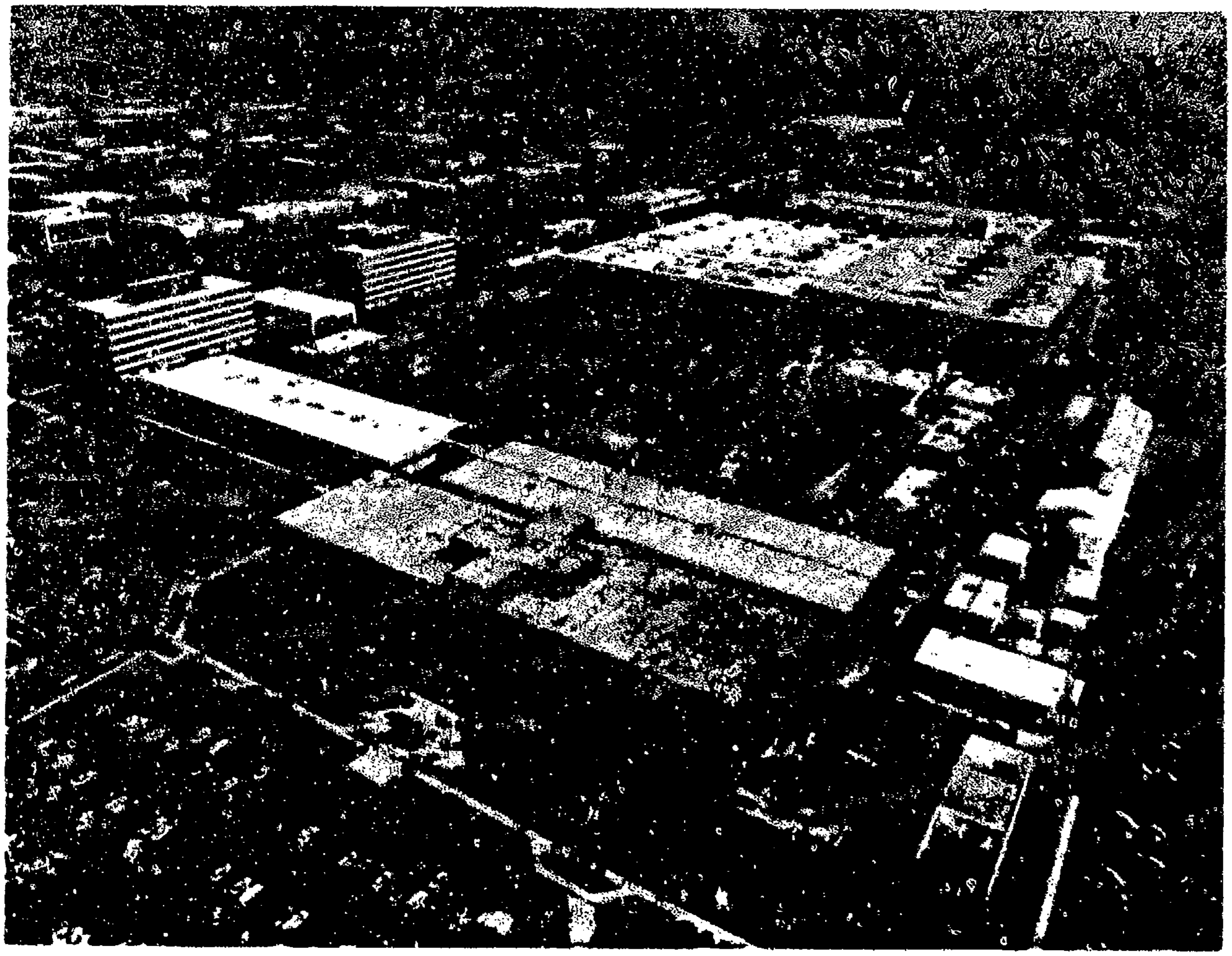

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Figure 6-1 General Dynamics Kearny Mesa Plant

6.1.2 SPACECRAFT RESPONSIBILITIES — Each spacecraft and mission has unique requirements. Interested Atlas users are encouraged to discuss their particular needs with General Dynamics. Appendix B, spacecraft information requirements, can be used as a guide to initiating dialogue. Items in bold in Appendix B should be used as the basis for the first face-to-face meeting between General Dynamics and the potential user to assist in determining spacecraft/launch vehicle compatibility. Customers are encouraged to contact General Dynamics to verify the latest launch information, including:

- Hardware status and plans
- Launch and launch complex schedules
- Hardware production schedule and costs

6.1.3 INTEGRATION MANAGEMENT — For each Atlas mission, General Dynamics Commercial Launch Services assigns a mission program manager. The mission manager is responsible for overall management of the particular customer activities at San Diego, California and at CCAFS, Florida. He is the principal interface with the customer for all technical and launch vehicle/satellite interface and integration matters. The CLS staff (Figure 6-2) supports the mission program manager.

The mission chief engineer assigns a dedicated mission integrator for each Atlas mission. The mission integrator is responsible for the timely engineering integration of the spacecraft with the Atlas launch vehicle.

The Commercial Launch Services organization is aligned to provide low-risk launch services. As illustrated by Figure 6-3, CLS responds to a customer order by arranging services from several other organizations. General Dynamics Space Systems Division is the subcontractor responsible for Atlas production, launch, and mission-peculiar integration. CLS has contracts in place with NASA for use of the CCAFS launch complex and payload integration facilities, and with the Air Force for range and launch site services. Astrotech payload integration facilities are contracted for commercial Atlas

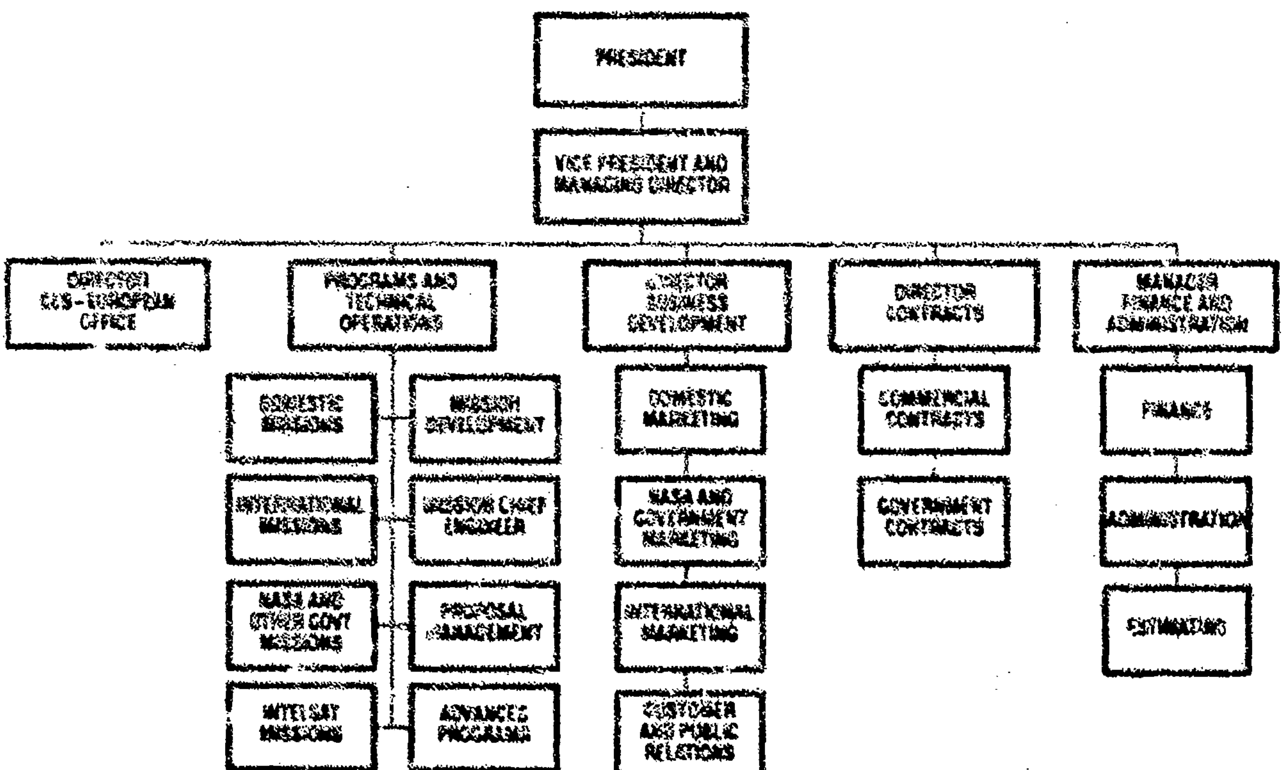

Figure 6-2 Commercial Launch Services organization

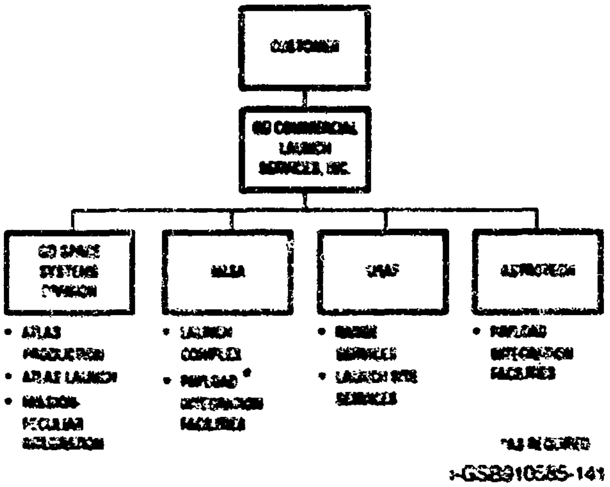

Figure 6-3 Commercial Launch Services implements required services

launches, with additional support from CCAFS when required.

To provide maximum efficiency for management of the many CCAFS operations, a launch operations manager is assigned to each mission. He represents the mission manager during development, integration, and installation of all spacecraft-peculiar items at the launch site and arrival of the spacecraft. This program organization concept has been used successfully for all major Atlas programs.

General Dynamics' approach to integration management is through establishment of a formal Interface Control Document (ICD) agreement and formal configuration control following ICD signature. Existing ICDs may be adapted to reduce the development time. Coordination of the tasks required to develop and maintain the ICD is accomplished through management and technical working groups.

6.1.4 WORKING GROUPS AND RESPONSIBILITIES — In all phases of the mission, from go-ahead to launch, interface activities between the spacecraft and launch vehicle contractors are coordinated by specialized working groups. These groups, which include the spacecraft contractor as an active participant, develop schedules, monitor progress, and ensure that the technical and management tasks are accomplished properly and on time. Figure 6-4 shows the typical working groups and their responsibilities.

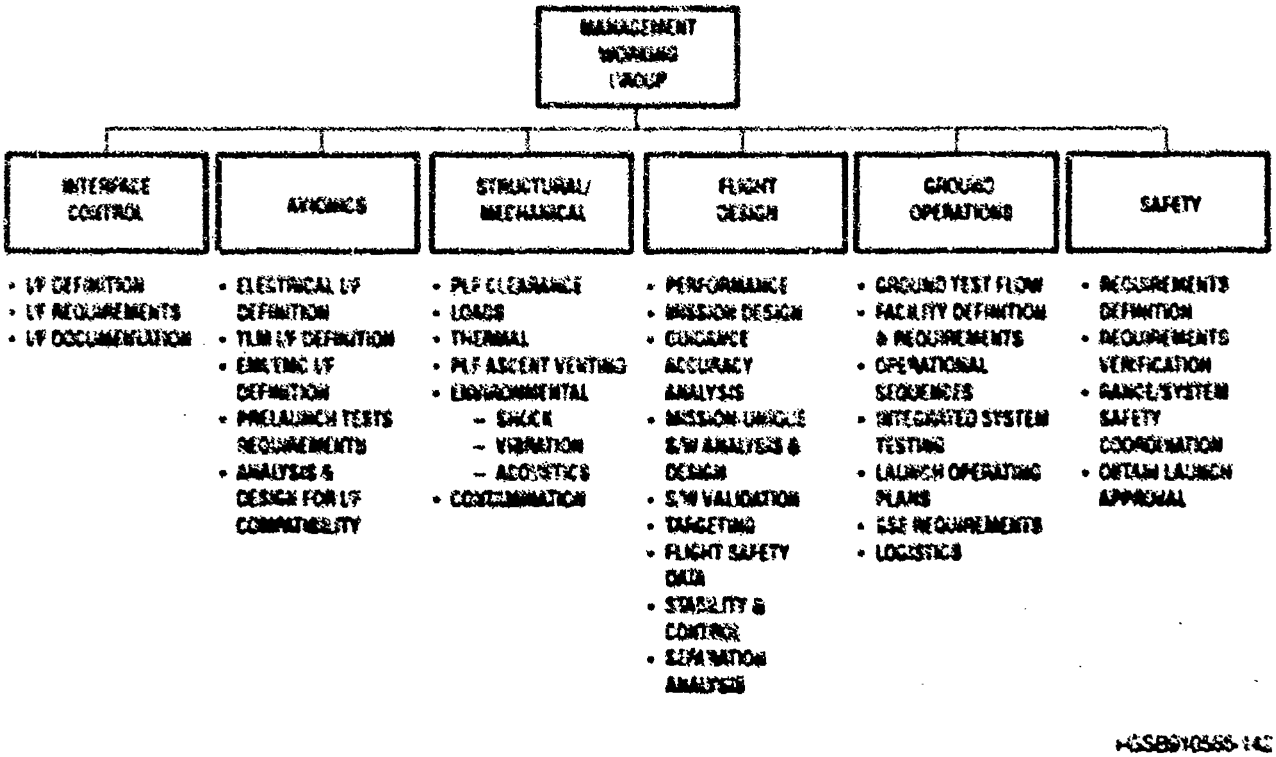

Figure 6-4 Typical working groups and responsibilities

The technical working groups can be tailored to support unique payload integration requirements. It should be noted that technical personnel frequently participate in multiple working groups, where qualifications allow, to minimize the personnel requirements of the participating contractor.

The contractor and the working groups exchange information through minutes and action items developed during the meetings and through control drawings and other documentation.

6.1.4.1 Management Working Group (MWG) — The technical and program managers comprise the Management Working Group (MWG). This group is responsible for coordinating and managing the efforts of all mission participants throughout all phases of the mission. This group establishes policy and provides guidelines to the integration effort. Where interface incompatibilities are not resolved within the working group structure, the MWG provides the direction required to achieve spacecraft/launch vehicle system interface compatibility. The group also manages the mission integration schedule, which monitors integration status including interface documentation, analyses documentation, hardware interchange, and systems and launch operations integrated tests.

6.1.4.2 Technical Working Groups — Technical working groups are organized by function and operate under MWG authority. For items that require specialized attention for resolution, technical interchange meetings (TIMs) can be called by either the MWG or a working group.

6.1.5 MANAGEMENT WORKING GROUP MEETINGS — The MWG convenes quarterly and is attended by representatives from GD, the spacecraft contractor, and the launch services customer. The GD MWG senior member is the mission manager, who is supported by members of the GD staff as required. The MWG meeting provides an overall program review, including review of the Master Schedule and Interface Scheduling Document (ISD), approval of program documentation, and review of all outstanding action items and action item closures since the previous MWG.

In general, the technical working groups convene as required to coordinate the integration tasks and schedules. Working group meetings are attended by the technical representatives from each organization. Meeting objectives include: development of data exchange lists, a review of outstanding action items, a status of scheduled activities, and a discussion of outstanding issues or concerns.

GD also participates (either in San Diego or at the spacecraft facility) in senior management meetings held every six months. These meetings review the overall status of the launch vehicle, spacecraft, integration activities, and other matters that mutually interest spacecraft and launch vehicle contractors.

6.1.6 INTEGRATION PROGRAM REVIEWS — During the integration process, reviews are held to focus management attention on significant milestones during the launch system design and launch preparation process. As with the working group meetings, these reviews can be tailored to the user requirements; however, for a first-time launch, they include at least a preliminary and critical mission-peculiar design review and a mission readiness review.

In recognition of a valuable "check and balance" provided in the past by NASA, CLS has established an independent technical oversight function. This group of senior personnel participates in technical and mission readiness reviews and participates in the formal review processes: Engineering Review Board (ERB), Preliminary Design Review (PDR), Critical Design Review (CDR), and final launch readiness.

6.1.6.1 Mission-Peculiar Design Reviews — Prior to initiating detailed mission-peculiar design, GD holds a PDR to demonstrate compatibility between the design and mission-peculiar requirements. Prior to committing the design to production, GD conducts a CDR to ensure the released design meets the mission-peculiar requirements. GD prepares and presents the reviews with participation from the spacecraft contractor, launch services customer, and launch vehicle management.

6.1.6.2 Mission Readiness Review — This review, conducted approximately one week prior to launch, provides a final prelaunch assessment of the integrated spacecraft/launch vehicle system and launch facility readiness. The Mission Readiness Review provides the forum for final assessment of all launch system preparations and for the contractors' individual certifications of launch readiness.

6.1.7 INTEGRATION CONTROL DOCUMENTATION

6.1.7.1 Mission Integration Schedule — This top-level schedule is prepared by GD and monitored by the MWG. It maintains visibility and control of all major program milestone requirements including working group meetings, major integrated reviews, design and analysis requirements, and major launch operations tests. It is developed from the tasks and schedule requirements identified during the initial integration meetings and is used by all participating organizations and working groups to develop and update sub-tier schedules.

6.1.7.2 Interface Requirements Documents (IRD) — The customer creates the IRD to define technical and functional requirements imposed by the spacecraft on the launch vehicle system. The document contains the applicable spacecraft data identified in Appendix B. Information typically given includes:

- Mission requirements, including orbit parameters, launch window parameters, preseparation and separation functions, and any special trajectory requirements such as thermal maneuvers and separation over a telemetry and tracking ground station
- Spacecraft characteristics, including physical envelope, mass properties, dynamic characteristics, contamination requirements, acoustic and shock requirements, thermal requirements, and any special safety issues
- Mechanical and electrical interfaces, including spacecraft mounting constraints, spacecraft access requirements, umbilical power, command and telemetry, electrical bonding, and EMC requirements
- Mechanical and electrical requirements for ground equipment and facilities including: spacecraft handling equipment, checkout and support services, prelaunch and launch environmental requirements, spacecraft gases and propellants, spacecraft RF power, and monitor and control requirements
- Test operations, including spacecraft integrated testing, countdown operations, and checkout and launch support

6.1.7.3 Interface Control Document (ICD) — This document defines spacecraft-to-launch vehicle and launch complex interfaces. All mission-peculiar requirements are documented in the ICD. The ICD is prepared by GD for the Interface Control Working Group and is under configuration control after formal sign-off. The document contains the technical and functional requirements contained in the IRD, and any additional requirements developed during the integration process. The ICD supersedes the IRD and is approved with signature by both CLS and the launch service customer.

6.1.7.4 Interface Scheduling Document (ISD) — This document contains the schedules and planning data necessary to accomplish integration of the spacecraft with the Atlas launch system. The document is prepared by GD for the MWG, with inputs from the launch service customer. Signatory approval is required from both GD and the customer. The ISD reflects actions and agreements made by the MWG, which has responsibility for monitoring and reporting schedule status. ISD events include interface documentation, analyses documentation, hardware interchange, combined systems tests, and launch operations integrated tests.

6.1.7.5 Mission Design and Analysis Documentation — Mission design and analysis requirements are defined in the ICD. Integration analyses generally include:

- Flight vehicle trajectory analysis through spacecraft separation, guidance accuracy analysis, and separation analysis
- Spacecraft and launch vehicle coupled loads analysis utilizing integrated dynamic models
- SC-to-LV clearance analysis
- Software development and targeting
- Mission trajectory-dependent analysis, such as payload fairing venting
- Launch vehicle stability and control analysis
- Analysis and hardware design to incorporate mission-peculiar changes to the baseline vehicle/launch complex
- Radio frequency compatibility and EED analysis
- RF link analysis
- EMI/EMC compatibility analysis

Mission analyses and design are scheduled in the mission integration schedule and are formally documented in reports generated by GD. Updates are provided as required and agreed to during the integration process. Figure 6-5 is a typical mission integration schedule showing when data is exchanged and analysis is performed.

6.2 VEHICLE INTEGRATION/LAUNCH OPERATIONS

6.2.1 ORGANIZATION AND RESPONSIBILITIES — General Dynamics provides complete vehicle integration and launch services for its customers. A system of facilities, equipment, and personnel trained in launch vehicle/spacecraft integration and launch operations is in place. The following subsections summarize the types of support and services available. Figure 6-6 shows typical factory-to-launch operations flow.

6.2.1.1 Vehicle/Spacecraft Integration — GD performs launch vehicle/spacecraft integration and interface verification testing. This testing includes:

a. Matchmate testing of interface hardware at the spacecraft contractor's facility:
   - Prototype items
     - For early verification of design
     - For accessibility to install equipment
     - For development of handling/installation procedures
   - Flight items
     - For verification of critical mating interfaces prior to hardware delivery to launch site
     - Separation system installation
     - Bolt hole pattern alignments/indexing
     - Mating surface flatness checks
     - Electrical conductivity checks
     - Electrical harness cable lengths
     - Electrical connector mechanical interface compatibilities
b. Avionics/electrical system interface testing in the Systems Integration Laboratory (SIL) at San Diego, using a spacecraft simulator or prototype test items for verifying functional compatibility of:
   - Data/instrumentation interfaces
   - Flight control signal interfaces
   - Pyrotechnic signal interfaces

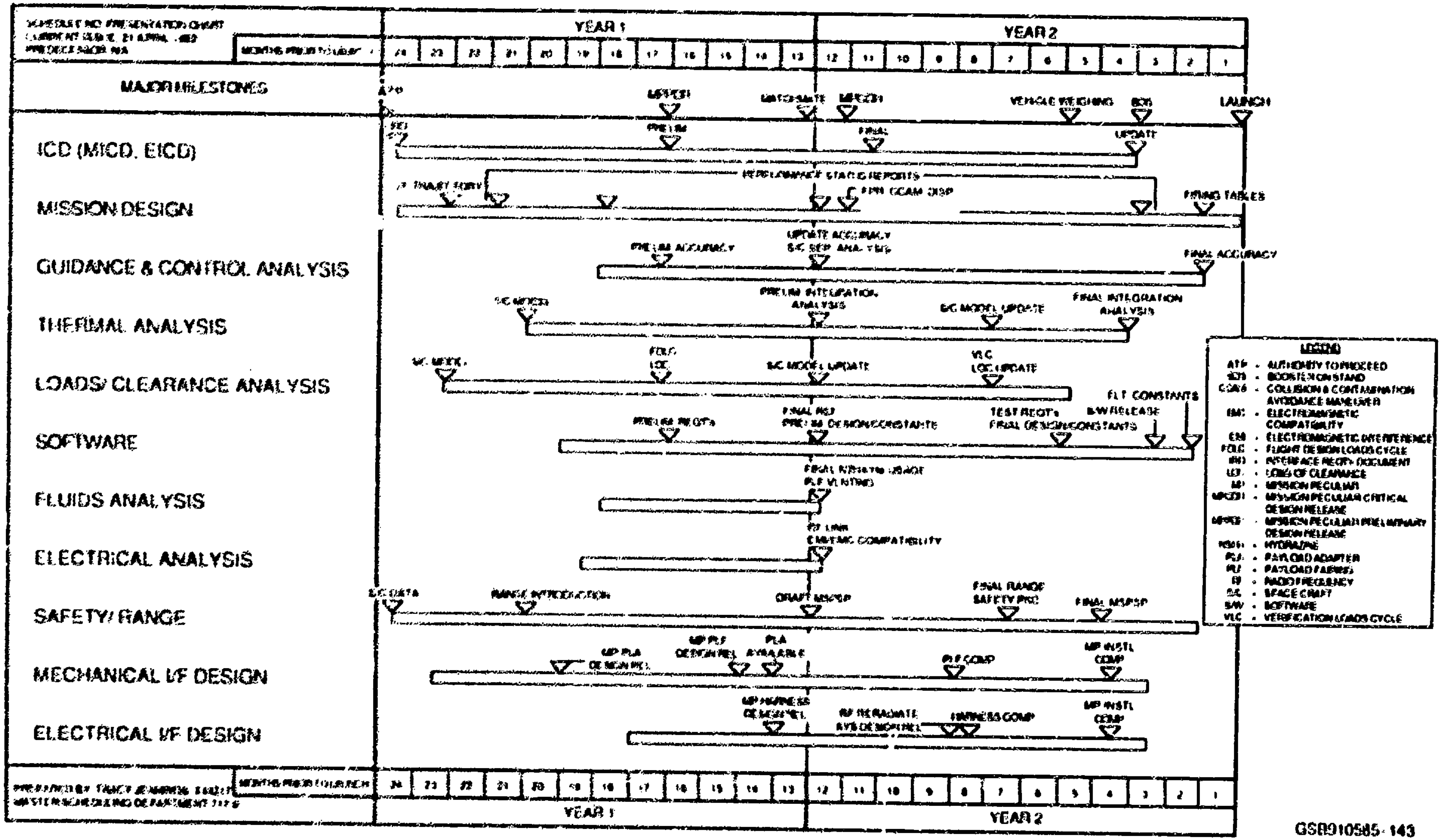

Figure 6-3 Typical mission analysis and design flow.

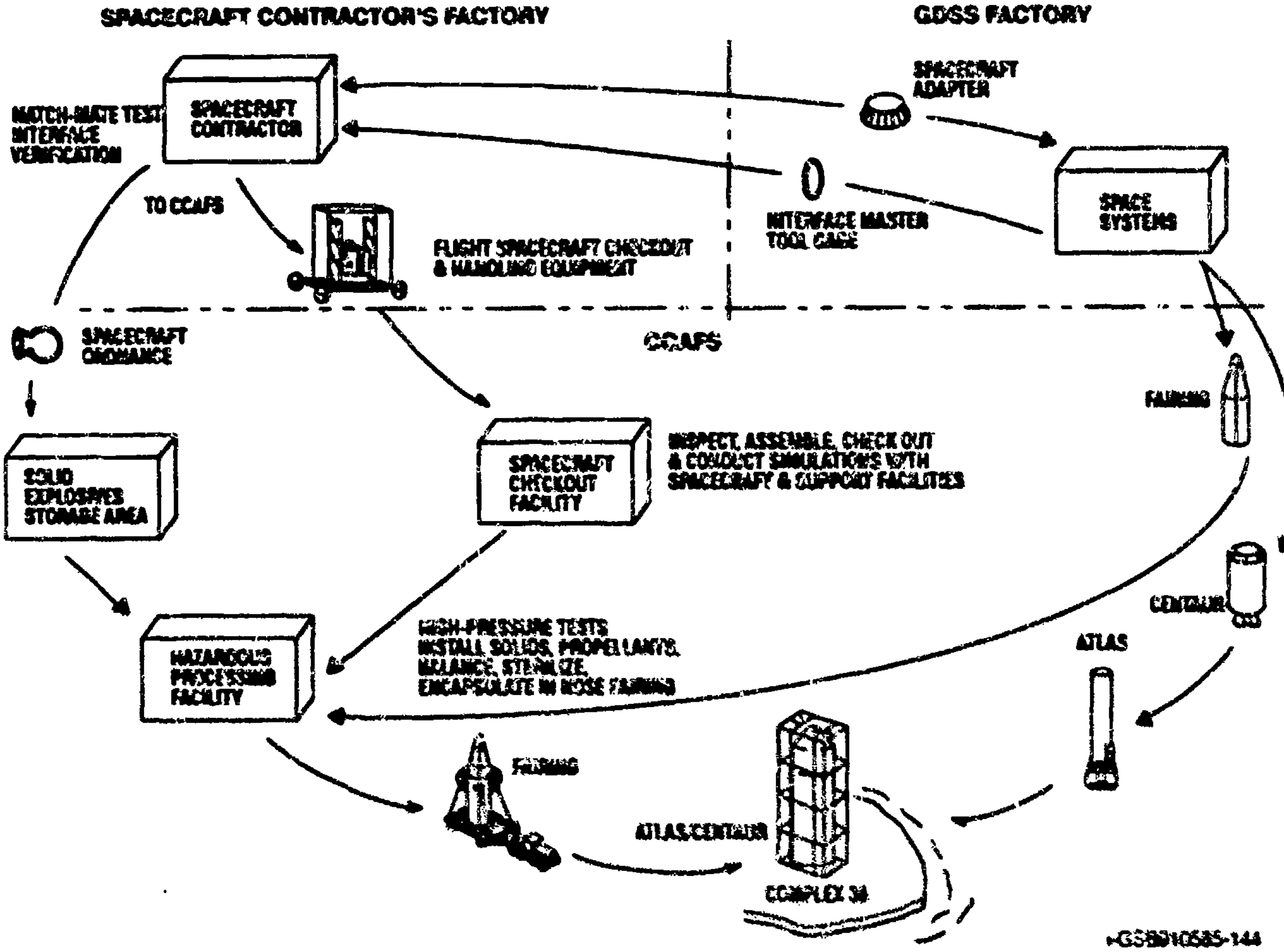

Figure 6-6. Typical factory-to-launch operations.

c. Special development tests at the launch site:

- Spacecraft data flow tests at launch pad (to verify spacecraft mission-peculiar command, control, and/or data return circuits, both hardline and/or RF)
- Electromagnetic compatibility (EMC) testing at the launch pad (to verify spacecraft, launch vehicle, and launch pad combined EMC compatibility)

In addition to integration/interface verification test capabilities, General Dynamics uses test facilities in San Diego to perform system development and qualification testing. Facilities include an integrated acoustic and thermal cycling test facility capable of performing tests on large space vehicles (see Figure 6-7). Other test facilities include:

- Vibration test laboratory
- Hydraulic test laboratory
- Pneumatic high pressure and gas flow laboratories
- Propellant tanking test stands
- Structural test stands for static and modal testing
- Combined environment vibration, acceleration and temperature centrifuge
- Thermal vacuum test chambers
- RF radiation laboratory.

6.2.1.3 Launch Services — In addition to its basic responsibilities for Atlas design, manufacture, checkout, and launch, GD offers the following operations integration and documentation services in support of prelaunch and launch operations.

ACOUSTIC CHAMBER

- DIMENSIONS: 33 FT X 40 FT (H)
  (10.1 X 12.2 X 15.2 m)
- VOLUME: 65,000 FT³ (1840.3 M³)
- FREQUENCY RANGE: 25 TO 10,000 Hz
- SOUND PRESSURE LEVEL: 154 dB
- DATA ACQUISITION: 250-CHANNEL DIGITAL ACOUSTIC SYSTEM
- 22,000 GAL (83.3 m³) CRYOGENIC TANKING ABILITY

CVG860722

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MAIN STRUCTURE

- DIMENSIONS: 130 FT X 80 FT X 60 FT (H)
  (39.6 X 24.4 X 18.3 m)
- SQUARE FOOTAGE: 10,400 FT² (966.2 m²)
- OVERHEAD CRANE: 10-TON (9,071.8 Kg) TRAVELING BRIDGE CRANE
- MAIN CONTROL ROOM: 41 FT X 46 FT (12.5 X 14 m)
- MAIN ENTRY: 30 FT X 50 FT (H) (9.1 X 15.2 m)

THERMAL CHAMBER

- DIMENSIONS: 30 FT X 31 FT X 50 FT (H)
  (9.1 X 9.4 X 15.2 m)
- VOLUME: 46,500 FT³ (1,316 m³)
- TEMPERATURE RANGE: -40°F TO +185°F (-40 to +85°C) WITH HUMIDITY CONTROL
- TEMPERATURE RATE CHANGE: 1°F/MINUTE (0.6°C/MIN)
- DATA ACQUISITION: 200-CHANNEL GRAPHIC DISPLAY PLOTS

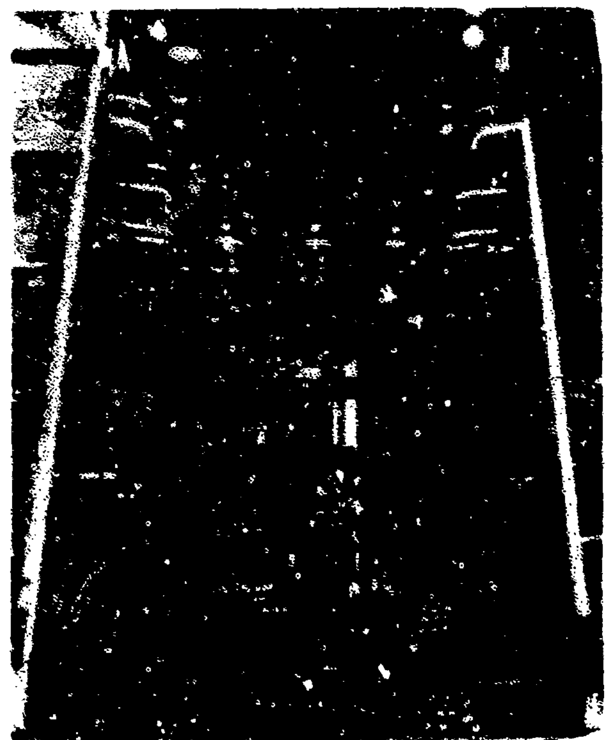

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Figure 6-7. Integrated acoustic and thermal test facility, General Dynamics Space Systems Division, San Diego, California.

a. Launch site operations support:

- Prelaunch preparation of the GD-supplied payload adapter, nose fairing, and other spacecraft support hardware
- Transport of the encapsulated spacecraft to the launch pad and mating of the encapsulated assembly to the launch vehicle
- Support of launch vehicle/spacecraft interface tests
- Support of spacecraft on-stand launch readiness tests (if requested)
- Prepare for and conduct the joint launch countdown

b. Provide basic facility services and assistance in installation of spacecraft ground support equipment at the launch site. This typically includes:

- Installation of spacecraft power, instrumentation, and control equipment in the launch services building and blockhouse
- Provision of electrical power, water, gases (helium and GN_2) long-run cable circuits, and on-stand communications
- Supply of on-stand payload air conditioning
- Provision of a spacecraft RF repeater system in the mobile service tower (permitting on-stand spacecraft RF testing)

c. Coordination, preparation, and maintenance of required range support documents:

- Air Force System Command required documents — Required whenever support by any element of the Air Force Satellite Control Facility (AFSCF) is requested
  - Orbital Requirements Document (ORD) — Details all requirements for support from the AFSCF remote tracking stations (RTS) and/or satellite test center (STC) during on-orbit flight operations
- Range ground safety and flight safety documentation as required by Range Safety Regulation ESMCR-127-1
  - Missile System Prelaunch Safety Package (MSPSP) — Provides detailed technical data on all launch vehicle and spacecraft hazardous items, which forms the basis for CCAFS approval of hazardous ground operations at the launch site
  - Flight Data Safety Package — Compiles detailed trajectory and vehicle performance data (nominal and dispersed trajectories, instantaneous impact data, 3-sigma maximum turn rate data, etc.), which forms the basis for CCAFS approval of mission-unique targeted trajectory

d. Flight status reporting during launch ascent — Real-time data processing of upper stage flight telemetry data to provide the following in near-real time:

- Mark event voice callouts of major flight events throughout launch ascent
- Orbital parameters of attained parking and transfer orbits (from upper stage guidance data)
- Confirmation of spacecraft separation, time of separation, and spacecraft attitude at separation

e. Transmission of spacecraft data via upper stage telemetry (as an option) — Interleaving a limited amount of spacecraft data into the upper stage telemetry format and downlinking it as part of the upper stage flight data stream

f. Postflight processing of launch vehicle flight data — Quick-look, preliminary, and final flight evaluation reports of selected flight data on a timeline and quantitative basis, as negotiated with the customer

6.2.1.3 Propellants, Gases, and Ordnance — Minor quantities of GN_2, liquid nitrogen, GHe, isopropyl alcohol, freon TF, and deionized water are provided prior to propellant loading. A hazardous materials disposal service is also provided. Spacecraft propellants are available at the CCAFS fuel storage depot. The United States National Aerospace Standards and U.S. Military Specifications that they meet are described in Table 6-1. If the spacecraft uses any of these propellants, they must be supplied by CCAFS.

- Sampling and Analysis — Analysis of fluid and gas samples is provided as specified in the Interface Control Document (ICD).
- Propellant Handling and Storage — Short-term storage and delivery to the HPF of spacecraft propellants.
- Ordnance Storage, Handling, and Test — Spacecraft ordnance and solid motors receiving inspection, bridge wire check, leak test, motor buildup, motor cold soak safe and arm check, X-ray, and delivery to HPF. Flight units may be stored for approximately three months and spares may be stored for up to six months. Other long-term storage is provided on a space-available basis and must be arranged for in advance. In addition, there is a safe facility for test and checkout (receiving, inspection, lot verification testing) of ordnance devices.

Table 6-1 Hypergolic propellants available at CCAFS fuel storage depot.

Propellant, Hydrazine, Standard Grade, MIL-P-26536
Propellant, Hydrazine, Monopropellant Grade, MIL-P-26536
Propellant, Hydrazine/Uns-dimethylhydrazine, MIL-P-27402
Monopropellant, High Purity Hydrazine, MIL-P-26536D
Propellant, Monomethylhydrazine, MIL-P-27404
Propellant, Uns-dimethylhydrazine, MIL-P-25604
Propellant, Nitrogen Tetroxide (NTO), NAS3620
Propellant, Nitrogen Tetroxide (MON-1), NAS3620
Propellant, Nitrogen Tetroxide (MON-3), NAS3620
Propellant, Mixed Oxides of Nitrogen (MON-10), MIL-P-27408
Propellant, Nitrogen Tetroxide (MON-3, Low Iron), NAS3620

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6.2.2 INTEGRATED TEST PLAN (ITP) — All testing performed during Atlas design, development, manufacture, launch site checkout, and launch operations is planned and controlled through the Atlas Integrated Test Plan (ITP). This encompasses all launch vehicle testing, including the spacecraft mission-peculiar equipment and launch vehicle/spacecraft integrated tests.

The ITP documents all phases of testing in an organized, structured format. It provides the visibility necessary to formulate an integrated test program that satisfies overall technical requirements and provides a management tool to control test program implementation.

The ITP consists of an introductory section (defining test concepts, philosophy, and management policies), a summary section (providing a system-by-system listing of all tests, requirements, and constraints for hardware development), and seven sections designated for seven different phases of testing (i.e., design evaluation, qualification, components, flight acceptance, launch site, etc. (see Figure 6-8)).

Subsections within these headings consist of the individual test plans for each Atlas component, system, and integrated system and provide the detailed test requirements and parameters necessary to achieve desired test objectives. Each subsection is issued as a unique stand-alone document, permitting its review, approval, and implementation to be accomplished independently from the parent document. Signatory approval is required from both GD and the customer for all launch vehicle and spacecraft integrated tests.

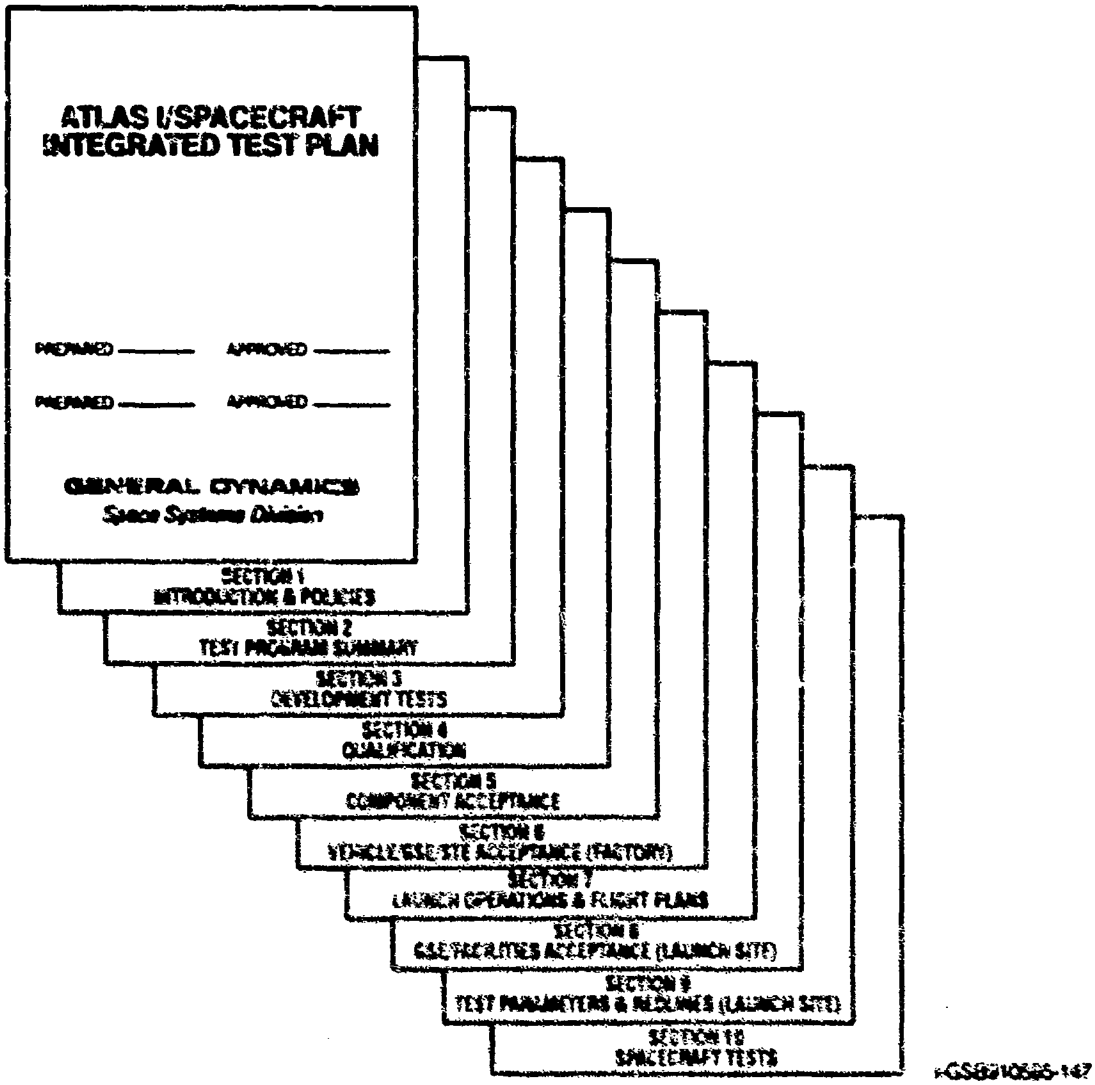

Figure 6-8 Integrated test plan organization

6.2.3 ATLAS AND CENTAUR TEST PROCEDURES — All test operations are performed in accordance with documented test procedures prepared by test operations personnel using the approved ITP subsections together with engineering drawings and specifications. The procedures for testing of Atlas flight hardware are formally reviewed, approved, and released prior to test. The procedures are verified as properly performed by inspection and made a part of each vehicle's permanent history file for use in determining acceptance for flight and final launch readiness.

Test procedures are also documented for spacecraft mission-peculiar hardware and for joint launch vehicle/spacecraft integrated tests and operations. Customers are urged to discuss their needs with GD early in the mission planning phase so that the various interface and hardware tests can be identified and planned. Customer personnel review and approve mission-peculiar test procedures and participate as required in LV/SC integrated tests.

6.2.4 LAUNCH VEHICLE TASKS — The following paragraphs provide an overview of the typical sequence of tests and activities performed during manufacture, prelaunch checkout, major launch readiness operations, and launch countdown of the Atlas launch vehicle. The purpose is to provide customers with an overview to the general flow and overall scope of activities typically performed.

6.2.4.1 Factory Tests — Flight vehicle acceptance (or factory) tests are performed after final assembly is complete. Functional testing is typically performed at the system level: low-pressure and leak checks of the propellant tanks and intermediate bulkhead, checkout of propellant-level sensing probes, verification of all electrical harnesses, and high-pressure pneumatic checks.

6.2.4.2 Launch Site Prelaunch Operations — Figure 6-9 shows a typical Atlas checkout and launch operations sequence. Upon arrival at the launch site, all launch vehicle items are inspected prior to erection on the launch pad.

Following erection of the Atlas and connection of ground umbilical lines, subsystem and system-level tests are performed to verify compatibility between airborne systems and associated ground support equipment in preparation for subsequent integrated system tests.

The payload fairing halves and payload adapter are prepared for spacecraft encapsulation in the HPF (see Figure 6-10). Two major tests are performed before the launch vehicle and launch pad are prepared to accept the spacecraft and start integrated operations.

- *Launch Vehicle Simulated Flight* — This first major launch vehicle test verifies that all integrated Atlas and Centaur ground and airborne electrical systems are compatible and capable of proper integrated system operation throughout a simulated launch countdown and plus-count flight sequence.
- *Wet Dress Rehearsal (WDR)* — The WDR is a tanking test to verify the readiness of all ground and airborne hardware, all support functions, the launch countdown procedure, and all Atlas and spacecraft system launch operations personnel assigned launch countdown responsibilities. Although pad operations and selected system responses are simulated, the WDR demonstrates that the integrated Atlas ground, airborne, and associated launch support functions, including range operations, are ready to support launch operations.

The customer supports the launch vehicle simulated flight and WDR operations. Although spacecraft and launch vehicle activities are simulated, participation enhances efficiency of personnel and procedures in subsequent integrated tests and the launch operation.

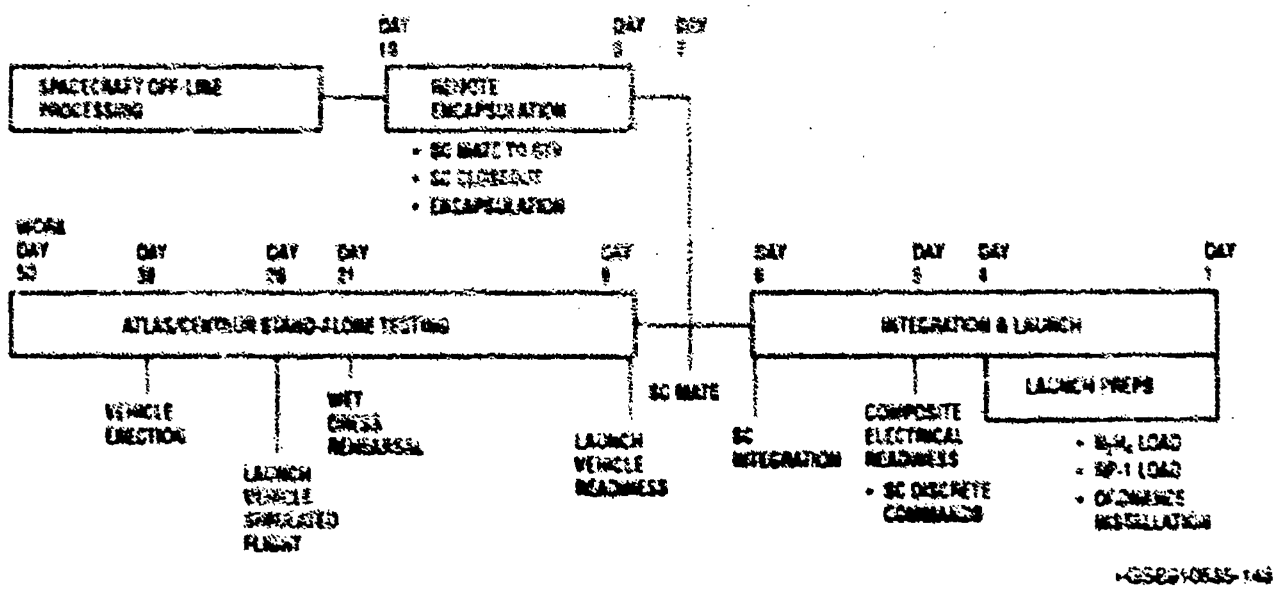

Figure 6-9 Typical launch operations sequence

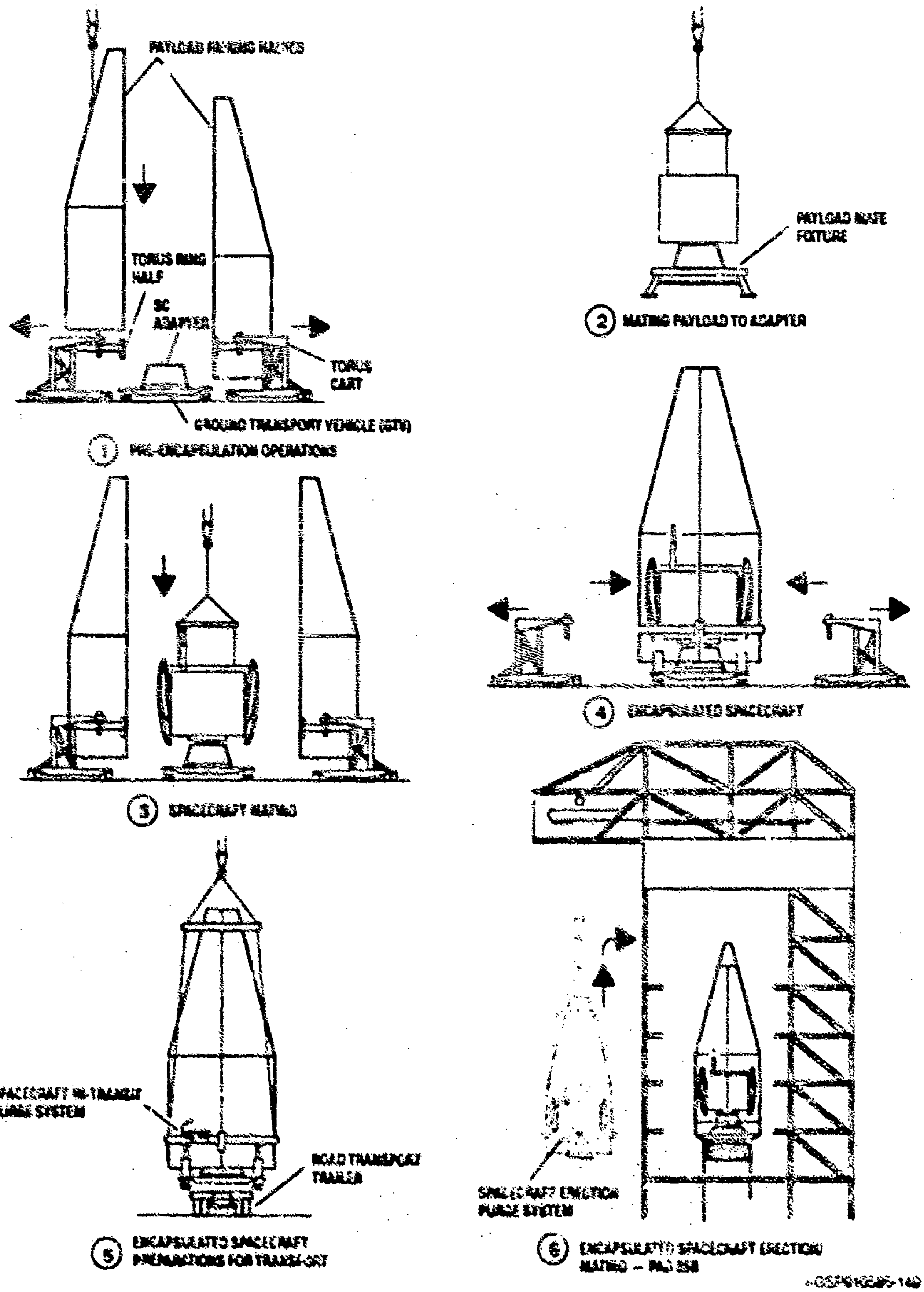

Figure 6-10. Payload fairing and spacecraft processing are identical to previous Atlas/Centaur operations.

6.2.4.3 Integrated Operations — Following successful WDR, the launch vehicle and launch pad are prepared to accept the spacecraft and commencement of integrated operations. Major prelaunch integrated test and operations include:

- Encapsulated spacecraft positioning in mobile service tower (MST) and mate to the launch vehicle.
- Performance of spacecraft flight readiness and system functional tests, including mission-peculiar command, control, and data return circuits, both hardline and RF.
- Composite Electrical Readiness Test (CERT), which includes space vehicle and launch vehicle operating through an integrated simulation of the final minutes of launch countdown and the plus-count flight sequence.
- Final launch readiness preparations, including the following major tasks:
  - Centaur N_2H_4 tanking
  - Atlas booster N_2H_4 tanking
  - Atlas RP-1 tanking
  - Pyrotechnic installation
- L-1 Day operations, consisting of integrated testing and final readiness tasks, including:
  - Spacecraft servicing as required
  - Launch vehicle closeout
  - Centaur C-band radar transponder and S-band telemetry tests
  - Flight termination system test
  - Atlas final ordnance tasks.

For all operations following spacecraft mate, the following conditions impose operational constraints:

- RF silence: RF silence is required for eight hours, off-shift, at approximately L-3 Day, for installation of Atlas ordnance and for three hours on L-1 Day ordnance electrical connection. RF silence can be scheduled during the launch countdown to support spacecraft ordnance connection.
- Hydrazine loading: Access to the MST is restricted during the off-shift hours between L-3 and L-4 Day during Atlas and Centaur hydrazine storage tank fueling operations.
- Lightning: Cloud-to-ground lightning within five miles of the launch complex after fuel or ordnance is installed requires pad clearing of all personnel.
- Hurricanes: The MST is designed and procedures are in place to secure the launch vehicle and MST for hurricane conditions.

A summary listing of the tasks versus their approximate launch-day schedule (launch days are calendar days before launch) is provided in Table 6-2.

6.2.4.4 Launch Countdown Operations — The Atlas launch countdown consists of an approximate nine- to ten-hour count, which includes two built-in holds — one at T-90 minutes (for 30 minutes) and the second at T-5 minutes (for 10 minutes) — to enhance the launch-on-time capability (see Figure 6-11).

GD's launch conductor performs the overall launch countdown for the total vehicle. The launch management is designed for customers and General Dynamics efficiencies and control elements (see Figure 6-12).

Spacecraft operations during the countdown should be controlled by a spacecraft test conductor

Table 6-2 Launch-day task schedule

	L-day
• Ground and airborne systems readiness tests	L-4
• Air-conditioning flow test (optional for S/C)	L-4
• Hydrazine loading of Centaur reaction control system	L-4/L-3
• Atlas RP-1 tanking	L-3
• Pyrotechnic installations	L-3
• Atlas and Centaur battery installations	L-2
• Ordnance final installations and hookups	L-1
• Launch vehicle closeout tasks	L-1

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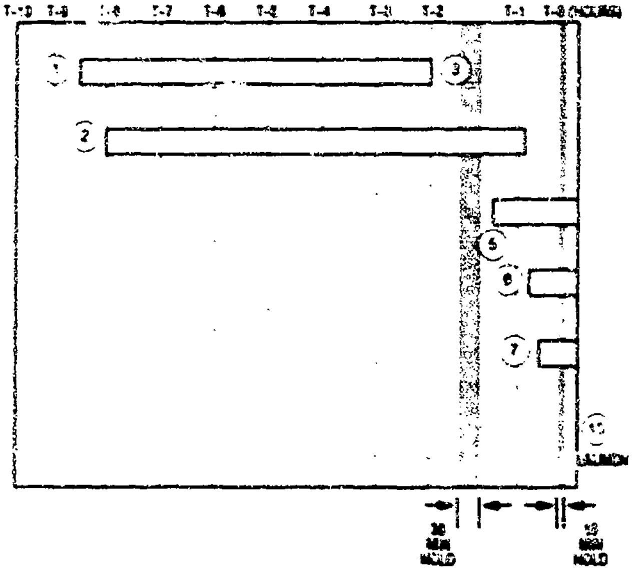

1.	START COMPLEX 36B MECHANICAL & ELECTRICAL PREPS	T-515 MINUTES
2.	START S/C INITIALIZATION PREPS	T-450 MINUTES
3.	START TOWER REMOVAL	T-120 MINUTES
4.	BUILT-IN HOLD FOR 30 MINUTES	T-90 MINUTES
5.	START CENTAUR LO_2 TANKING	T-75 MINUTES
6.	ARM SPACECRAFT S&A/START ATLAS LO_2 TANKING	T-55 MINUTES
7.	START CENTAUR LH_2 TANKING	T-43 MINUTES
8.	T-5 & HOLDING FOR 10 MINUTES	T-5 MINUTES
9.	ATLAS TO INTERNAL POWER	T-4:30 MINUTES
10.	CENTAUR TO INTERNAL POWER	T-2:00 MINUTES
11.	PRESSURIZATION TO INTERNAL	T-70 SECONDS
12.	START LAUNCH AUTOMATIC SEQUENCE	T-31 SECONDS
13.	LIFTOFF	T-0

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Figure 6-11 Launch countdown summary

located either in the Complex 36 Blockhouse or at some other spacecraft control center (e.g., in the spacecraft checkout facility) at the option of the spacecraft customer.

As launch pad integrator, GD prepares the overall countdown procedure for launch of the vehicle.

Typically, however, the spacecraft agency prepares its own launch countdown procedure for controlling spacecraft operations. The two procedures are then integrated in a manner that satisfies the operations and safety requirements of both and permits a synchronization of tasks through periodic status checks at predetermined times early in the count and a complete mesh of operations during the final steps leading to final committal to launch.

6.2.5 LAUNCH CAPABILITY — In addition to the scheduled 30-minute and 10-minute countdown holds, additional hold time can be scheduled for up to two additional hours under normal environmental conditions or until end of the scheduled launch window, whichever comes first.

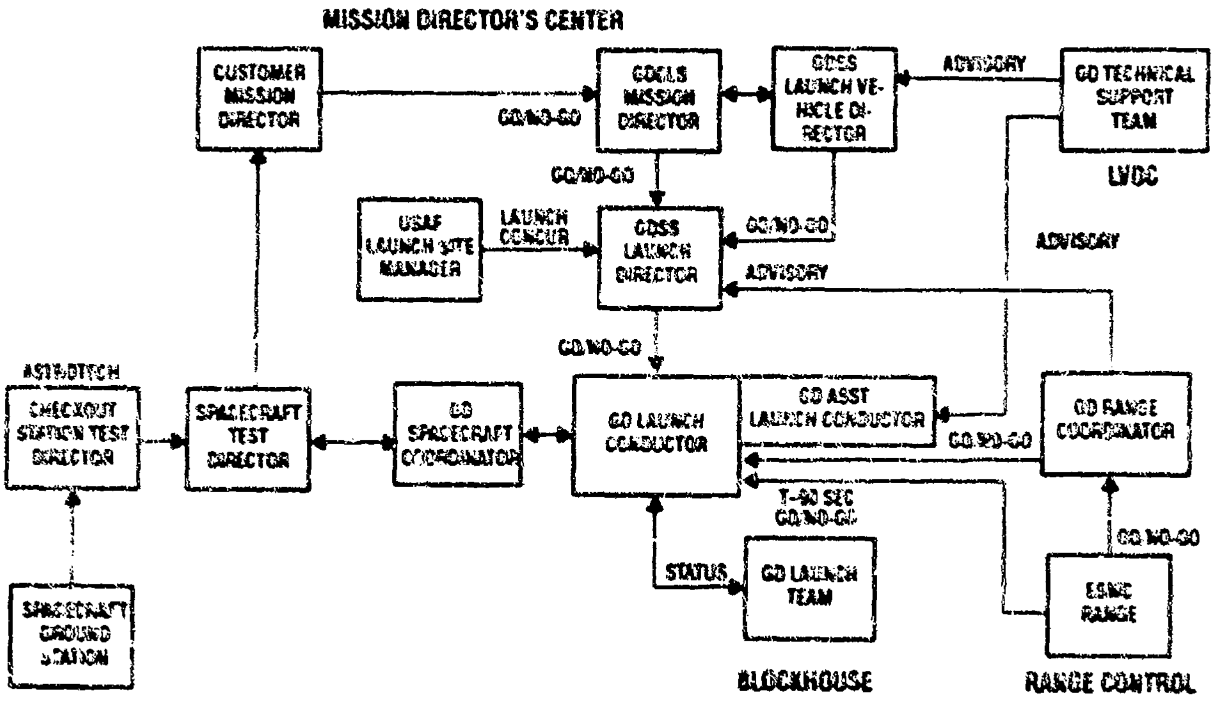

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Figure 6-12 Final launch commit is initiated by senior customer and launch service management

Launch window restrictions have typically been determined by the spacecraft mission requirements. The Atlas launch vehicle essentially does not have launch window constraints beyond those of the mission.

6.2.6 LAUNCH POSTPONEMENTS

6.2.6.1 Launch Abort/Launch Vehicle 24-Hour Recycle Capability — Prior to T−4 seconds (when the upper stage aft panel is ejected), the launch vehicle has a 24-hour turnaround capability following a launch abort due to a non-launch vehicle/GSE problem. If the abort occurs after securing the interstage adapter area, access to it will be required to service the N_2H_4 system.

6.2.6.2 Launch Abort/Launch Vehicle 48-Hour Recycle Requirement — A launch abort after T−4 seconds and prior to T−0.7 seconds requires a 48-hour recycle. The principal reason for a 48-hour recycle versus a 24-hour recycle is the added time requirement for replacing the upper stage aft panel (ejected at T−4 seconds) and the removal and replacement of the propellant pressurization lines pyro valves (fired at T−2 seconds).

7 • ATLAS ENHANCEMENT OPTIONS

General Dynamics has identified the vehicle modifications for the Atlas family to perform alternate types of missions. In addition, we are developing a performance enhancement package for the Atlas IIA and IIAS vehicles, which will be available in the second quarter of 1993.

This section addresses the following modifications:

- Block 1 performance enhancements
- Centaur extended mission kit
- Structural uprates for heavy payloads
- Lengthened 14-foot payload fairing

7.1 BLOCK 1 PERFORMANCE ENHANCEMENTS

General Dynamics is developing a performance enhancement package to increase the performance of the Atlas IIA and IIAS vehicles. These enhanced configurations will be available in the second quarter of 1993. The enhancement package uses vehicle modifications that are straightforward and do not affect system reliability or operability. The enhanced Atlas IIA and IIAS configurations feature an uprated Centaur engine, the RL10A-4-1. This engine offers a thrust increase of 1,500 lb (22,300 lb thrust) and an I_{sp} increase of 2 seconds (451 sec I_{sp}). The Centaur forward and aft bulkheads will incorporate lighter weight and more efficient insulation materials. The air-lit solid rocket motors on the Atlas IIAS are being modified to decrease the nozzle cant angle and increase the component of axial thrust. In addition, several flight software and mission design enhancements are being developed to further optimize the ascent phase. The performance yields for the enhanced Atlas IIA and IIAS are included in Section 2.

If an initial performance assessment indicates that a particular mission's requirements are not satisfied, please contact us to discuss additional performance enhancement options.

7.2 CENTAUR EXTENDED MISSION KIT

The majority of previous Atlas/Centaur missions have been to GTO. This type of mission typically requires two Centaur burns and a relatively short, 15-minute parking-orbit coast. For other types of missions, such as low Earth orbit (LEO), high Earth orbit (HEO), and planetary missions, the parking orbit coast period can be much longer. General Dynamics has investigated the issues associated with Centaur performing an extended parking orbit coast. These include propellant management, thermal control of components, and space vehicle power requirements. We are developing a mission-peculiar kit that will allow Centaur to perform parking orbit coasts of up to 90 minutes. The Centaur is modified to incorporate an additional helium bottle, a balanced vent system for the liquid oxygen tank, and radiation shielding on the LO_2 tank sidewall (Figure 7-1). The Centaur's avionic and electrical components will be covered with special thermal paints and tapes, and additional radiation shielding to maintain their operating temperatures (Figure 7-2). This kit will be available in the second quarter of 1993 for use on the Atlas IIA and IIAS vehicles.

7.3 HEAVY PAYLOAD UPRATES

For LEO missions, the vehicle performance capability exceeds the structural capability of Centaur's equipment module and payload adapters. General Dynamics has investigated the vehicle modifications required to launch payloads in the 10,000 to 18,000 lb (4550 to 8200 kg) range. Based on the configuration of the spacecraft, either a 62-inch (1575 mm) or a 105-inch (2667 mm) diameter mechanical interface can be provided with minor Centaur modifications.

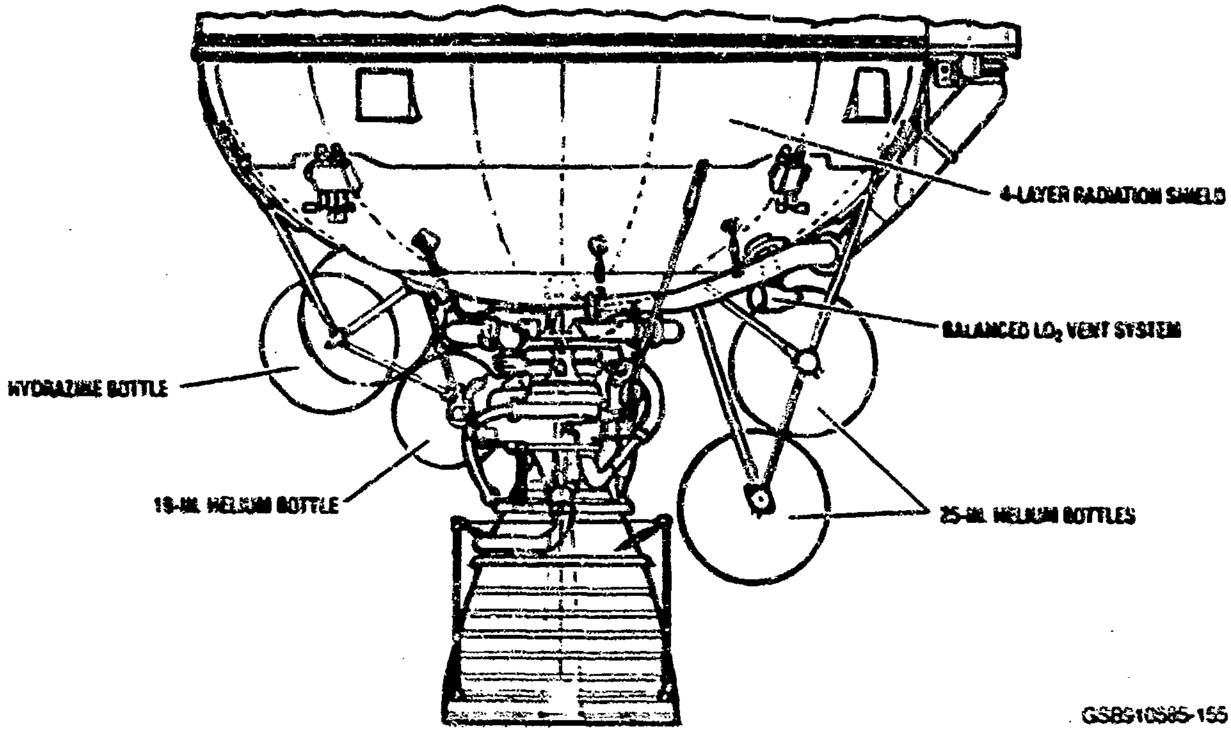

Figure 7-1. Centaur aft bulkhead for extended coasts.

The 62-inch (1575 mm) diameter interface would be provided by strengthening the current equipment module. The changes are straightforward and include increasing the cross sectional area of the forward ring and increasing the gauge of the stringers.

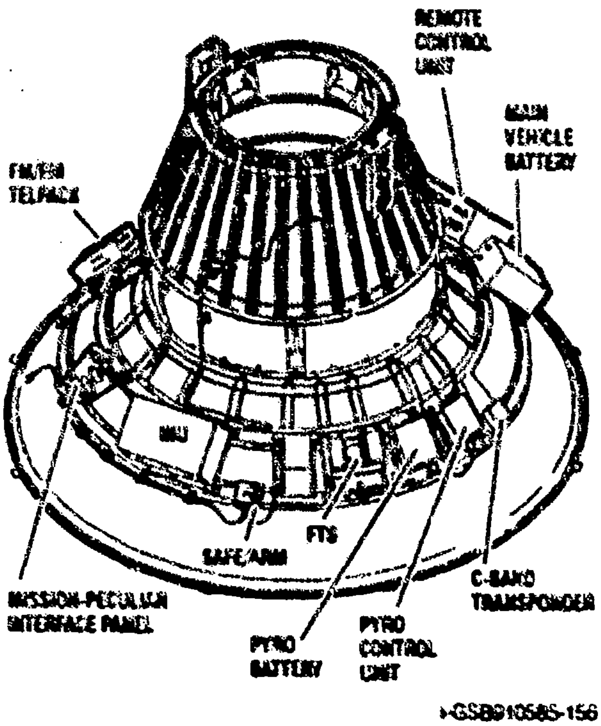

Figure 7-2. Typical Centaur equipment module for extended coasts.

This option offers the same bolted interface and stayout zones as the existing equipment module. The structural capability of the strengthened equipment module is shown in Figure 7-3. It should be noted that if an application exceeds this curve, additional strengthening of the equipment module is possible. A mission-unique payload adapter and separation system mates with the top of the equipment module and can be provided y either General Dynamics or the spacecraft manufacturer.

For extremely large or heavy spacecraft, the 105-inch (2667 mm) diameter interface offers greater stiffness and structural capability than the 62-inch (1575 mm) interface. This concept features a truss adapter, which is similar in design to those flown on NASA's Viking program (Figure 7-4).

The design incorporates an aft torque box and multiple hard points on the Centaur equipment module and an intercostal in the stub adapter (Figure 7-5). A series of struts attach to the hard points and support a forward torque box. The forward torque

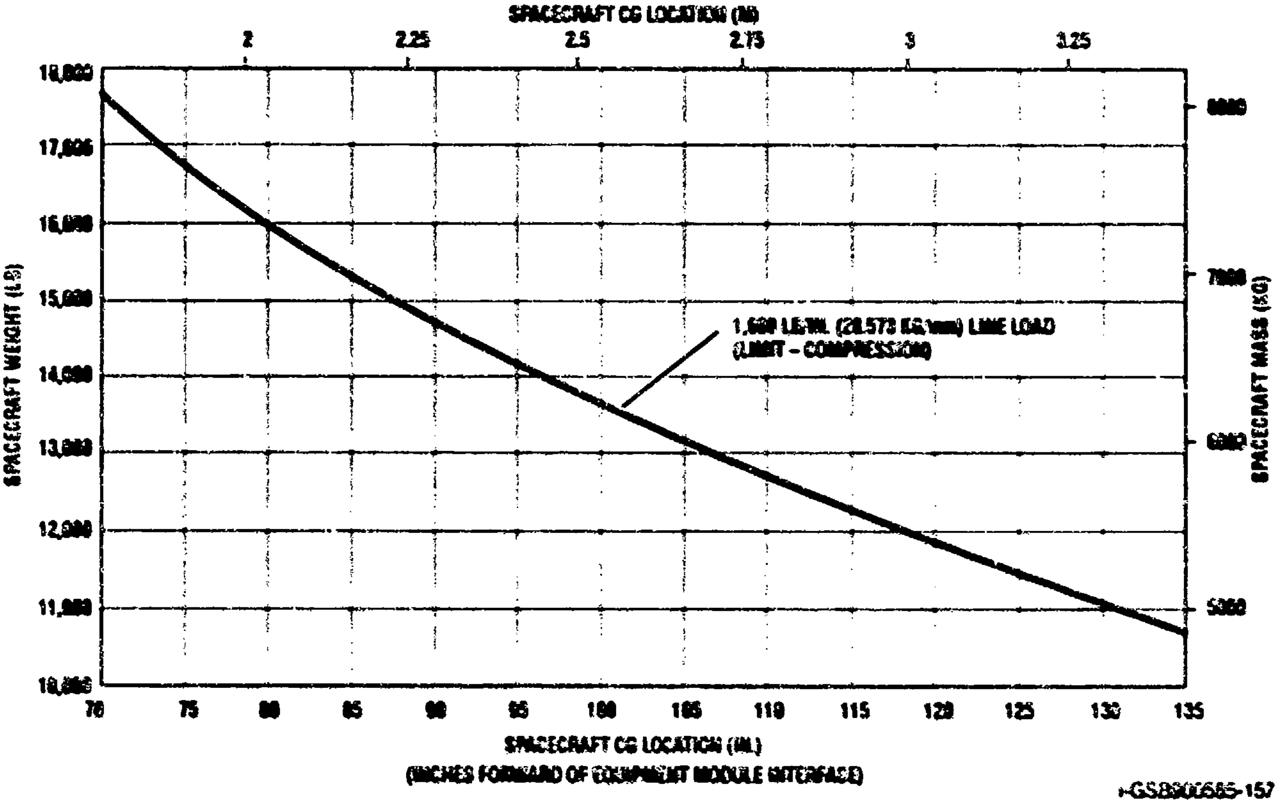

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Figure 7-3. Strengthened equipment module structural capability.

box provides a bolted interface. The structural capability of the truss adapter is shown in Figure 7-6. It should again be noted that, if an application exceeds this curve, the design can be strengthened to match a particular application. The mission-unique payload adapter and separation system that mates with the top of the truss can be provided by either General Dynamics or the spacecraft manufacturer.

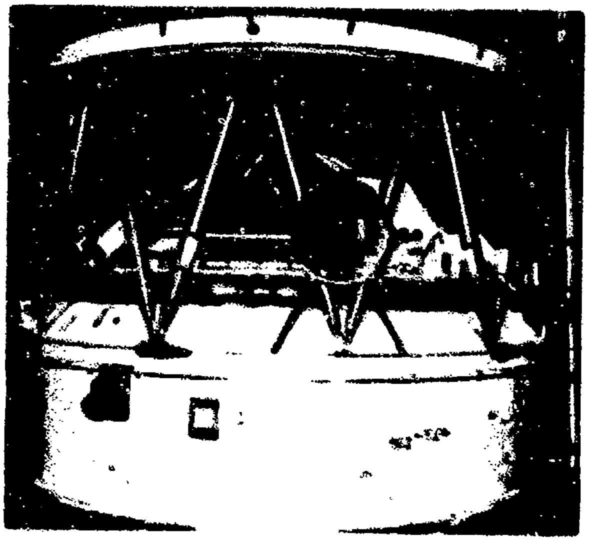

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Figure 7-4. Viking truss adapter for heavy spacecraft.

7.4 LENGTHENED 14-FOOT PAYLOAD FAIRING

For spacecraft that exceed the height of the standard 14-foot (4.2 m) payload fairing, we can develop a lengthened fairing. The fairing can be readily lengthened up to 3 feet (914.4 mm) by incorporating a spacer at the forward end of the cylindrical section (Figure 7-7). This concept has been used previously on other Atlas payload fairings. The launch availability of the lengthened vehicle is maintained through minor modifications to the Centaur. The payload envelope for a lengthened fairing is shown in Figure 7-8.

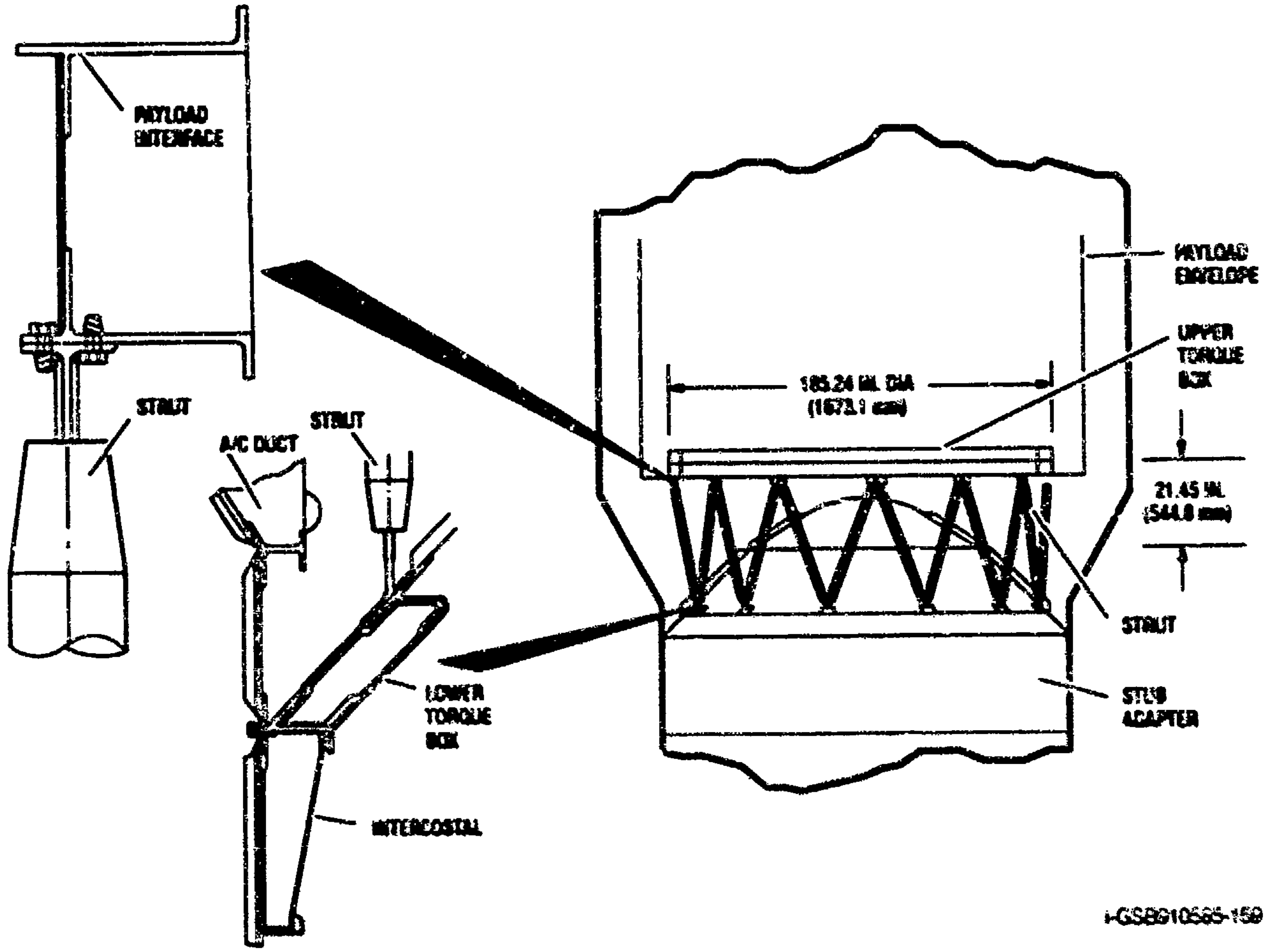

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Figure 7-5. Truss adapter for heavy payloads.

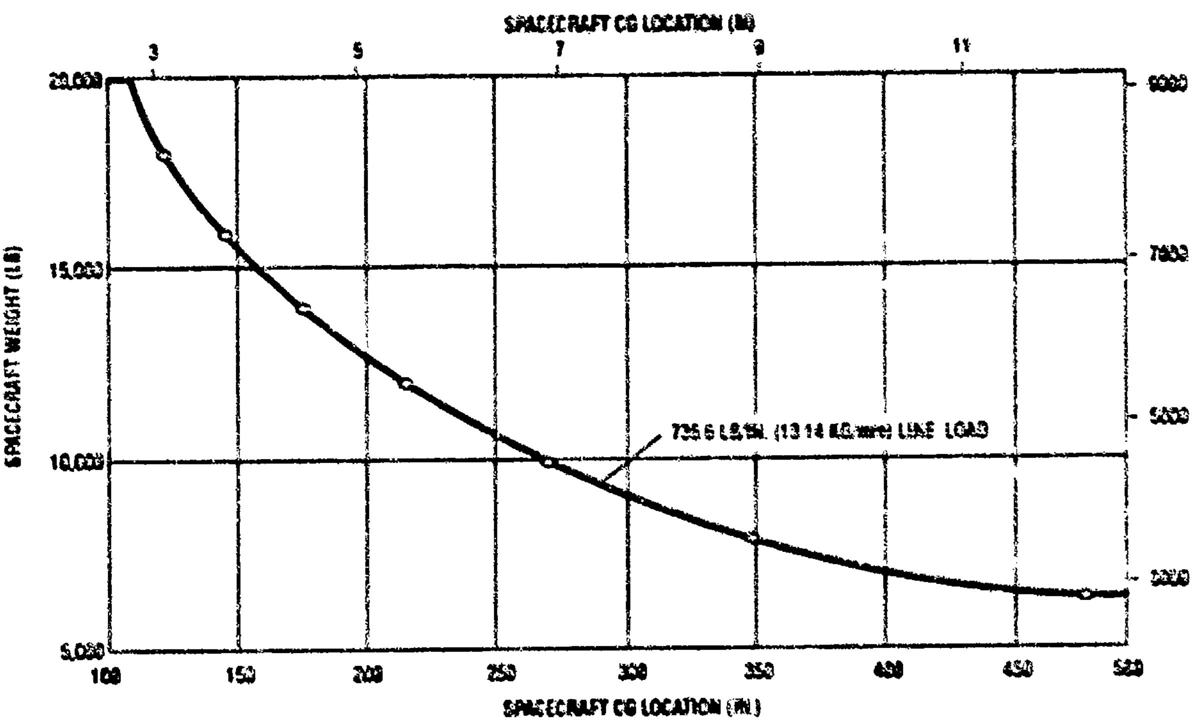

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Figure 7-6. Truss adapter structural capability

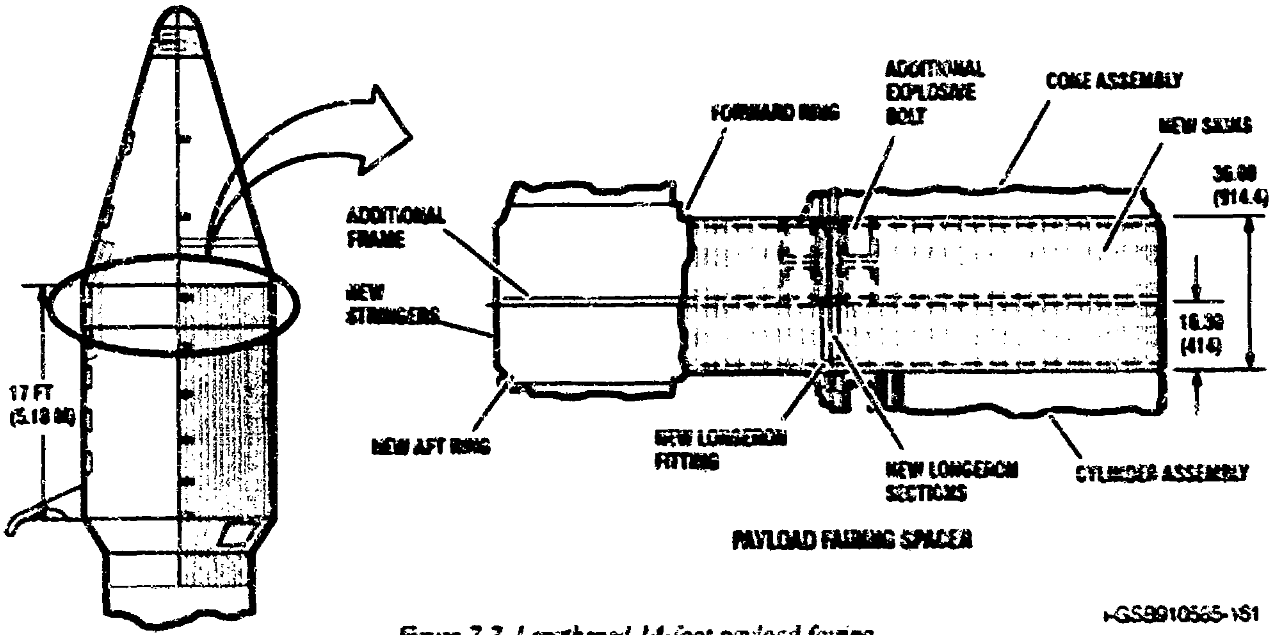

Figure 7-7. Lengthened 14-foot payload fairing.

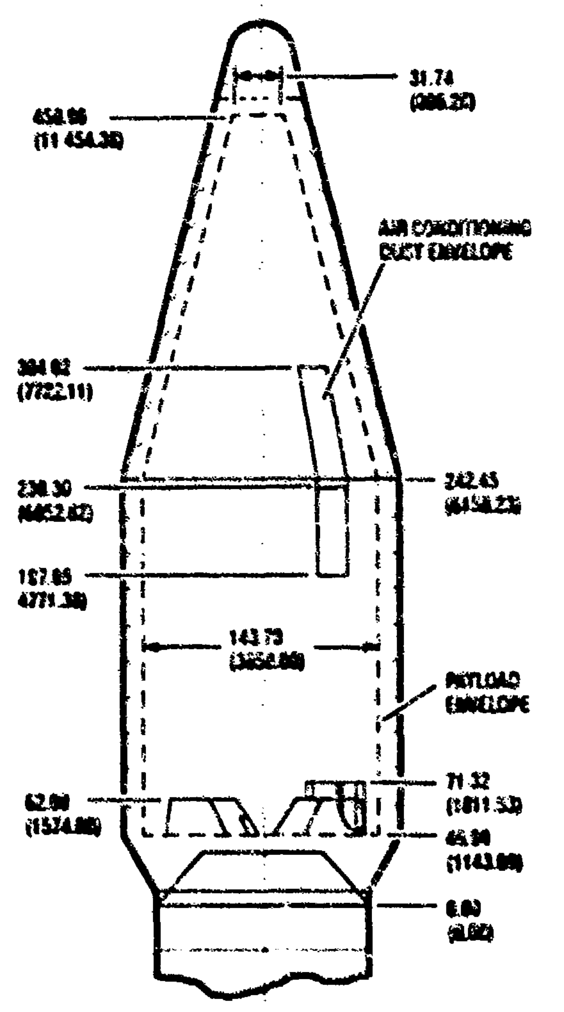

Figure 7-8. Payload envelope for lengthened payload fairing.

8 • WEST COAST ATLAS LAUNCH CAPABILITY

Missions to Sun-synchronous, polar, and stable 63.4-degree inclined orbits are more readily launched from a West Coast launch facility than from Cape Canaveral. Planar and near-planar flight azimuths can be employed and range safety restrictions are minimized from a West Coast site. General Dynamics has supported preliminary United States Air Force (USAF) planning to place Atlas/Centaur capability at Vandenberg Air Force Base (VAFB) in California. Plans have been defined for modification of Space Launch Complex 3E (SLC-3E) at VAFB for the Atlas II family. This section describes the Atlas West Coast launch capability.

It should be noted that capability to launch the Atlas II, IIA, and IIAS from VAFB does not currently exist, but the USAF has a plan to develop this capability. When this capability becomes a reality, projected for 1996, the facility will be owned by the USAF and will be intended to be used for U.S. Government missions (USAF, NASA). Launches for customers other than the U.S. Government will require USAF concurrence.

SLC-3E modifications, as summarized in the following pages, have been discussed in detail with the USAF. Over the last two decades, NASA and the Air Force have shared the use of SLC-3 facilities for the TIROS-N, SEASAT, GEOSAT, and NOAA programs. Continued NASA-USAF cooperation is expected for future West Coast launches with the Atlas II family of launch vehicles.

8.1 FACILITY LOCATIONS

Atlas launch facilities at VAFB are located at Space Launch Complex 3 (SLC-3) as illustrated in Figure 8-1. SLC-3E is located approximately seven miles from the Base Industrial Area on North VAFB and approximately four miles from NASA Building 836. For reference, the locations of SLC-4, SLC-6, and Buildings 8510 and 7525 are shown. These areas are potential operational support facilities for the Atlas

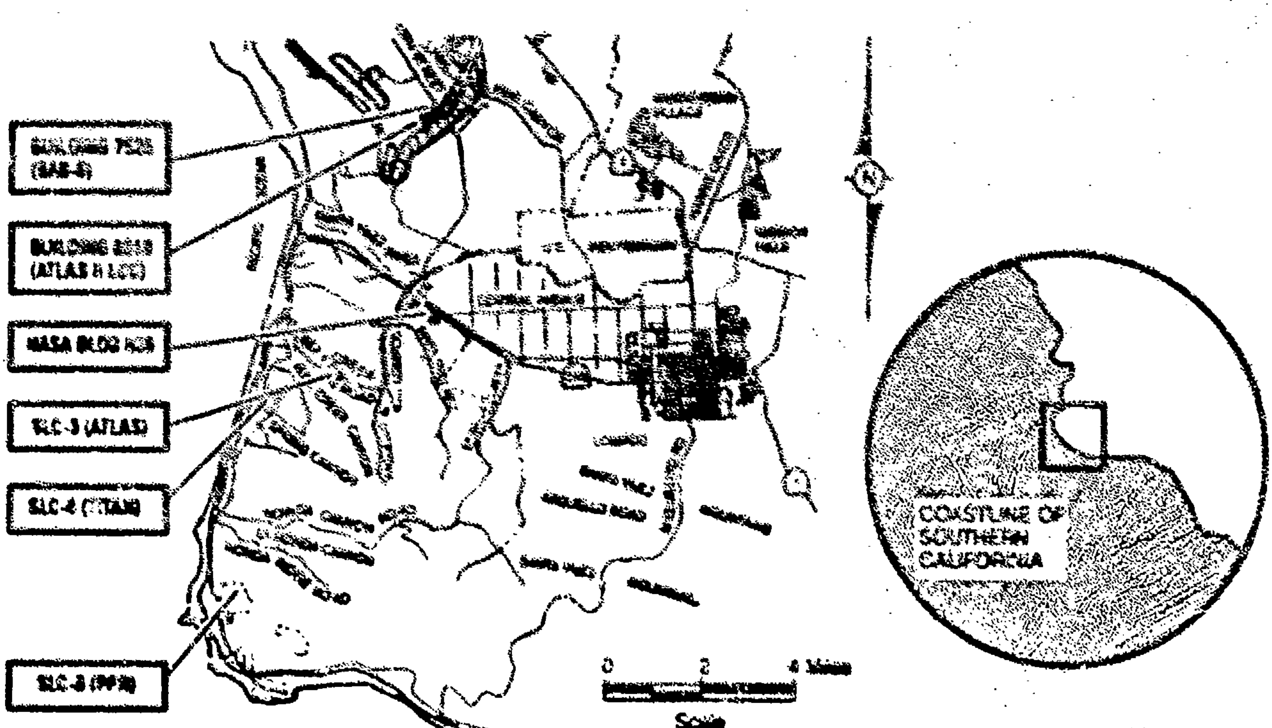

Figure 8-1. Location of launch facilities at VAFB

II family of launch vehicles and/or for payload operations.

8.2 ATLAS HISTORY AT SLC-3

Originally designated as Point Arguello Launch Complex 1 (PALC-1), the SLC-3 launch facility was developed by the U.S. Navy to launch the Atlas D booster with an Agena upper stage. Using the dual-pad capability, a total of 18 Atlas/Agena vehicles were launched from PALC-1 between 1960 and 1963. Upon completion of this Atlas program, Pad 1 was converted to launch Thor/Agena vehicles and Pad 2 was placed in a minimum caretaker status.

In 1965, Pad 2 was configured to launch the Atlas Standard Launch Vehicle (SLV-3), a space launch vehicle design based on the Atlas D ICBM system. In 1966, the U.S. Navy Point Arguello facility was assigned to the U.S. Air Force along with responsibility for all space and missile launch activities. Subsequently, PALC-1 was redesignated by the USAF as Space Launch Complex 3, and Pads 1 and 2 were redesignated SLC-3W (West) and SLC-3E (East), respectively.

Between 1963 and 1972, 38 Thor/Agena vehicles were launched from SLC-3W. In 1972, modification of SLC-3W was initiated to accommodate Atlas E/F ICBM vehicles refurbished for space launch operations. SLC-3W remains in this configuration with additional Atlas E launches planned through 1993.

In 1966, three additional Atlas/Agena vehicles were launched from SLC-3E, and between 1966 and 1968, four SLV-3s were launched. Following the final SLV-3 launch, SLC-3E was returned to minimum caretaker status until 1975, at which time it was modified to launch Atlas E/F vehicles in support of the Global Positioning System (GPS) program. SLC-3E was subsequently returned to the Atlas D configuration in 1982 to support the USAF Atlas H program. Between 1983 and 1987, five Atlas H vehicles were successfully launched from SLC-3E, followed by return of the launch pad to minimum caretaker status (current status).

The history of SLC-3 is summarized in Figure 8-2.

8.3 ATLAS II SITE DEVELOPMENT

8.3.1 CONCEPT STUDIES — In 1989, General Dynamics recognized that the planned DoD space launch capability at VAFB had a significant performance gap that, if filled by the Atlas II family, would significantly benefit payload mission planning. With an Atlas IIAS launch capability at VAFB,

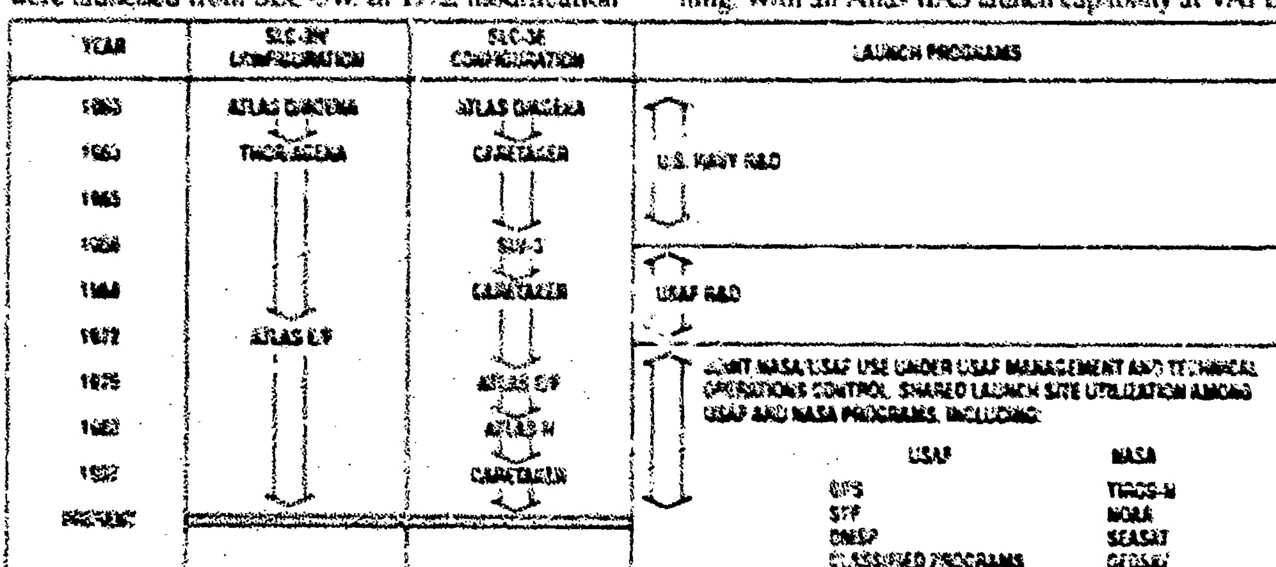

Figure 8-2. SLC-3 launch site history

a payload of 7300 kg (16,100 lb) could be delivered into a low polar orbit.

This launch vehicle performance assessment resulted in initial General Dynamics studies to define the SLC-3E modification requirements that would be necessary to establish the Atlas II West Coast capability. Top-level requirements, technical approaches, cost estimates, and site activation schedules were developed and provided to the USAF. Subsequent USAF interest in the Atlas II capability resulted in General Dynamics development of a launch feasibility study.

8.3.2 DEVELOPMENT STATUS — Throughout 1990, General Dynamics developed system-level and detailed requirements for SLC-3E activation. Coincident with this General Dynamics activity, USAF initiated the required environmental impact/environmental approval (EA) process with Santa Barbara County in October 1990, and final EA approval was received in July 1991.

Activity during 1990 included range safety studies for the Atlas IIAS. Flight safety ascent event sequencing was developed to ensure that solid rocket booster jettison events would result in SRB casing impacts clear of all land masses, including the Santa Barbara Channel Islands. In addition, explosive siting studies verified that Atlas IIAS quantity-distance (QD) safety criteria were in compliance with the existing SLC-3E QD explosive siting envelope. Recent activities have included developing requirements documents for the facility and ground support equipment, plans for use of the existing facility, and activation schedules.

8.4 REQUIRED FACILITY MODIFICATIONS

The upgrade of SLC-3E from the existing Atlas II configuration to an Atlas IIAS launch pad involves design and installation of new hardware, modification of some existing hardware, and use of some existing hardware.

The activation plan for SLC-3E includes relocation of the Atlas Launch Control Center (LCC) from the existing Blockhouse to Building 8510. Building 7525 has adequate space for Centaur receiving inspection to be performed in this facility. The SLC-4 gaseous nitrogen (GN_2) pipeline is planned to be extended to SLC-3, thus providing a high-volume, high-pressure supply to the SLC-3 site.

8.4.1 OVERALL DESCRIPTION — Figure 8-3 provides an overall description of the launch site upgrades required to achieve SLC-3 launch capability for the Atlas II family of vehicles. New and modified equipment is identified in these figures by shading. Some of the more significant upgrades include:

- New mobile service tower (MST)
- New umbilical tower
- Modified exhaust duct with contoured deflection angle similar to SLC-36
- New Centaur LH_2 storage area
- Modified LO_2 storage area
- New GN_2 pipeline from SLC-4 with receiving/distribution station
- New MST bridge, embankment, tracks, and tiedowns
- New flare stack for LH_2 storage tank boiloff disposal

8.5 SITE ACTIVATION SCHEDULE

General Dynamics studies and current schedules are intended to support a 1996 launch capability.

8.6 MISSION DESIGN

The Range Safety requirements of a West Coast launch site permit high-inclination orbits not easily achieved from Florida. Highly inclined orbits, including Sun-synchronous, polar, and stable

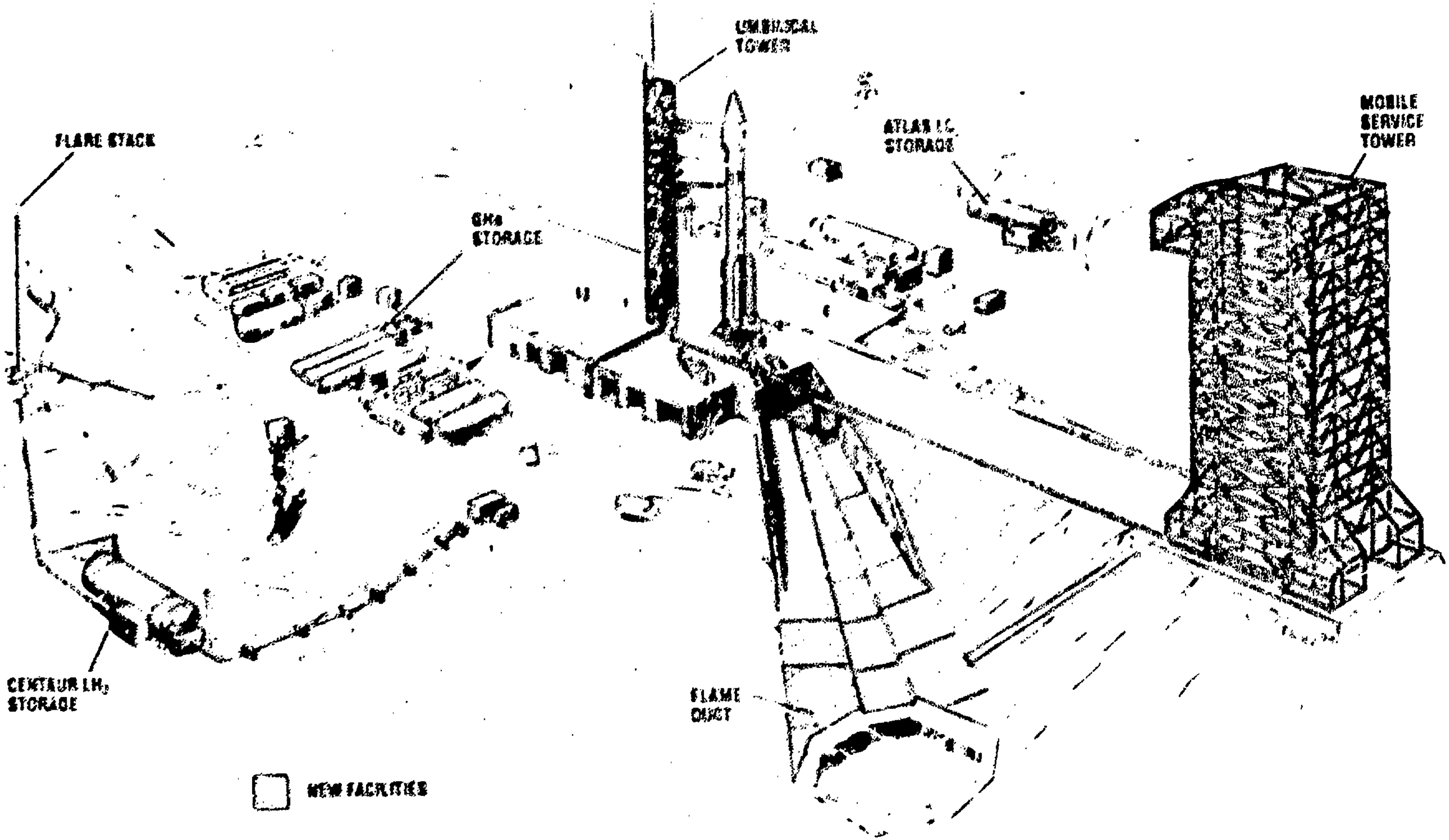

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Figure 8-1 SLC-36 modifications for the Atlas II family

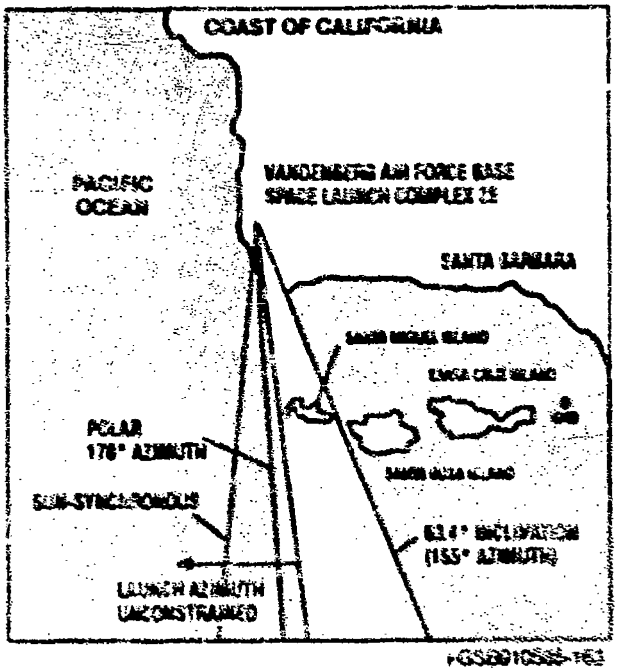

Figure 8-4 Available launch azimuths

63.4-degree inclination orbits, are readily achieved from VAFB with near-due south launch azimuths.

Trajectory Design — Atlas launches from VAFB will use ascent profile sequencing and trajectory design (direct ascent, parking orbit ascent) similar to their East Coast counterparts. Ascent mission sequencing parameters will be altered to reflect safety, land overflight, and hardware jettison constraints imposed by the U.S. Air Force 30th Space Wing. Figure 8-4 illustrates several candidate launch azimuths associated with desired inclination final orbits. Figure 8-5 illustrates ground traces for a reference direct ascent and parking orbit ascent mission to a low Earth orbit. As a note, parking orbit ascent missions to LEO utilize short-burn constraints and the long-coast kit as described in Sections 2.3.4 and 2.3.5.

8.7 ATLAS PERFORMANCE CAPABILITY

This section describes Atlas performance capability from VAFB for both circular and elliptical orbits. The performance curves represent the maximum capability to the specified orbits with a three-sigma (99.87%) probability. Once specific spacecraft mass properties data, propulsion data, mission objectives, and mission constraints are identified, additional performance information can be developed to specifically address the unique mission requirements.

Payload Systems Weight — As noted in Section 2, performance capabilities quoted throughout this document are presented in terms of payload systems weight. *Payload systems weight (PSW) is defined as the total mass delivered to the target orbit, including the separated spacecraft, any spacecraft-to-launch vehicle adapter and all other hardware required on the launch vehicle to support the payload (payload flight termination system, harnessing, etc.)* **The weight of the heavy**

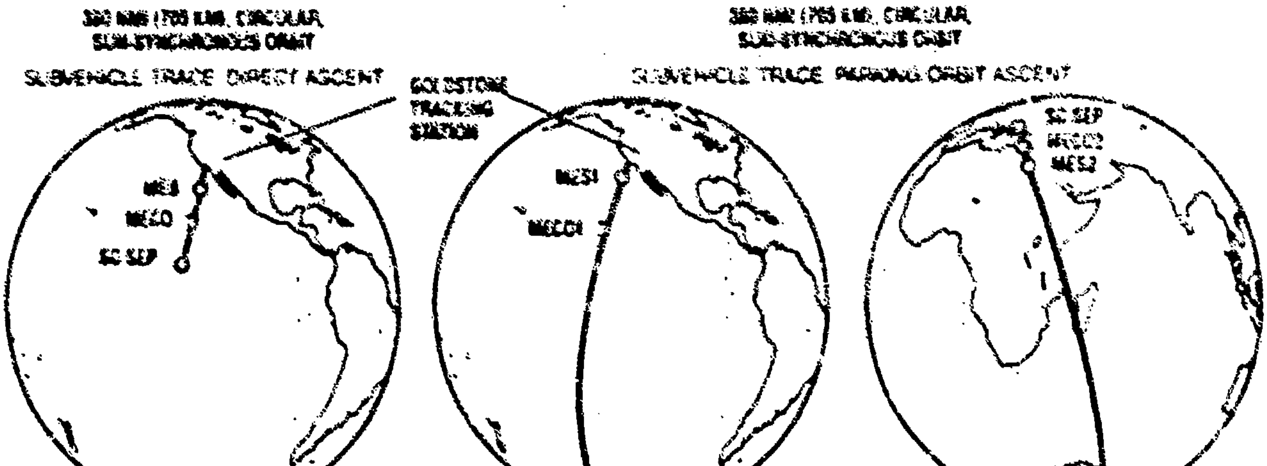

Figure 8-5 Ground traces for LEO missions on Atlas

payload adapter system, including changes to the Centaur forward equipment module, stub adapter, and the truss adapter, are considered part of the payload required hardware and must be subtracted from payload systems weight to determine spacecraft mass capability. Payload hardware requirements vary from mission to mission and depend upon payload interface requirements and payload design constraints. To further develop the Atlas heavy-payload mission options, a reference payload hardware list has been developed and is shown in Table 8-1. This table also includes an example of PSW and separated spacecraft weight capabilities.

This list is provided to allow Atlas mission design efforts to begin with a uniform set of requirements. As each mission developed will require its own set of launch vehicle interface and support systems, this list will vary to meet individual payload requirements. Atlas performance capabilities are based on the ground rules listed in the next section.

Table 8-1. Performance effects of payload support items for Atlas launches of LEO spacecraft (example only).

Payload Support Item	kg	lb
Centaur LVMP*		
FM/FM telepak	12	27
Telemetry interface unit	14	31
Harnessing and connectors	1	3
Pyro control relay	3	6
Payload destruct and antisafe device	5	10
Payload adapter	59	130
Strengthened equipment module	45	100
	139	307
Payload fairing LVMP		
Two payload fairing doors	18	40
Reradiating antenna	6	12
Acoustic blankets	101	223
	125	275
Performance		
Example payload systems weight	6350	14,000
Centaur required payload hardware	139	307
Performance effect of PF LVMP	17	38
Separated spacecraft mass	6194	13,655

* LVMP = Launch vehicle mission-peculiar

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Performance Ground Rules — The Atlas performance capabilities presented in this section are based on the trajectory design ground rules listed in Table 8-2 and shown on each plot.

Performance is shown in Figures 8-6 through 8-8. Figures 8-6a and 8-6b show Atlas II performance capabilities. Figures 8-7a and 8-7b show Atlas IIA launch vehicle performance to the specified orbits. Figures 8-8a and 8-8b show Atlas IIAS performance capabilities. Within each figure, the first and second curves show direct ascent performance to elliptical orbits, and the third and fourth show both direct and parking orbit ascent performance to circular orbits.

Many of the trajectory design constraints, including jettison criteria, booster staging level, ground telemetry constraints, and parking and transfer orbit perigee altitudes can be modified to satisfy specific mission requirements. The performance effects of adjusting these parameters are mission-dependent.

8.7.1 ONE-BURN ELLIPTICAL ORBIT CAPABILITY — As discussed previously, Atlas can place payloads into elliptical orbits with low perigee altitudes and desired apogee altitudes via the direct ascent trajectory design. Figures 8-6 through 8-8 show direct ascent elliptical orbit performance for Atlas II, Atlas IIA, and Atlas IIAS, respectively. With this trajectory design, Centaur places the payload into transfer orbit with a perigee altitude of 100 nmi (185 km) and an argument of perigee of approximately 178 degrees. Performance for various inclinations is shown on each plot. The elliptical orbit performance curves labeled Sun-synchronous assume that the spacecraft circularizes the orbit at apogee into the final Sun-synchronous orbit. The parking orbit ascent design becomes the better choice with orbit parameter targets difficult to reach with a direct ascent design.

Table 8-2. Atlas trajectory design ground rules.

Payload Fairing Size – All performance curves assume the standard large payload fairing

Heavy Payload Adapter – *User must reduce performance (PSW) to include effects of heavy payload interface*

Flight Performance Reserve (FPR) – Centaur propellant has been allocated for a 99.87% assurance (3-sigma) of achieving the quoted performance capability

Parking Orbit Perigee Altitude – The minimum parking orbit perigee altitude is 80 nmi (148 km). This constraint applies to parking orbit ascent only.

Payload Fairing Jettison Criteria – Jettison occurs only after the 3-sigma qV $\leq$ 360 Btu/ft·hr^2 (1135 W/m^2)

Booster Staging Acceleration Level – Booster phase is terminated when the nominal axial acceleration level reaches target levels. For LEO missions, BECO occurs at 5.2g for Atlas IIA and 5.0g for Atlas IIAS.

Range Safety – The Instantaneous Impact Point (IIP) trace and impact points of jettisoned items are constrained to clear land mass by distances that satisfy VAFB range safety requirements

Launch Azimuths – 63.0- and 60.0-degree inclined orbit cases use launch azimuth constraint of 155 degrees, with yaw turn

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8.7.2 ONE-BURN CIRCULAR ORBIT CAPABILITY – The single burn to circular orbit is achieved with Centaur targeting directly into the desired circular orbit. From Figures 8-6 through 8-8, it is clear that the direct ascent to circular orbit is advantageous for low-altitude orbits. As altitude increases, it becomes more advantageous to use the parking orbit ascent design.

8.7.3 TWO-BURN CIRCULAR ORBIT CAPABILITY – Parking orbit ascent to circular orbit missions utilize a parking orbit design to achieve maximum mission performance while meeting Centaur operational constraints. The performance quoted in this section assumes a parking orbit perigee altitude of 80 nmi (148 km). Since parking orbit coast times are increased for higher-altitude circular orbits, an extended mission kit was added to Centaur to meet the higher RCS, tank pressurization, and electrical power requirements associated with the longer coast mission.

8.8 GUIDANCE/SEPARATION POINTING ACCURACIES

The Atlas II family's combination of precision guidance hardware with flexible guidance software provides accurate payload injection conditions for a wide variety of missions. The minimal data required to specify targeted end conditions provides for rapid preflight retargeting in response to changing mission requirements. These functional capabilities have been demonstrated on many LEO, geosynchronous orbit, lunar, and interplanetary missions.

Accuracy for a variety of Earth-orbital missions is displayed in Table 8-3 and is typical of the three-sigma accuracies following final upper stage burn.

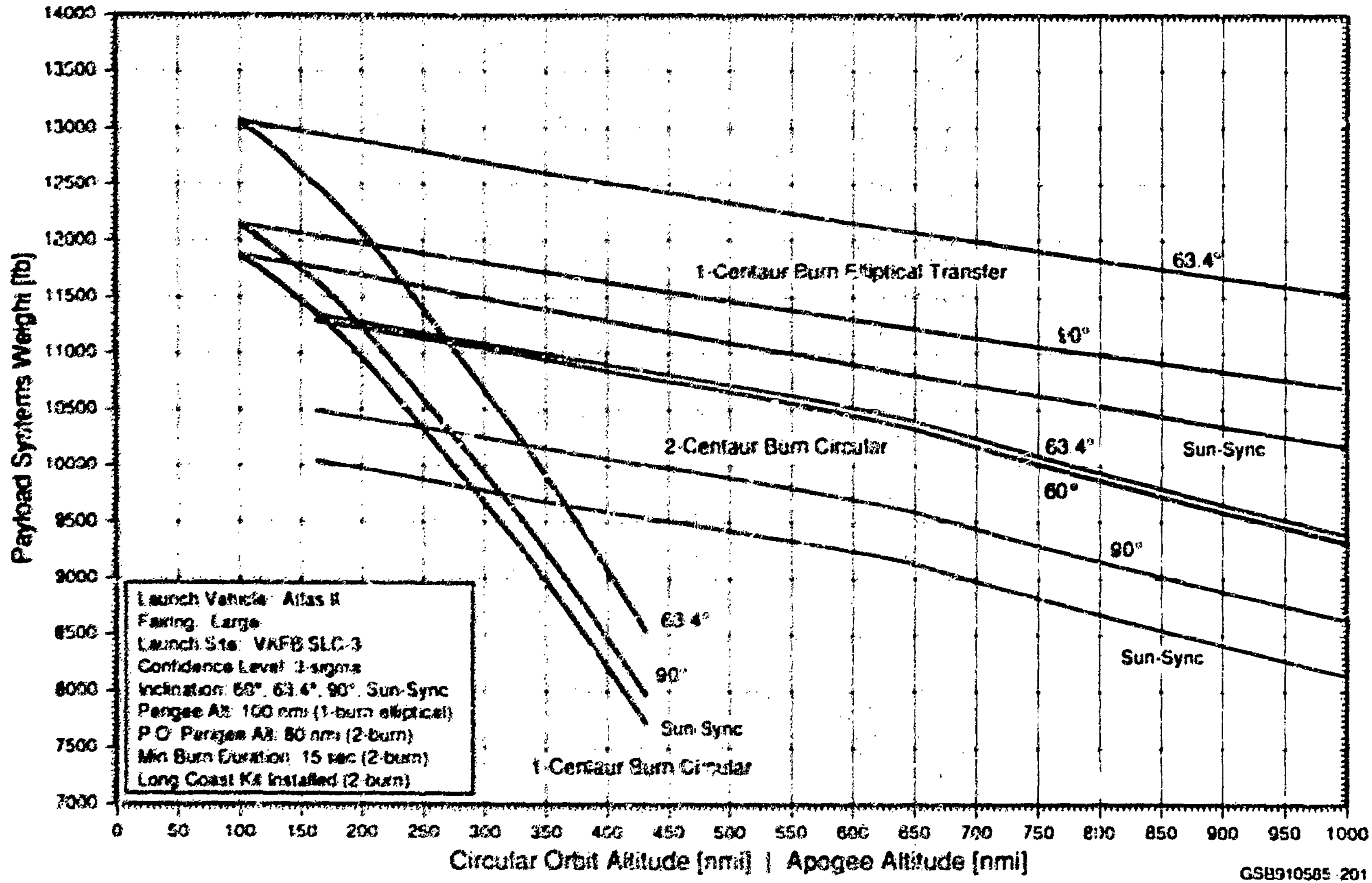

Figure 8-6a. Atlas II low Earth orbit performance

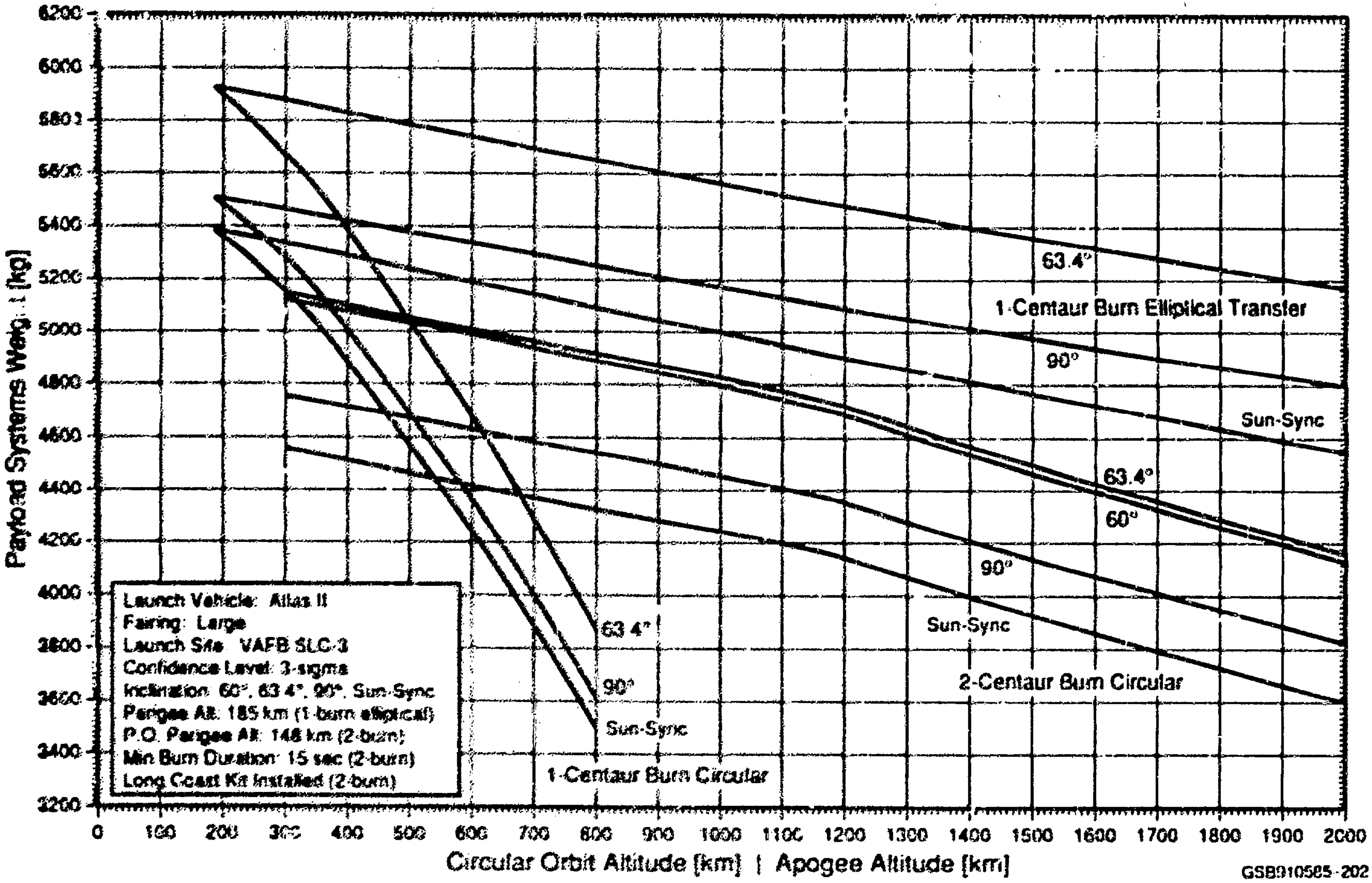

Figure 8-6b. Atlas II low Earth performance (metric)

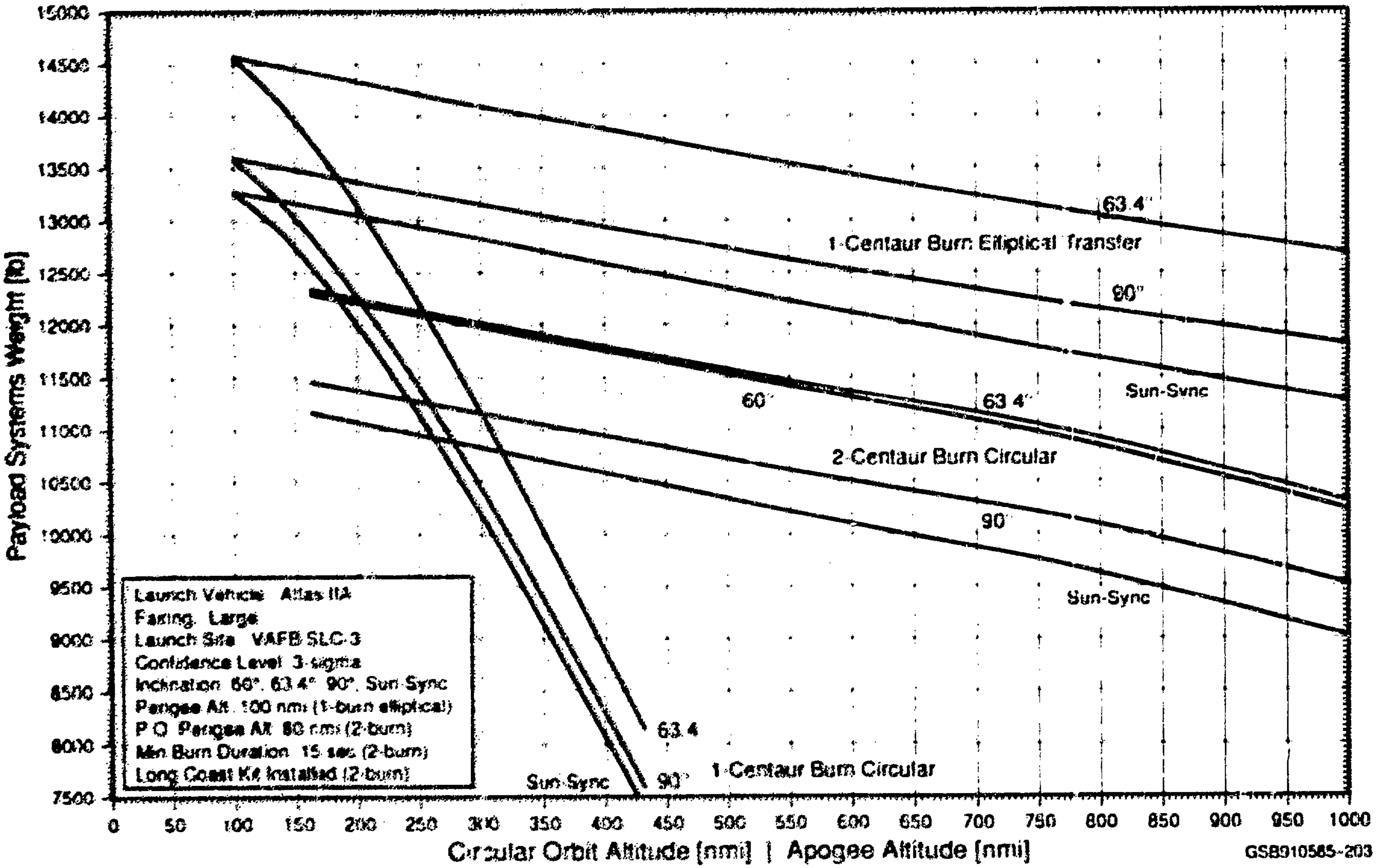

Figure 8-7a. Atlas IIA low Earth orbit performance.

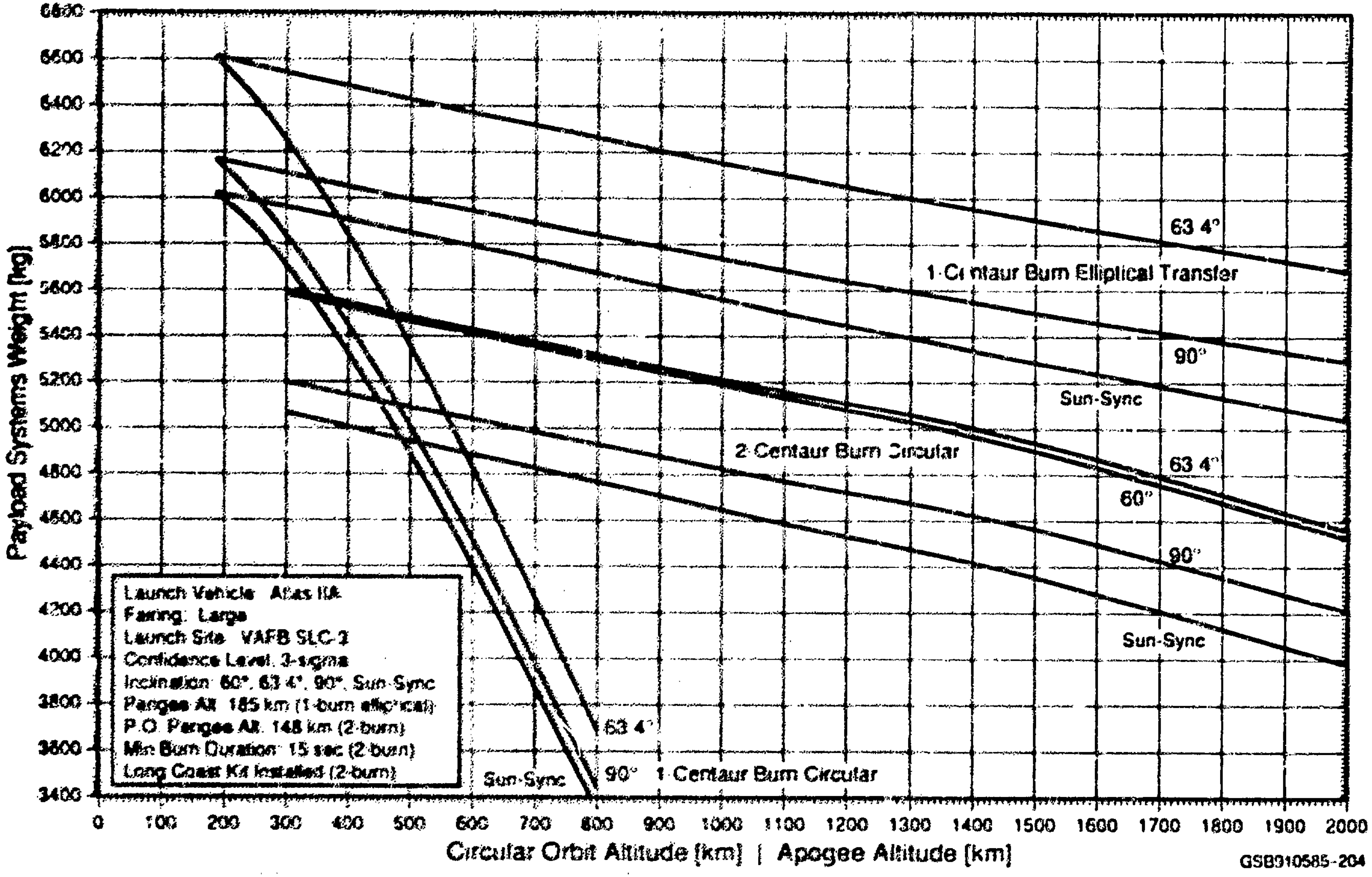

Figure 8-7b Atlas IIA low Earth performance (metric)

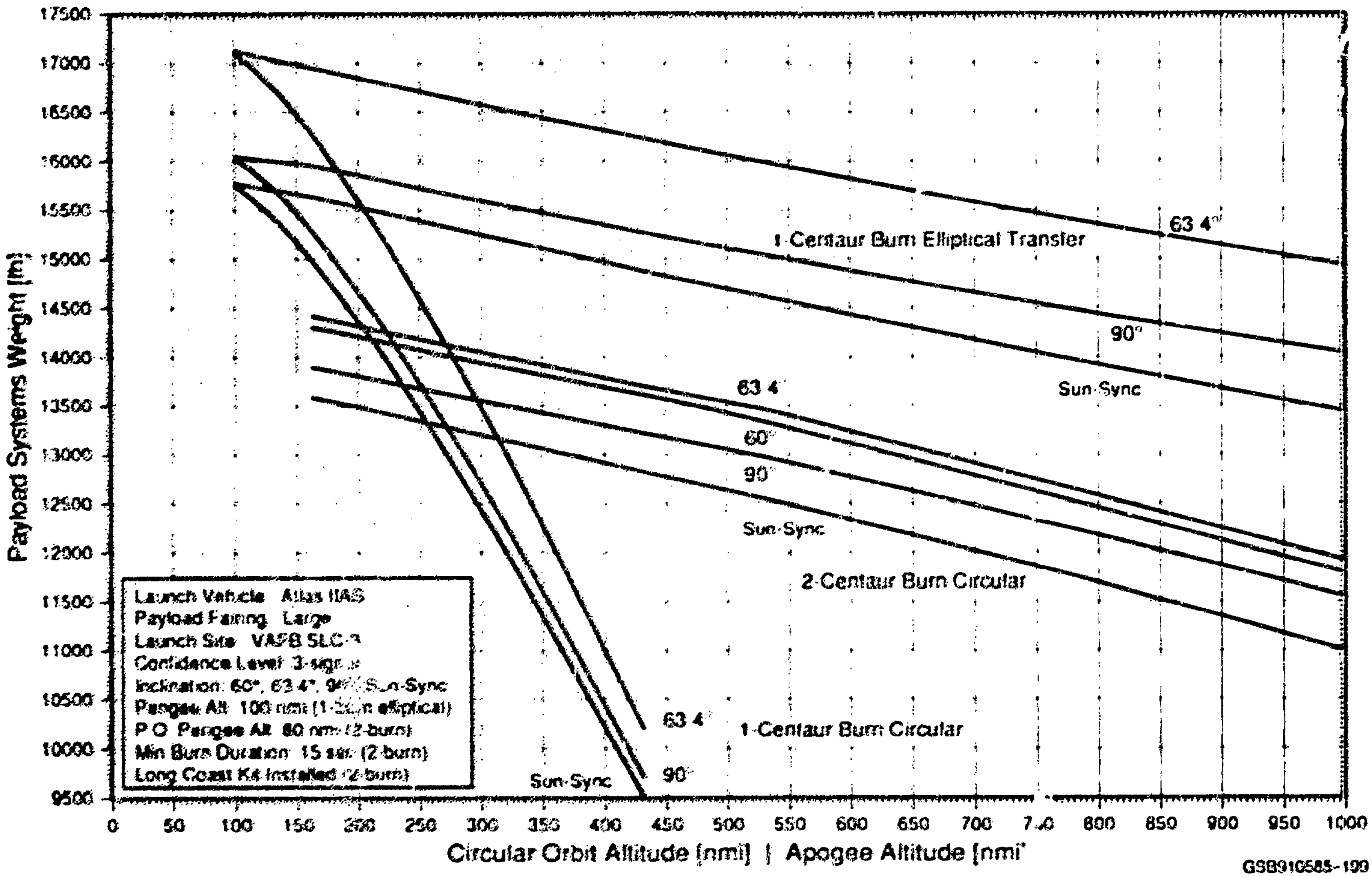

Figure 8-8a. Atlas IIAS low Earth orbit performance

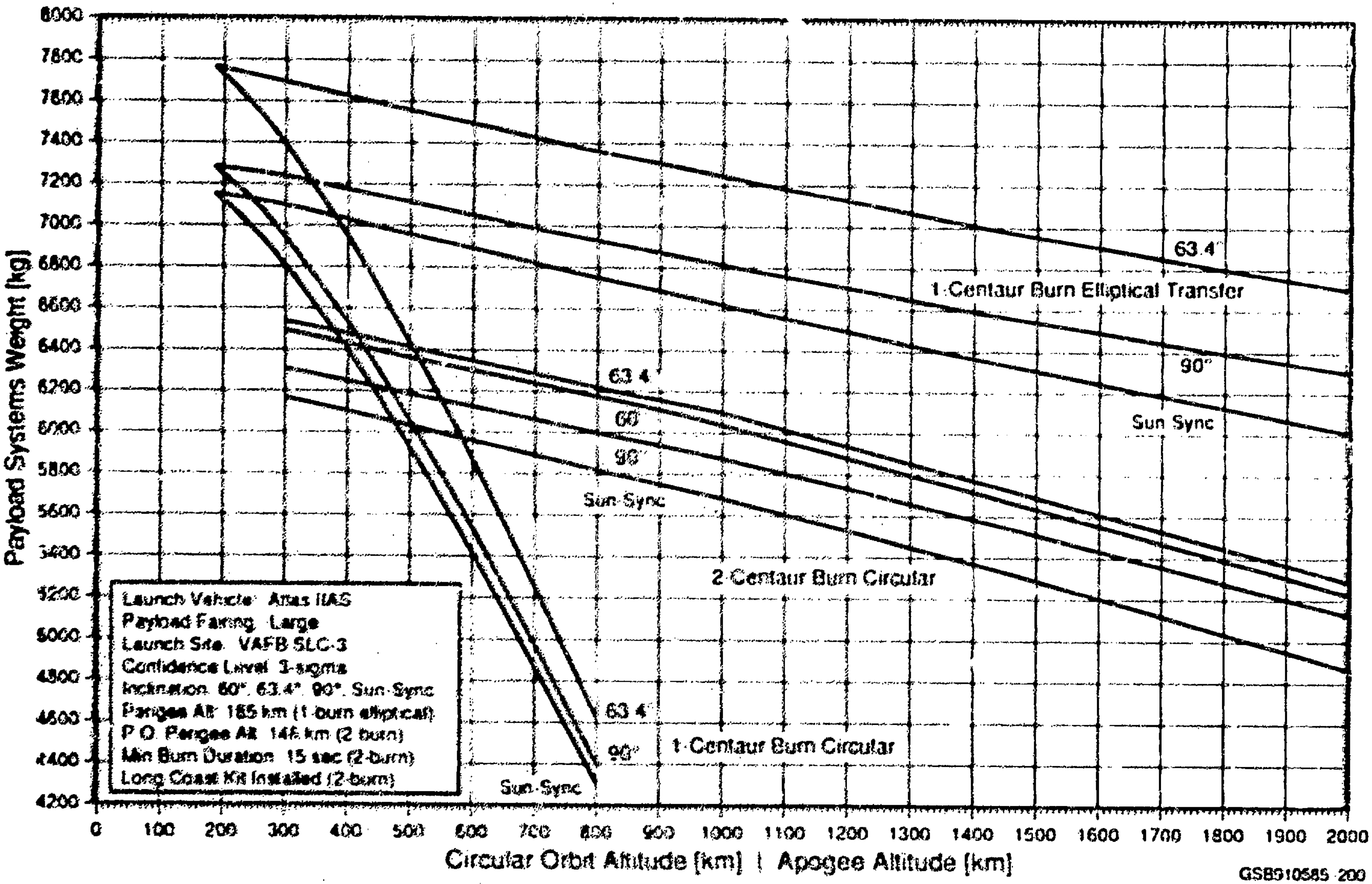

Figure 8-8b. Atlas IIAS low Earth performance (metr

Table 8-3. Typical injection accuracies at spacecraft separation.

Orbit at Centaur/Spacecraft Separation			± Three-Sigma Errors				
Mission	Apogee nmi (km)	Inclination (deg)	Apogee nmi (km)	Perigee nmi (km)	Inclination (deg)	Argument of Perigee (deg)	RAAN (deg)
LEO (Elliptical)[1]	416 (770)	90.0	3.2 (5.9)	0.5 (0.9)	0.07	0.40	0.05
LEO (Elliptical)[2]	380 (705)	98.2	3.1 (5.7)	0.7 (1.3)	0.07	0.40	0.05
LEO (Circular)	216 (400)	60.0	2.4 (4.4)	2.4 (4.4)	0.06	N/A	0.06
LEO (Circular)	380 (705)	98.2	2.7 (5.0)	2.7 (5.0)	0.07	N/A	0.05

NA – Not Applicable

[1] 150-nmi (278 km) perigee altitude

[2] 100-nmi (185 km) perigee altitude

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APPENDIX A • ATLAS HISTORY

A.1 BACKGROUND

Atlas is built by General Dynamics Space Systems Division. Major subcontractors include Honeywell for the guidance/navigation system, and Pratt & Whitney and Rocketdyne for the propulsion systems. Principal vehicle characteristics include:

- Efficient pressure-stabilized stainless steel structure for high-stage mass fraction
- High-energy liquid hydrogen and oxygen propellant upper stage
- Advanced inertial guidance and control hardware and software for high accuracy and flexibility
- Proven record of reliability and launch dependability for lunar, planetary, and Earth-orbit missions.

There have been 503 Atlas flights since the first research and development launch in 1957. Centaur (the Atlas upper stage) has flown 79 times, including seven missions on the Titan IIIE booster.

The first successful flight of Centaur atop Atlas occurred in November 1963. This was the world's first in-flight ignition of a hydrogen-powered vehicle. Three years later, Centaur performed the first successful space restart of liquid hydrogen engines in October 1966. With this flight, the Centaur research and development phase was completed and Centaur became fully operational.

Over a 28-year period, 73 spacecraft of diverse types and applications have been integrated and launched with Atlas/Centaur. Although the moon and planets have been special domains for Atlas and Centaur, and a wide range of Earth orbiting programs have been accommodated. Atlas/Centaur's recent usage has been principally for commercial and government communications satellite missions.

Figure A-1 illustrates the diverse range of missions, mission types, and customers of the 73 Atlas/Centaurs.

Vehicle reliability has shown a steady growth. Today's Atlas has demonstrated a flight reliability record of 94% based on the Duane methodology.

For the U.S. and European governments, this record has signified dependability in the entrustment of vitally important scientific missions to Atlas and Centaur. For prior and new commercial customers, it ensures on-time, successful launches and satellite system revenue. For General Dynamics, it repre-

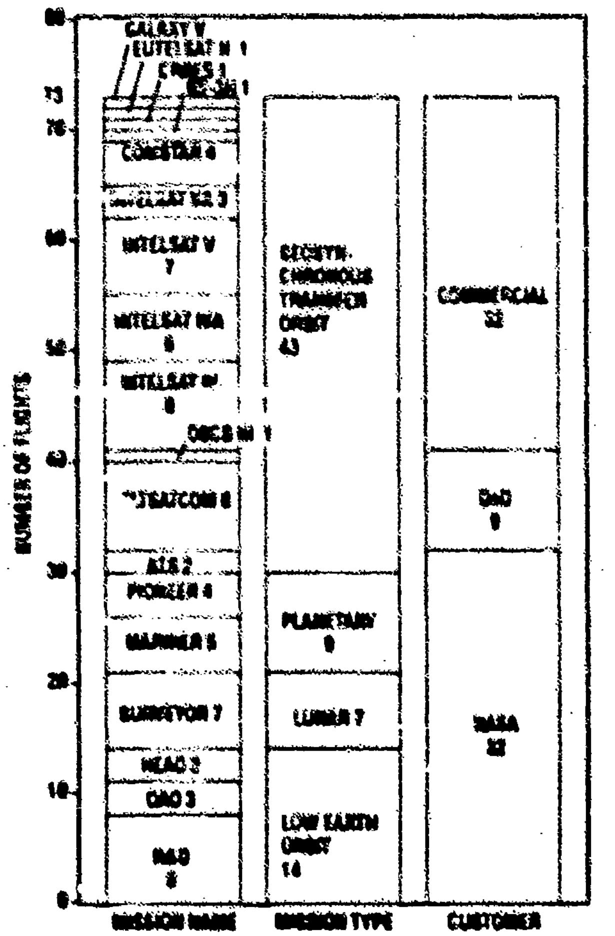

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Figure A-1 Diverse payloads demonstrate Atlas versatility and adaptability sents pride in playing an integral part in successful space programs for more than 30 years.

A.2 DEVELOPMENT

General Dynamics' launch vehicle system development experience dates from the mid-1940s, when initial studies began to explore the feasibility of long-range ballistic missiles. Atlas evolved through various Air Force and NASA programs toward its present role as a highly efficient space launch vehicle (see Figure A-2).

Versions of Atlas were built specifically for manned and unmanned space missions and as a booster for Centaur, the high-energy upper stage designed to carry NASA spacecraft on lunar and planetary missions. As payload weights increased, requirements were met by Atlas improvements, including increased propellant tank sizes and uprated engine performance (Figure A-3).

In 1981, the Atlas G booster for Centaur was developed to improve Atlas/Centaur performance by increasing propellant capacity and uprating engine thrust. Since that evolutionary step, Atlas G has had six successful flights. This is the baseline vehicle, which is being uprated to provide the Atlas I, II, IIA, and IIAS family.

Centaur, our high-energy upper stage, has also followed an evolutionary development to reach its Centaur II and IIA configurations (see Figure A-4). Development began on Centaur in 1958 as a means to carry NASA spacecraft on lunar and planetary missions. It was the first upper stage to use liquid hydrogen fuel. It also required the development of improved avionics capable of guiding its initial payloads on lunar missions. Throughout the development history of Centaur, the avionics have continually been upgraded to provide outstanding orbital insertion accuracy while balancing cost and weight considerations.

Centaur II performance improvements include design changes to stretch the tank structure, which, like Atlas, is a stainless steel pressure-stabilized structure. Shuttle/Centaur and Titan IV/Centaur configurations required a lengthening of the liquid oxygen (aft) tank and expansion of the diameter of the liquid hydrogen tank to 170 inches (4.32 meters) from the 120-inch (3.05-meter) diameter to increase fuel capacity.

A.3 MISSION INTEGRATION

The most challenging part of our ongoing effort is mission integration. We have integrated over 50 different types of spacecraft with both Atlas and Cen-

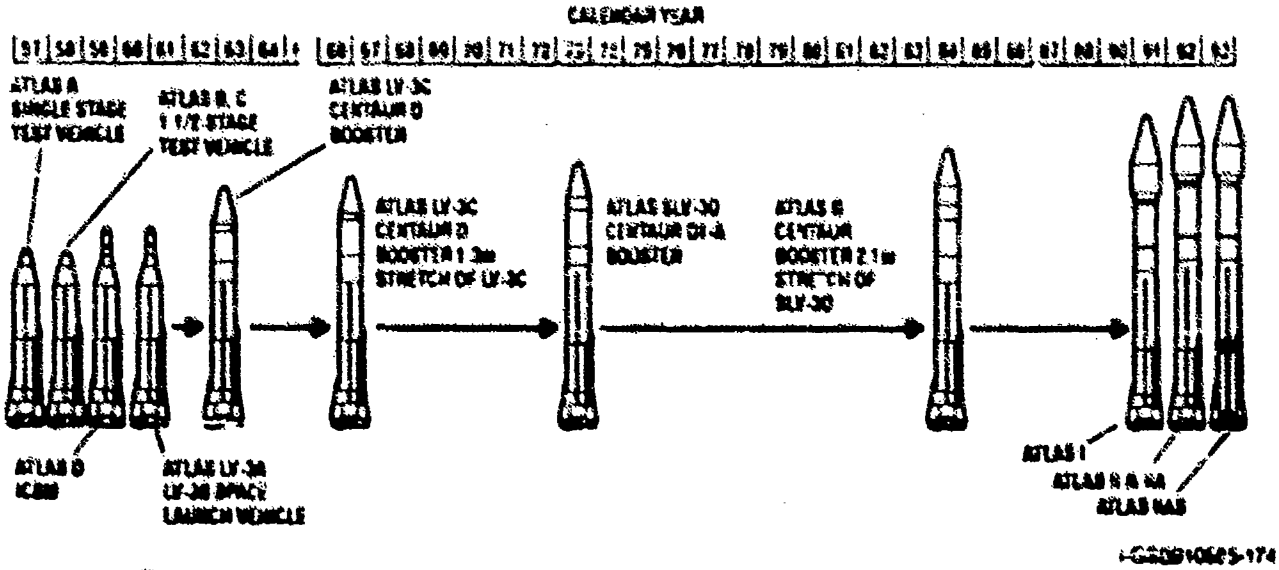

Figure A-2 Atlas and Centaur have successively evolved to satisfy requirements of many missions

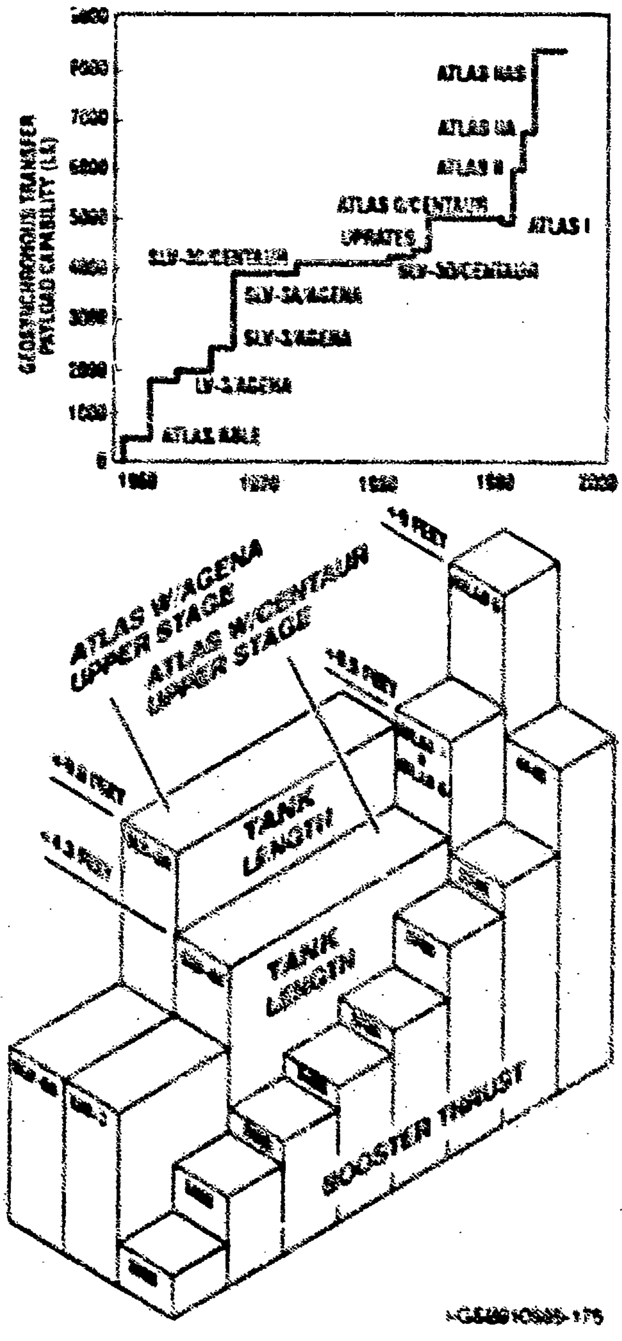

Figure A-3 Atlas performance has increased with spacecraft requirements

taur, providing mission-peculiar analyses and design, verification testing, interface definition and documentation, and other services as needed for a multitude of scientific and commercial payloads. The General Dynamics team has developed the skill required to integrate payloads and accomplish the mission-to-mission changes that are typically required to accommodate evolving needs.

A.4 FLIGHT HISTORY

The quality of Atlas and Centaur launch services, as demonstrated by mission success, is excellent, with a continuous Atlas space launch success record for 34 vehicles over the last 14 years. Centaur had a string of 42 operational successes between 1971 and 1984, on both Atlas and Titan boosters. Table A-1 tabulates the total operational record of Atlas/Centaur as a combined vehicle, after eight R&D flights. Corrective actions for flight failures were incorporated, and no failures were repeated on subsequent flights.

A.5 RELIABILITY

Many of the same vehicle enhancements that increase performance (Figure A-3) also increase reliability. Comprehensive failure investigation and corrective action following each failure have increased reliability predicted by the J.T. Duane reliability growth model from 85% to 94% since 1971, when our current Product Assurance program was implemented.

The conservative Duane Reliability Growth Model is based on observations of the reduction in failure rate as the cumulative number of missions increases. When mean missions between flight failures are plotted on log-log scale against the cumulative missions flown, the data points fall approximately in a straight line. The steepness of the slope of this line, called α, gives a measure of the growth rate of improvements in reliability over time. In our case, the high value of α achieved is associated with performance upgrades of our operational launch vehicles. The growth rate is due to our rigorous failure analysis and corrective action system, and our focus on controlling process failure causes. We expect a high probability of mission success through our use of only proven hardware of high inherent design reliability and processes that are controlled and stable.

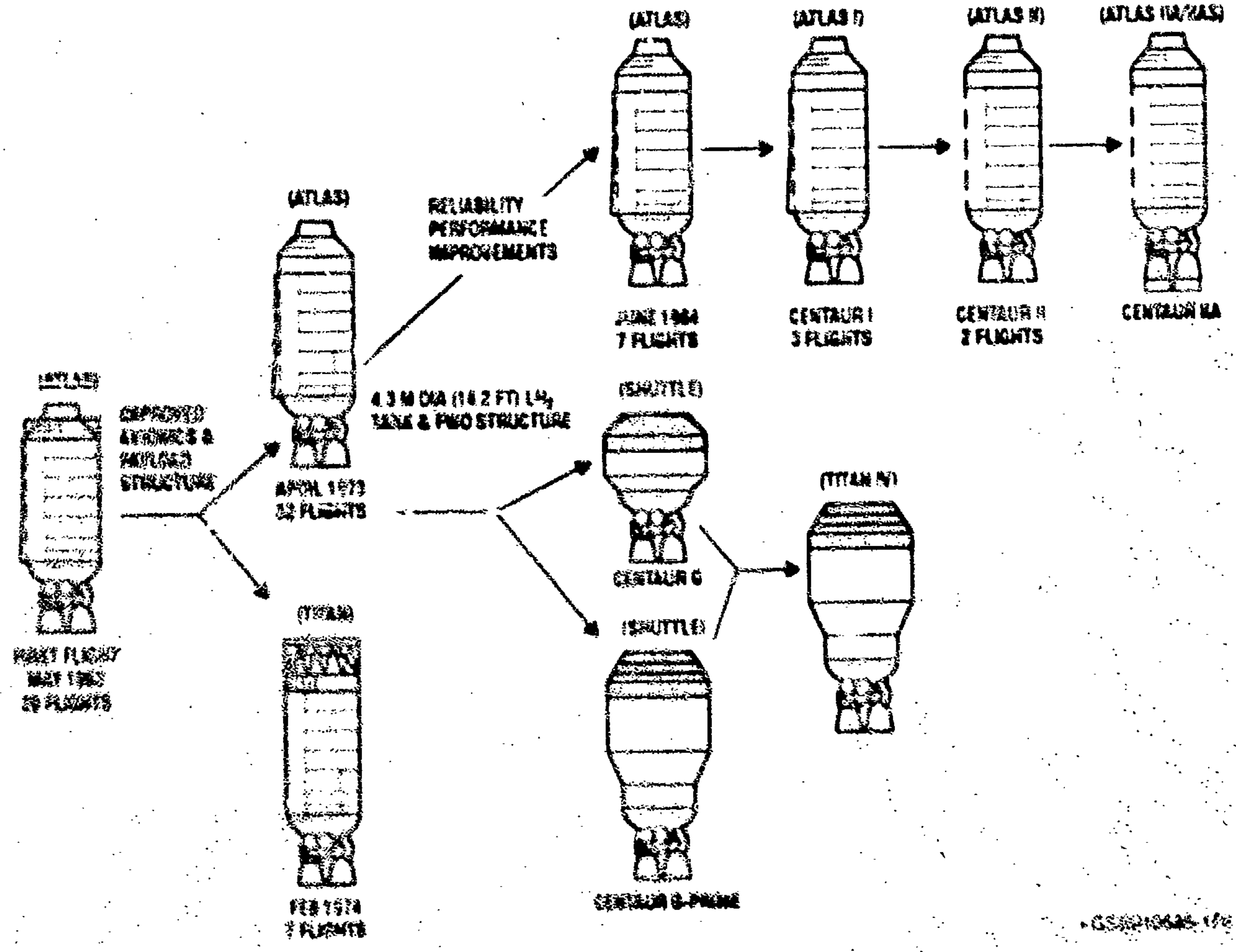

Figure A-4 Centaur evolution reflects changes to accommodate other boost vehicles as well as performance and reliability improvements

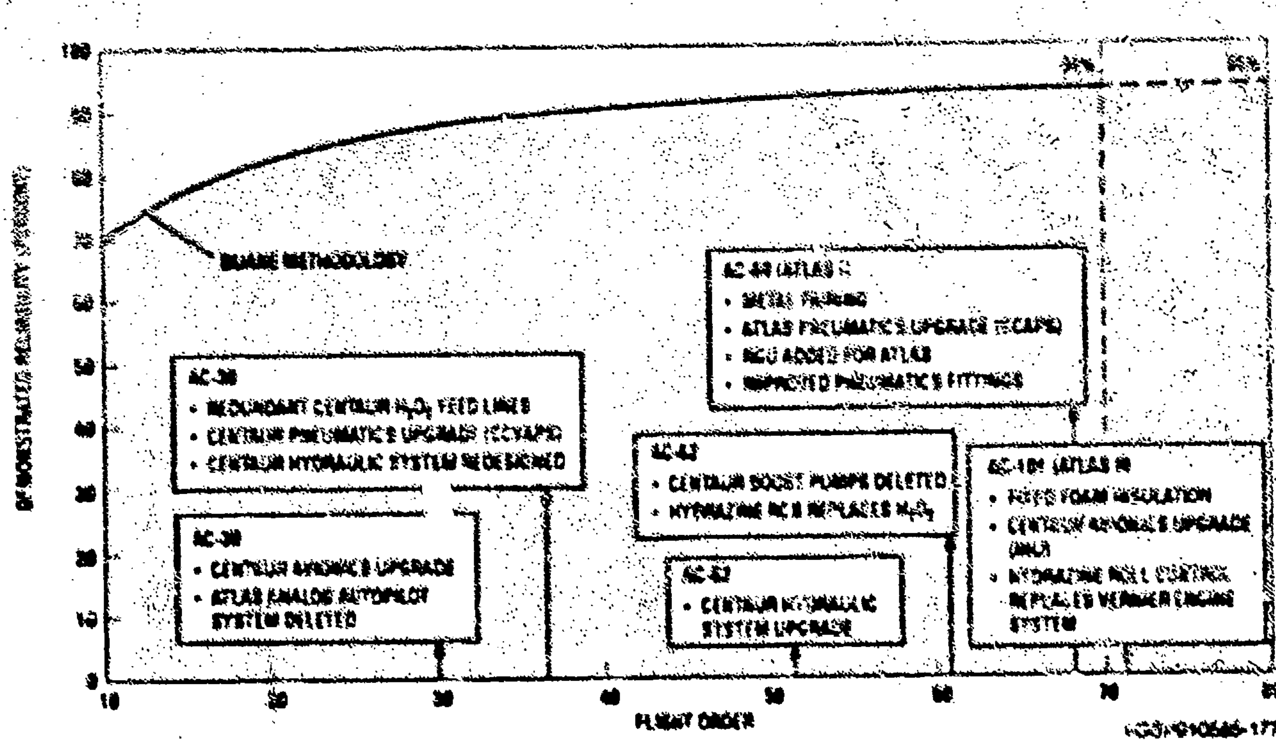

Figure A-5 Atlas reliability enhancements and performance growth are concurrent

Table A-1. Summary of Atlas/Centaur operational launch history.

Date	Mission	Vehicle	LV Results
1966			
May 30	Surveyor	AC-10	Success
Sept 20	Surveyor	AC-7	Success
1967			
Apr 17	Surveyor	AC-12	Success
Jul 14	Surveyor	AC-11	Success
Sep 8	Surveyor	AC-13	Success
Nov 7	Surveyor	AC-14	Success
1968			
Jan 7	Surveyor	AC-15	Success
Aug 10	ATS-D	AC-17	Centaur 2nd burn failure
Dec 7	OAO-A	AC-16	Success
1969			
Feb 24	Mariner Mars	AC-20	Success
Mar 27	Mariner Mars	AC-19	Success
Aug 12	ATS-E	AC-18	Success
1970			
Nov 30	OAO-B	AC-21	Shroud jettison failure
1971			
Jan 25	INTELSAT IV	AC-25	Success
May 8	Mariner Mars	AC-24	Centaur flight control failure
May 30	Mariner Mars	AC-23	Success
Dec 19	INTELSAT IV	AC-26	Success
1972			
Jan 22	INTELSAT IV	AC-28	Success
Mar 2	Pioneer F	AC-27	Success
Jun 13	INTELSAT IV	AC-29	Success
Aug 21	OAO-C	AC-22	Success
1973			
Apr 5	Pioneer G	AC-30	Success
Aug 23	INTELSAT IV	AC-31	Success
Nov 3	Mariner Venus/Mercury	AC-34	Success
1974			
Nov 21	INTELSAT IV	AC-32	Success
1975			
Feb 20	INTELSAT IV	AC-33	Atlas electrical failure
May 22	INTELSAT IV	AC-35	Success
Sept 25	INTELSAT IVA	AC-36	Success
1976			
Jan 29	INTELSAT IVA	AC-37	Success
May 13	COMSTAR	AC-38	Success
Jul 22	COMSTAR	AC-40	Success
1977			
May 26	INTELSAT IVA	AC-39	Success
Aug 12	HEAO-A	AC-45	Success
Sept 29	INTELSAT IVA	AC-43	Atlas propulsion failure
1978			
Jan 6	INTELSAT IVA	AC-46	Success
Feb 9	FLTSATCOM	AC-44	Success
Mar 31	INTELSAT IVA	AC-48	Success
May 20	Pioneer Venus	AC-50	Success
Jun 29	COMSTAR	AC-41	Success
Aug 8	Pioneer Venus	AC-51	Success
Nov 13	HEAO-B	AC-52	Success
1979			
May 4	FLTSATCOM	AC-47	Success
Sept 20	HEAO-C	AC-53	Success
1980			
Jan 17	FLTSATCOM	AC-49	Success
Oct 30	FLTSATCOM	AC-57	Success
Dec 6	INTELSAT V	AC-54	Success
1981			
Feb 21	COMSTAR	AC-42	Success
May 23	INTELSAT V	AC-56	Success
Aug 5	FLTSATCOM	AC-59	Success
Dec 15	INTELSAT V	AC-55	Success
1982			
Mar 4	INTELSAT V	AC-58	Success
Sept 28	INTELSAT V	AC-60	Success
1983			
May 19	INTELSAT V	AC-61	Success
1984			
Jun 9	INTELSAT VA	AC-62	Centaur tank leak failure
1985			
Mar 22	INTELSAT VA	AC-63	Success
Jun 29	INTELSAT VA	AC-64	Success
Sept 28	INTELSAT VA	AC-65	Success
1986			
Dec 4	FLTSATCOM	AC-66	Success
1987			
Mar 26	FLTSATCOM	AC-67	No trial (lightning strike)
1989			
Sept 25	FLTSATCOM	AC-68	Success
1990			
July 25	CRRES	AC-69	Success
1991			
Apr 18	BS-3H	AC-70	Foreign object damage in Centaur engine
Dec 7	EUTELSAT II	AC-102	Success
1992			
Feb 10	DSCS III	AC-101	Success
Mar 13	Galaxy V	AC-72	Success

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APPENDIX B • SPACECRAFT DATA REQUIREMENTS

The items listed in this Appendix are representative of the information required for spacecraft integration and launch activities. The data usually is provided by the customer in the form of an Interface Requirements Document (IRD) and is the basis for preparation of the Interface Control Document (ICD). Additional information may be required for specific spacecraft.

B.1 SPACECRAFT DATA

Table B-1 indicates the spacecraft information required to assess the compatibility of the spacecraft with the Atlas. Items in bold should be provided for a preliminary compatibility assessment while all items should be completed for a detailed assessment. This information is typically supplied for the spacecraft prior to a proposal being offered by General Dynamics Commercial Launch Services.

Tables B-2 through B-8 indicate spacecraft data required after contract signature to start integration of the spacecraft. The asterisks in these tables indicate data required at an initial meeting between General Dynamics and the customer. This data will provide the detailed information required to fully integrate the spacecraft in order to determine such items as optimum mission trajectory as well as to verify compatibility of the launch vehicle environments and interfaces.

B.2 SPACECRAFT DESIGN REQUIREMENTS

Table B-9 lists specific requirements that should be certified by analysis and/or test by the spacecraft agency to be compatible for launch with the Atlas. Should the spacecraft not meet any of these requirements, General Dynamics will work with the customer to resolve the incompatibility.

B.3 CAD DATA TRANSFER REQUIREMENTS

Where the transfer of computer-aided design (CAD) information is required or appropriate, the user must provide that data in accordance with specified formats. The following are the three types of data formats that can be used for the transfer of CAD data to GDSS Computervision CADDSTATION systems

1. Computervision Format (from CADDSTATION systems) — Data can be supplied on 1/2-inch magnetic tape directly from Computervision CADDSTATION systems as single precision data bases. The _pd files and drawing files should be written to tape with the default blocksize of 512 using the tar command syntax as follows:

   tar cvf /dev/rmt0 partfile

2. Computervision Format (from CGOS systems) — Data can be transferred from Computervision CGOS systems. The &pd files and drawing files should be written to 1/2-inch 9-track magnetic tape in CVASCII format with the Futil copy command as follows:

   copy partfile :mt/label = 'partname'

3. IGES Format (from other CAD systems) — Data can be transferred from other CAD systems using IGES 4.0 format. Solids should be converted to 3D wireframe and surface entities in the IGES file. The maximum number of entities in the model should be 65,500. Curves making up bounded planes should be converted to be independent entities. The entities and views in the model should be unblanked. Entities should be as close as possible to the origin of the model coordinate system. Subfigures contained in the model should be included in the IGES file. Critical sculpted surfaces should be accompanied by a mesh of check-points lying on the surface at a minimum density of at least three points between each patch boundary. The files should be written to 1/2-inch magnetic tape at 1600 BPI, 7-bit ASCII, record size of 80, and a block size of 800. The tape should be labeled with these parameters. The following command syntax can be used from UNIX systems:

   dd if = igesfile of = /dev/rmt0 cbs = 80 bs = 800 conv = block

For each of the cases above, the spacecraft contractor should verify that the tapes contain the correct data by reading the tapes back into the originating CAD system prior to the tape transmittal. In addition. all data transfers should have the following information included with the tape.

- Data format (ASCII. CVASCII. etc.)
- Blocking factor
- Number of records
- Size of IGES file (on Kbytes)
- Numbers of lines in IGES file
- Entity list with a count of each entity
- Name and phone number of the computer system administrator/operator
- Name and phone number of contact person if problems or questions arise
- A multi-view plot of the drawing

As an alternative to tape transfers. spacecraft contractors can electronically transmit the CAD data using FTP via the Internet network.

Table B-1. Spacecraft information worksheet.

For a preliminary compatibility assessment, all items in bold print should be completed. For a detailed compatibility assessment, all items should be completed.

Spacecraft Name:
Spacecraft Owner:
Name of Principal Contact:
Telephone Number:
Date:

Spacecraft Manufacturer:
Spacecraft Model Number:
Number of Launches:
Dates of Launches:

Spacecraft Design Parameter	SI Units	English Units
TRAJECTORY REQUIREMENTS		
1) Satellite Mass	___ kg	___ lbm
2) Minimum Satellite Lifetime	___ Years	___ Years
3) Final Orbit Apogee	___ km	___ nmi
4) Final Orbit Perigee	___ km	___ nmi
5) Final Orbit Inclination	___ degrees	___ degrees
6) Propulsion – propellant type, orbit insertion		
7) Propulsion – propellant type, stationkeeping		
8) Propulsion – multiple burn capability (Y/N)		
9) Propulsion – propellant mass	___ kg	___ lbm
10) Propulsion – Effective Isp	___ sec	___ sec
11) Maximum Apogee Allowable	___ km	___ nmi
12) Minimum Perigee Allowable	___ km	___ nmi
13) Argument of Perigee Requirement	___ degrees	___ degrees
14) Right Ascension of the Ascending Node Requirement	___ degrees	___ degrees
15) Apogee Accuracy Requirement	___ km	___ nmi
16) Perigee Accuracy Requirement	___ km	___ nmi
17) Inclination Accuracy Requirement	___ degrees	___ degrees
18) Argument of Perigee Accuracy Requirement	___ degrees	___ degrees
19) Right Ascension of the Ascending Node Accuracy Requirement	___ degrees	___ degrees
MECHANICAL INTERFACE		
20) Spacecraft Mechanical Drawing (Launch Configuration)		
21) Spacecraft Effective Diameter	___ mm	___ in.
22) Spacecraft Height	___ mm	___ in.
23) Spacecraft/Launch Vehicle Interface Diameter	___ mm	___ in.
24) Payload Separation System Supplier (S/C or LV)		
25) Maximum Spacecraft Cross-sectional Area	___ m^2	___ ft^2
26) Number and Size of Payload Fairing Access Doors	___ mm x mm	___ in. x in.
27) Pre-separation RF Transmission Requirement	___ band	___ band
ELECTRICAL INTERFACE		
28) Spacecraft Electrical Drawing		
29) Number of Launch Vehicle Signals Required		
30) Number of Separation Discretes Required		
31) Number of Umbilicals and Pins/Umbilical		
32) Curve of S/C-Induced Electric Field Radiation		
33) Number of Instrumentation Analogs Required		

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Table B-1. Spacecraft information worksheet (continued).

Spacecraft Design Parameter	SI Units	English Units
THERMAL ENVIRONMENT		
34) Prelaunch Ground Transport Temperature Range	___ C	___ F
35) Prelaunch Launch Pad Temperature Range	___ C	___ F
36) Maximum Prelaunch Gas Impingement Velocity	___ m/sec	___ ft/sec
37) Maximum Ascent Heat Flux	___ W/m^2	___ Btu/hr ft^2
38) Maximum Free-Molecular Heat Flux	___ W/m^2	___ Btu/hr ft^2
39) Maximum Fairing Ascent Depressurization Rate	___ mbar/sec	___ psi/sec
40) Prelaunch Relative Humidity Range	___ %	___ %
41) Maximum Prelaunch Air Conditioning System Noise	___ dB	___ dB
42) Pre-separation Spacecraft Power Dissipation	___ W	___ Btu/hr
43) Maximum Payload Fairing Ascent Differential Pressure	___ mbar	___ psid
44) Maximum Free-stream Dynamic Pressure	___ mbar	___ psi
DYNAMIC ENVIRONMENT		
45) Maximum Allowable Flight Acoustics	___ dB OA	___ dB OA
46) Allowable Acoustics Curve		
47) Maximum Allowable Random Vibration	___ G_{rms}	___ G_{rms}
48) Allowable Random Vibration Curve		
49) Maximum Allowable Sine Vibration	___ G_{axis}	___ G_{axis}
50) Allowable Sine Vibration Curve		
51) Maximum Allowable Shock	___ G	___ G
52) Allowable Shock Curve		
53) Maximum Acceleration (Static + Dynamic) Lateral	___ G	___ G
54) Maximum Acceleration (Static + Dynamic) Longitudinal	___ G	___ G
55) Fundamental Natural Frequency -- Lateral	___ Hz	___ Hz
56) Fundamental Natural Frequency -- Longitudinal	___ Hz	___ Hz
57) CG -- Thrust Axis (origin at separation plane)	___ mm	___ in.
58) CG - Y axis	___ mm	___ in.
59) CG - Z axis	___ mm	___ in.
60) CG Tolerance - Thrust Axis	___ ± mm	___ ± in.
61) CG Tolerance - Y Axis	___ ± mm	___ ± in
62) CG Tolerance - Z Axis	___ ± mm	___ ± in.
CONTAMINATION REQUIREMENTS		
63) Fairing Air Cleanliness	___ Class	___ Class
64) Maximum Deposition on Spacecraft Surfaces	___ mg/m^2	___ mg/m^2
65) Outgassing -- Total Weight Loss	___ %	___ %
66) Outgassing -- Volatile Condensable Material Weight Loss	___ %	___ %
SPACECRAFT DESIGN SAFETY FACTORS		
67) Airborne Pressure Vessel Burst Safety Factor		
68) Airborne Pressure System Burst Safety Factor		
69) Structural Limit (Yield) Safety Factor		
70) Structural Ultimate Safety Factor		
71) Battery Burst Safety Factor		
SPACECRAFT QUALIFICATION TEST PROGRAM		
72) Acoustic Qualification	___ + dB	___ + dB
73) Random Vibration Qualification Safety Factor		
74) Sine Vibration Qualification Safety Factor		
75) Shock Qualification Safety Factor		
76) Loads Qualification Safety Factor		

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Table B-1. Spacecraft information worksheet (continued).

Spacecraft Design Parameter	SI Units	English Units
GUIDANCE		
77) **Range of Separation Velocity**	**___ m/sec**	**___ ft/sec**
78) **Maximum Angular Rate at Separation – Roll**	**___ rpm**	**___ rpm**
79) Maximum Angular Rate Uncertainty – Roll	___ ± rpm	___ ± rpm
80) Maximum Angular Rate at Separation – Pitch and Yaw	___ rpm	___ rpm
81) Maximum Angular Rate Uncertainty – Pitch and Yaw	___ ± rpm	___ ± rpm
82) Maximum Angular Acceleration	___ rad/sec²	___ rad/sec²
83) Maximum Pointing Error Requirement	___ deg	___ deg
84) Maximum Allowable Tip-off Rate	___ deg/sec	___ deg/sec
85) Coefficients of Inertia – Ixx (x = thrust axis)	___ kg m²	___ slug ft²
86) Coefficients of Inertia – Ixx Tolerance	___ ± kg m²	___ ± slug ft²
87) Coefficients of Inertia – Iyy	___ kg m²	___ slug ft²
88) Coefficients of Inertia – Iyy Tolerance	___ ± kg m²	___ ± slug ft²
89) Coefficients of Inertia – Izz	___ kg m²	___ slug ft²
90) Coefficients of Inertia – Izz Tolerance	___ ± kg m²	___ ± slug ft²
91) Coefficients of Inertia – Ixy	___ kg m²	___ slug ft²
92) Coefficients of Inertia – Ixy Tolerance	___ ± kg m²	___ ± slug ft²
93) Coefficients of Inertia – Iyz	___ kg m²	___ slug ft²
94) Coefficients of Inertia – Iyz Tolerance	___ ± kg m²	___ ± slug ft²
95) Coefficients of Inertia – Ixz	___ kg m²	___ slug ft²
96) Coefficients of Inertia – Ixz Tolerance	___ ± kg m²	___ ± slug ft²

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Table B-2. Mission requirements.

Type of Data	Scope of Data
• Number of launches	
• Frequency of launches	
• Spacecraft orbit parameters including tolerances (park orbit, transfer orbit)	1. Apogee altitude 2. Perigee altitude 3. Inclination 4. Eccentricity 5. Argument of perigee 6. RAAN
Launch window constraints	
• Preseparation function	1. Pre-arm 2. Arm 3. Spacecraft equipment deployment timing and constraints 4. Acceleration constraints (pitch, yaw, roll) 5. Attitude constraints 6. Spinup requirements
• Separation parameters (including tolerances)	1. Angular rate of spacecraft 2. Orientation (pitch, yaw, and roll axis) 3. Acceleration constraints
Any special trajectory requirements	1. Boost phase 2. Coast phase 3. Free molecular heating constraints 4. Thermal maneuvers 5. Separation over telemetry and tracking ground station 6. Maximum or minimum position drift rate (synchronous orbit) 7. Telemetry dipout maneuvers

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Table B-3. Spacecraft characteristics.

Type of Data	Scope of Data
*Configuration drawings	1. Drawings showing the configuration, shape, dimensions and protrusions into the mounting adapter (ground launch and deployment configurations) 2. Coordinates (spacecraft relative to launch vehicle) 3. Special clearance requirements
*Apogee kick motor	1. Manufacturer's designation 2. Thrust 3. Specific impulse 4. Burn action time 5. Propellant off-load limit
* Mass properties (launch and orbit configurations)	* 1. Weight - Specify total, separable and retained masses 2. Center of gravity -- Specify in three orthogonal coordinates parallel to the booster roll, pitch, and yaw axes, for total, separable, and retained masses 3. Changes in cg due to deployment of appendages 4. Propellant slosh models
Moments of inertia (launch and orbit configurations)	Specify about the axes through the spacecraft cg that are parallel to the Atlas roll, pitch, and yaw axes for total, separable, and retained masses.
Structural characteristics	Spring ratio of structure, elastic deflection constants, shear stiffness, dynamic model, bending moments, and shear loads at Atlas/Centaur/spacecraft interface and limitations to include acoustic, shock acceleration, temperature, and bending moments.
Dynamic model for 3-D loads analysis	1. Generalized stiffness matrix (see Appendix C for details) 2. Generalized mass matrix 3. Description of the model, geometry and coordinate system 4. Loads transformation matrix Note: Models must include rigid body and normal modes.
Handling constraints	1. Spacecraft orientation during ground transport 2. Spacecraft handling limits (e.g., acceleration constraints)
Spacecraft critical orientation during	1 Location and direction of antennas checkout, prelaunch and orbit 2. Location, look angle and frequency of sensors 3. Location and size of solar arrays
Safety items	1. Cryogenics 2. Corrosive fluids 3. High-pressure gases 4. Radiation (radioactive materials) 5. Ordnance items a. Location and number b. Type, manufacturing part number, and hazard classification per ESMCM 127-1 c. No-fire and all-fire current levels d. Installation (e.g., who, when, where) e. RF susceptibility (e.g., pin-to-case, pin-to-pin, bridge wire and input frequency) f. Electrostatic sensitivity data 6. AKM flight termination system description
Thermal characteristics	1. Spacecraft thermal math model (if available) 2 Emissivity 3. Conductivity 4. Resistivity 5. Thermal constraints (maximum and minimum allowable temperature) 6. Heat generation (e.g., sources, heat, time of operation)

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Table B-3. Spacecraft characteristics (continued).

Type of Data	Scope of Data
Contamination control	1. Requirements for ground-supplied services 2. In-flight conditions (e.g., during ascent and after PLF jettison 3. Surface sensitivity (e.g., susceptibility to propellants, gases and exhaust products)
RF radiation	1. Characteristics (e.g., power levels, frequency and duration) for checkout and flight configuration 2. Locations (e.g., location of receivers and spacecraft when radiating) 3. Checkout requirements (e.g., open-loop, closed-loop, prelaunch, ascent trajectory)

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Table B-4. Aerospace vehicle equipment (AVE) requirements (mechanical).

Type of Data	Scope of Data
• Mechanical interfaces	• 1. Base diameter of spacecraft interface • 2. Structural attachments at S/C interface • 3. Required accessibility to spacecraft in mated condition • 4. Extent of equipment remaining with adapter after spacecraft separation 5. Degree of environmental control required 6. Spacecraft pressurization and fueling system connector type and location and timeline for pressure/fuel system operation 7. Spacecraft/adapter venting requirements
Payload fairing requirements	1. Heating constraints 2. Venting characteristics (e.g., quantity, timing and nature of gases vented from payload) 3. RF transport windows (e.g., size, location, etc., if required) 4. PLF separation (e.g., altitude, cleanliness, shock, aeroheating and airload constraints) 5. Acoustic environment constraints 6. Special environmental requirements
Preflight environment	1. Requirements a. Cleanliness b. Temperature and relative humidity c. Air conditioning d. Air impingement limits 2. Monitoring and verification requirements
• Umbilical requirements	1. Separation from launch vehicle 2. Flyaway at launch 3. Manual disconnect (including when)
Materials	1. Special compatibility requirements 2. Outgassing requirements

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Table B-5. Aerospace vehicle equipment (AVE) requirements (electrical).

Type of Data	Scope of Data
* Power requirements (current, duration, function time and tolerances)	1. 28 Vdc power
	2. Other power
	3. Overcurrent protection
* Command discrete signals	* 1. Number
	2. Sequence
	3. Timing (including duration, tolerance, repetition rate, etc.)
	4. Voltage (nominal and tolerance)
	5. Frequency (nominal and tolerance)
	6. Current (nominal and tolerance)
	7. When discretes are for EED activation, specify minimum, maximum and nominal fire current, minimum and maximum resistance, minimum fire time, operating temperature range and manufacturer's identification of device
Other command and status signals	1. Status displays
	2. Abort signals
	3. Range safety destruct
	4. Inadvertent separation destruct
Ordnance circuits	Safe/arm requirements
* Telemetry requirements	1. Quantity of spacecraft measurements required to be transmitted by Atlas telemetry and type (e.g., temperature, vibration, pressure, etc.) and details concerned with related systems including operating characteristics (response definition of system) and locations and anticipated time of operation
	2. Impedance, capacitance, operating range and full-scale range of each measurement
	3. Signal conditioning requirements (e.g., input impedance, impedance circuit load limits, overcurrent protection and signal-to-noise ratio)
	4. Discrete events (bi-level)
	5. Analog measurements
	6. Transducers required to be furnished by launch vehicle contractor
	7. Minimum acceptable frequency response for each measurement
	8. Minimum acceptable system error for each measurement. (Sampling rate is also governed by this requirement.)
	* 9. Period of flight for which data from each measurement is of interest (e.g., from liftoff to space vehicle separation)
	10. Atlas flight data required by spacecraft contractor
Bonding	1. Bonding requirements at interface
	2. Material and finishes at interface
EMC	1. Grounding philosophy (e.g., MIL-B-5087B)
	2. EMC protection philosophy for low-power, high-power and pyrotechnic circuits
Grounding philosophy	1. Structure (e.g., use of structural as ground and current levels)
	2. Electrical equipment (e.g., grounding technique for "black boxes" and power supplies)
	3. Single-point ground (e.g., location and related equipment)
* Interface connectors	* 1. Connector item (e.g., location and function)
	2. Connector details
	3. Electrical characteristics of signal on each pin
Shielding requirements	1. Each conductor
	2. Overall
	3. Grounding

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Table B-6. AGE/facility requirements (mechanical).

Type of Data	Scope of Data
Spacecraft launch vehicle integration	1. Sequence from spacecraft delivery through mating with the LV 2. Handling equipment required 3. GDSS-provided protective covers or work shields required 4. Identify the space envelope, installation, clearance, and work area requirements 5. Any special encapsulation requirements 6. Support services required
Spacecraft checkout AGE and cabinet data	1. List of all AGE and location where used (e.g., storage requirements on the launch pad) 2. Installation criteria for AGE items: a. Size and weight b. Mounting provisions c. Grounding and bonding requirements d. Proximity to the spacecraft station in use e. Period of usage f. Environmental requirements g. Compatibility with range safety requirements and LV propellants h. Access space to cabinets required for work area, door swing, slideout panels, etc. i. Cable entry provisions and terminal board types in cabinets and/or interface receptacle locations and types j. Power requirements and characteristics of power for each cabinet
Spacecraft environmental protection (preflight)	1. Environmental protection requirements by area, including cleanliness requirements: a. Spacecraft room b. Transport to launch pad c. Mating d. Inside payload fairing e. During countdown 2. Air-conditioning requirements for applicable area (pad area) by: a. Temperature range b. Humidity range c. Particle limitation d. Impingement velocity limit e. Flow rate 3. Indicate if space vehicle is not compatible with LV propellants and what safety measures will be required 4. Environmental monitoring and verification requirements
Space access requirements	1. Access for spacecraft mating and checkout 2. Access during transportation to the launch pad and erection onto the Atlas 3. Access for checkout and achieving readiness prior to fairing installation 4. Access after fairing installation. State location, size of opening and inside reach required 5. Access during the final countdown, if any 6. AGE requirements for emergency removal
Umbilicals	1. Ground servicing umbilicals by function and location in excess of Atlas/Centaur baseline 2. Structural support requirements and retraction mechanisms 3. Installation (e.g., when and by whom supplied and installed)
Commodities required for both spacecraft, AGE and personnel	1. Gases, propellants, chilled water, and cryogenics 2. Source (e.g., spacecraft or launch vehicle) 3. Commodities for personnel (e.g., work areas, desks, phones)
Miscellaneous space	Spacecraft guidance alignment requirements

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Table B-7. AGE/facility requirements (electrical).

Type of Data	Scope of Data
Space vehicle electrical conductor data	Spacecraft system schematic showing all connectors required between spacecraft equipment and spacecraft terminal board position or receptacle pin assigned to each conductor, electrical characteristics of each connector including maximum end-to-end resistance, shielding, capacitance, etc. and spare conductors
Electrical power (AGE and facility)	1. Frequency, voltage, watts, tolerance, source 2. Isolation requirements 3. Identify if values are steady or peak loads 4. High-voltage transient susceptibility
RF transmission	1. Antenna requirements (e.g., function, location, physical characteristics, beam width and direction and line-of-sight) 2. Frequency and power transmission 3. Operation
Cabling	Any cabling, ducting, or conduits to be installed on the mobile service tower. Who will supply, install, check out, and remove
Monitors and control	1. Specify which signals from the spacecraft are to be monitored during readiness and countdown and the power source: spacecraft, Atlas, Centaur, or AGE 2. Transmission method (e.g., spacecraft TLM, LV TLM, landline, or launch vehicle readiness monitor) 3. Location of data evaluation center, evaluation responsibility measurement limits and go/no-go constraints. Identify where in the operational sequence measurements are to be monitored and evaluated. Specify frequency and duration of measurements 4. Video output characteristics of telepaks (if available) for closed-loop prelaunch checkout at the launch pad. Data to include location and type of interface connector(s) and characteristics of signal at source, including voltage level, output impedance, output current limitation, maximum frequency of data train, and output loading requirements

[illegible]

Table B-8 Test operations.

Type of Data	Scope of Data
* Hardware needs (including dates)	1. Electrical simulators
	2. Structural simulators
	* 3. Master drill gage
Interface test requirements	1. Structural test
	2. Fit test
	3. Compatibility testing of interfaces (functional)
	4. EMC demonstration
	5. LV/spacecraft RF interference test
	6. Environmental demonstration test
Launch operations	1. Detailed sequence and time span of all spacecraft-related launch site activities, including AGE installation, facility installation and activities, spacecraft testing, and spacecraft servicing
	2. Recycle requirements
	3. Restrictions to include launch site activity limitations, constraints on launch vehicle operations, security requirements and personnel access limitations, and safety precautions
	4. Special requirements such as handling of radioactive materials, security, and access control
	5. Support requirements to include personnel, communications, and data reduction
	6. Launch and flight requirements for real-time data readout, postflight data analysis, data distribution, postflight facilities

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Table B-9. Spacecraft design requirements.

Spacecraft design requirement	Comment
Mechanical	
1 Payload fairing envelope per Figures 4-3 and 4-4	
2 Payload adapter envelope per Figures 4-5 through 4-11	
3 Payload adapter interface per Figures 4-13 and 4-14	If P/L adapter is S/C-provided, interface can be a field joint at the equipment module ring or on the top of the spacer adapter (Type C)
Electrical	
4 Two or fewer separation commands (4.1.3)	
5 Sixteen or fewer control commands (28 V discretes or dry loop) (4.1.3)	
6 Instrumentation I/F: two or fewer inputs for S/C separation detection, four or fewer analog inputs for general use, ten or fewer command feedback discretes, two or fewer serial data interfaces for down-linking S/C data (4.1.3)	
7 Two umbilical connectors at S/C interface (Figure 4-19)	
Structure and loads	
8 Design load factors per Tables 3-2 and 3-3	
9 First lateral modes above 10 Hz and first axial mode above 15 Hz (3.2.1)	
10 S/C weight vs cg range per Figure 4-17	
11 Design factors of safety (FS) (see item 22)	

Design factors of safety (FS)

Item	Yield	Ultimate	Proof
Quasi-static loads			
- Primary and secondary structure verified by analysis and static test	1.10	1.25	1.1
- Primary and secondary structure verified by analysis only	1.25	2.00	--
Lines and fittings			
- Less than 0.762 inches	--	4.0	2.0
- 0.762 inches or greater	--	1.5	1.2
Pressure vessels	1.10	1.25	1.06
Reservoirs, actuation cylinders, valves, filters, switches	--	2.0	1.5

Spacecraft design requirement	Comment
Environment	
12 Quasi-sinusoidal vibration per Figure 3-10	
13 Random vibration levels per Figure 3-11	
14 Acoustic levels in the payload fairing per Figure 3-8	
15 Shock induced by payload fairing jettison and payload separation per Figure 3-12	
16 Payload compartment pressures and depressurization rates per Figures 3-15 through 3-18	
17 Gas velocity across S/C components < 7 ft/sec	
18 Electric fields per Figures 3-3 through 3-6	
19 S/C radiation per Figure 3-7 levels	
20 EM environment at launch range per TOR-0064 (4338-42)-1 Reissue B per 3.1.2.3	
Safety	
21 All S/C propellant fill and drain valves and all pressurant fill and vent valves readily accessible when S/C is fully assembled and serviced in launch configuration	For new S/C design, it is advisable to accommodate normal servicing/deservicing as well as potential emergency backout situations
22 Requirements in ESMCR range safety Regulation 127-1	
Miscellaneous	
23 See Atlas Launch Services Facilities Guide for spacecraft propellants and specifications available at CCAFS fuel storage depot	

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APPENDIX C • ANALYSES IN SUPPORT OF MISSION INTEGRATION

The following analyses are typically conducted in support of mission integration activities. The analyses generally are two-phased. The preliminary and final analyses are typically scheduled as shown in Figure 6-5 of this guide.

C.1 TRAJECTORY ANALYSIS

The Atlas trajectory from launch through spacecraft injection into the final mission orbit is designed to optimize the customer's stated mission objective. Two typical objectives are to maximize spacecraft weight into the desired final orbit or maximize on-orbit lifetime for a specified spacecraft weight. Along with the objective, mission requirements and constraints must be clearly established in order to create this optimum trajectory design. Typical spacecraft requirements include:

- Telemetry data requirements for specific events
- Thermal attitude control during coast phase
- Spacecraft pointing at separation
- Spacecraft roll rates at separation
- Acceptable solar exposure durations and attitudes
- Maximum acceptable aerodynamic heating rate at payload fairing jettison
- Maximum acceptable static acceleration level (occurs at booster package jettison)

In addition to the spacecraft requirements, all trajectory design analyses consider the range safety requirements and policies established by the 45th SPW.

C.2 GUIDANCE ANALYSIS

Analyses are performed to demonstrate that the guidance and navigation requirements are satisfied. Analyses include targeting, standard vehicle dispersions, extreme vehicle dispersions, and guidance accuracy. The targeting analysis verifies that the guidance program achieves all the mission requirements throughout the launch opportunities and across the daily launch windows. Standard vehicle dispersion analysis demonstrates that the guidance algorithms are insensitive to 3-sigma launch vehicle dispersions in that the guidance program compensates for these dispersions while minimizing orbit insertion errors. The extreme launch vehicle dispersions are abnormal dispersions (e.g., 10-sigma) and failure modes. These cases are selected to stress the guidance program and demonstrate that the guidance software capabilities far exceed the vehicle capabilities.

The guidance accuracy analysis combines vehicle dispersions and guidance hardware and software error models to evaluate total guidance system injection accuracy. Hardware errors model the off-nominal effects of the guidance system gyros and accelerometers. Software errors include computation errors and vehicle dispersion effects. These vehicle dispersions include independent vehicle and atmospheric dispersions that perturb Atlas and Centaur performance. This accuracy analysis will include the noise, and twist and sway effects on guidance system alignment during gyro compassing, as well as the covariance error analysis of the guidance hardware.

C.3 SPACECRAFT SEPARATION ANALYSIS

Six-degree-of-freedom simulation of the Centaur/spacecraft separation event will be performed using finalized spacecraft mass properties to verify that Centaur will not recontact the spacecraft following separation system release. This analysis will demonstrate clearances, preseparation rate, separation nutation, and momentum pointing using the minimum relative separation velocity of 1.0 ft/sec. This ensures that adequate separation distance will be achieved before initiating the collision and contamination avoidance maneuver (CCAM).

C.4 DYNAMIC COUPLED LOADS ANALYSIS

GDSS performs mission-peculiar dynamic coupled loads analysis to determine spacecraft loads, deflections, and accelerations during Atlas transient flight events.

To calculate spacecraft loads during flight, General Dynamics needs a dynamic model of the spacecraft and output transformation matrices (OTMs). The spacecraft dynamic model should consist of generalized mass and stiffness matrices along with a recommended modal damping schedule. The desired format is Craig-Brampton, constrained at the Centaur interface in terms of spacecraft modal coordinates and six (single-point) discrete Centaur interface degrees of freedom. The dynamic model should have an upper frequency cutoff of 50 to 60 Hz. The OTMs should be in the form that, when multiplied by the spacecraft modal acceleration or displacement time histories, they will recover the desired accelerations, displacements, or internal loads. One of the OTMs should contain data that will allow calculation of loss of clearance between the payload fairing and extreme points on the spacecraft.

The coupled loads analysis are performed in loads cycles phased to support spacecraft design and test schedules. Typically, the load cycles are preliminary (early version, design of all components may not be finalized), final (analytical model of completed design), and verification (test verified model).

General Dynamics calculates spacecraft loads for three events: 1) gust/flight winds (gusts and steady state winds at time of maximum dynamic pressure); 2) BECO/BPJ (Atlas booster engine cutoff/booster package jettison, which also envelopes liftoff); and 3) MECO (final upper stage main engine cutoff). Gust/flight wind is a low-frequency event (< 12 Hz) that produces maximum loss of clearance between the spacecraft and payload fairing and high loads near the base of the spacecraft primary structure. BECO/BPJ excites all frequencies (3 to 40 Hz) and produces the majority of the maximum loads throughout the spacecraft. MECO excites all frequencies and produces the highest tension (negative axial) loads and sometimes maximum loads on secondary structure.

Spacecraft data recovery is performed with contractor-provided output transformation matrices. Typically, the size of the OTMs are 200–400 rows for accelerations, 50–100 rows for displacements, and 300–800 rows for internal loads. The output can consist of maximum/minimum response listings, time history plots, shock spectrum plots, and modal participation of maximum responses. Maximum/minimum listings and time history plots of selected responses are the most commonly desired output. At a Technical Interchange Meeting early in the program, the output request can be finalized.

C.5 PAYLOAD FAIRING VENTING, JETTISON, AND LOSS OF CLEARANCE ANALYSES

Payload fairing venting analyses are performed for first-of-a-kind missions to evaluate and verify that spacecraft depressurization rate requirements are met. This analysis is based on the mission trajectory and spacecraft geometry and volume.

Verification of payload clearance during payload fairing jettison is performed as part of the loss of clearance. The effects of thermal preload, disconnect forces, shear pin forces, actuator forces, and dynamic response are all included in a fully three-dimensional nonlinear analysis.

C.6 TYPICAL THERMAL ANALYSIS

New mission requirements, including trajectories, timelines, and air conditioning, are examined to determine if these are compatible with the spacecraft.

Typical thermal analysis performed for each new payload include:

1. Pad air conditioning analysis to validate that all unique spacecraft requirements (velocity, temperature range) are met.
2. Computation of free molecular heating on spacecraft surfaces normal and parallel to the vehicle longitudinal axis for the worst-case depressed trajectory.
3. Integrated thermal analysis of the spacecraft and Centaur to demonstrate that the prelaunch, ascent, and preflight thermal environments are compatible with the spacecraft's requirements.

In addition to these specific tasks, General Dynamics supplies a nominal level of effort to ensure correct vehicle configuration, integration support, and launch activity support.

C.7 RF LINK AND COMPATIBILITY ANALYSES

GDSS analyzes the LV telemetry system, range safety command system, and C-band tracking system RF links from launch through spacecraft separation to determine adequate link margins.. Analysis will be performed for nominal, lofted, and depressed trajectories for the mission using antenna pattern data taken from range testing of one-fourth scale models of the Atlas.

A frequency compatibility analysis is performed to demonstrate transmitter/receiver compatibility between the Atlas and spacecraft and considers the LV C-band tracking system, telemetry transmitters and range safety receivers, and space vehicle transmitters and receivers. Worst-case intentional radiation impingement on the spacecraft (from the LV) and on the LV (from the SC) is addressed under the EMC analysis.

Prelaunch RF links to the spacecraft ground checkout facilities are analyzed for the spacecraft to determine adequate link margin. Typically, link margin analysis is conducted utilizing the standard LV-provided reradiating RF system (using antenna couplers and tower reradiating antennas).

C.8 ELECTROMAGNETIC COMPATIBILITY

The EMC plan defines how compatibility within Atlas vehicle and between Atlas and spacecraft and ground support equipment is achieved. It includes system descriptions and tailors EMC specification applications, covers electrical bonding/grounding/isolation, wire routing/shielding, interference emissions/susceptibility, and short-duration transients within the system and at system interfaces. The document defines EMI/EMC testing and analysis processes. Technical operating reports are used to document results of system EMC acceptance tests, the EMC test plans and procedures. Based on information contained in all of the foregoing, an EMC analysis is prepared to demonstrate that design requirements defined in the ICD are met.

ABBREVIATIONS, ACRONYMS, AND DEFINITIONS

A

AB	Airborne
A/C	Air Conditioning
ac	Alternating Current
AF	Air Force
AFR	Air Force Regulation
AFSCF	Air Force Satellite Control Facility
AGE	Aerospace Ground Equipment
AVE	Aerospace Vehicle Equipment

B

BECO	Booster Engine Cutoff (Atlas)
BPJ	Booster Package Jettison
BTU	British Thermal Units

C

C	Centigrade
CCAFS	Cape Canaveral Air Force Station
CCAM	Collision and Contamination Avoidance Maneuver
CCLS	Computer-Controlled Launch Set
CCTV	Closed-Circuit Television
CDR	Critical Design Review
CERT	Composite Electrical Readiness Test
CG	Center of Gravity

D

DAS	Data Acquisition System
dB	Decibel
dc	Direct Current
deg	Degree
DoD	Department of Defense
DSN	Deep Space Network

E

EAGE	Electrical Aerospace Ground Equipment
ECS	Environmental Control System
EED	Electro-Explosive Device
EM	Electromagnetic
EMC	Electromagnetic Compatibility
EMI	Electromagnetic Interference
ERB	Engineering Review Board

F

F	Farenheit
FED	Flight Event Demonstration
FPR	Flight Performance Reserve
fps	Feet per Second
ft	Feet
FTS	Flight Termination System

G

G	Gravity
GN_2	Gaseous Nitrogen
GDCLS	General Dynamics Commercial Launch Services
GDSS	General Dynamics Space Systems
GG	Gas Generator
GHe	Gaseous Helium
GN&C	Guidance, Navigation, and Control
GSE	Ground Support Equipment
GSO	Geosynchronous Stationary Orbit
GSTDN	Goddard Space Tracking and Data Acquisition Network
GTO	Geosynchronous Transfer Orbit

H

HPF	Hazardous Processing Facility
Hz	Hertz
hr	Hour

I

IB	Intermediate Bulkhead
ICD	Interface Control Document
IIP	Instantaneous Impact Point
IMP	Integration Management Panel
INCL	Inclination
INU	Inertial Navigation Unit
IPJ	Insulation Panel Jettison
IRD	Interface Requirements Document
ISD	Interface Scheduling Document
ITP	Integrated Test Plan

J

JPL	Jet Propulsion Laboratory

K

kg	Kilograms
KM	Kearny Mesa
km	Kilometers
kN	Kilonewtons
KSC	Kennedy Space Center

L

lb	Pound
LH_2	Liquid Hydrogen
LN_2	Liquid Nitrogen
LO_2	Liquid Oxygen
L/O	Liftoff
LEO	Low Earth Orbit
LHe	Liquid Helium
LM	Lumens
LO/FP	Launch Operations and Flight Plan
LPF	Large Payload Fairing
LV	Launch Vehicle
LVC	Launch Vehicle Contingency

M

mV/m	Millivolts per Meter
MAB	Margin Allocation Board
MDU	Master Data Unit
MEP	Management Effectiveness Program
MECO	Main Engine Cutoff (Centaur)
MES	Main Engine Start (Centaur)
Mfg	Manufacturing
MHz	Megahertz
mm	Millimeters
M	Meter
MPF	Medium Payload Fairing
MRR	Mission Readiness Review
MSPSP	Missile Systems Prelaunch Safety Package
MST	Mobile Service Tower

N	
N	Newtons
NASA	National Aeronautics and Space Administration
NASCOM	NASA Communications Network
nmi	Nautical Mile
O	
Ω	Ohm
Oct	Octave
OIS	Orbit Insertion Stage
OR	Operations Requirements
ORD	Orbital Requirements Document
OTM	Output Transformation Matrix
P	
Pa	Pascal
P&W	Pratt & Whitney
PCM	Pulse Code Modulation
PCU	Power Control Unit
PDR	Preliminary Design Review
PFJ	Payload Fairing Jettison
PI	Program Introduction
Pk	Peak
PLF	Payload Fairing
PPF	Payload Processing Facility
PRD	Program Requirements Document
PSD	Power Spectral Density
psi	Pounds per Square Inch
PSW	Payload Systems Weight
PVA	Perigee Velocity Augmentation
PYC	Pyrotechnic Control Unit
Q	
QA	Quality Assurance
R	
RAAN	Right Ascension of Ascending Node
R&D	Research and Development
RCS	Reaction Control System
RCU	Remote Control Unit
RDU	Remote Data Unit
RF	Radio Frequency
rms	Root Mean Square
rpm	Revolutions per Minute
RSC	Range Safety Command
rss	Root Sum Square
RTS	Remote Tracking System
S	
S&A	Safe and Arm
SC	Spacecraft
SCU	Sequence Control Unit
sec	Seconds
SECO	Sustainer Engine Cutoff (Atlas)
SIL	Systems Integration Laboratory
SIRD	Support Instrumentation Requirements Document
SMFLT	Simulated Flight (Launch Vehicle)
SSE	System Security Engineering
SSP	System Security Plan
STA	Station
STC	Satellite Test Center
T	
TCD	Terminal Countdown Demonstration
TDRSS	Tracking and Data Relay Satellite System
TIM	Technical Interchange Meeting
TLM	Telemetry
U	
UPS	Uninterruptable Power System
USAF	United States Air Force
UT	Umbilical Tower
V	
V	Volt
VAT	Vertical Assembly Tower
VDC	Volts Direct Current
W	
W	Watt
WDR	Wet Dress Rehearsal