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User's Handbook

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Revision A

TITAN IV

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Foreword

This document is submitted in accordance with the requirements of Contract F04701-85-C-0019, Contract Data Requirements List Number 011A2, for an integrated edition of the Titan IV User's Handbook addressing the following configurations: Centaur Upper Stage, SS-ELV-401; Inertial Upper Stage, SS-ELV-402; No Upper Stage, SS-ELV-403; and No Upper Stage, SS-ELV-405. The SS-ELV-404 NUS is not addressed in this document

A questionnaire for potential spacecraft users of the Titan IV launch system is included as Appendix A. The questions are representative of the type of information needed to determine the degree of compatibility between a specific spacecraft and the Titan IV launch vehicle system as well as the data necessary to provide the Titan IV/spacecraft integration services.

The questionnaire is to be completed by the Launch System Integration Contractor (LSIC) and validated by the Space Division Spacecraft Program Office. The questionnaire is then to be transmitted to the Titan System Program Office, Program Manager for Integration (MEI).

This Address is: Department of the Air Force
Headquarters Space Division, AFSC
Los Angeles Air Force Station
P.O. Box 92960
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Los Angeles, CA 90009
Attn: MEI

For Assistance in completing the questionnaire or if more detailed information is needed for a specific application, please contact:

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Thank you for your interest in the Titan IV program.

Introduction

The purpose of the Titan IV User's Handbook is to present to potential Spacecraft Users an overview of the Titan IV Launch Vehicle configurations, performance and the various Spacecraft interfaces. Information is also provided necessary for the Spacecraft User to gain perspective into his roles, direction and information sources for the accomplishment of payload integration and launch implementation.

This document presents the Spacecraft /Titan IV Launch Vehicle integration elements, the Titan IV Launch Vehicle major configuration elements and performance, the Upper Stage and No Upper Stage Spacecraft interfaces to Titan IV, the operational environments, launch facilities support and launch operations.

Also presented is a history of the Titan IV family of launch vehicles; and the following appendices: Spacecraft User Questionnaire, Independent Verification and Validation, Spacecraft Chargeable Weight Items, EMC/EMI Interface Guidelines for Titan IV Payloads, and a Glossary.

The Titan IV Launch Vehicle configurations presented are the Centaur Upper Stage (SS-ELV-401) and Inertial Upper Stage (SS-ELV-402) both currently to be launched from Cape Canaveral Air Force Station, Florida; the No Upper Stage (SS-ELV-403) scheduled to be launched from Vandenberg Air Force Station, California; and, the No Upper Stage (SS-ELV-405) scheduled to be launched from Cape Canaveral Air Force Station, Florida.

This document was prepared as the first update to the Titan IV User's Handbook. It describes the essential programmatic and technical data necessary to provide a preliminary compatibility assessment of the Titan IV system to prospective users. This revision supercedes the December 1988 issue.

Revisions to the User's Handbook are significant. Updated programmatic information, revised environments and inclusion of preliminary SRMU data were incorporated to insure maximum utility for the user community. Updates will continue to be provided periodically as required.

Note: The terms spacecraft, payload and satellite vehicle as used in this document are essentially synonymous. The context in which the term is used often dictates the suitable terminology.

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Autogenous Pressurization Flow	Titan Stage I and II gas from engine turbine inlet (fuel) and from heat exchanger in turbine exhaust stack (oxidizer) to the main propellant tanks to provide pressurization gas in the ullage volume
BL	Butt Line – lateral reference datum used with mass property location data. Butt line 0 is coincident with the vehicle yaw axis. Measurements in inches are positive (+) from BL = 0 toward SRM 2 (right wing) and negative (-) from BL = 0 toward SRM 1 (left wing) when viewed looking forward and downrange
Booster Vehicle	Contains two Stage 0 SRMs, Titan Stages I and II, and Centaur
BOW	Total Centaur Burn-Out Weight (after main engine decay) including spacecraft weight and propellant excess.
Cant Angle	Angle between engine nozzle centerline and vehicle centerline with zero nozzle deflection
CSIC	Common Standard Inlet Conditions – set of reference pressure and temperature inlet conditions for liquid rocket engine performance comparisons
FPR	Flight Performance Reserve. Centaur main impulse propellants held in reserve to ensure the achievement of mission injection conditions in the event of 3-sigma low performance
Frangible	Breakable
Fuel	Main engine fuel of Titan Stages I and II, a blend of hydrazine and Unsymmetrical Dimethylhydrazine (UDMH)
Fuel Bias	Extra amount of fuel loaded on Titan Stages I or II in order to minimize the second moment of the outage distribution function about zero outage
EPDM	EPDM rubber (Ethylene Propylene Diene Monomer) insulation used in IUS SRMs
GN_2	Gaseous Nitrogen – used as a pressurant in various liquid propellant tanks
GSO	Geo-Synchronous Orbit – circuit equatorial orbit having a period of 23.934 hours
HTPB	Hydroxl-Terminated Poly-Butadiene – Propellant type used on IUS SRMs

Inert Weight	Ablatives – usually refer to internal coating of the engine nozzles which ablate during engine operation providing nozzle temperature control; also Stage 0 and IUS internal insulation expended (not additive to engine thrust)
IUS	Inertial Upper Stage – BAC two-stage vehicle consisting of two SRM steps and associated guidance and control systems required to perform orbital transfers
LKA	Antenna look angle between the negative vehicle roll axis and the slant range vector
LKB	Antenna look angle between the projection of the slant range vector in the vehicle roll plane and the positive vehicle yaw axis, positive yaw axis positive counter-clockwise when viewing the vehicle from the rear as measured from the positive yaw axis
MECO	Main Engine Cutoff
Minimum Ullage	The smallest volume of gas in the propellant tanks which is large enough to: (A) Provide required Net Positive Suction Head (NPSH) at the propellant pump and maintain structural integrity of the propellant tanks during an engine start transient (B) Provide required hold time capability by allowing for thermal expansion of the propellant while maintaining the ullage gas pressure within required limits
Mission-Peculiar Interfaces	All Program unique interfaces
MPJETW	Centaur mission-peculiar jettison weight, excluding PE. the minimum Centaur jettison weight for the benchmark conditions
MR	Engine Mixture Ratio – mass ratio at which oxidizer is burned proportional to fuel in Titan Steps 1 and 2, and ratio of LO_2/LH_2 burned during Centaur flight
Outage	Nonusable propellant remaining in one tank after all the usable propellant in the other tank has been consumed
Oxidizer	Main engine oxidizer in Titan Stages I and II, and Stage 0 TVC propellant, nitrogen tetroxide (N_2O_4)

Pathfinder	A non-recurring activity for verification of compatibility of the Titan IV Booster Vehicle elements and the Booster Vehicle and PLF with the Ground Facilities and AGE. Pathfinder is a prerequisite to processing and Launch Site modification buyoff and initial site validation of readiness to receive and support a Titan IV configuration consisting of a Booster Vehicle and PLF.
Payload	P/L – Same as Spacecraft; however, Spacecraft is preferred
Payload Fairing	PLF – The trisector payload fairing used for ascent environmental protection of the SV and US
PE	Propellant Excess. Total Centaur burnable propellants after main engine decay
RN	Roll Nozzle – gas generator exhaust gas is ducted overboard to provide Titan Stage II roll control through deflection of the roll nozzle
Satellite Vehicle	SV – The Spacecraft plus its interface adapter to mate with the Launch Vehicle
Satellite Vehicle System	SVS – the SV and its Aerospace Ground Equipment (AGE)
SECO	Sustainer Engine Cut-Off – shutdown signal for Titan Step 2 which is sometimes referred to as the sustainer engine
Spacecraft	SC – That portion which stays on-orbit to perform a mission. Also known as Payload (P/L)
SRM	Solid Rocket Motor – Titan Stage 0 – two identical 120-in. dia motors mounted 180 deg apart on the core vehicle in the yaw plane. Each motor consists of a nose cone and core attach ring; a forward closure; seven 10-ft motor segments; an aft closure and a nozzle with a fixed 6 deg outboard cant. An aft skirt transmits thrust loads to the core vehicle. A liquid injected TVC system, staging rockets and instrumentation are also included. The PBAN propellant formulation is 84 percent solids, including 16 percent powdered aluminum fuel, ammonium perchlorate oxidizer and synthetic rubber binder. IUS Stages I and II – two filament-wound Kevlar-49 motors insulated with EPDM rubber using propellant of tubular perforation grain design that can be off-loaded by machining. The propellant formulation for both motors is an HTPB with 86 percent solids, including 18 percent aluminum. Both IUS stages utilize partially submerged nozzles

SRMU	Solid Rocket Motor Upgrade – Titan Stage 0 – two identical 126-in. dia motors mounted 180 deg apart on the core vehicle in the yaw plane. Each motor consists of a forward segment, core attach ring, one center segment and an aft segment with a gimballed nozzle assembly. An aft skirt transmits thrust loads to the core vehicle. A gas generator/turbo hydraulic TVC system, staging rockets and instrumentation are also included. The HTPB propellant formulation is 88 percent solids, including 19 percent powdered aluminum fuel with an ammonium perchlorate oxidizer and polymeric binder.
SRM WAT	SRM Web Action Time for Titan Stage 0 – The start of the web action time is that point in the forward-end chamber pressure time history at which the pressure has reached 75 percent of maximum ignition pressure. The end of Web Action Time occurs when the time integral of chamber-pressure achieves 96.3 percent of the value occurring at 50 psia during tail-off
Stage or Stg	The segment of the vehicle consisting of the operating step, remaining steps, and payload
Step	A contained segment of the vehicle bounded by staging separation points
TCA	Thrust Chamber Assembly – refers to liquid rocket engines to Titan Stages I and II
TCPS	Thrust Chamber Pressure Switch – pressure switch on Titan Stages I and II Liquid Rocket Engines activated by decaying thrust chamber pressure, thereby sending a shutdown signal to the engines
TFS	Temperature measured at the fuel pump inlet of Titan Stages I and II Engines
TPS	Transient Power Switch
Trailblazer	A non-recurring activity (for each different Satellite Vehicle) that evaluates form and fit relative to Satellite Vehicle interfaces with a Titan IV Upper Stage, No Upper Stage configuration, Payload Fairing, Ground Facilities, and AGE. Trailblazer provides for facility design verification, procedures, development/validation, and training. A space Vehicle simulator is supplied by SVC/LSIC.

TVC	Thrust Vector Control – Titan Stage 0 – on command from the guidance and control system, pressurized liquid nitrogen tetroixide is injected into the nozzle cone through 24 injector valves per SRM to deflect the direction of the rocket exhaust gases Titan Stages I and II – hydraulic actuators pivot liquid propellant engine nozzles
Ullage	The gas-filled volume above the liquid propellant in a propellant tank
VS	Vehicle Station – Vertical position in inches measured to the nearest one thousandth of an inch. The various key vertical stations are indicated on the vehicle outboard profile drawings
WL	Waterline – vertical reference datum used with mass property location data. Water line 0 is 60 inches from the vehicle pitch axis (+Y) along the target-axis or yaw axis (+Z).It is perpendicular to the target-axis and parallel to the pitch-axis.Measurements in inches are negative (–) from WL = 0 toward target, and positive (+) from WL = 0 away from target

A	Ampere(s), Analog
A/C	Air Conditioning
ac	Alternating Current
ACCEL	Acceleration
ACRBC	Acceptance, Checkout, Retest and Backout Criteria
ACS	Attitude Control System
A/CST	Acceptance Combined Systems Test
ACT	Activation
ADC	Analog Digital Convertor
ADS	Advanced Decom System
AFB	Air Force Base
AFLC	Air Force Launch Controller
AFNC	Air Force Network Controller
AFOSH	Air Force Occupational Safety and Health
AFTC	Air Force Test Controller
AFTD	Air Force Test Director
AFSC	Air Force Systems Command
AFSCF	Air Force Satellite Control Facility
AFWTR	Air Force Western Test Range
AGC	Automatic Gain Control
AGE	Aerospace Ground Equipment
AH	Ampere Hour(s)
AHI	Aerodynamic Heating Indicator
AKM	Apogee Kick Motor
ALT	Altitude
AM	Amplitude Modulation
AMP, amp	Ampere(s)
AMPS	Amplifiers
ANT	Antenna
APS	Accessory Power System, Airborne Power Supply
ARIA	Advanced Range Instrumentation Aircraft
ARG	Argument
A&S	Arm and Safe
A/S	Arm/Safe

ASC	Aerospace Corporation
ASSY	Assembly
ASTG	Aerospace Test Group
ATC	Aerojet TechSystems Company
ATG	Aerospace Test Group
AV	Avionics
AVG	Average
AZ	Azimuth
BA	Boeing Aerospace
BA	Breathing Air
BAC	Boeing Aerospace Company
BATT,BAT	Battery
B/B	Broadband
BCC	Booster Countdown Conductor
BL	Butt Line
BLCU	Battery Liquid Cooling Unit
Bldg	Building
BLK	Block
bps	Bit(s) per Second
BTU	British Thermal Unit(s)
BU, B/U	Backup
BV	Boost Vehicle
BVCC	Booster Vehicle Countdown Controller
CA	California
CAM	Collision Avoidance Maneuver
CC	Command Control
CC,C/C	Control Center
CCAFS	Cape Canaveral Air Force Station
CCAM	Contamination and Collision Avoidance Maneuver
CCB	Change Control Board
CCPS	Command Control Power System
CCU	Central Control Unit
CCW	Counter Clockwise

C/D	Countdown
CDR	Critical Design Review
CDRL	Contract Data Requirements List
CE	Cincinnati Electronics
CEC	Cincinnati Electronics Corporation
CELV	Consolidated Expendable Launch Vehicle
CERT	Certification
CG, C.G., c.g.	Center of Gravity
CH, CHAN	Channel
CHG	Charge
Chkout	Checkout
CI	Configuration Item
CIRC	Circulator
C/L, C_L	Centerline
Clear	Clearance
CM	Centimeter(s)
CMA	Contractor Maintenance Area
CMCC	Command Management Control Center
CMD	Command
Comb	Combined
COMP, COMPT	Compartment
C/O	Checkout
COND	Conductor
CONN	Connector
Corp	Corporation
COS	Checkout Station
CPDP	Computer Program Development Plan
CPU	Computer Processing Unit
CR	Capability Ratio
CRD	Command Receiver/Decoder
CRT	Cathode Ray Tube
CRU	Converter Regulator Unit
CSD	Chemical Systems Division
CST	Combined Systems Tests
CSTC	Consolidated Space Test Center

CSTSS	Combined System Test Simulator Set
CTO	Canaveral Test Order
CTR	Canaveral Test Report
CU	Computer Unit
CU	Convertor Unit
CW	Clockwise
CW	Continuous Wave
CW	Coolant Water
CYL	Cylinder
DAS	Data Acquisition System
dB	Decibel(s)
dBμA	Decibel(s), referenced to one microamp
dBμA/MHz	Decibel(s), referenced to one microamp per Megahertz
dBμA/m/Mhz	Decibel(s), referenced to one microamp per meter per Megahertz
dBW	Decibel(s), referenced to one Watt
DC, dc	Direct Current
DC-DC	Direct Current to Direct Current
DCU	Digital Computer Unit
DDIF	Digital Data Interface
°F	Degree(s) Fahrenheit
DEG, Deg, deg	Degree(s)
DEST, DESTR	Destruct
DETN	Detonation
DEVEL	Development
DEMODS	Demodulators
DFCS	Digital Flight Control System
DIA, dia	Diameter
DMS	Data Management System
DN	Down
DOD	Department of Defense
DOF	Degrees of Freedom
DOL	Day of Launch
DOP	Detail Operating Procedure

DSCS	Defense Systems Communication Satellite
DSO	Delco Systems Operation (Division of DELCO Corp)
DSP	Defense Satellite Program
DTD	Dated
DTS	Data Transmission System
E	Emissivity
E	Modulus of Elasticity
EA	Each
ECC	Eccentricity
ECP	Engineering Change Proposal
ECPTR	Encryptor
ECS	Environmental Control System
EEC	Extendible Exit Cone
EED	Electro-Explosive Device
EEE	Electrical, Electronic and Electromechanical
EFP	Explosively Formed Projectile
EIVE	Electrical Interface Verification Equipment
EL	Elevation
ELEC	Electric (trical, tricity, tronic)
ELS	Eastern Launch Site
EMC	Electromagnetic Compatibility
EMI	Electromagnetic Interference
EMISM	EMI Safety Margin
EMV	Electromechanical Valve
ENGR	Engineer
EQUIP	Equipment
ES	Environmental Shelter
ESMC	Eastern Space and Missile Center
ESS	Equipment Support Section
ETA	Explosive Transfer Assembly
ETR	Eastern Test Range
EVAL	Evaluation
EVCF	Eastern Vehicle Checkout Facility
EXT	Exterior

FBR	Forward Bearing Reactor
fc	Center Frequency
FCP	Flight Control Processor
FCR	Facility
FCS	Flight Control System
FECP	Facility Engineering Change Proposal
FFT	Fast Fourier Transform
Fl	Frequency, Lower Band Edge
FIB	Functional Instrumentation Box
FIS	Functional Interface Specification
FL	Florida
FLT, Flt	Flight
FM	Frequency Modulated
FMH	Free Molecular Heating
fpm	Foot (feet) per Minute
FPS, fps	Foot (feet) per Second
FREQ	Frequency
FT, ft	Foot (feet)
ft^2	Square Foot (feet)
ft/sec	Foot (feet) per Second
FTS	Flight Termination System
FTS MOD	Flight Termination System Module
FTSS	Flight Test Support System
fu	Frequency, Upper Band Edge
FWD	Forward
FWG	Facility Working Group
G, g	Gravitational acceleration
GAL	Gallon(s)
GCMG	Guidance Control Monitor Group
GCS	Ground Computer System
GD	General Dynamics
GDSS	General Dynamics Space Systems
GDSSD	General Dynamics Space Systems Division
GEN	Generator

GEO	Geostationary Earth Orbit
GFE	Government Furnished Equipment
GFP	Government Furnished Property
GH, GH_2	Gaseous Hydrogen
GHe	Gaseous Helium
GHz	Gigahertz
GIE	Ground Instrumentation Equipment
G&N	Guidance and Navigation
GN	Gaseous Nitrogen
GN_2	Gaseous Nitrogen
GN&C	Guidance, Navigation, and Control
GND, Gnd	Ground
GOAS	Guidance Optical Alignment System
GOWG	Ground Operations Working Group
GPIB	General Purpose Instrumentation Bus
GPS	Global Positioning Satellite
GPS	Ground Power System
GSE	Ground Support Equipment
GSO	Geosynchronous Orbit
GSP	Gyro Stabilized Platform
GTO	Geosynchronous Transfer Orbit
GTS	Ground Telemetry System
Gyro	Gyroscope
ha	Height of Apogee
HAT	Hardware Acceptance Team
HDQS	Headquarters
HDW	Hardware
He	Helium
HEFU	High Energy Firing Unit
HEPA	High Efficiency Particulate Air
HF	High Frequency
HI	Hercules Incorporated
HORIZ	Horizontal
HP	Hard Point

HP	Hewlett-Packard
hp	Height of Perigee
hr	Hour(s)
HTBP	Hydroxyl-Terminated Poly-Butadiene
HTF	Horizontal Test Facility
HTRS	Heaters
H/W	Hardware
Hz	Hertz
ICD	Interface Control Document
ICBM	Intercontinental Ballistic Missile
ICN	Interface Change Notice
ICNP	Interface Change Notice Proposal
ICWG	Interface Control Working Group
I.D.	Inside Diameter
I/F, I.F	Interface
IGN	Ignition
IGPS	Inertial Guidance Power System
IGS	Inertial Guidance System
IIP	Instantaneous Impact Point
IMG	Inertial Measuring Group
IMU	Inertial Measurement Unit
IN., in.	Inch(es)
IN.^{2}, in.^{2}	Square Inch(es)
INC	Inclination
INIT	Initiator
INST, INSTL	Installation
INST, INSTR	Instrumentation
INSTRUMT	Instrument
INT	Interior
INU	Inertial Navigation Unit
INV, Inv	Inverter
I/O	Input/Output
IPS	Interface Problem Summary
IPS	Instrumentation Power System

IRR	Independent Readiness Review
IRU	Inertial Reference Unit
ISDS	Inadvertent Separation Destruct System
ISL	Inside Skin Line
ISP	Specific Impulse
IST	Integrated System Test
ITL	Integrate-Transfer – Launch
IUS	Inertial Upper Stage
IVCR	Interface Verification Certification Report
IVE	Interface Verification Equipment
IV&V	Independent Verification and Validation
JB, J-box	Junction box
JUNC	Junction
K	Kilo
kbps	Kilobit(s) per Second
kHz	Kilohertz
KIPS	KiloPounds
KSC	Kennedy Space Center
Ksps	Kilo-Symbol(s) per Second
lb	Pound(s)
LBF, lbf	Pound(s) Force
LBM, lbm	Pound(s) Mass
LC	Launch Complex, Launch Controller
LCA	Launch Control Assembly
LCC	Launch Control Center
LCD	Launch Constraints Document
LEA	Linear Explosive Device
LEO	Low-Earth Orbit
LF	Low Frequency
LH_2	Liquid Hydrogen
LHe	Liquid Helium
LLSSS	Low Level Shutdown Sensor System

LMCC	Launch Management Control Center
LN_2	Liquid Nitrogen
LOB	Launch Operations Building
LO_2	Liquid Oxygen
LOC	Location
LOCC	Launch Operations Control Center
LOE	Level of Effort
LOIS	Lift Off Instrumentation System
LOX	Liquid Oxygen
LP	Liquid Propane
LPG	Liquid Petroleum Gas
LRE	Liquid Rocket Engine
LRR	Launch Readiness Review
LRU	Line Replacement Unit
LSB	Launch and Service Building
LSB	Launch Support Building
LSIC	Launch Systems Integration Contractor
LSMC	Lockheed Space & Missile Center
LTD	Launch Test Directive
LTWG	Launch Test Working Group
LV	Launch Vehicle
LVDT	Linear Variable Displacement Transformer
LVIC	Launch Vehicle Integrating Contractor
LVS	Launch Vehicle System
m	Meter(s)
M	Mega
MARS	Martin Automatic Reporting System
MAX	Maximum
MCC	Motor Case Cutter
MDSSC	McDonnell Douglas Space Systems Company
MDU	Master Data Unit
ME	Main Engine
MEAS	Measurement
MECH	Mechanical

MECO	Main Engine Cutoff
MEG	Megabit(s)
MES	Main Engine Start
MFG	Manufacturing
MGC	Missile Guidance Computer
Mgt, MGMT	Management
MHz	Megahertz
MILA	Merrit Island Launch Area
MILS	Missile Impact Location System
MIN	Minute(s)
MIN, min	Minimum
MIS	Motor Inert Storage
MIVE	Mechanical Interface Verification Equipment
MLI	Multi Layer Insulation
mm	Millimeter(s)
MMH	Monomethyl Hydrazine
MMC	Martin Marietta Corporation
MMAG	Martin Marietta Astronautics Group
MMCO	Martin Marietta Canaveral Operations
Mods	Modifications
MOD KIT	Modification Kit
MOI	Moment of Inertia
MOL	Manned Orbital Laboratory
MPG	Multiple Point Ground
MSBLS	Microwave Scanning Beam Landing System
msec	Millisecond(s)
MST	Mobile Service Tower
MVB	Main Vehicle Battery
MUXD	Multiplexer-Demultiplexer
MWG	Management Working Group
MΩ	Milliohm(s)
μS	Microsecond(s)
N/A	Not Applicable
NASA	National Astronautics and Space Administration

NB	Narrow Band
N_2H_2	Monopropellant Hydrazine
N_2H_4	Hydrazine
NMI, nmi	Nautical Mile(s)
No.	Number
N_2O_4	Nitrogen Tetroxide
NPC	Nitrogen Pressure Controller
NRZ	Non-Return to Zero
NRZ-M	Non-Return to Zero Mark
NSI	NASA Standard Initiator
NUS	No Upper Stage
O&C	Operation and Checkout
O.D.	Outside Diameter
OD	Operations Directive
OPS, Ops	Operations
OR	Operations Requirements
ORD, Ord	Ordnance
ORD	Operational Requirements Document
OSC	Oscillator
OSL	Outside Skin Line
OSS	Overpressure Suppression System
OTA	Ordnance Transfer Assembly
OX, Ox	Oxidizer
OXID	Oxidizer
PACE	Programmable Aerospace Control Equipment
PAFB	Patrick Air Force Base
PAM	Payload Assist Module
PAM	Pulse-Amplitude Modulation
PBAN	Poly-Butadiene Acrylic Nitrile
PCB	Printed Circuit Board
PCM	Pulse Code Modulation
PCU	Power Control Unit
PDR	Preliminary Design Review

PDU	Power Distribution Unit
PE	Project Engineer
PEM	Prototype Equipment Modification
PEO	Program Executive Officer
PHS	Procedure History Sheet
P/L	Payload
PL	Place
PLB	Payload Battery
PLCS	Places
PLF	Payload Fairing
PLFPF	Payload Fairing Preparation Facility
PLTD	Payload Test Director
PM	Post Modulation
POI	Park Orbit Insertion/Injection
PPF	Payload Processing Facility
ppm	Part(s) per Million
PPS	Pulse(s) per Second
PRD	Program Requirements Document
PREPS	Preparations
PRO	Proposed
PROP	Propellant
PSC	Payload Support Contractor
PSE	Payload Support Equipment
PSE	Peculiar Support Equipment
PSF, psf	Pound(s) per Square Foot
psi/sec	Pound(s) per Square Inch per Second
PSIA	Pound(s) per Square Inch, Absolute
PSK	Phase Shift Keying
PSP	Program Support Plans
PSSS	Payload Simulator Support Structure
PSU	Propellant Servicing Unit
PSU	Pyro Switching Unit
PTU	Power Transfer Unit
PU	Propellant Utilization
PWG	Payload Working Group

PWR, Pwr	Power
PYC	Pyrotechnic Control Units
Q	Dynamic Pressure; Amplification Factor
Quad	Quadrant
R	Radius
R-Count	Readiness Count
RAAN	Right Ascension of the Ascending Node
RAM	Random Access Memory
RCS	Reaction Control System
RCV	Receiver
Rec	Receive
REC	Recorder
REC	Recurring
REF	Reference
RDU	Remote Data Unit
REM	Rocket Engine Module
Req'ts	Requirements
RF	Radio Frequency
R/G	Rate Gyro
RH	Relative Humidity
RIMU	Redundant Inertial Measurement Unit
RIS	Receiving Inspection and Storage
RM	Room
RMIS	Remote Multiplexed Instrumentation System
RMU	Remote Multiplexer Unit
RP	Resistor Rack
RRC	Rocket Research Corporation
RS	Range Safety
RSO	Range Safety Officer
RTN	Return
RTS	Remote Tracking Station
RV	Re-entry Vehicle

σ	Sigma
S&A	Safe and Arm
SAB	Satellite Assembly Building
SAGE	Strategic Air-Ground Equipment
SAO-67	Smithsonian Astrophysical Observatory Special Report 264 29 Dec 1967
SATCOM	Satellite Communications
SBX	S-Band Transmitter
S/C SC	Spacecraft
SCC	Satellite Countdown Controller
SCI	SCI Technologies Incorporated
SCSI	Small Computer Standard Interface
SCU	Signal Conditioning Unit
SD	Space Division
S/D	Shutdown
SDRC	Structural Dynamics Research Corporation
SDRL	Subcontractor Data Requirements List
SEC, sec	Second(s)
SECO	Sustaining Engine Cutoff
SEP, Sep	Separation
SER	Service
SEU	System Electronics Unit
SFC	Squib Fire Circuit
SFC/RP	Squib Fire Circuit/Resistor Pack
SGLS	Space Ground Link System
SID	Standard Interface Document
SIG	Signal
SIMS	Simulators
SIU	Servo Inverter Unit
SKD	Skip on Discrete
SLC	Space Launch Complex
SMAB	Solid Motor Assembly Building
SMARF	Solid Motor Assembly and Readiness Facility
SMILS	Sonobuoy Missile Impact Location System
SOP	Standard Operating Procedures

SOW	Statement of Work
SPG	Single Point Ground
SPLTR	Splitter
SPS	Sample(s) per Second
SPO	System Program Office
SPW	South Park West
S/R	Series Regulator
SRM	Solid Rocket Motor
SRMU	Solid Rocket Motor Upgrade
S/S	Subsystem
SSD	Space Systems Division
STA	Service Task Authorization
STA	Station
STE	Special Test Equipment
STD	Standard
STG, Stg	Stage
STA	Service Task Authorization
STO	System Test Objectives
STS	Space Transportation System
Subassy	Subassembly
SV	Space Vehicle
S/V	Satellite Vehicle
S/VAS	Satellite Vehicle Avionics Simulator
S/V	Satellite Vehicle
S/VC	Satellite Vehicle Contractor
SVTC	Spacecraft Vehicle Test Conductor
SW	Switch
S/W	Software
SWG	Safety Working Group
SYNC	Synchronizer
T34D	Titan 34D
TACAN	Tactical Air Navigation
TAG	Test(ed) and Guaranteed(d)
TBD	To Be Determined

TC	Test Conductor
T/C	Titan Centaur
TCA	Test Code Article
TCPS	Thrust Chamber Pressure Switch
TDR	Transient Data Recorder
TDSP	Twisted Double Shielded Pair
TECH	Technical
tf	Fall Time
TF&S	Tracking and Flight Safety
TGSF	Test Group Support Facility
TIM	Technical Interchange Meeting
TIPS	Telemetry Integrating Processing System
TIU	Telemetry Interface Unit
TIU	Titan (Ordnance) Interface Unit
TIV	Titan IV
TKS	Tanks
TLM	Telemetry
TLM/CMD	Telemetry/Command
TM	Telemetry
TP	Twisted Pair
TP	Technical Power
TPE	Telemetry Processing Equipment
TPS	Transient Power Supply
tr	Rise Time
Trans	Transport
TSE	Test Support Equipment
TSP	Twisted Shielded Pair
TT&C	Telemetry, Tracking and Command
TVC	Thrust Vector Control
TYP	Typical
TW	Twisted
TWS	Twisted and Single Shielded
TX	Transmitter
UDMH	Unsymmetrical Dimethylhydrazine

UDS	Universal Document System
UES	Universal Environmental Shelter
UFS	Upper Flight Section
UHF	Ultra-High-Frequency
UMB, Umb	Umbilical
UP	Utility Power
UPI	Unified Payload Integration
UPS	Uninterruptible Power Supply
US	Upper Stage
U.S.	United States
USAF	United States Air Force
UT	Umbilical Tower
UTC	United Technologies Corporation
UTIL	Utility
UT/CSD	United Technologies/Chemical Systems Division
V	Volt(s)
V_I	Inertial Velocity
VA	Volt-Ampere(s)
Vac	Volt(s) Alternating Current
VAB	Vehicle Assembly Building
VAFB	Vandenberg Air Force Base
VCA	Vehicle Checkout Assembly
VCO	Voltage Controlled Oscillator
VDC, Vdc	Volt(s) Direct Current
VDU	Voltage Distribution Units
VEH	Vehicle
VIB	Vertical Integration Building
VLC	Verification Loads Cycle
V/m	Volt(s) per Meter
VPDC	Van Power Distribution Control
VPS	Vehicle Power Supply
VS	Vehicle Station
VTS	Vandenberg Tracking Station
VTRS	Vandenberg Telemetry

V&V	Validation and Verification
W	Watt(s)
W	Windsonde
W/	With
W/B	Wideband
WBDL	Wideband Data Link
WBDS	Wideband Data Set
WCR	Winds Conference Room at South Park West (Denver)
WIS	Wideband Instrumentation System
WL	Water Line
W/m^2	Watt(s) per Square Meter
WPM	Words Per Minute
WSMC	Western Space and Missile Center
WT	Weight
WTR	Western Test Range
Xdcr	Transducer
XMTR, xmitter	Transmitter
Xpdr	Transponder
Xptr	Transporter

Chapter 1

Titan Launch Vehicle Family

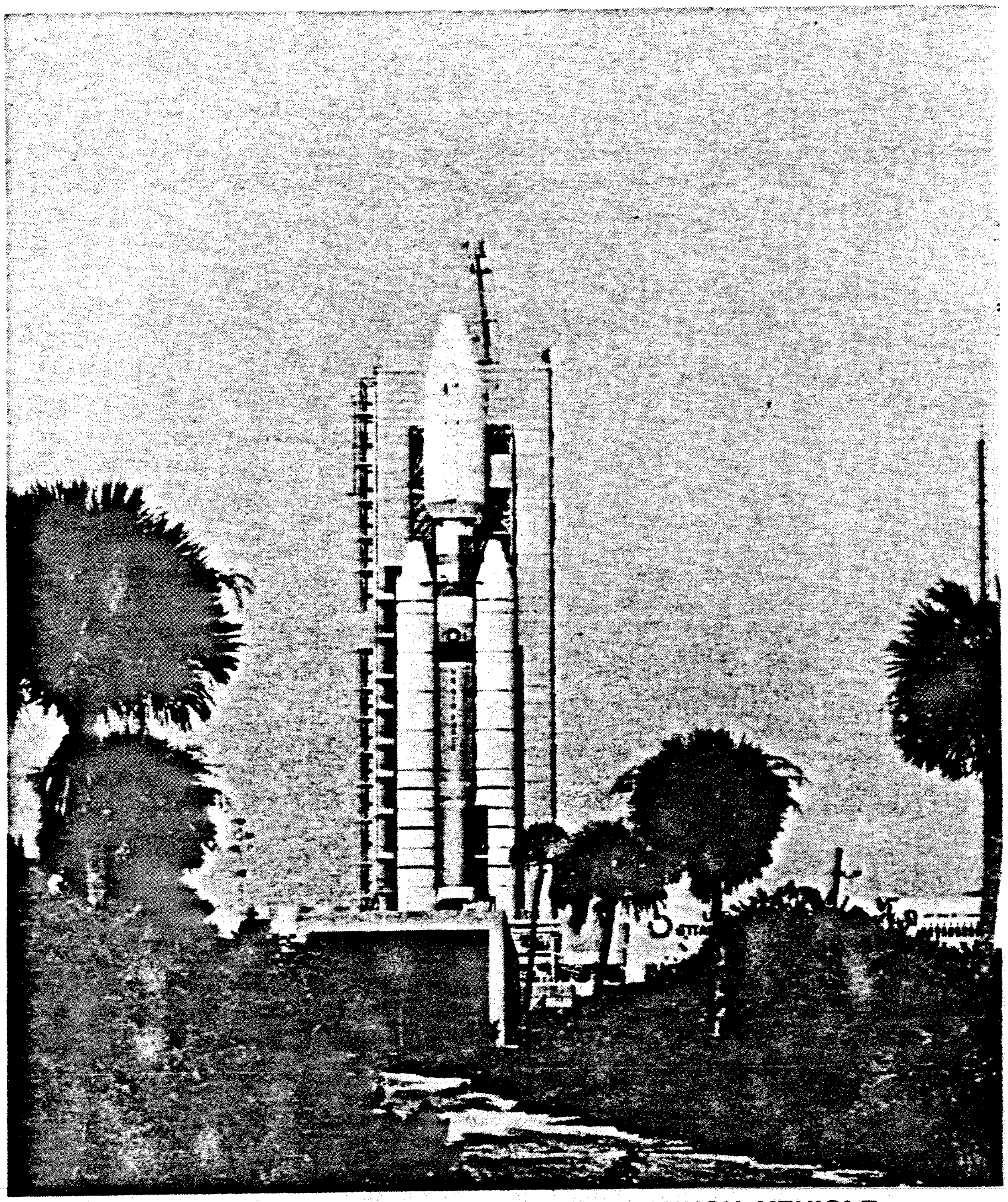


FIGURE 1-1 THE FIRST TITAN IV LAUNCH VEHICLE

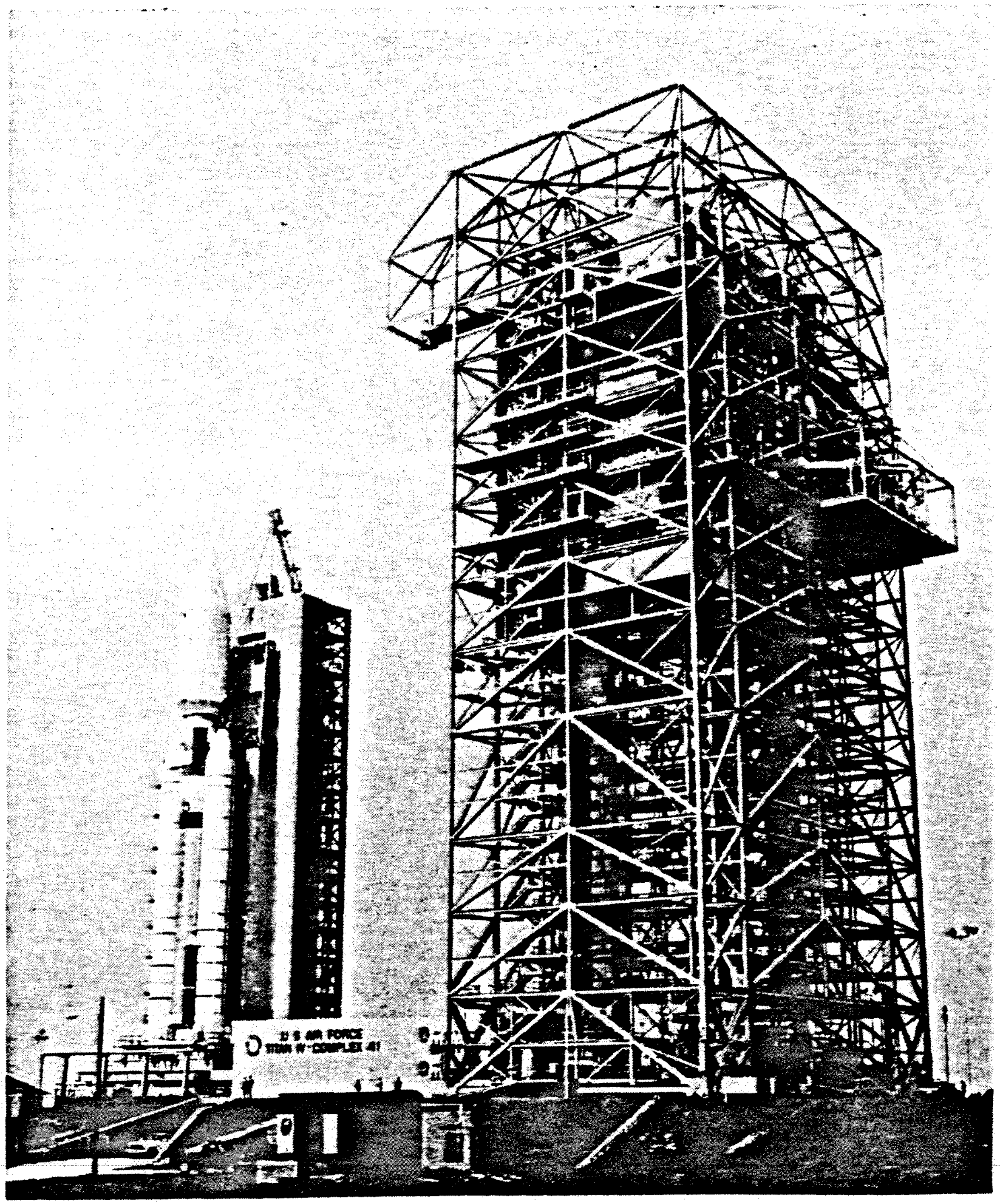


FIGURE 1-2 THE FIRST TITAN IV LAUNCH VEHICLE

1.0 TITAN LAUNCH VEHICLE FAMILY

1.1 Introduction

Titan launch vehicles have been contributing to our national space objectives for more than 25 years. As the American space program has grown, the Titan family has expanded to meet the changing requirements. The dependability and versatility of Titan vehicles have been demonstrated by their selection for missions ranging from strategic intercontinental ballistic missile weapon systems, to the manned Gemini space flights, to NASA interplanetary missions, to Viking Mars missions, to critical national security programs.

Figure 1-3 depicts the profile outlines and proportionate size of the principal launch vehicles in the Titan development history. Current, active program vehicles are shaded.

A brief summary of the vehicle history for each Titan program follows.

1.2 Titan Configuration Evolution

1.2.1 Titan I ICBM

This was the first Titan program. It was a two-stage Intercontinental Ballistic Missile (ICBM). Development began in 1955 with the first launch occurring in February 1959 and the last launch in March 1965.

1.2.2 Titan II ICBM

Titan II was Martin Marietta's second ICBM program. Development began in 1960 with the first launch in March 1962. This was the first strategic missile to use storable hypergolic propellants and the first to use an inertial guidance system. The last Titan II launch was in June 1976. Deactivation was completed in 1986.

1.2.3 Titan II/Gemini

The Titan II ICBM was also converted into the man-rated Gemini launch vehicle and served as a significant step in the evolution of the Apollo programs. There were 12 successful Gemini flights between April 1964 and November 1966. Ten of these 12 flights were manned.

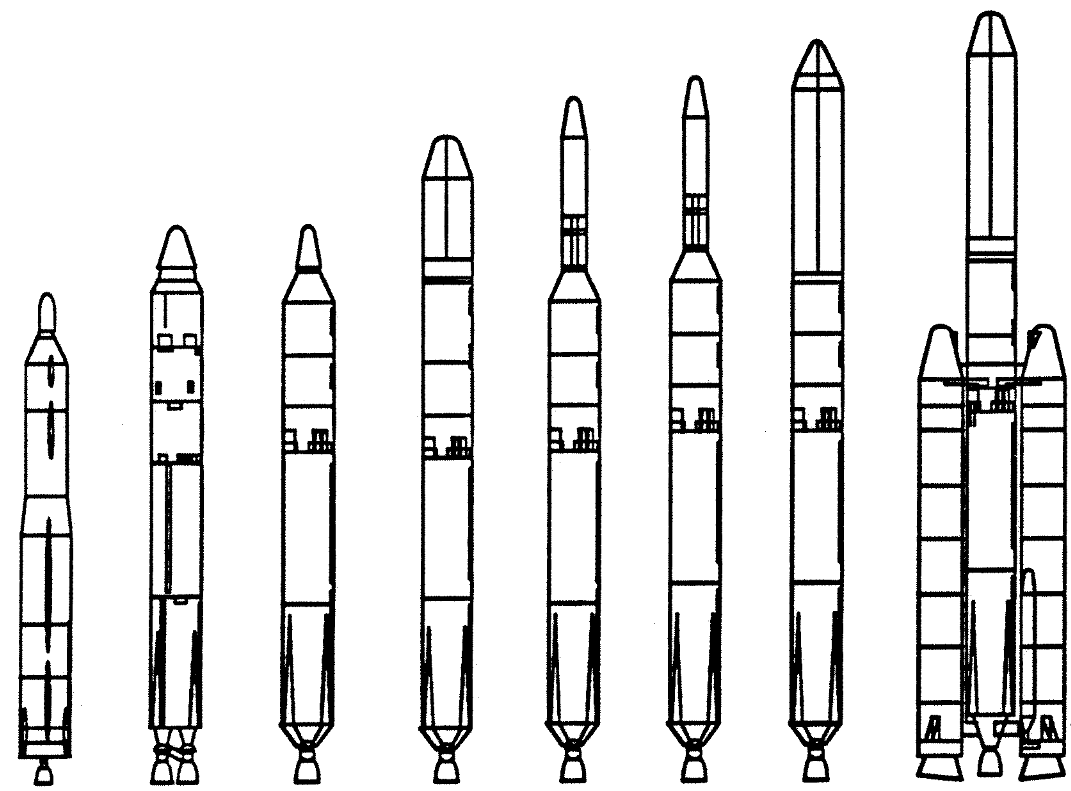


FIGURE 1-3 TITAN LAUNCH VEHICLE FAMILY (SHEET 1 OF 2)

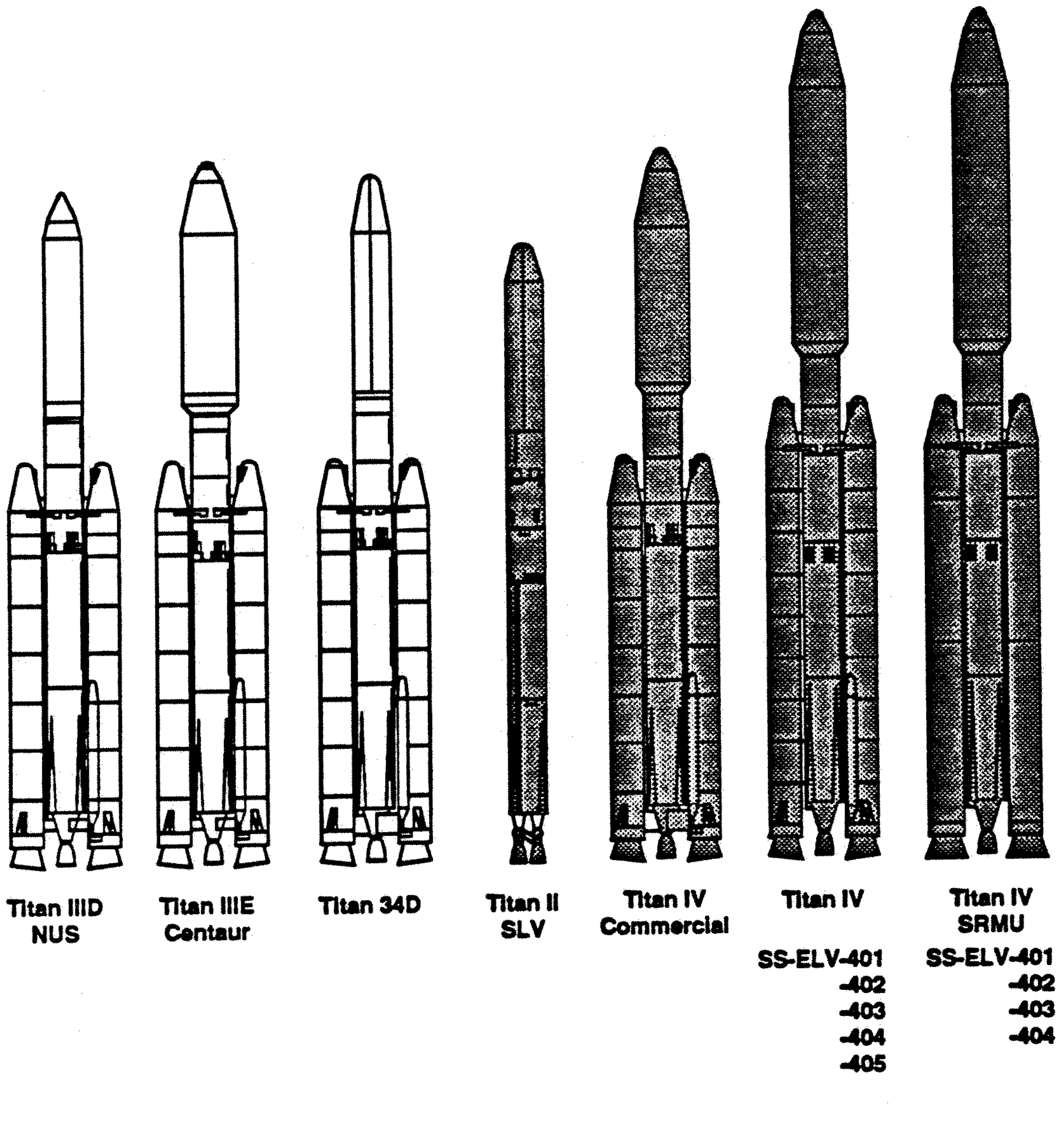


FIGURE 1-3 TITAN LAUNCH VEHICLE FAMILY (SHEET 2 OF 2)

1.2.4 Titan II Space Launch Vehicle

The Titan II ICBM is undergoing conversion into a fleet of space launch vehicles for relatively small U.S. government payloads to be launched from Vandenberg Air Force Base (VAFB), California at the rate of 3 per year. It utilizes a 10-foot payload fairing. The goal of this program is to maximize the use of Titan II weapon system materials and to minimize launch site modifications and life-cycle costs - all at a level consistent with mission success objectives. To meet these objectives, extensive use has been made of technology and hardware developed during the Titan III program. A total of 55 Titan II ICBMs are available to be converted into space launch vehicles with 14 currently on contract for the conversion. Refurbishment activities are performed at Martin Marietta's Denver facilities. Unmodified Titan II ICBMs are stored at Norton Air Force Base, California awaiting transportation to Denver to meet the refurbishment schedule.

1.2.5 Titan III Programs

The Titan III program began with the development of the Titan IIIA, which was a two-stage liquid rocket engine, core only (no solid rocket motor strap-ons) vehicle using a Transtage as the third stage. The first Titan IIIA was launched from Vandenberg Air Force Base, California 1 September 1964. Core-only vehicle development continued with the Titan IIIB program. These vehicles were also launched from VAFB, California with the first launch being 29 July 1966. Of the 68 Titan IIIB vehicles launched, 67 were successful. The Titan IIIC evolved from the IIIA with the addition of two 5-segment Solid Rocket Motors (SRMs) to the core vehicle. Since the first launch 18 June 1965, the IIIC successfully deployed numerous payloads covering a broad range of flight plans, spacecraft and spacecraft deployment modes. The Titan IIIC missions were flown from both Cape Canaveral Air Force Station (CCAFS) and Vandenberg Air Force Base. Several of the flights involved multiple-payload missions with from four to eight spacecraft and as many as three different spacecraft manufacturers. On 30 May 1974, a Titan IIIC launched the most powerful NASA Communications satellite as of that date (ATS-F). The last Titan IIIC flight was 6 March 1982. The Titan IIID configuration was a no-upper-stage version of the IIIC developed for west coast launches. The first Titan IIID launch was 15 June 1971. The IIID achieved a 100-percent mission success rate. The Titan IIIE configuration included a Centaur D-IT upper stage, an inertial guidance system and a 14-foot diameter payload fairing. The IIIE was used to boost the highly successful Viking spacecraft to Mars and the two Voyager spacecraft to Jupiter and Saturn. Martin Marietta designed and built the Mars Lander Vehicles. Martin also designed and built the Voyager's propulsion system and associated control electronics. The first Titan IIIE was launched 11 February 1974 from CCAFS.

1.2.5 Titan III Programs (Continued)

The Titan 34B and 34D vehicles are derivatives of the Titan III class and they are outgrowths of the Titan IIIM, originally designed to launch the Air Force Manned Orbiting Laboratory (MOL) from Vandenberg AFB. Even though the MOL program was cancelled, much of the design effort was transferable to the Titan 34 vehicles. The 34B had a 10-foot diameter payload fairing with a length up to 57.9 feet. This vehicle accommodated a variety of upper stage and payload combinations that were launched into polar and other high-inclination orbits.

The 34D is a stretched version of the 34B core vehicle and it incorporates two five-and-one-half segment Solid Rocket Motors. This vehicle has been integrated with several upper stages (IUS, Agena, Transtage), payload fairings, and guidance configurations. Missions from the Eastern Space and Missile Center (ESMC) included Geosynchronous and Low-Earth Orbits (LEO)with low and high inclinations. Launches from the Western Space and Missile Center (WSMC) included putting 27,600 pound payloads into a 100-mile polar orbit.

1.2.6 Titan IV

The Titan IV is designed to complement the National Space Transportation System (shuttle, orbiter). It is an independent launch vehicle system to assist in assuring DOD access to space.

The Titan IV is essentially a derivative of the 34D with structural extensions and other modifications to the core vehicle design to accommodate two seven-segment SRMs or two three-segment Solid Rocket Motor Upgrades (SRMUs). The Titan IV provides STS-equivalent and greater payload-to-orbit performance, thus enabling it to meet DOD unique requirements.

1.2.7 Titan III Commercial Launch Vehicle

The Titan III and III-T (Transtage), the newest members of the family heritage, provide launch services for both commercial and government payloads to either Low-Earth Orbit (LEO) or Geosynchronous Transfer Orbit (GTO) from Launch Complex 40 at Cape Canaveral Air Force Station, Florida. This configuration is similar to the Titan T-34D, but offers greater flexibility through a larger Payload Fairing (PLF).

Chapter 2

Payload Integration

2.0 PAYLOAD INTEGRATION

2.1 Overview

Payload Integration is the process of integrating the user's payload (also used interchangeably with satellite or spacecraft) onto the Titan IV Launch Vehicle System as represented in Figure 2.1-1. In addition to acquiring mission specific hardware and conducting technical analyses, it also encompasses management, finances, planning and other programmatic considerations.

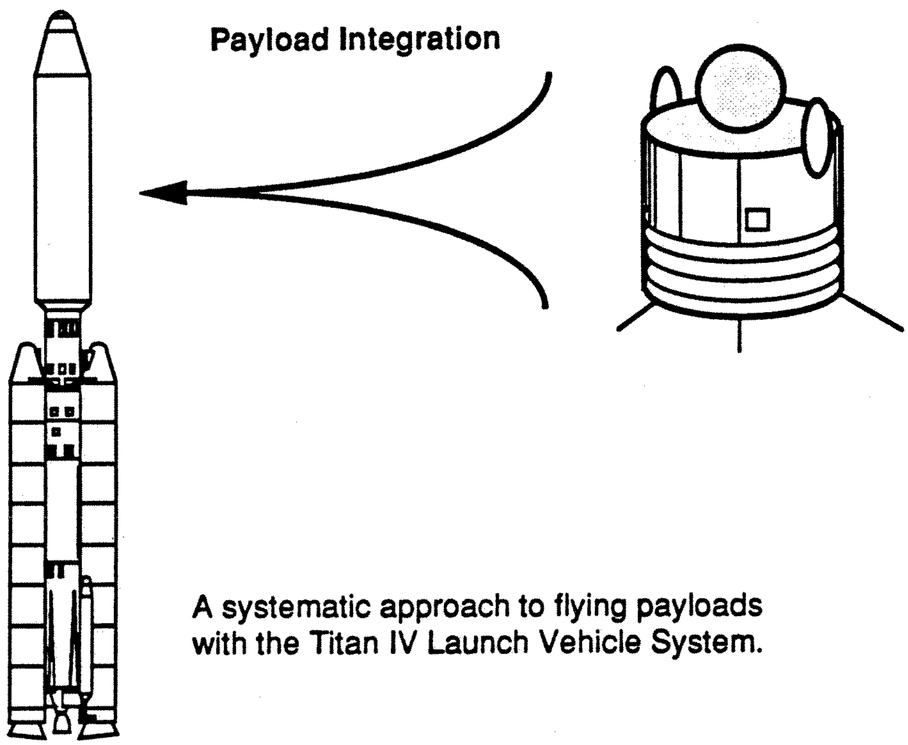


FIGURE 2.1-1 PAYLOAD INTEGRATION

Effective integration with the Titan IV Launch Vehicle System requires that all government agencies and contractor organizations possess a clear understanding of their roles and responsibilities. All participants must implement effective plans to manage and control the integration process. Early identification of payload requirements is the key to a smooth, timely payload integration implementation.

This chapter deals with the Payload Integration Implementation Elements, Payload Interface Verification Process and Independent Verification and Validation.

2.2 Payload Integration Implementation Elements

2.2.1 Introduction

This section discusses the organization, process, control, roles and responsibilities for integrating a payload onto the Titan IV at either CCAFS or VAFB. It is a top-level guide for Titan IV and Payload managers in tailoring and establishing the integration approach for each payload. The section is sub-divided as follows:

2.2.1 Introduction
2.2.2 Organizations
2.2.3 Integration Process Overview
2.2.4 The Working Groups
2.2.5 Payload Integration Documentation
2.2.6 Payload Interface Verification
2.2.7 Scheduling Support
2.2.8 Payload Support Equipment
2.2.9 The Unified Payload Integration (UPI) Effort

2.2.2 Organizations

The Titan IV payload integration team is comprised of a number of Air Force and contractor organizations. A typical organizational structure is summarized in Figure 2.2.2-1; however, this may be modified, depending on the payload program. In general, for the Titan IV/Centaur and the Titan IV/NUS launch vehicles, payload integration is the joint responsibility of the Titan IV System Program Office (SPO) (ME) and the respective Payload SPOs. This joint responsibility continues through park orbit for the Titan IV/NUS. For the Titan IV/Centaur, it continues through transfer orbit to spacecraft separation and subsequent completion of a satisfactory separation "collision and contamination" avoidance maneuver. Finally, for the Titan IV/IUS, this joint responsibility is expanded to include the Upper Stages SPO (CLU), and is shared through park orbit insertion of the spacecraft/IUS combination.

2.2.2.1 Program Executive Office

The Program Executive Office (PEO) is responsible for research, development and acquisition of military space systems. The PEO reports directly to the Secretary of Defense. In this role, it is responsible for the Titan IV program and various DOD payload programs. The PEO is currently located at Los Angeles Air Force Base, California, but will relocate to Washington, DC in the fall of 1990.

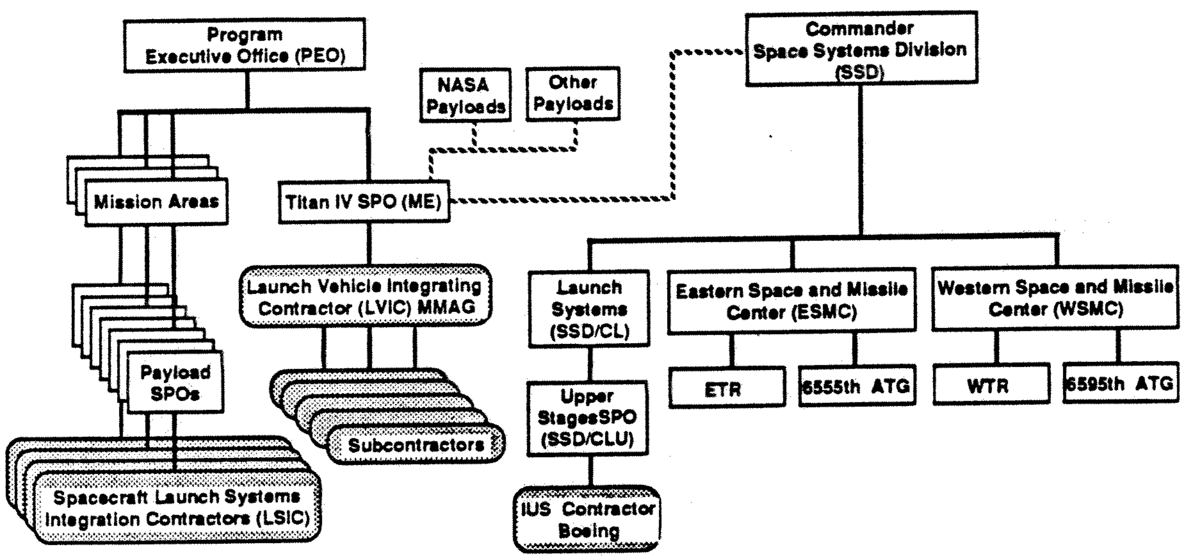


FIGURE 2.2.2-1 TYPICAL TITAN IV OVERALL AIR FORCE/ CONTRACTOR ORGANIZATION

2.2.2.1.1 Titan IV System Program Office

The Titan IV SPO (ME) is responsible for the Titan IV Launch Vehicle System. This responsibility includes the Titan's management, program control, procurement, development, integration, operation and mission success. The procurement function includes Titan IV systems analyses, services and launch vehicle mission peculiar hardware modifications to the Titan IV system for meeting specific spacecraft integration requirements. The Titan IV SPO also reports through and gets support from the Space Systems Division.

2.2.2.1.2 Payload (Satellite/Spacecraft) System Program Offices

These offices, manage and direct Air Force spacecraft programs. Also, the SPO or its equivalent may reside within a DOD, NASA, or other government organizations. The SPO is responsible for total mission management of the Spacecraft (SC) including contractual and technical direction. This includes developing, procuring and operating the SC. The SPO also provides contractual and technical direction to the Launch Systems Integration Contractor (LSIC), if one is required.

2.2.2.2 Space Systems Division

Space Systems Division (SSD) which is part of the Air Force Systems Command, is responsible for research, development, acquisition and launch of military space systems. In this role, it is responsible for the management of the launch bases which provide support to the Titan IV program. The Space Systems Division is headquartered at Los Angeles Air Force Base, California, with subordinate organizations located elsewhere in the United States and overseas.

2.2.2 2.1 Eastern Space and Missile Center

The Eastern Space and Missile Center (ESMC) is composed of various organizations, two of which are the Eastern Test Range (ETR) and the 6555th Aerospace Test Group (ATG). ESMC operates the CCAFS and provides support to various tenant organizations at Patrick AFB. ESMC supports all launches from the Eastern Launch Site (ELS). ESMC Safety Engineering is responsible for flight and range safety, and for the safety of all ESMC and user personnel. Additionally, ESMC provides safety oversight and review of all prelaunch and launch procedures and facility operations.

2.2.2.2.1.1 Eastern Test Range

The ETR provides launch and tracking facilities, base support services, safety procedures, and data acquisition, evaluation and processing. ETR supports Army, Navy (including submarine-launched ICBMs), NASA, commercial, foreign government, and Air Force manned and unmanned space, missile and launch vehicle programs. This support begins at CCAFS (for all Titan IV missions), Kennedy Space Center (KSC) or from a down-range location, and continues through the Caribbean and into the South Atlantic Ocean for a distance of over 10,000 miles.

2.2.2.2.2 Western Space and Missile Center

The Western Space and Missile Center (WSMC) organizations include the Western Test Range (WTR), the 6595th ATG and the 6595th Missile Test Group. WSMC supports all launches from VAFB. The WSMC Safety Engineering is responsible for range safety, flight safety and for the safety of all WSMC and user personnel. WSMC provides safety approval for all prelaunch and launch procedures and for facility operations.

2.2.2.2.2.1 Western Test Range

The WTR is responsible for all range resources in support of all launch operations at VAFB or down-range locations. WTR supports a wide range of launch vehicle and missile systems including the Peacekeeper ICBMs, which is under the control of the 6595th Missile Test Group. Services provided by WTR include weather forecasting, launch scheduling, communications, hazard area clearance, tracking, data acquisition and analysis, ICBM impact scoring, recovery of reentry vehicles and data packages, and facility maintenance. WTR extends from the California Coast to 90 deg longitude in the Indian Ocean where it meets ETR.

2.2.2.2.3 6555th and 6595th Aerospace Test Group

The 6555th ATG and 6595th ATG are the designated test groups for ESMC and WSMC respectively. Under each ATG is a Launch Vehicle Division to represent the Titan IV SPO (CLD) in all matters concerning Titan activities at their respective locations. Also under each ATG are Payload Divisions representing the Payload SPOs. Some of the functions of the ATGs include providing the required facilities, communications, tracking, and command and control facilities.

2.2.2.2.4 Aerospace Corporation

The Aerospace Corporation specializes in space systems and related technologies. It provides technical support and general systems engineering and integration for the PEO/Titan IV and payload SPOs, and for SSD and its 6555th and 6595th Aerospace Test Groups.

2.2.2.3 The Titan IV Contractor Team

The contractor team for the Titan IV is shown in Figure 2.2.2.2-1 with each contractor's area of responsibility. Martin Marietta is the Launch Vehicle Integrating Contractor (LVIC), and together with the Payload SPO's Launch Systems Integration Contractor (LSIC) are responsible for accomplishing the payload integration.

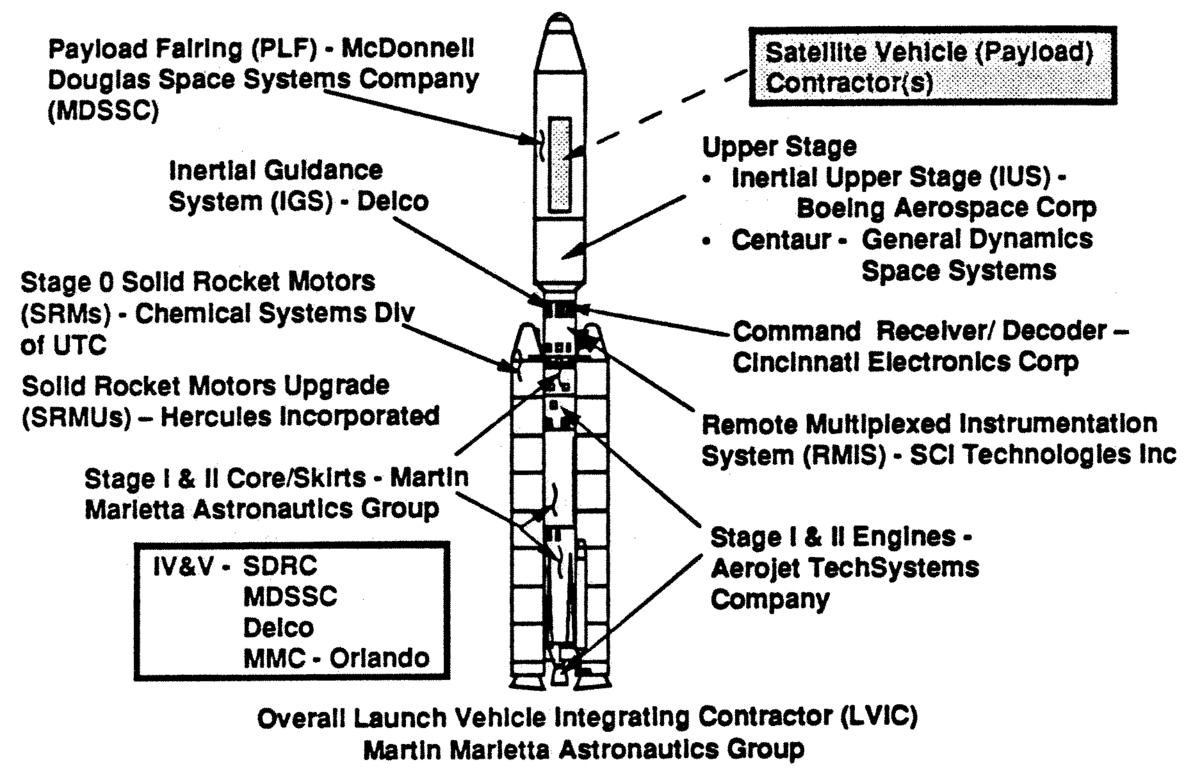


FIGURE 2.2.2.2-1 LAUNCH VEHICLE INTEGRATION TEAM

2.2.2.3.1 Martin Marietta Astronautics Group

As prime contractor, Martin Marietta Astronautics Group (MMAG) has overall responsibility for design, development, procurement, test and launch of the Titan IV. Martin Marietta as the LVIC, together with the LSICs has overall contractual responsibility for all integration analyses and services required to integrate a spacecraft onto the Titan IV/Centaur and Titan IV/NUS launch vehicles. For the Titan IV/IUS, Boeing shares this responsibility. For the Titan IV system, Martin Marietta will subcontract to the Titan IV element contractors those special integration analyses and services which require their input or action.

Additionally, Martin Marietta is responsible for managing the facilities and activities within the Integrate-Transfer-Launch (ITL) area at CCAFS and the Titan launch facilities at VAFB. This includes: 1) receipt through launch processing of the various Titan IV elements at both CCAFS and VAFB; 2) launch operations in support of the various types of missions and different user program requirements; and 3) refurbishment and maintenance of the launch preparation and launch facilities, and the core vehicle transporters at CCAFS. Martin Marietta is also responsible for modifying these facilities, such as the Mobile Service Towers, to meet Titan IV program and payload/mission peculiar requirements.

2.2.2.3.2 General Dynamics Space Systems

General Dynamics Space Systems (GDSS) is a subcontractor to Martin Marietta and is responsible for the design, development, testing and delivery of the Centaur upper stage. GDSS also provides analyses, mission-peculiar hardware kits and services to support the payload integration activities.

2.2.2.3.3 Boeing Aerospace Corporation

For the Titan IV/IUS missions, Boeing Aerospace Corporation (BAC) and Martin Marietta jointly share Payload (P/L) integration responsibilities as associate contractors. Boeing designs, builds, delivers, checks out and maintains the IUS and provides associated support equipment.

2.2.2.3.4 McDonnell Douglas Space Systems Company

McDonnell Douglas Space Systems Company (MDSSC) is a subcontractor to Martin Marietta and is responsible for the design, development, testing and delivery of the PLF. MDSSC also provides analyses and mission-peculiar kits, such as unique access doors or thermal protection, as required to support the payload integration activities.

2.2.2.3.5 Other Titan IV Subcontractors

Chemical Systems Division (CSD), Hercules Incorporated (HI), Aerojet TechSystems Company (ATC), Delco Systems Operations, SCI Systems, Inc. and Cincinnati Electronics Corporation (CEC) are subcontractors to Martin Marietta and have respective element responsibilities as shown in Figure 2.2.2.2-1. Generally, these element contractors are not involved in the payload integration process, but they are available to provide any special payload/mission requirements. Also, note the Independent Verification and Validation (IV&V) contractors shown in Figure 2.2.2.2-1. These IV&V contractors perform an independent re-verification of the Titan IV data models and analytical tools. The IV&V efforts are further discussed in Appendix B.

2.2.2.3.6 Spacecraft 's Launch Systems Integration Contractor

The Payload SPO may designate a spacecraft contractor/representative as the LSIC. The LSIC is responsible for Satellite Vehicle (S/V) integration. This responsibility includes: defining SC and mission requirements, specifying launch base and range support requirements, verifying interface requirements, developing and maintaining integration schedules and plans, and coordinating integration activities with the LVIC.

2.2.3 Integration Process Overview

2.2.3.1 Requirements Flow and Direction

The goal of the integration process is to place a spacecraft vehicle into a predefined orbit/orientation, subject to a variety of system requirements and constraints. The Payload SPO has the responsibility for identifying these requirements and constraints through its LSIC as well as directly to the Titan IV SPO -- and for a Titan IV/IUS mission, to the Upper Stages SPO. The flow of integration requirements and the program direction are summarized in Figure 2.2.3.1-1.

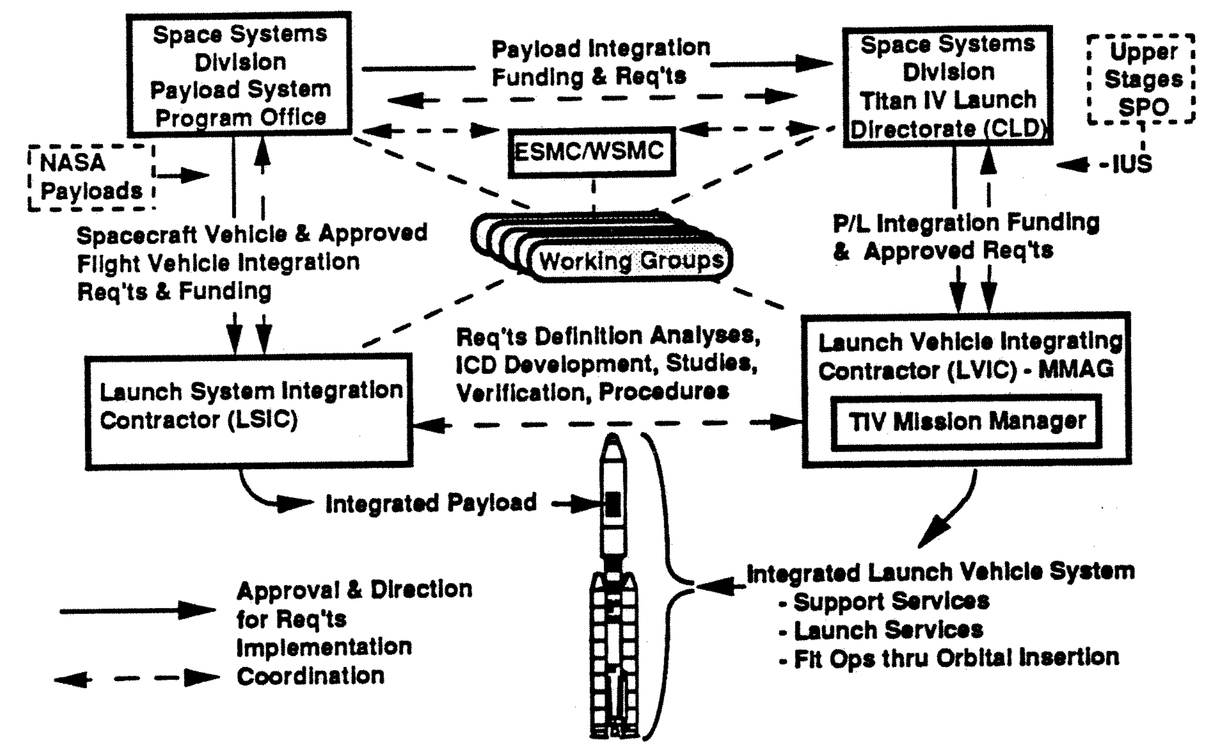


FIGURE 2.2.3.1-1 FLOW OF INTEGRATION REQUIREMENTS AND DIRECTION

For each payload program, Martin Marietta identifies a Titan IV Mission Manager. This individual is the LVIC's focal point for the overall integration of the spacecraft into the Titan IV system. Similarly, the LSIC is the focal point for payload integration activities.

A free-exchange of ideas and issue/problem solutions is required by all participants to the integration process to help identify and coordinate the requirements. Key to this coordination are the Working Groups shown in the above figure; they will be discussed further in Paragraph 2.2.4.

2.2.3.2 Payload Integration Tasks

Typical Martin Marietta payload integration tasks are shown in Figure 2.2.3.2-1. These include assigning a dedicated Titan IV Mission Manager, whose responsibilities include providing technical engineering support in the areas of loads, performance, aerodynamics, thermal, EMC/interference and environmental analyses. Martin Marietta also conducts development testing, defines ground equipment and test requirements, assesses and implements safety and security plans, schedules and integrates all spacecraft requirements, conducts mission analyses and provides payload support equipment and simulators. In summary the real measure of success of the integration process is to provide a safe, successful launch and flight for the spacecraft to its designated orbital injection.

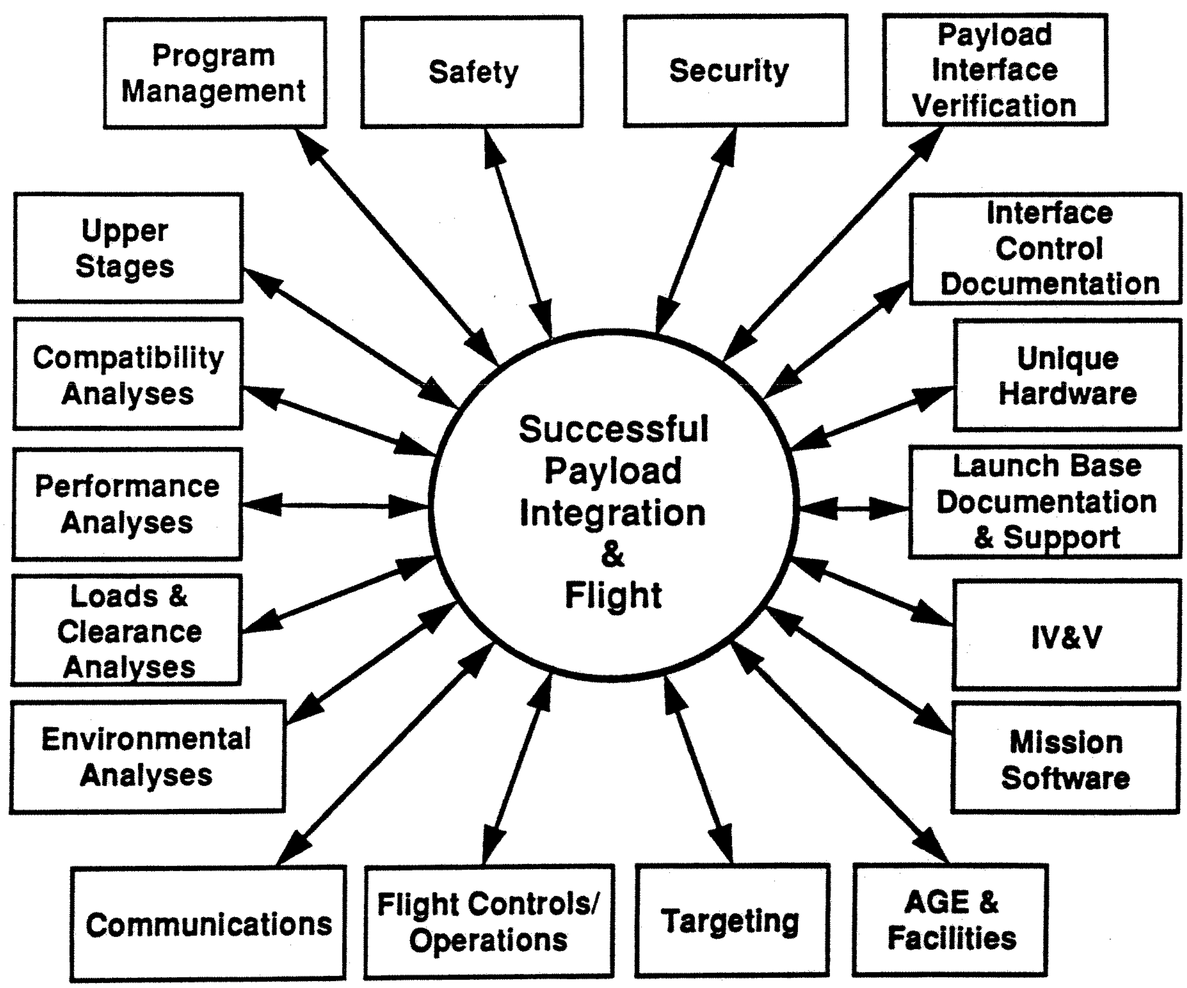


FIGURE 2.2.3.2-1 TYPICAL LVIC PAYLOAD INTEGRATION TASKS

2.2.3.3 Payload Integration Phases

Shown in Figure 2.2.3.3-1 is an "ideal" picture of the payload integration process in a time span ranging from 21 to 48 months. This represents a first time payload user. The time of the actual integration for any new spacecraft is a function of spacecraft and upper stage complexity and their impact to the Titan IV in terms of analyses and hardware modifications, and relative national priority. The distinction between phases is sometimes merged depending on payload requirements but the three-phase view still provides a technical model for describing the maturity of the integration process and the activities within these phases that must be accomplished.

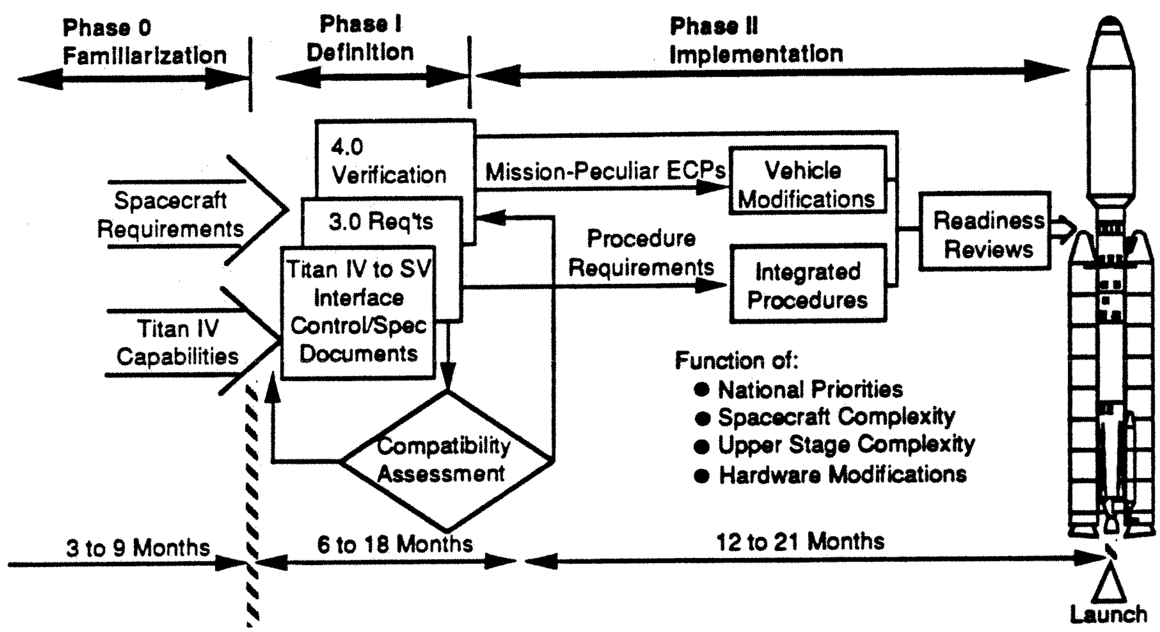


FIGURE 2.2.3.3-1 TITAN IV INTEGRATION PROCESS FOR FIRST TIME USER (PHASE VIEW)

2.2.3.3.1 Phase 0 – Familiarization

A Titan IV mission feasibility assessment is performed for the new spacecraft. Titan IV capability data are provided to the Payload SPO and his contractor(s) through the LSIC, and preliminary spacecraft requirements are assessed against these Titan capabilities. A three-dimensional reference trajectory is generated, and preliminary design loads are calculated. Trade studies are performed to resolve any major incompatibilities between spacecraft requirements and Titan IV capabilities.

2.2.3.3.1 Phase 0 – Familiarization (Continued)

By the end of Phase 0, the scope and general tasks required in the subsequent Phase I and Phase II -- or the UPI effort -- must be defined. This definition then enables the LVIC to generate and submit proposal(s) for accomplishing the hardware modifications, documentation, and other activities of payload integration. In summary, the major outputs of the Phase 0 are:

- Completion of the Payload Questionnaire (see Appendix A)
- Contractor sign-off of Interface Control Documentation (ICDs) by the LVIC-LSIC ("two-party")
- Titan IV Mission Plan for the subsequent UPI effort (see Paragraph 2.2.9)
- Management Plan defining the responsibilities of LVIC, LSIC and Customers

2.2.3.3.2 Phase I – Definition

Additional design analyses are performed for the Titan IV with the integrated payload. These analyses include loads, stability, thermal, contamination, performance, electrical, avionics and communications, guidance accuracy, electromagnetic compatibility, acoustics, vibration, shock and venting. A spacecraft vehicle/payload fairing rattlespace (dynamic clearance) analysis is normally performed and a preliminary safety analysis is started. Also, the Titan IV flight software is generated and tailored for the particular spacecraft requirements and constraints.

The "two-party" contractor-signed ICDs from Phase 0 are baselined and approved by adding the Titan IV and Payload SPOs' signatures. If the ICDs require any payload/mission peculiar hardware modifications to the Titan IV launch system, Engineering Change Proposals (ECPs) are processed and formalized as a class I proposal for SSD approval. Also, procurement of long-lead Phase II items should be initiated during Phase I (or earlier).

As coordinated with the Titan IV and user SPO, this submittal may propose to contractually implement the agreed to ICD or an upper level Functional Interface Specification (FIS) which broadly defines the detailed requirements of the ICD. Typical vehicle modifications include fabricating special access doors and RF windows in the PLF, and adding unique wiring for the payload.

2.2.3.3.3 Phase II – Implementation

Vehicle modifications are performed, integrated procedures are developed and Payload/Titan IV verification is completed. The safety analyses are completed and the analytical predictions made during Phase I are verified. Integrated procedures are developed for testing and checkout of subsystems and systems, such as installation of the payload, installation of the PLF, payload combined systems testing, launch vehicle combined systems test, R-count and the final countdown.

2.2.4 The Working Groups

An initial integration task in Phase 0 is to establish the overall management approach for the particular payload program. The key is to provide a clear and concise management approach that gives the integration team an understanding of how integration will be controlled. The Management Working Group (MWG) implements the management approach and is co-chaired by the Titan IV and Payload SPOs. It has overall responsibility for the working group process that helps guide and track payload integration.

The Management Working Group establishes other working groups as required, comprised of the various Air Force agencies and contractors, to identify and assign actions for resolving schedule and technical integration issues. In addition to the Management Working Group, other working groups may typically include:

- Interface Control
- Payload
- Payload Ground Operations
- Flight Design
- Structural, Mechanical & Environmental
- Electrical & Avionics
- System Engineering

Representative functions for several working groups are as follows:

Interface Control Working Group – The Interface Control Working Group (ICWG) is responsible for reviewing and identifying payload-peculiar interface requirements, and recommending changes to the ICDs. Also, overall program schedules are reviewed, data exchange requirements identified and, in general, technical and schedule incompatibilities and conflicts resolved/identified. Meetings are co-chaired by the Titan IV and Payload SPOs -- and for a Titan IV/IUS integration, also by the Upper Stages SPO.

Payload Working Group – The Payload Working Group (PWG) plans and coordinates support for the payload element at VAFB and CCAFS. The group is generally chaired by the respective launch site Aerospace Test Group (6555th or 6595th ATG). This working group works closely with, or may be a part of, the Ground Operations Working Group (GOWG) which performs a similar function in planning and coordinating launch site program integration activities for the Titan IV Launch Vehicle System elements.

Technical Area (Structural, Electrical and Environmental) Working Groups – These groups are formed to monitor a particular technical or functional area of special interest or concern to payload integration. For example, a payload may have unique air conditioning and thermal constraints, as well as special cleanliness requirements. An environmental working group would then be established to monitor and track concerns in these areas.

2.2.4 The Working Groups (Continued)

Except for the Management Working Group, working groups are formed, as required, and when no longer needed, dissolved or combine with other working groups. Also, sub-working groups can be formed to resolved a unique or more complex integration issue. Each working group establishes a charter identifying the general scope of its intended activities, and this charter is provided to the Management Working Group.

Additionally, Technical Interchange Meetings (TIMs) are called to discuss technical and programmatic issues of special technical complexity or single interest. Generally, the results of the TIMs will be integrated into the working group process.

2.2.5 Payload Integration Documentation

The ICDs are the primary documentation instrument for identifying the Payload-to-Titan IV interface requirements. They define the operational, physical, functional and environmental interfaces between the payload-to-booster and other affected launch site facilities. The ICDs identify such items as the payload/booster attachment point configuration including tolerances, the voltage and current from various interface connectors, air conditioning temperatures and flow-rates, vibration and acoustic envelopes, interfacing facility platform locations, timelines of various events critical to the payload (e.g., PLF separation), the PLF skin temperature profile during flight and its emissivity, and any unique PLF access door requirements.

As noted in Paragraph 2.2.3.3.1, "two-party" ICDs are one of the major outputs of Phase 0. The initial sources and inputs for the Payload-to-Titan IV ICDs are shown in Figure 2.2.5-1. As shown in this figure, the primary format and source used in generating the ICDs are the generic (or "boiler plate") ICDs developed for East and West Coast Titan IV missions. Other inputs are the Martin Marietta payload integration briefing, the payload questionnaire, the Titan IV Payload Support Capabilities Document, other Titan IV documentation, and the mission needs and requirements of the payload program. These ICDs contain both parametric and drawing interfaces, with Section 3.0 in each ICD addressing the requirements, while Section 4.0 states how those requirements are verified (e.g., by test, demonstration, inspection, similarity or analysis).

Once the ICDs are approved and baselined by SSD, they are under formal configuration control. Their maintenance and any change processing are defined by the Martin Marietta Configuration Management Plan for the Titan IV Program, MCR-85-2510, which implements SSD document, Format and Requirements for Interface Documents, SAMSO-STD 77-4, Revision A. An overview of the flow for change processing after the ICDs are baselined is shown in Figure 2.2.5-2.

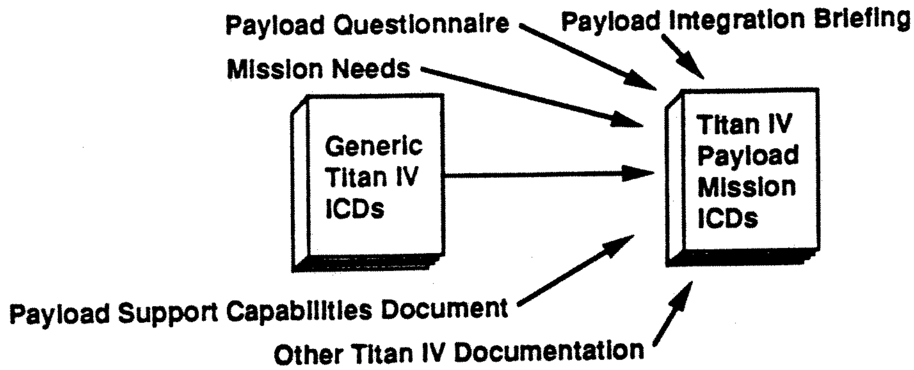


FIGURE 2.2.5-1 BASIS FOR A PAYLOAD-TO-TITAN IV ICD

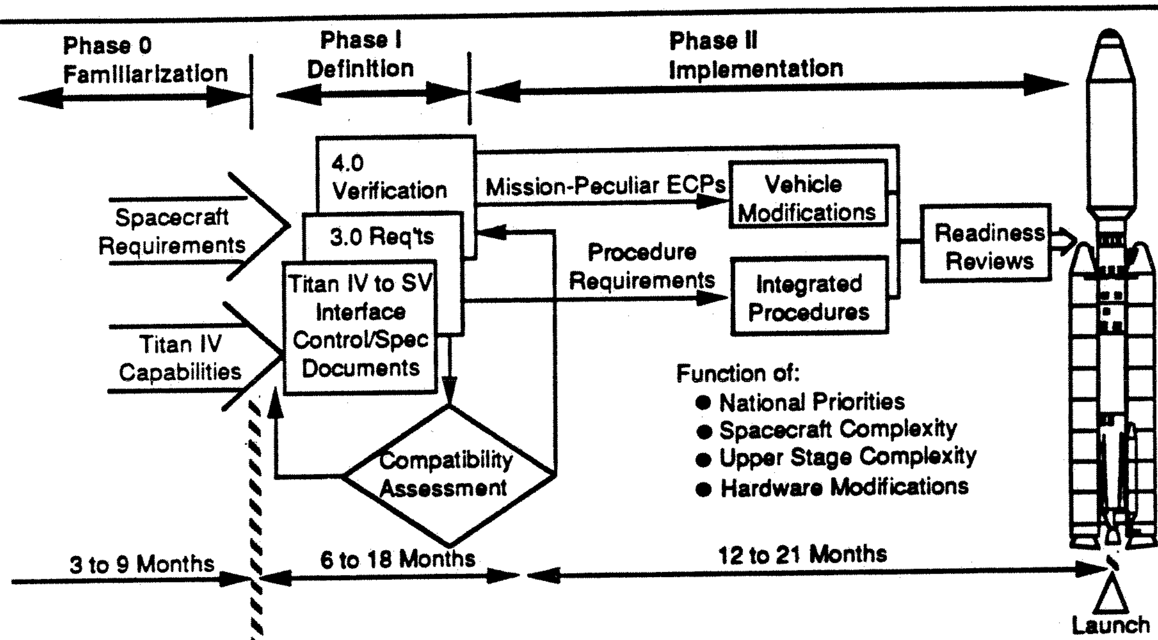


FIGURE 2.2.5-2 INTERFACE CHANGE PROCESS (AFTER BASELINED)

2.2.5 Payload Integration Documentation (Continued)

Referring to Figure 2.2.5-2, the procedures for updating the baselined ICDs in resolving interface incompatibilities and problems are as follows:

1. After the necessary analyses, an interface resolution action statement is entered on an Interface Problem Sheet (IPS). The IPS is a tracking sheet for the specific problem.

2. An Interface Change Notice Proposal (ICNP) is prepared utilizing SAMSO-STD 77-4 as a guide and is signed by the LVIC and LSIC contractors.

2.2.5 Payload Integration Documentation (Continued)

3. The contractors simultaneously submit proposals to their respective customers at SSD to enable an assessment of the total impact of the proposed change. If the ICNP (which is a proposed change only against the ICD itself) also results in a hardware, schedule, cost, test, or other contract impact, then the submitted proposal must not only include the ICNP but also an ECP.

4. These proposals are then reviewed by the SSD's Configuration Control Board (CCB) for technical approval.

5. When the ICNP is approved by SSD, the ICNP is issued as an Interface Change Notice (ICN) to all applicable agencies.

2.2.6 Payload Interface Verification

The Titan IV-S/V interface requirements identified in Section 3.0 of the ICD are verified by procedures in Section 4.0. The verification approach is developed and coordinated with the applicable working group. This information is documented in the planning section of the Interface Verification Certification Report (IVCR). As Titan IV and S/V processing occur at the factory and launch site, the verifications are accomplished and documented in the certification section of the IVCR. The LVIC and LSIC are required to sign off each IVCR. The completed IVCRs are then provided to the government for review prior to flight.

2.2.7 Scheduling Support

Planners support payload integration by providing detailed and unique mission-by-mission, integrated schedules. They support the Titan IV Mission Managers with complete and thorough understanding of all scheduled items, "what if" impacts, cause/effect analyses and proposal activity. These payload integration planners also provide coordination efforts between the various Working Groups to ensure that there is no duplication of effort or inconsistencies in information presented or received. The Titan IV Interface Engineers, Integration Payload Engineers and Verification Engineers are supported with the planner's overall and detailed mission plans, which assist in the development of requirements.

2.2.8 Payload Support Equipment

Payload Support Equipment (PSE) is available to the Titan IV payload community to satisfy interface verification requirements at the LSIC manufacturing, test or launch site facilities. This equipment includes hardware used in the Titan IV core vehicle development program. The available PSE is described in Table 2.2.8-1. Each LSIC should identify early in the program his payload requirements for this PSE to the LVIC's Mission Manager. The Titan IV Management Plan for Payload Support Equipment, MCR-87-2591, provides further description of the PSE available.

TABLE 2.2.8-1 PAYLOAD SIMULATOR EQUIPMENT HARDWARE

PSE Hardware Includes:

PSE Hardware	
• Payload Fairings	Test Code Hardware
– TCA #1 Forward Module	Test Code Hardware
– TCA #1 Base Module	Test Code Hardware
– TCA #2 Forward Module	Test Code Hardware
– TCA #2 Base Module	Test Code Hardware
– TCA #2 Fairing Handling Equipment	Test Code Hardware
• 2490 Skirts	Test Code Hardware
• 2491 Skirts	Test Code Hardware
• 2492 Skirts	Test Code Hardware
• 2500 Skirts	Test Code Hardware
• Mechanical Interface Verification Equipment (MIVE)	
• Centaur Forward Adapter	
• Payload Simulator Support Structure (PSSS)	
• Vertical Test Fixture (Payload Fairing Mockup)	
• Master Gauges	

2.2.9 The Unified Payload Integration Effort

Once the Phase 0 familiarization/feasibility effort is completed, the major portion of the payload integration effort begins and is carried out under UPI. As noted in Paragraph 2.2.3.3.1, Phase 0 must provide enough definition to enable Martin Marietta to prepare proposal(s) for submittal to SSD for completing the hardware payload integration process. The major products of Phase 0 that are especially key in defining the UPI activities are the contractors "two-party" ICDs and a Titan IV UPI Mission Plan.

The UPI effort provides the integration services but not hardware build/modifications required for integrating a specific payload into the Titan IV system. Should there be any required payload-peculiar hardware modifications, or build (for testing purposes, etc.) then their procurement will be by proposed action, separate from UPI. One pictorial view of the relationship between Phase 0, UPI and the separate hardware changes driven by mission-peculiar requirements is shown in Figure 2.2.9-1.

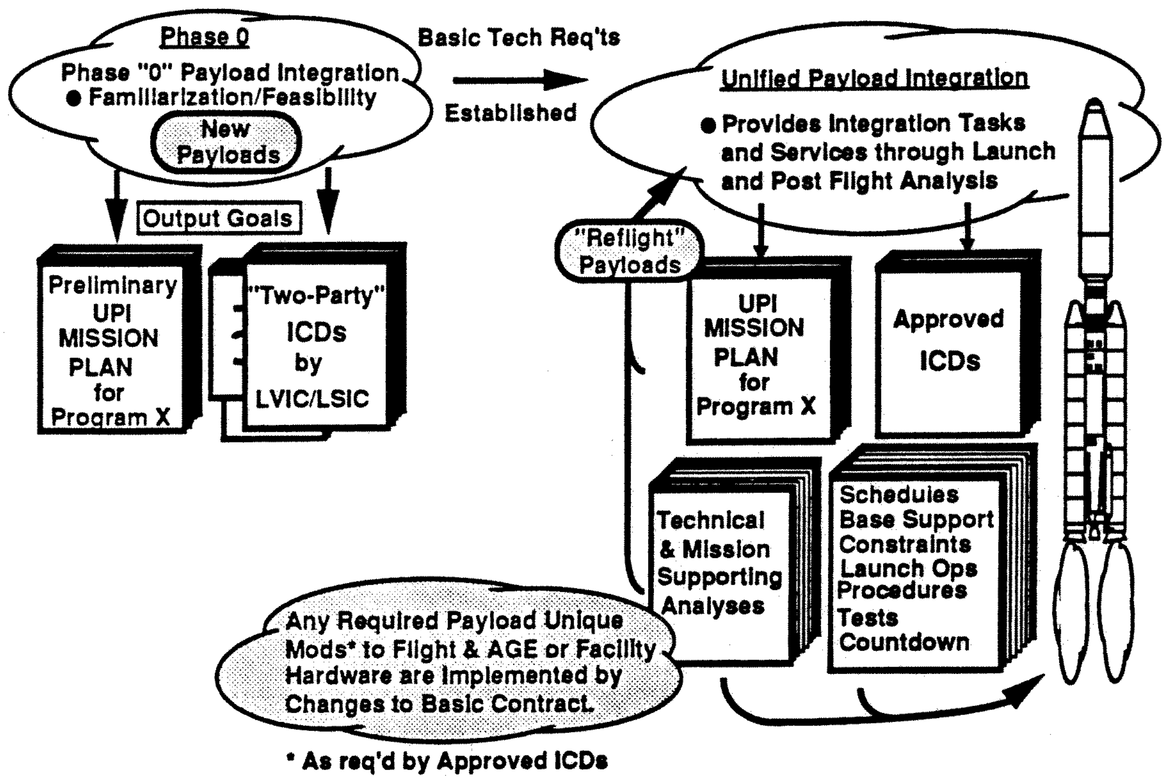
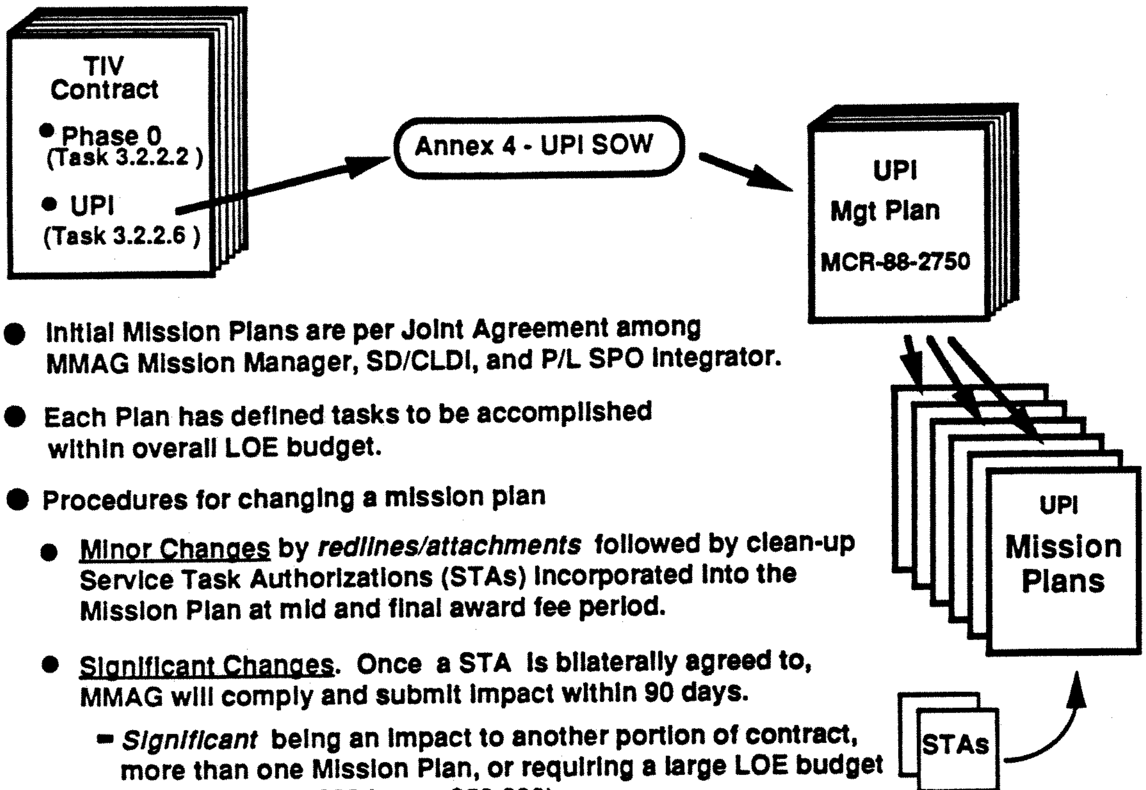


FIGURE 2.2.9-1 OVERVIEW OF PAYLOAD INTEGRATION IMPLEMENTATION

2.2.9 The Unified Payload Integration Effort (Continued)

Under UPI, each payload will have its own unique payload integration Mission Plan. Identified within this Mission Plan, which is developed jointly in Phase 0 between Martin Marietta and Space Systems Division, are the various tasks the contractor will accomplish. The generation and maintenance of these UPI Mission Plans are per the UPI Management Plan, MCR-88-2750, referenced in the UPI SOW Annex to the Titan IV contract. This relationship and further details on generating and modifying a Mission Plan is shown in Figure 2.2.9-2.

In summary, payload integration is the most visible portion of the Titan IV program that a payload user will see and deal with. All participating parties -- government and contractor -- must insure that effective plans are implemented early to insure full participation in the integration process: from the initial Phase 0 familiarization studies into the larger Unified Payload Integration effort that culminates in launch and flight. Finally, although payload requirements and needs are defined in various documents, the Interface Control Documentation is the cornerstone for identifying the physical, electrical and environmental requirements and constraints -- and requires a team effort to insure their timely definition and implementation.



- Initial Mission Plans are per Joint Agreement among MMAG Mission Manager, SD/CLDI, and P/L SPO Integrator.
- Each Plan has defined tasks to be accomplished within overall LOE budget.
- Procedures for changing a mission plan
  - <u>Minor Changes</u> by *redlines/attachments* followed by clean-up Service Task Authorizations (STAs) incorporated into the Mission Plan at mid and final award fee period.
  - <u>Significant Changes</u>. Once a STA is bilaterally agreed to, MMAG will comply and submit impact within 90 days.
    - *Significant* being an impact to another portion of contract, more than one Mission Plan, or requiring a large LOE budget adjustment (>1,000 hrs or $50,000).

FIGURE 2.2.9-2 IMPLEMENTATION OF UPI THROUGH THE MISSION PLAN

2.3 <u>Independent Verification and Validation (IV&V)</u>

Martin Marietta, consistent with its Launch System responsibilities, shall comply with the intent of Space Divisions Commander's policy, SAMSO regulation 550-1. This policy directs that Independent Verification and Validation be performed by an Independent Contractor (other than the developer) on flight critical airborne software and flight critical vehicle loads analyses. Figure 2.3-1 depicts the IV&V process. Table 2.3-2 identifies overall responsibilities. Reference Appendix B for a more detailed breakdown of IV&V validator's responsibilities.

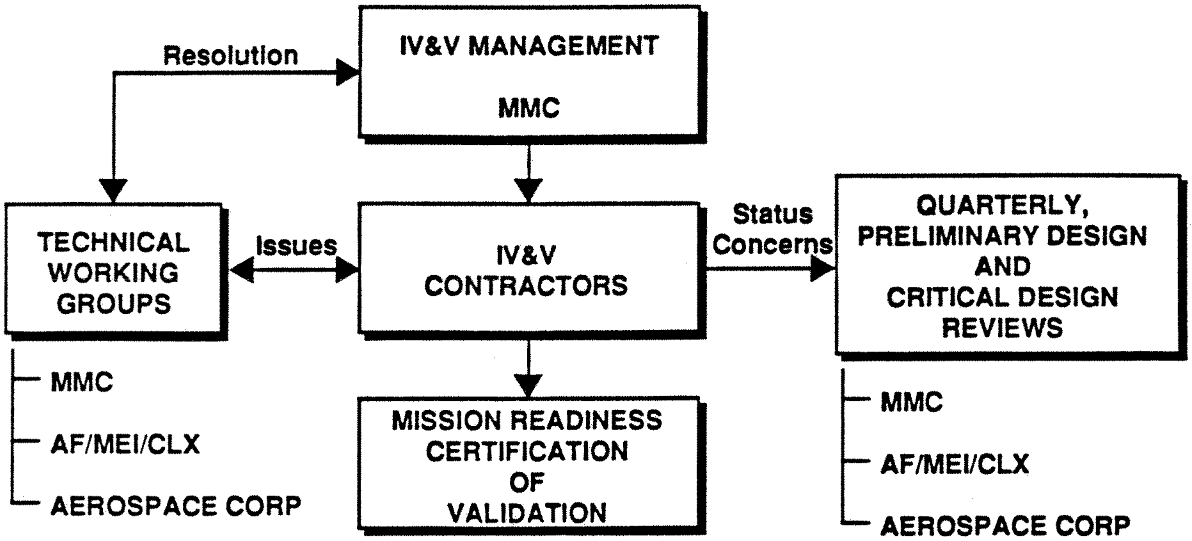


FIGURE 2.3-1 IV&V PROCESS

TABLE 2.3-2 IV&V RESPONSIBILITIES

Function	Developer	Validator
<u>Contract IV&V</u>		
Titan IV Booster Software	MMC	Delco
Centaur Software	GDSS	Analex
Titan IV Loads and Dynamics	MMC	MDSSC
Centaur Loads and Dynamics	GDSS	SDRC
Booster Targeting Software	MMC	Delco
Centaur Targeting Software	GDSS	Analex
Command Receiver Software	Cincinnati Electronics	MMC
General Centaur Engineering Support	Analex	——
<u>Mission Critical Analyses (Typical)</u>		
Booster Auto Pilot Stability	MMC	MMC/Orlando
Centaur Auto Pilot Stability	GDSS	Analex
Booster Guidance and Navigation	MMC	MMC/Orlando
Centaur Guidance and Navigation	GDSS	MMC
Booster Environments	MMC/CSD/MDSSC	MMC
Booster Stress	CSD/MDSSC	MMC
Centaur Stress	GDSS	MMC

Note: LSIC and Aerospace Corp are participating in selected validations

Chapter 3

Titan IV SLV

3.0 TITAN IV SLV

3.1 Introduction

This chapter describes the major elements that comprise the Launch Vehicle (LV). This includes the Titan IV core with its two liquid fueled stages, engines and forward skirts; the SRMs; the PLFs; and the Upper Stages and No Upper Stage (NUS) configurations.

The material presented is a description of the structures, propulsion and electrical systems and the performance of the major LV elements.

The Titan IV Boost Vehicle has the capability to autonomously place the Titan IV Upper Stage/NUS/Payload to the required mission position before Upper Stage/NUS/Payload separation; but, is capable of utilizing the Upper Stage or P/L avionics to accomplish the boost phase.

The Configuration Management Plan for Titan IV Program is presented in MCR-85-2510.

3.1.1 Nomenclature – Titan IV Configurations

The various nomenclature commonly used to refer to the Titan IV configurations include the Flight Vehicle, LV, Boost Vehicle, Titan IV System, Titan IV Type I and the Titan IV Type II LV.

3.1.1.1 Flight Vehicle

The Flight Vehicle is the ready to launch Titan IV configuration consisting of a S/V and Adapter, PLF, Upper Stage and Adapter if applicable and the Titan IV Boost Vehicle.

3.1.1.2 Launch Vehicle

The Launch Vehicle is defined and identified as an SS-ELV-401, SS-ELV-402, SS-ELV-403, or SS-ELV-405 configuration. The SS-ELV-401, 402 and 405 are scheduled for launch from the ESMC LC-41 and the SS-ELV-403 configuration is scheduled for launch from the WSMC SLC-4E.

3.1.1.2.1 SS-ELV-401 Launch Vehicle

The SS-ELV-401 LV consists of a Centaur Upper Stage, Adapter, PLF and the Titan IV Boost Vehicle.

3.1.1.2.2 SS-ELV 402 Launch Vehicle

The SS-ELV-402 LV consists of a PLF, the Titan IV Boost Vehicle and is configured for an IUS.

3.1.1.2.3 SS-ELV-403/405 Launch Vehicle

The SS-ELV-403/SS-ELV-405 LV consist of the PLF, the Titan IV Boost Vehicle and is configured without an Upper Stage. The SV Adapter is Government Furnished Property (GFP).

Unique GFP is not part of the LV, but is identified as part of a Flight Vehicle.

3.1.1.3 Boost Vehicle Configuration

The Titan IV Boost Vehicle configuration consists of the Titan IV Core and the SRMs. The core provides the necessary interfaces for the SRMs, the Upper Stage or NUS and the PLF. The core includes the Stage II Forward Oxidizer Skirt CP2460, the Forward Skirt Extension CP2460 and an Adapter Skirt, CP2491 or CP2492.

3.1.1.4 Titan IV Configurations

The Titan IV/Centaur System consists of an SS-ELV-401 LV, and its associated facilities and Ground Support Equipment (GSE).

The Titan IV/IUS System consists of an SS-ELV-402 LV, GSE and facilities which provide the capability to accommodate an IUS System Interface using kitable hardware.

The Titan IV/NUS System consists of an SS-ELV-403 or 405 LV, associated GSE and facilities. Any Unique P/L Adapters are not part of the Titan IV/NUS System. If required, the Adapters are furnished as GFP.

3.1.1.5 Titan IV Type I/Type II Launch Vehicles

The Titan IV Type I LV uses the seven segment SRMs for Stage 0. This is also referred to as the standard configuration. The Titan IV Type II LV uses the three segment SRMUs for Stage 0. This arrangement is referred to as the SRMU Stage 0.

3.2 Titan IV Core

The Core assembly is mainly composed of structural, propulsion and avionics subsystems. The structure is made up of two stages (each consisting of fuel and oxidizer tanks) joined by a removable skirt. The forward end of the core structure includes the Forward Oxidizer Skirt, the Forward Skirt Extension and Adapter Skirts which allow interfacing with the various Upper Stage/NUS configurations. The core also provides attachment interfaces for the SRMs, Propulsion engines and PLF, reference Table 3.2.1.2-1.

3.2 Titan IV Core (Continued)

A propulsive gimballed thrust engine assembly is located on each of the two stages and Stage II has separation retrorockets for Stage II Upper Stage/NUS separation. The avionics components are installed on the Instrumentation Truss or the Guidance Truss located in compartment 2A of Stage II

General structural changes from the Titan 34D to Titan IV Core Vehicle include stretching Stage I and Stage II Fuel and Oxidizer tanks, addition of the Stage II Forward Skirt Extension (CP2490), structural modifications to the instrumentation truss to accommodate the Wideband Instrumentation System (WIS), and general structural enhancements to accommodate increased loads.

Some physical features of the Titan IV/Centaur Flight Vehicle are illustrated in Figure 3.2-1. Titan IV Core performance data is also shown in this figure. The illustration shows seven segment (standard) SRMs used with the Titan IV Booster. The three segment SRMUs are similarly applicable.

Figure 3.2-2 shows the application of the Stage/Step nomenclature found in mission performance descriptions.

3.2.1 Titan IV Core Structures

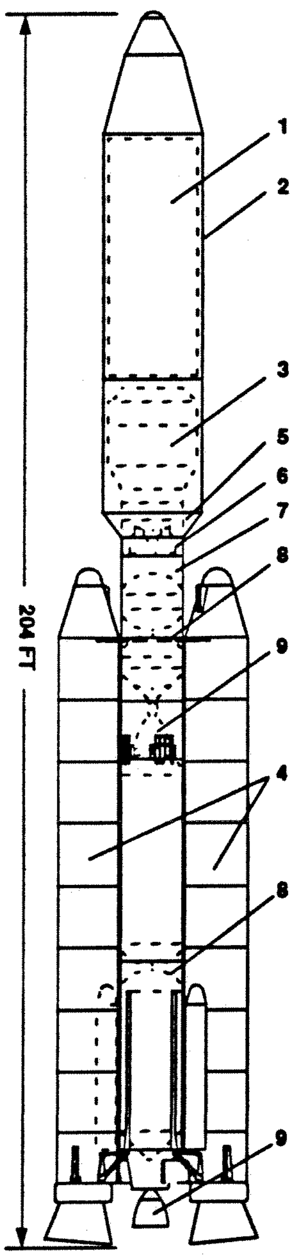
3.2.1.1 Titan IV Airframe

The Titan IV Core Vehicle is essentially an all-aluminum alloy (2014) structure designed to contain propellants and to provide structural support for the two SRMs (or SRMUs) the PLF, and the payload (NUS) or upper stage and payload (Centaur or IUS). Provisions are made for core subsystems including liquid engines, avionics, tank pressurization, cabling, and maintenance access.

Propellant tanks are of welded construction. Step I tank barrels incorporate integrally-machined longerons. Step two tank barrels incorporate integral isogrid. Skirts of skin-stringer-frame construction are welded to tank barrels. The forward oxidizer skirt extension is also of skin-stringer-frame construction.

Step I engine support is provided by an engine truss assembly. Step II engine support is provided by a stringer-reinforced aft fuel tank cone. Step 0 aft support is provided at the aft skirt frame, with additional load being carried through bolt-on longerons. Step 0 forward attachment is through outriggers at compartment 2B.

The Core structural design is based on Centaur loads. Loads at and aft of vehicle station 163 are not to exceed the capability of the Centaur configuration.



LEGEND:

- Titan IV Flight Vehicle
  1) Satellite Vehicle (S/V)
- Titan IV Launch Vehicle (LV)
  2) Payload Fairing (PLF)
  3) Centaur Upper Stage
- Titan IV Boost Vehicle (BV)
  4) Solid Rocket Motors (SRMs, Stage 0
- Titan IV Core Vehicle
  5) Centaur Adapter (2492)
  6) Forward Skirt Ext (2490)
  7) Stage II Forward Oxidizer Skirt (2460)
  8) Stages I and II
  9) Liquid Rocket Engines

FEATURES:

- 7 Segment SRMs
  - SRM Attach at Compt 2B and Base of Stage I
- 3 Segment SRMUs Not Shown
  - SRMU Attach at Compt 2B and Base of Stage I
- Separate Guidance & Instrumentation Trusses
- Stretched Core
  - Stage I – 95.0 In. Longer than 34D
  - Stage II – 17.0 In. Longer than 34D
- Added Forward Skirt Extension and Antennas
- Loads
  $P_{EQ} = 2.5 \times 10^6$ Lbs, Ultimate
- Assembly/Installation of Engines, Black Boxes at Site
- PLF Available in Lengths from 56' to 86' in 10' Increments

Performance Data/Titan IV Core:

	Stage I	Stage II
Thrust, pounds	542,600	103,500
Mixture Ratio	1.92	1.78
Specific Impulse	300.96	315.88

FIGURE 3.2-1 TITAN IV TYPE I FEATURES (CENTAUR FLIGHT VEHICLE CONFIGURATION)

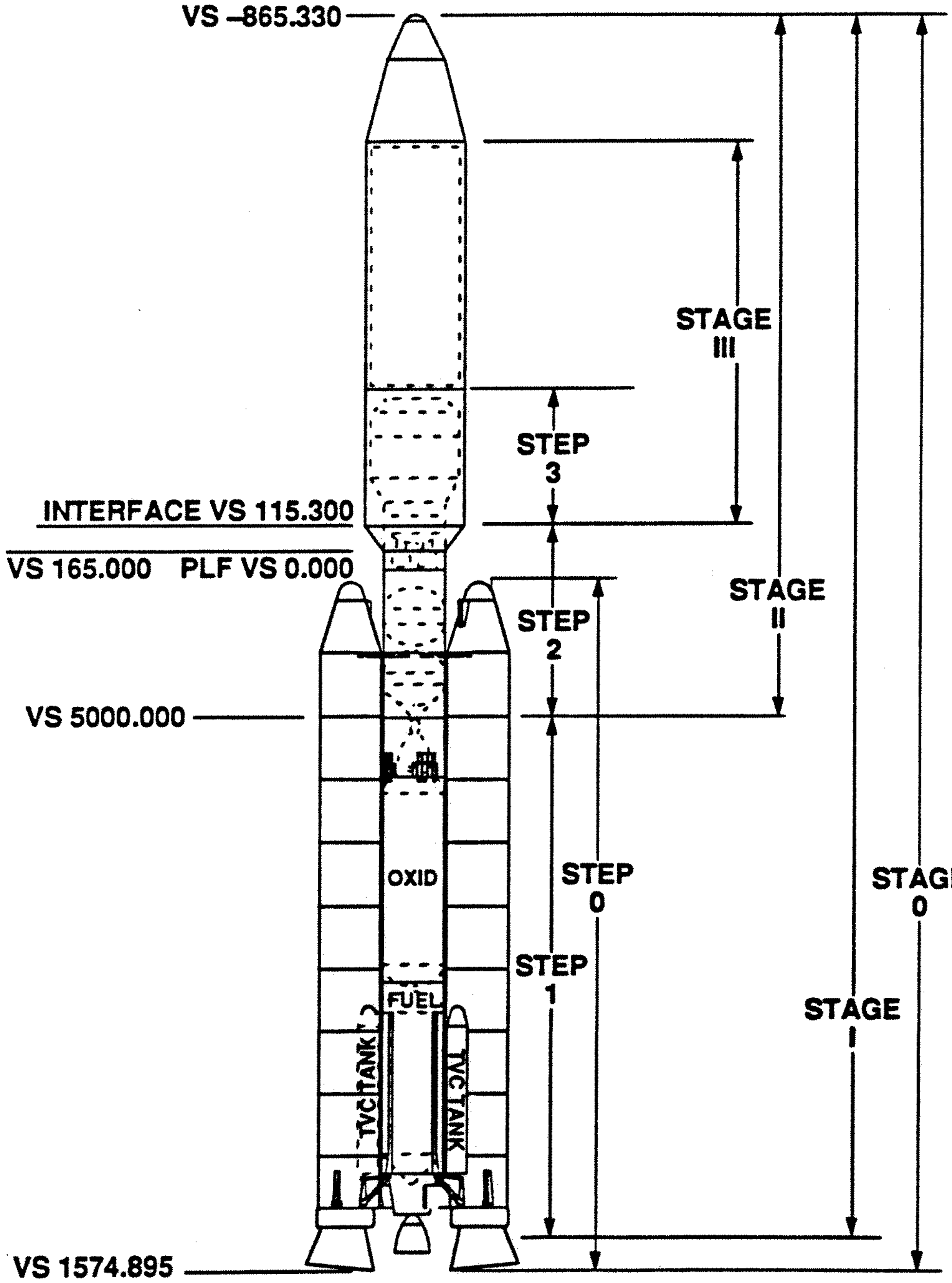


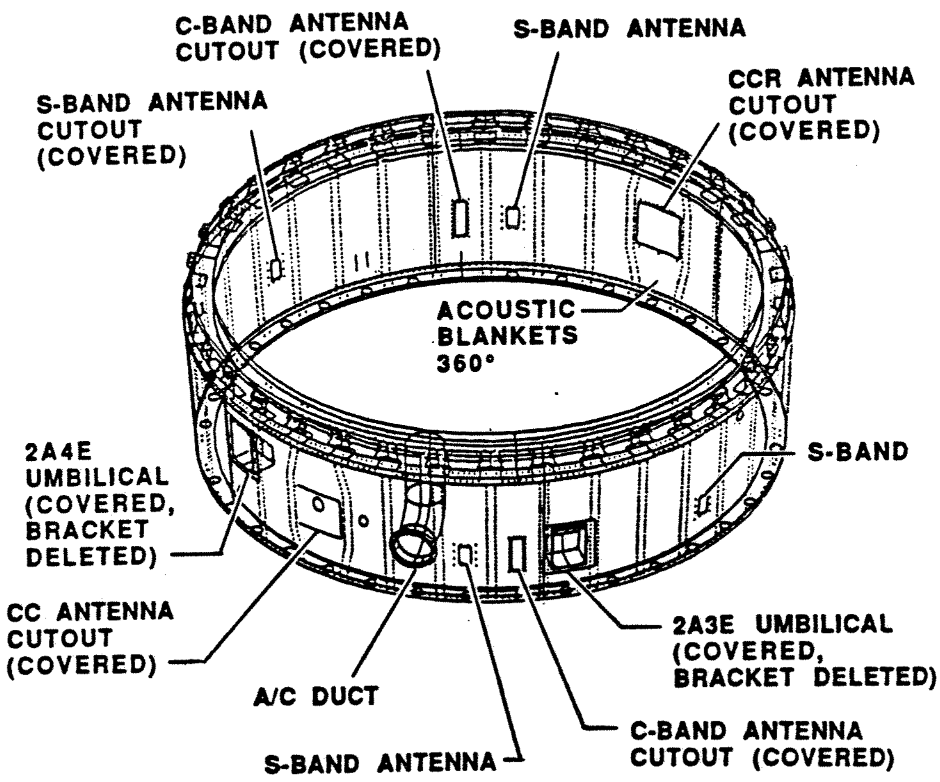
FIGURE 3.2-2 TITAN IV/CENTAUR VEHICLE (STAGE/STEP) MASS PROPERTIES REFERENCE DIAGRAM (401)

3.2.1.2 Titan IV Upper Stage/No Upper Stage Skirts/Adapters

All Titan IV configurations include a Forward Skirt Extension (CP2490). For the Centaur configuration there is a thermal barrier at the Stage II Forward Oxidizer Skirt (CP2460) and there is a Centaur Adapter Skirt (CP2492) between the CP2490 skirt and the Centaur. For the IUS configuration, there is an IUS Adapter Skirt (CP2491) between the CP2490 skirt and the IUS. Martin Marietta provides each of these skirts and the Centaur Thermal Barrier.

For NUS configurations, the spacecraft adapter/skirt between the Titan CP2490 skirt and the SC is GFP.

The CP2490 skirt typically contains antennas (up to eight if a WIS is present). Mission unique acoustic blanket configurations, flight termination system elements for some configurations and other mission peculiar items. Access doors, ducts and electrical cabling components are located per mission requirements as are temporary work platforms, rails and ladders, reference Figures 3.2.1.2-1, 3.2.1.2-2, 3.2.1.2-3, 3.2.1.2-4 and Table 3.2.1.2-1.



NOTE: 2A1E AND 2A2E UMBILICALS ARE LOCATED ON CP2460 (COMPARTMENT 2A)

FIGURE 3.2.1.2-1 FORWARD SKIRT EXTENSION (CP2490) COMMON CONFIGURATION

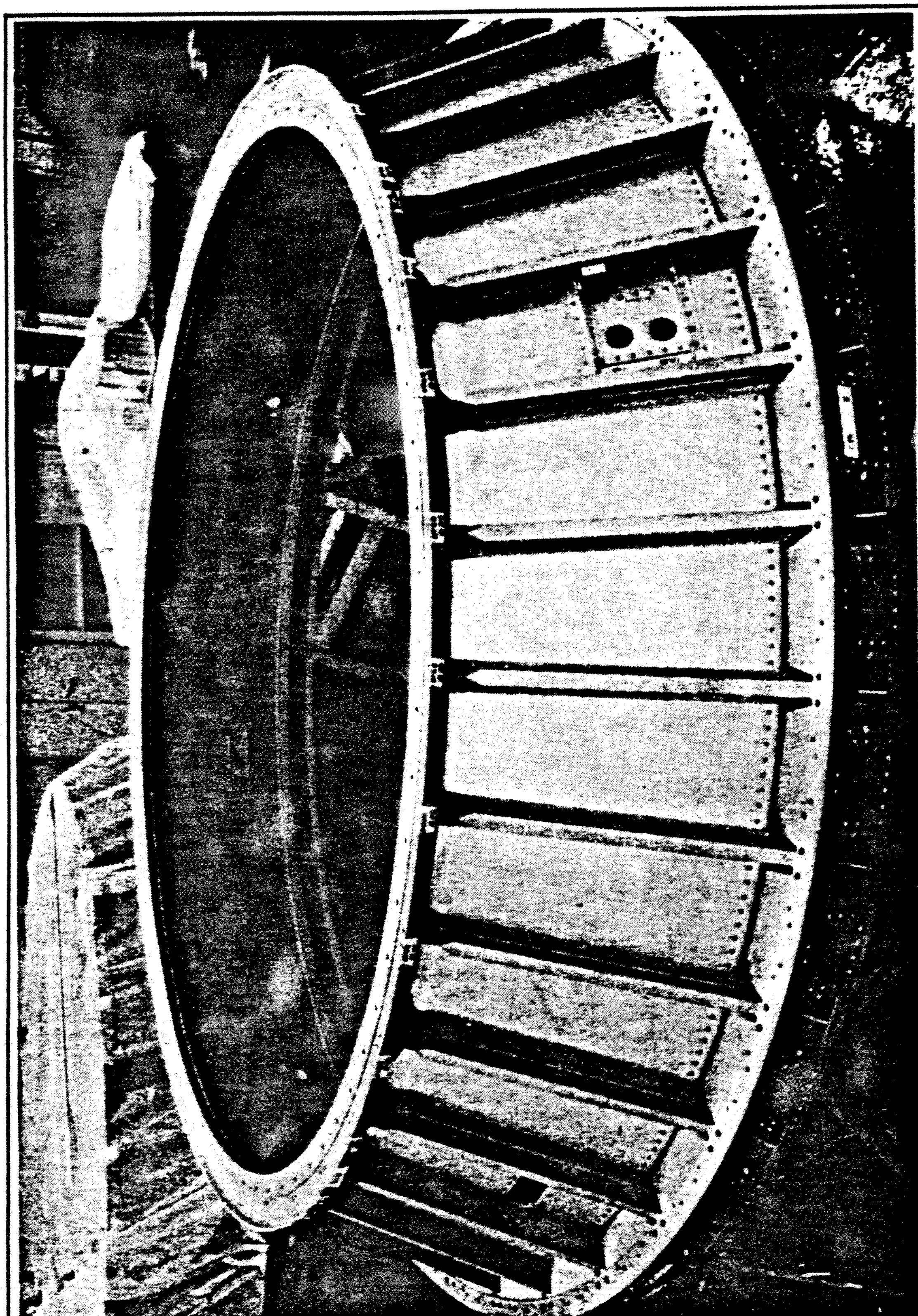
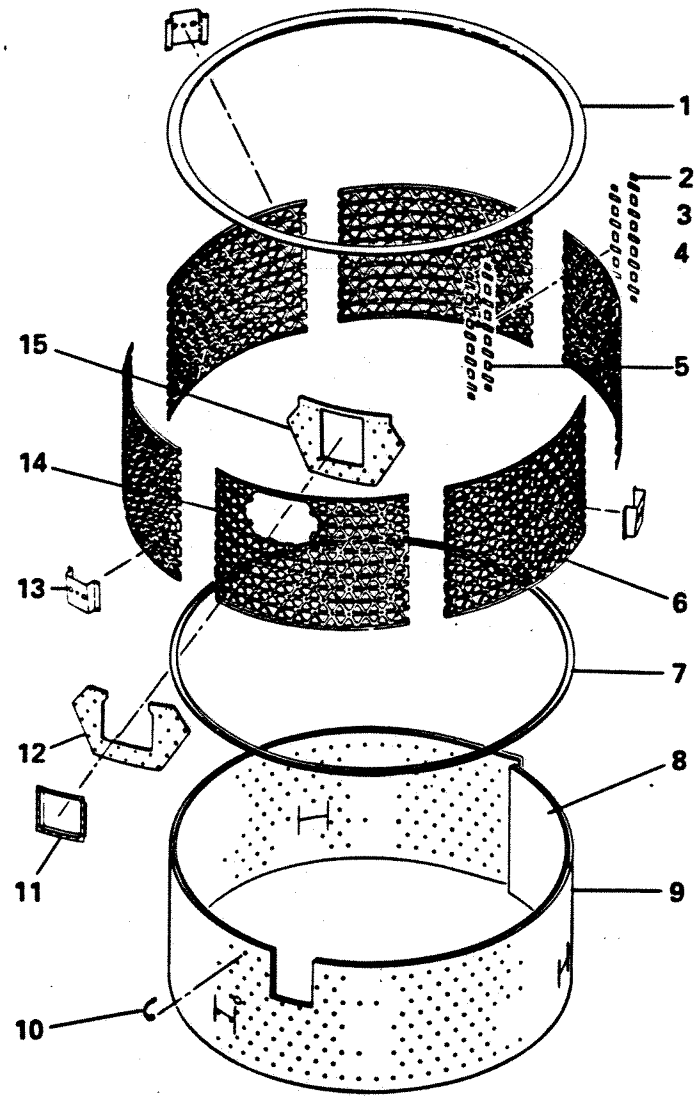
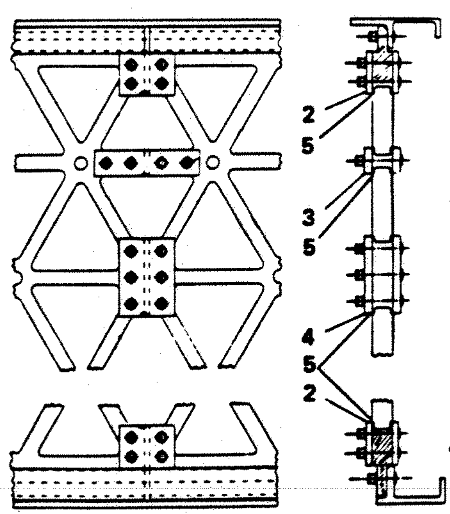


FIGURE 3.2.1.2-2 IUS ADAPTER SKIRT (CP2491)



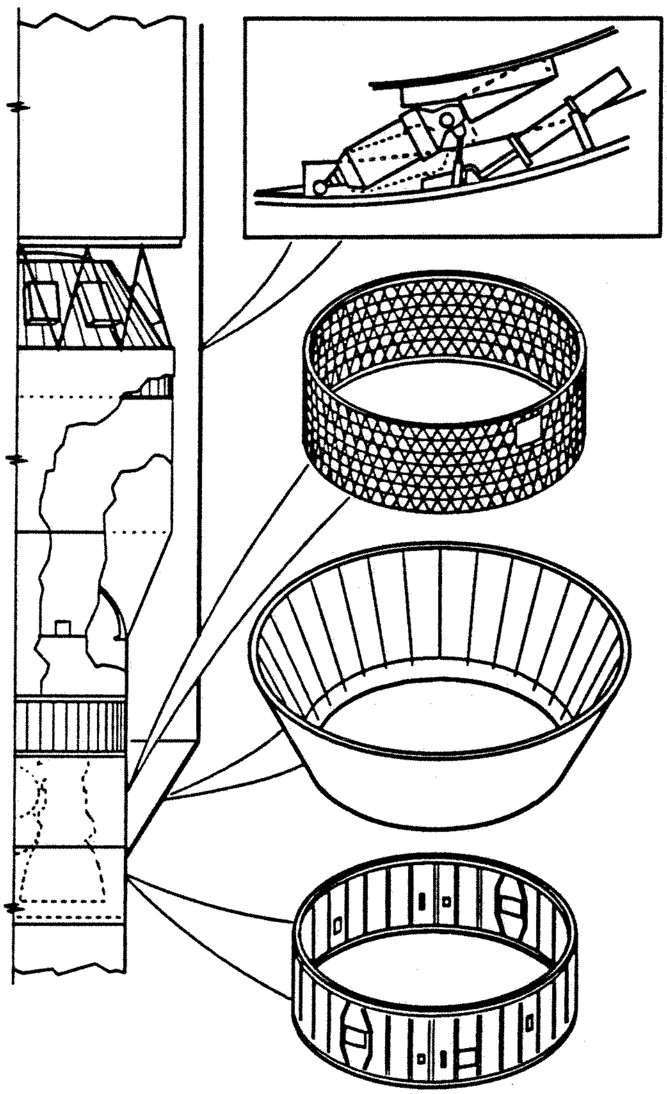
ADAPTER SKIRT, EXPLODED VIEW

FRONT AND CROSS-SECTION VIEWS OF TYPICAL ISO-GRID PATTERN



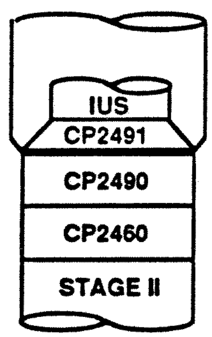
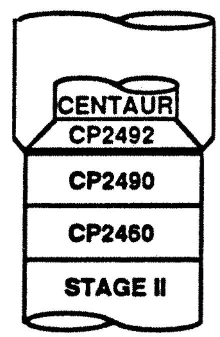
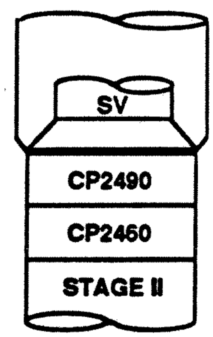
1. Forward Ring Frame
2. Inner and Outer Splice Plates (24 places)
3. Inner and Outer Radial Clip (60 places)
4. Inner and Outer Longitudinal Clip (48 places)
5. Shim (132 places)
6. Common Iso-Grid Panel (5 places)
7. Aft Ring Frame
8. Flow Control Barrier
9. Huygen Effect Barrier
10. Ground Strap
11. T-4 Frame Fitting
12. Outside Doubler
13. Electrical Panel Fairing (3 places)
14. T-4 Panel
15. Inside Doubler

FIGURE 3.2.1.2-3 CENTAUR ADAPTER SKIRT (CP2492) DETAILS



- FORWARD BEARING REACTOR
  - TRANSFERS LOAD TO FAIRING
  - PROVIDES SPACECRAFT CLEARANCE
- TITAN/CENTAUR ADAPTER SKIRT
  - SECONDARY LOAD PATH
  - MINIMIZES LOADS
  - 2492
- FAIRING BOATTAIL
  - PRIMARY LOAD PATH
  - LIMITS LOAD-SHARING CAPABILITY
- TITAN FORWARD SKIRT (2490)
  - FAIRING SEPARATION JOINT
  - LOADS COLLECTION POINT
  - 2490

FIGURE 3.2.1.2-4 TITAN IV STRUCTURAL INTEGRATION

TABLE 3.2.1.2-1 STRUCTURAL COMPARISON OF TITAN IV SKIRT KITS				
SV INTERFACE		IUS	CENTAUR	NUS
BOLT CIRCLE		93.500	121.140	117.645
NO. BOLTS		72	360	72
BOLT SIZE		3/8	1/4	3/8
SV AFT RING PROPERTIES	AREA	PER ICD	PER ICD	PER ICD
	MOMENT	PER ICD	PER ICD	PER ICD
I/F STATION		137.849	115.301	163.000
		 402	 401	 403/405

3.2.1.3 Step 1/Stage II Staging Release Set

This set is composed of studs, gas operated nuts, washer sets, gas pressure cartridges and nut catchers.

The staging release set is operated by a signal sent from the flight programmer. This signal simultaneously fires the staging set and the Stage II engine start cartridge initiator.

3.2.1.4 Titan IV Coordinate Systems

Reference Figure 3.2.1.4-1 Titan IV Coordinate Systems SLC-4E WSMC and Figure 3.2.1.4-2 Titan IV Coordinate Systems LC-41 ESMC.

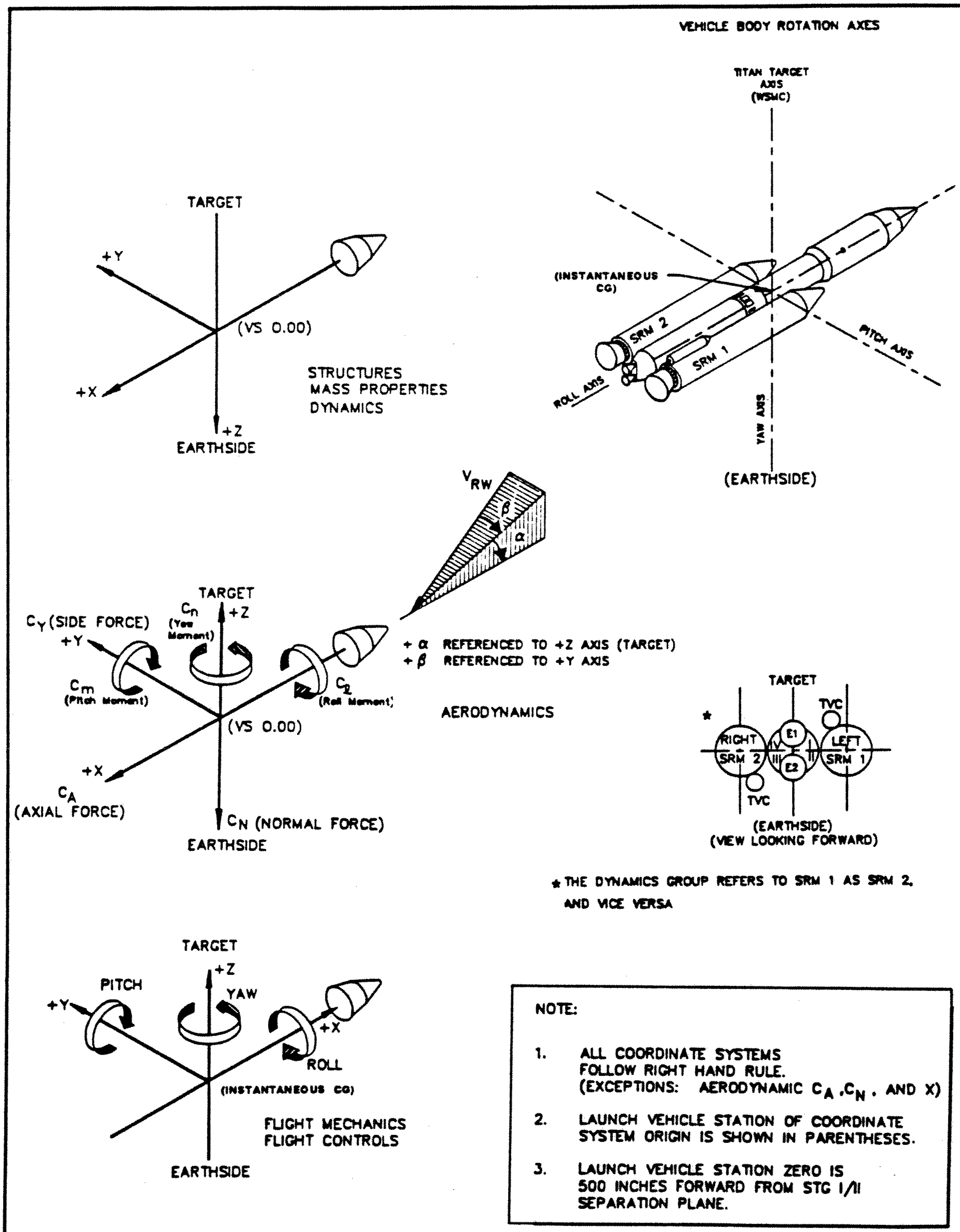
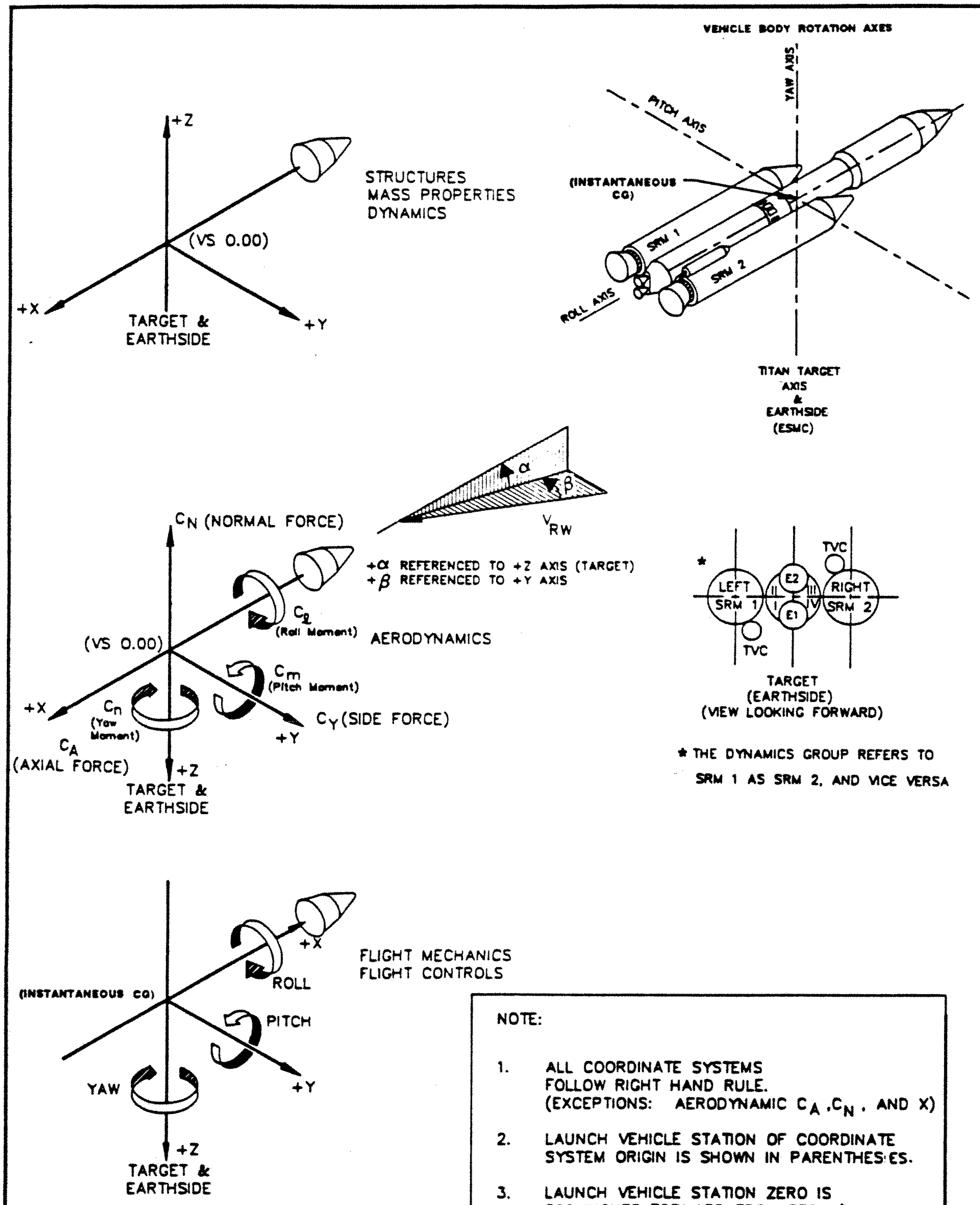


FIGURE 3.2.1.4-1 TITAN IV COORDINATE SYSTEMS (SLC-4E WSMC)



NOTE:

1. ALL COORDINATE SYSTEMS FOLLOW RIGHT HAND RULE. (EXCEPTIONS: AERODYNAMIC C_A, C_N, AND X)
2. LAUNCH VEHICLE STATION OF COORDINATE SYSTEM ORIGIN IS SHOWN IN PARENTHESES.
3. LAUNCH VEHICLE STATION ZERO IS 500 INCHES FORWARD FROM STG I/II SEPARATION PLANE.

FIGURE 3.2.1.4-2 TITAN IV COORDINATE SYSTEMS (ESMC)

3.2.2 Titan IV Stage I/II Propulsion System

3.2.2.1 Liquid Rocket Engines

3.2.2.1.1 Stage I Engine

The Booster vehicle has two Liquid Rocket Engine (LRE) assemblies, one for each stage. The Stage I engine Aerojet LR87-AJ-11) has two subassemblies that operate simultaneously and independently, but under a single/common control system. Each assembly contains fuel and oxidizer pumps that are geared together and are turbine driven. The turbine is powered by a gas generator which uses the same liquid propellants as the main combustion chambers. They receive the propellants from the pump discharge lines. Hydraulic actuators, in response to vehicle guidance commands, vector the thrust of the two combustion chambers to provide pitch, yaw and roll control. Hydraulic power is supplied by a single turbine-driven hydraulic pump mounted on the gearbox of subassembly two. The engine also provides hot gases to pressurize the main propellant tanks during engine operations. Each subassembly is started by a solid propellant start cartridge which drives the turbine until the gas generator "bootstraps" the engine to its steady-state operational level, reference Figure 3.2.2.1.1-1.

3.2.2 1.2 Stage II Engine

The Stage II engine, Aerojet LR91-AJ-11, operates in the same manner as the Stage I engine, except that it has only one combustion chamber. This chamber is hydraulically vectored to provide pitch and yaw control. The gas generator exhaust is ducted through a roll control nozzle, which is also hydraulically vectored to control vehicle roll, reference Figure 3.2.2.1.2-1.

3.2.2.1.3 Propellants

Both Stage I and Stage II engines use liquid hypergolic propellants that can remain aboard in a launch ready state for extended periods of time. The fuel (Aerozine-50) is nominally a 50-50 weight mixture of hydrazine and unsymmetrical dimethylhydrazine (UDMH) and the oxidizer is nitrogen tetroxide (N_2O_4). These propellants are storable at ambient temperature and pressure, thus eliminating temperature conditioning equipment such as is required when handing cryogenic propellants. The hypergolic property (spontaneous ignition upon fuel oxidizer contact) eliminates the need for an ignition system and related checkout and support equipment.

3.2.2.1.4 Engine Shutdown

There are shutdown systems available on Titan stages that allow only guidance command shutdowns, only level sensor shutdowns, only depletion shutdowns or a choice of both a level sensor and a depletion shutdown. The system selected for each stage is a function of the vehicle mission requirements.

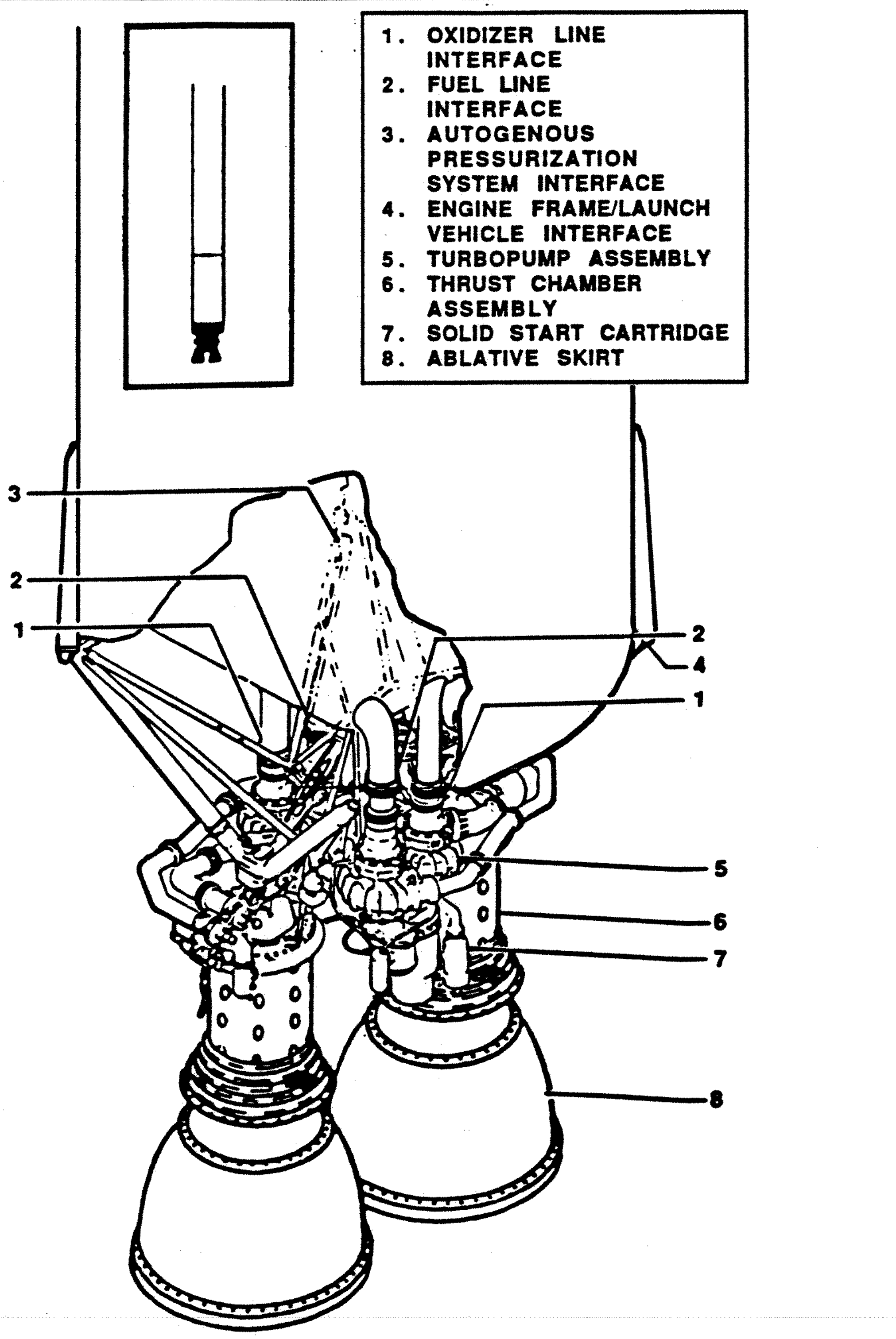


FIGURE 3.2.2.1.1-1 STAGE I ENGINE ATTACH TO FRAME AND LAUNCH VEHICLE

1. FUEL LINE INTERFACE
2. OXIDIZER LINE INTERFACE
3. AUTOGENOUS PRESSURIZATION SYSTEM INTERFACE
4. ELECTRICAL CONTROL HARNESS INTERFACE
5. ENGINE FRAME/LAUNCH VEHICLE INTERFACE
6. THRUST CHAMBER ASSEMBLY
7. ABLATIVE SKIRT
8. SOLID START CARTRIDGE
9. ROLL CONTROL ASSEMBLY
10. ROLL CONTROL/LAUNCH VEHICLE INTERFACE
11. TURBO PUMP ASSEMBLY
12. GAS GENERATOR ASSEMBLY

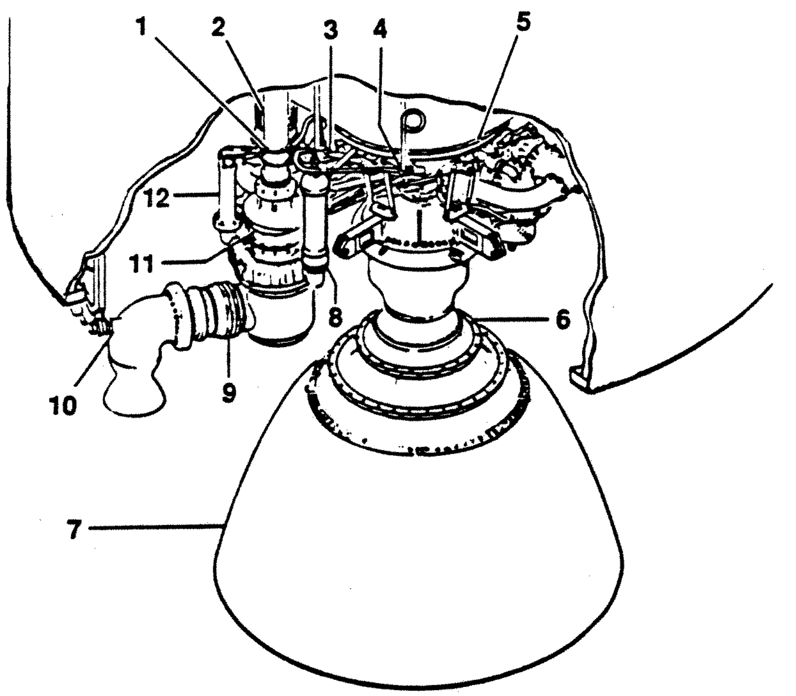


FIGURE 3.2.2.1.2-1 STAGE II ENGINE INSTALLATION

3.2.2.1.4 Engine Shutdown (Continued)

As an aid to the understanding of the propellant weight implications of each type of shutdown system Figures 3.2.2.1.4-1 thru 3.2.2.1.4-4 are included. Figures 3.2.2.1.4-1 and 3.2.2.1.4-2 show one example of the variation in the mean usable Stage I and II propellants for the different shutdown systems. In addition to propellant weight, the user must also consider whether the system requires carrying three-sigma performance margin for the stage when trying to determine what the performance impacts for a certain system will be. For example it is noted that a guidance command shutdown system and a Low Level Sensor Shutdown System (LLSSS) are comparable in terms of mean usable propellant weights, but the level sensor system provides significantly better performance since there is not a requirement to carry additional margin with the stage at the expense of orbit accuracy. It must also be emphasized that these relative propellant weights shown only indicate trends and every case can be different so the relative propellant weights must be assessed on a case-by-case basis. Figure 3.2.2.1.4-3 and -4 show the probabilities of an oxidizer depletion for the shutdown systems.

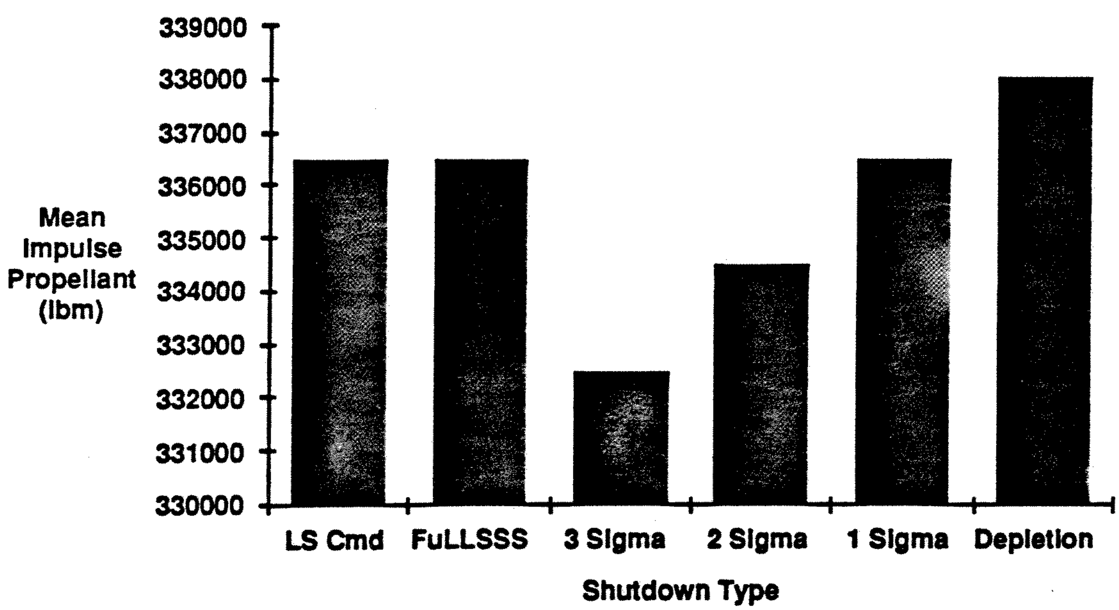


FIGURE 3.2.2.1.4-1 STAGE I PROPELLANT VARIATION WITH SHUTDOWN

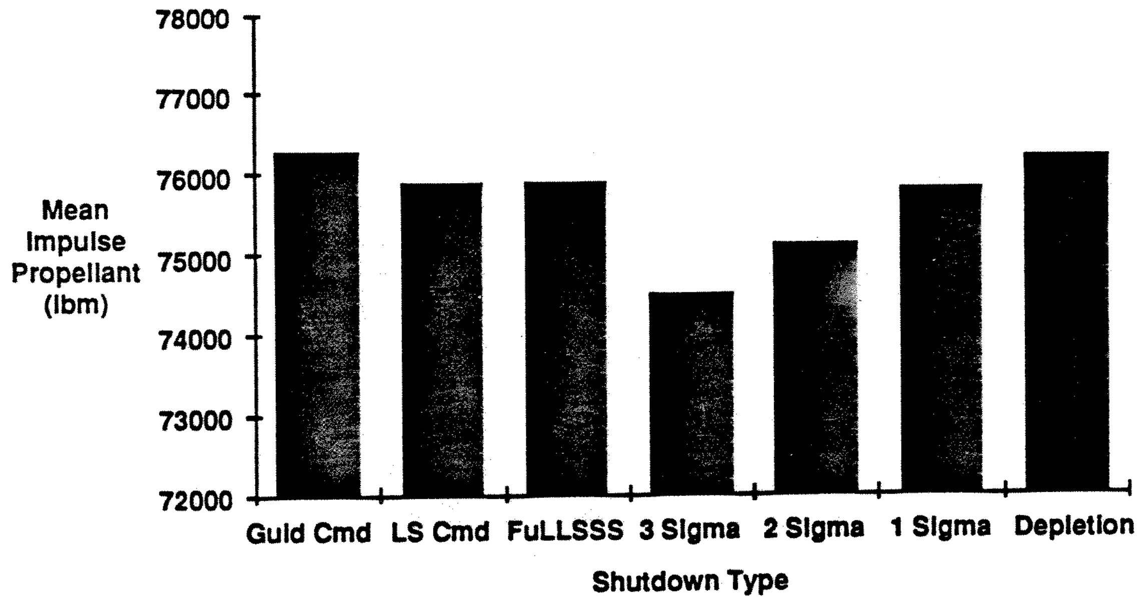


FIGURE 3.2.2.1.4-2 STAGE II PROPELLANT VARIATION WITH SHUTDOWN

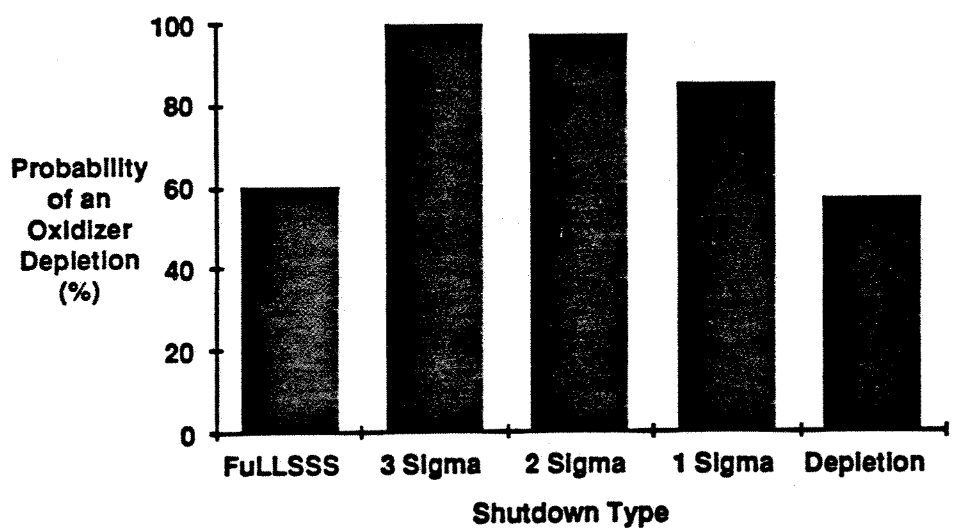


FIGURE 3.2.2.1.4-3 STAGE I OXIDIZER DEPLETION PROBABILITY

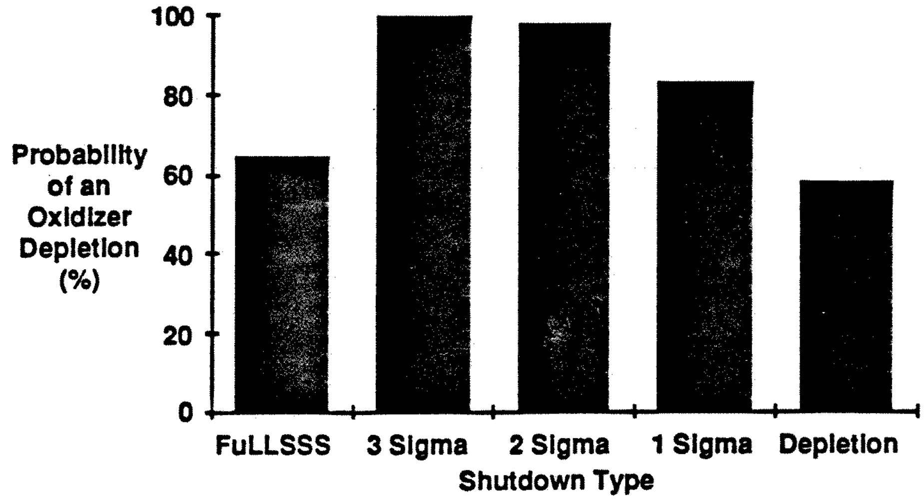


FIGURE 3.2.2.1.4-4 STAGE II OXIDIZER DEPLETION PROBABILITY

3.2.2.2 Retrorockets

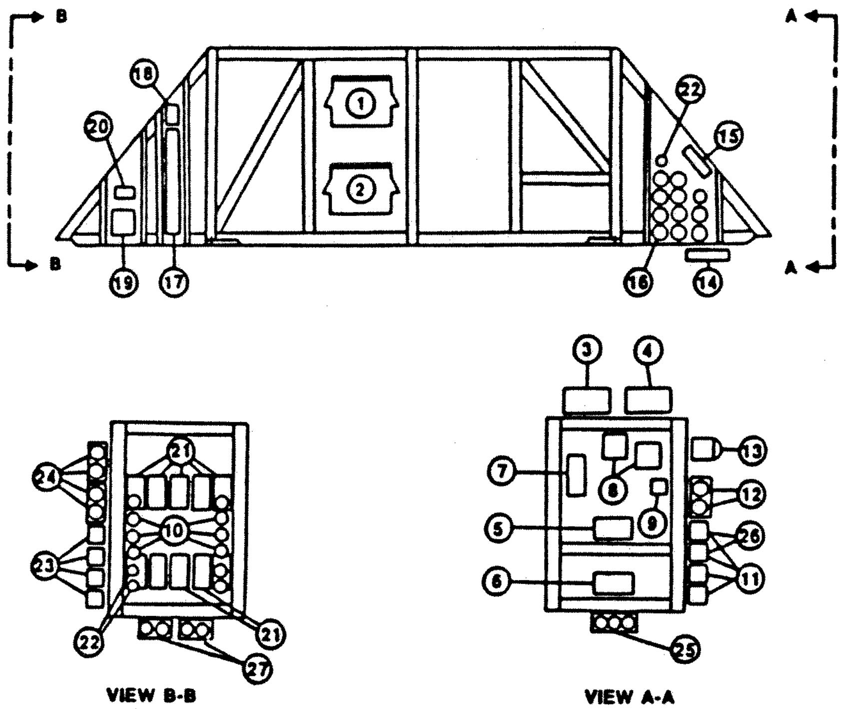
Four retrorocket motors with 440 lb thrust each, are mounted equally spaced around the aft edge of the Stage II Aft Oxidizer Skirt CP2420 with their thrust nozzles positioned forward. These rocket motors are fired to retard Stage II after Upper Stage or SV separation.

To preclude the retrorocket plume from contaminating the upper stage and/or SV on the IUS and NUS configurations, plume deflectors may be employed. When clean retrorockets (i.e., no more than 2% aluminum in the propellant) become available they will be installed to alleviate, or eliminate the plume degradation effects.

3.2.3 Titan IV Core Electrical Systems

3.2.3.1 Avionics

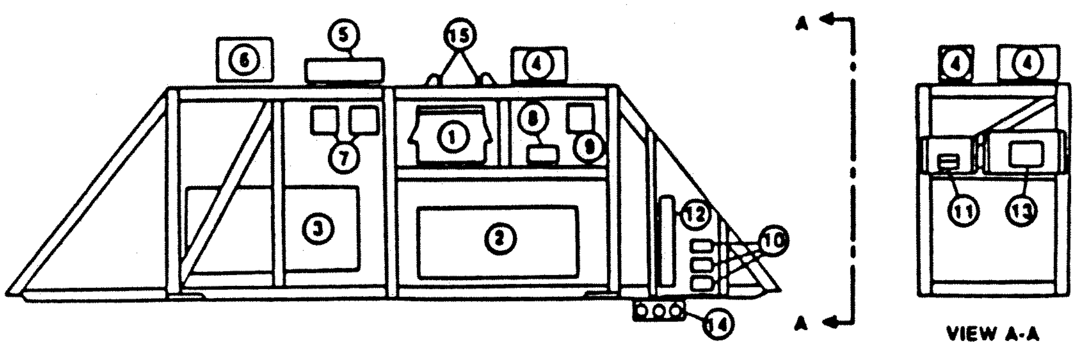
Avionics are utilized for a Flight Control System, a Telemetry Tracking and Command System, an Instrumentation System, a Flight Safety System and the Electrical Power Control System. The avionics hardware for the Titan IV core is located on the Instrumentation and Guidance Trusses in the Stage II compartment 2A (CP2460 Forward Oxidizer Skirt), reference Figures 3.2.3.1-1, 3.2.3.1-2 and 3.2.3.1-3.



1. 40 Ampere-Hour TPS 1 Battery
2. 40 Ampere-Hour TPS 2 Battery
3. 4 Ampere-Hour IPS Battery
4. 4 Ampere-Hour IPS Battery
5. Command Receiver/Decoder 2
6. Command Receiver/Decoder 1
7. S-Band Transmitter
8. Remote Multiplexer Unit (2)
9. PLF and Upper Stage Separation
10. Resistor Assemblies (10)
11. Command Destruct Diode Assembly
12. Command Destruct Resistor Assembly (2)
13. Command Shutdown Relay (2)
14. Hybrid Junction Assembly (08D)
15. RF Telemetry Diplexer (08D)
16. VDU and Signal Conditioners
17. Terminal Board (2)
18. Current Shunt (3)
19. TPS Power Transfer Switch (2)
20. TPS Bus Isolation Resistor (2)
21. PLF and Upper Stage Separation Switches
22. Diode Assembly (3)
23. PLF FBR Release Switch (4)
24. PLF FBR Release Resistor Assembly (4) (06D)
25. Test Connectors
26. Command Destruct Switch (2)
27. Truss to Truss Disconnect (3)
28. WIS Components (10 Flights Only) (Not Shown for Clarity)

FIGURE 3.2.3.1-1 INSTRUMENTATION TRUSS LAYOUT

GUIDANCE TRUSS LOCATED IN COMPARTMENT 2A (CP2460)



1. 40 AMPERE-HOUR VPS BATTERY
2. MISSILE GUIDANCE COMPUTER
3. INERTIAL MEASUREMENT UNIT
4. SEQUENCE RELAY ASSEMBLY (2)
5. FLIGHT CONTROL STATIC INVERTER
6. RMIS CONVERTER UNIT
7. REMOTE MULTIPLEXER UNIT (2)
8. IGS CAPACITOR (10, 606 MICROFARADS)
9. VPS POWER TRANSFER SWITCH
10. CURRENT SHUNT (3)
11. 30 WATT, WIRE WOUND RESISTOR (2)
12. TERMINAL BOARD
13. C-BAND TRANSPONDER*
14. TEST CONNECTORS
15. DIODE ASSEMBLIES AND SIGNAL CONDITIONERS

* FOR IUS/NUS ONLY

FIGURE 3.2.3.1-2 GUIDANCE TRUSS LAYOUT AND COMPONENTS

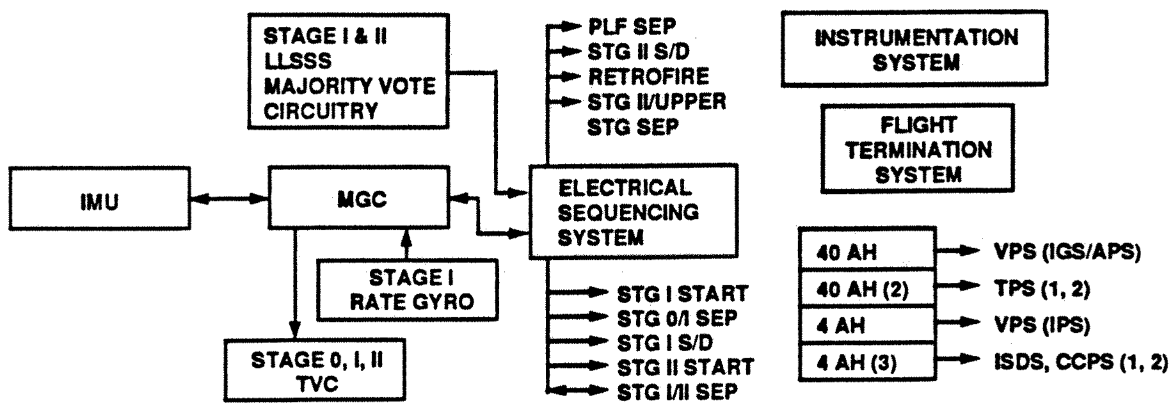


FIGURE 3.2.3.1-3 TITAN IV CORE AVIONICS

3.2.3.2 Flight Control System

The Flight Control System (FCS) consists of an IGS and a Digital Flight Control System (DFCS). The DFCS consists of that part of the Missile Guidance Computer (MGC) which functions as an autopilot, the Stage I Attitude Rate Sensing System, the SRM Thrust Vector Control (TVC) System, the Stage I Hydraulic Actuation System and the Stage II Hydraulic Actuation System.

The IGS guides Stage 0, I and II of the Titan IV vehicle on a planned trajectory and controls Stage II final velocity so that the P/L and powered vehicle arrives at the desired location in space traveling at the desired velocity, reference Figure 3.2.3.2-1.

The Titan IV IGS (by Delco Electronics) is made up of an IMU that consist of a four gimbal platform with gyros and accelerometers and an M-352 MGC. The M-352 is a 16, 384-word, 24-bit length, fixed point, coincident Core memory, parallel machine containing the specific capability to perform the Titan IV mission and vehicle functions.

The FCS software provides a demonstrated flexible launch capability that allows new SC requirements to be changed and compensated for without significant launch delay. The equations that control Titan IV Booster operations from launch to upper stage separation or SV separation are implemented in operational software. The basic boost-phase guidance equations are identical for virtually all configurations so that the trajectory retargeting modifies only software parameters.

3.2.3.2 Flight Control System (Continued)

Because of the broad range of vehicle inertia and center of mass location, bending frequency, fluid slosh and control thrust level conditions, the Titan IV autopilot is designed to accommodate a large range of gains and compensation. During powered flight, autopilot gains are changed linearly every 40 msec with the linear gain slopes being programmed in discrete segments. A filter compensation stage corresponds to each linear gain segment and autopilot filters are changed frequently during the flight by software control to ensure stability and satisfactory performance.

The load-relief channel of the autopilot utilizes derived lateral components of vehicle acceleration caused by wind gusts. The autopilot acts upon these signals to reduce the angle of attack, thereby reducing aerodynamic loads on the vehicle. The load-relief system is normally switched on at 35 to 40 sec and off at 85 sec after liftoff and is controlled by MGC parameters.

Issuance of various time and event-dependent discrete commands establish in-flight sequencing events which are internally programmed and commanded by the computer. Computer backup modes initiated by malfunction detection logic have been programmed to initiate corrective action commands to overcome performance anomalies by reissuing appropriate discrete commands.

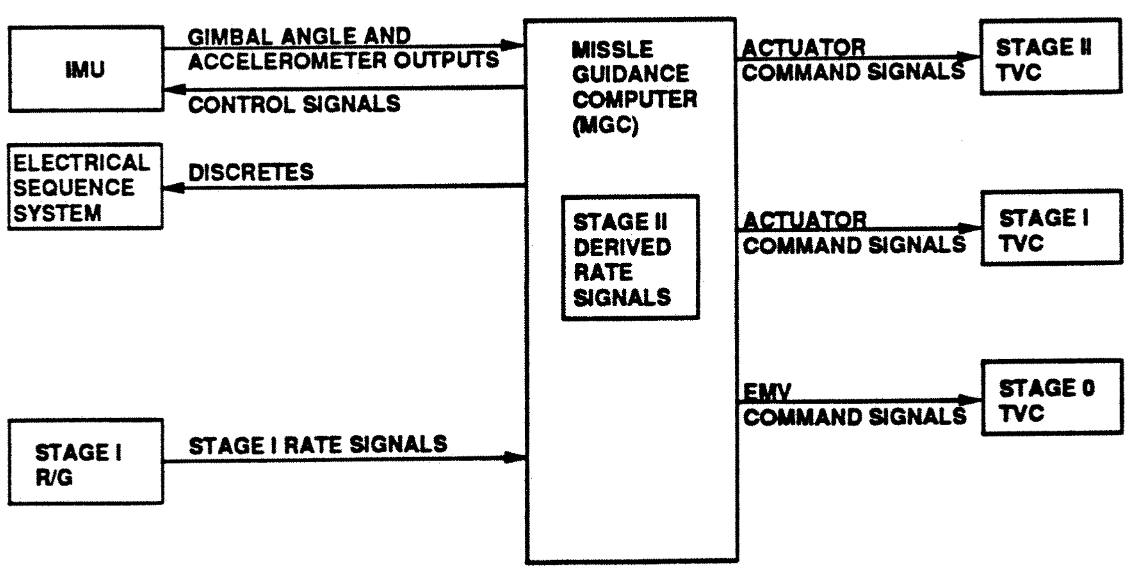


FIGURE 3.2.3.2-1 GUIDANCE AND CONTROL SYSTEM SIGNAL FLOW

3.2.3.3 Instrumentation and Telemetry

The Titan IV Core utilizes an RMIS and can accommodate a WIS.

The Titan IV Vehicle Measurement List, MCR-86-2543, contains RMIS & WIS measurement number allocations which include assignments for payload telemetry measurements.

3.2.3.3.1 Remote Multiplexed Instrumentation System

Titan IV measurements are processed through the RMIS which consists of a Converter Unit (CU) and up to 16 Remote Multiplexer Units (RMUs). The RMUs accept up to 32 analog measurements each, amplifies them and outputs them as Pulse Amplitude Modulated (PAM) signals. Bi-Level or discrete signals are wired directly into the CU.

The CU converts the input signals into a Pulse Coded Modulation (PCM) format and outputs the data as a Nonreturn-to-Zero (NRZ) PCM signal to the telemetry transmitter. The transmitter operates in the S-Band and has a minimum power output of 10 W. The telemetry antennas configuration can vary depending on mission requirements. The RMIS PCM signal is available to the land line system through a vehicle umbilical, reference Figure 3.2.3.3.1-1 IUS RMIS Instrumentation System Block Diagram.

3.2.3.3.2 Wideband Instrumentation System

The WIS is for the purpose of monitoring launch vehicle and payload environments for vibration, acoustic noise and strain measurements. The RMIS channel and frequency capability is not sufficient to accommodate these measurements.

The WIS telemetry signals are transmitted using a dedicated WIS S-Band transmitter and WIS antennas, all located on Titan IV stage II.

The WIS system is installed as a kit at the launch site, reference Figures 3.2.3.3.2-1, 4.4.3.3-3 and 4.5.3.2.1-1.

3.2.3.3.3 Lift-Off Instrumentation System

The Lift-Off Instrumentation System (LOIS) is a system for measuring Titan IV lift-off forcing functions during the first two seconds of the launch environment.

Pressure acoustic, strain and acceleration data from selected vehicle and ground points are obtained using various sensors connected to monitoring/recording ground equipment. The data are then analyzed to verify the validity of the modeled Titan IV lift-off environments, reference Figure 3.2.3.3.3-1.

All LOIS ground equipment is located near the launch pad to minimize cabling lengths.

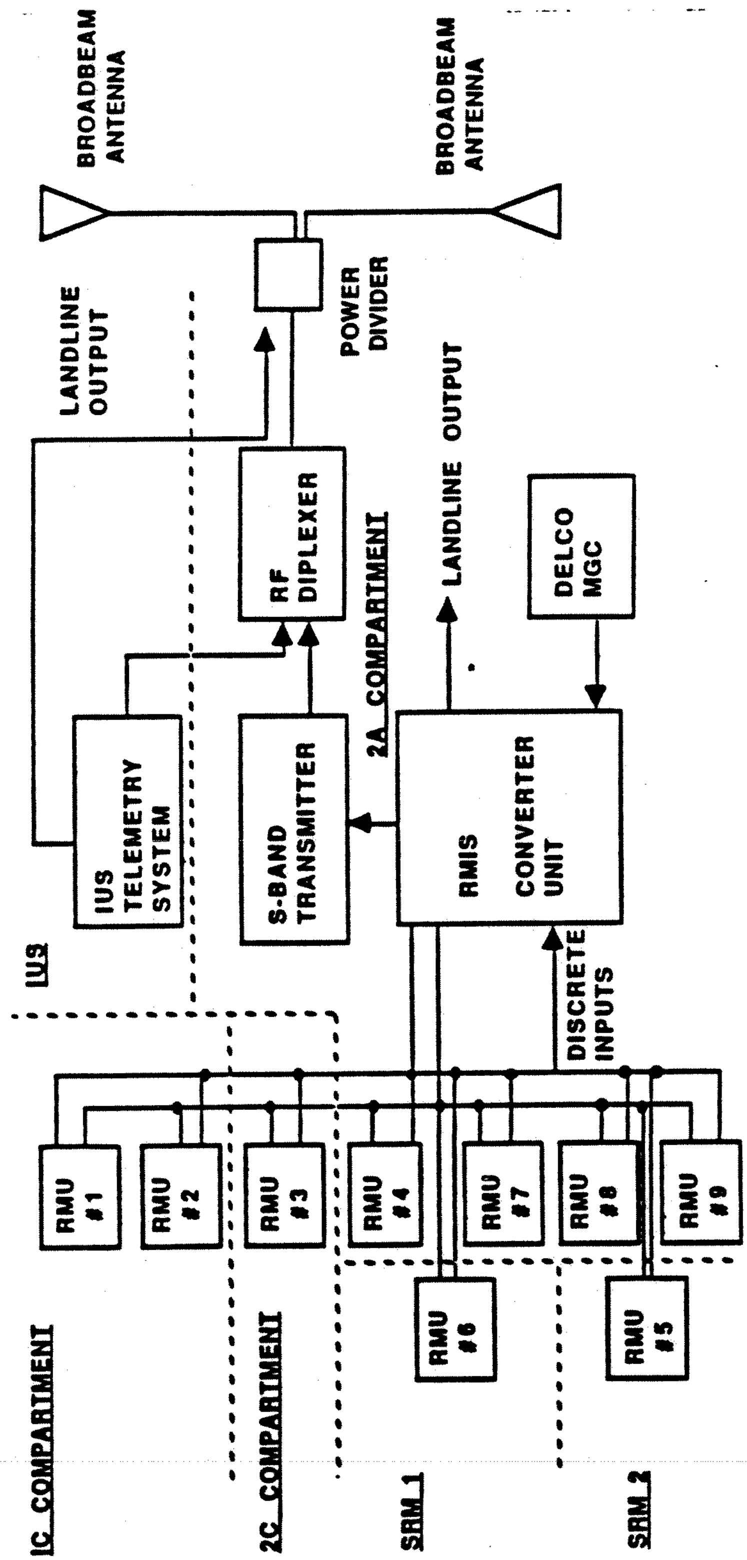


FIGURE 3.2.3.3.1-1 IUS RMIS INSTRUMENTATION SYSTEM BLOCK DIAGRAM

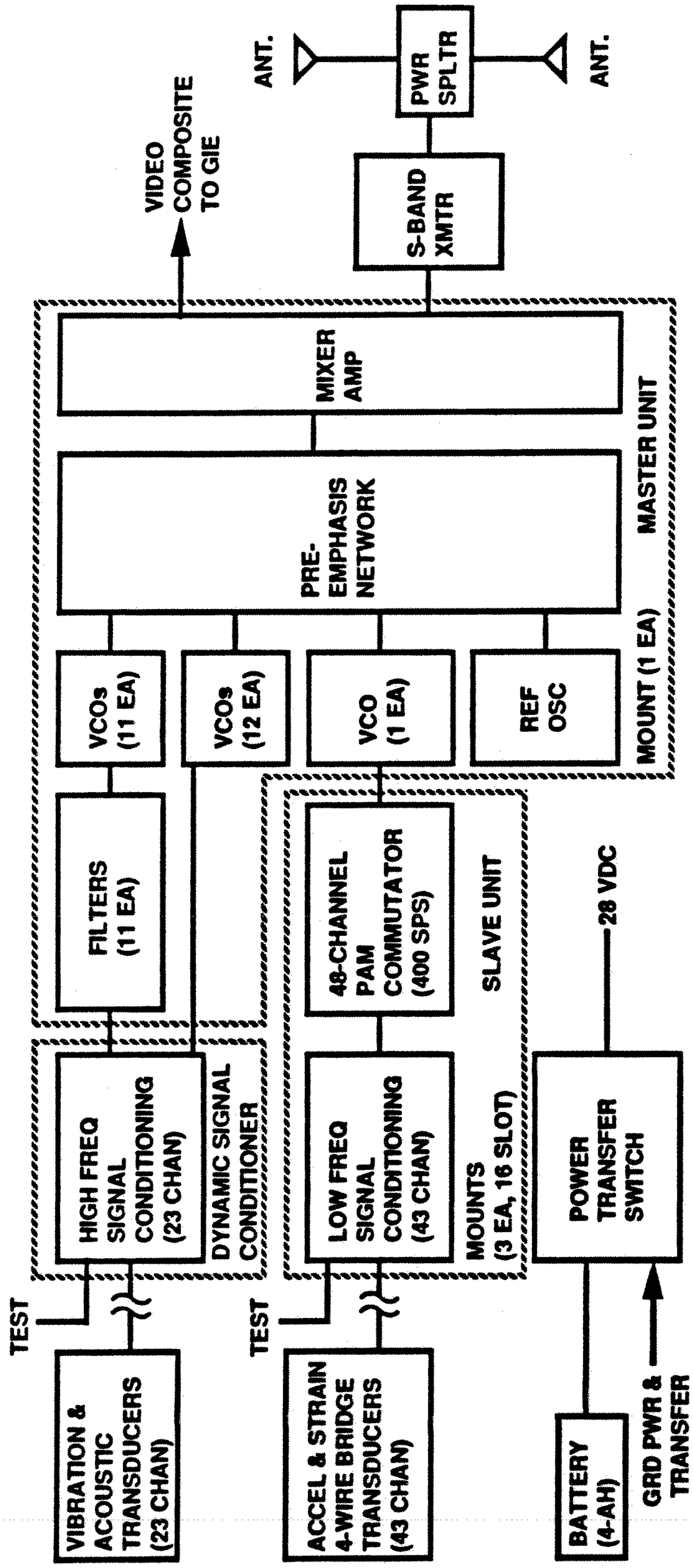


FIGURE 3.2.3.3.2-1 WIS BLOCK DIAGRAM – AIRBORNE

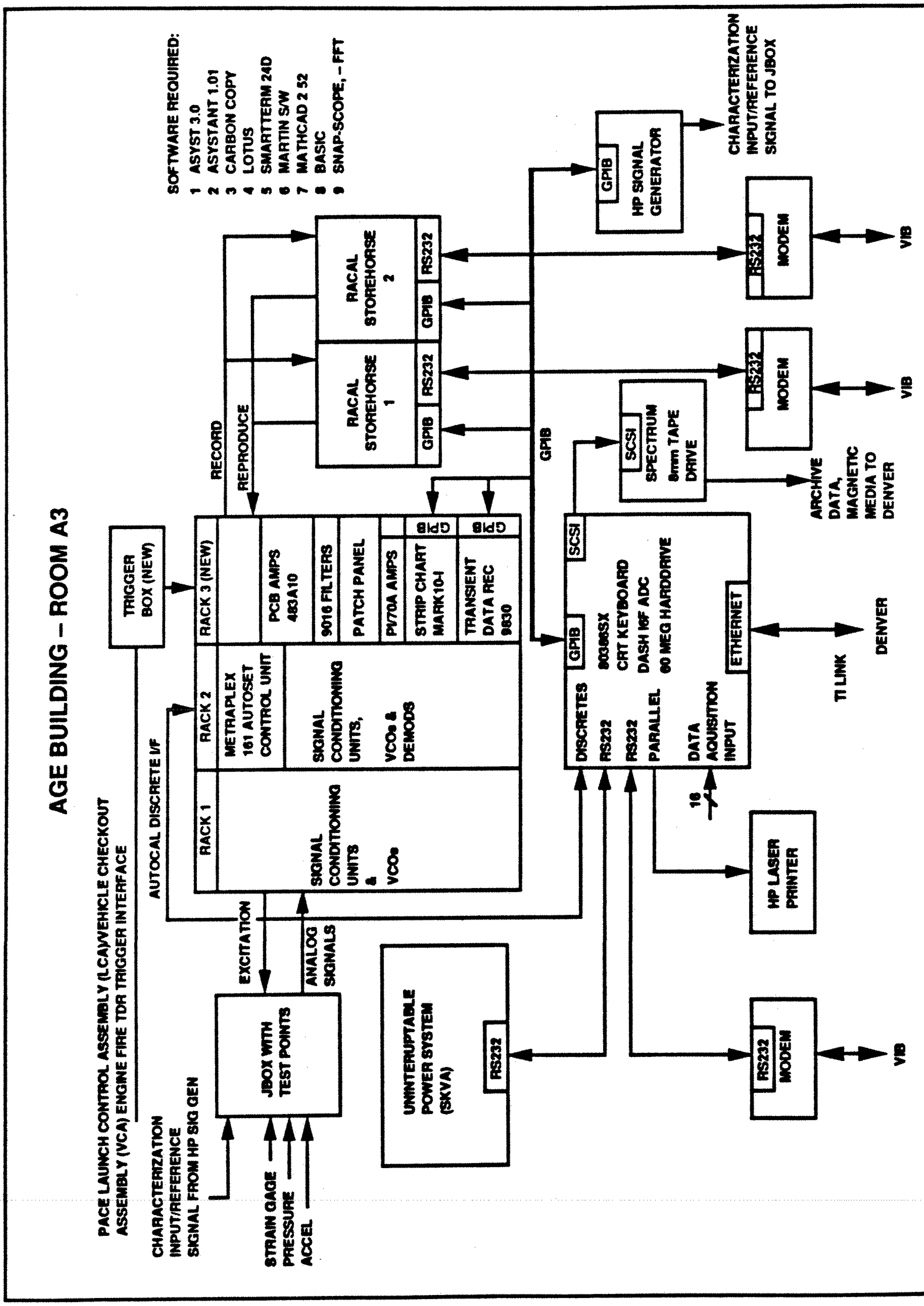


FIGURE 3.2.3.3–1 LOIS REMOTE BLOCK DIAGRAM

3.2.3.4 Tracking and Flight Safety

3.2.3.4.1 Avionics

The Tracking System avionics are mounted in the Titan Stage II guidance truss. This system consists of a C-band pulse beacon transponder, a radio frequency power divider and two antennas. The Titan IV carries a C-band transponder for each of the IUS and NUS flight configurations. The Titan IV/Centaur vehicle carries the flight vehicle C-band transponder on the Centaur Forward Adapter, reference Figure 4.2.3-1.

The Range Safety Officer monitors the vehicle flight path with the tracking data from the transponder and along with other tracking data will determine whether he should take action to command a zero thrust signal to the flight vehicle Flight Termination System (FTS).

3.2.3.4.2 Range Safety Compliance

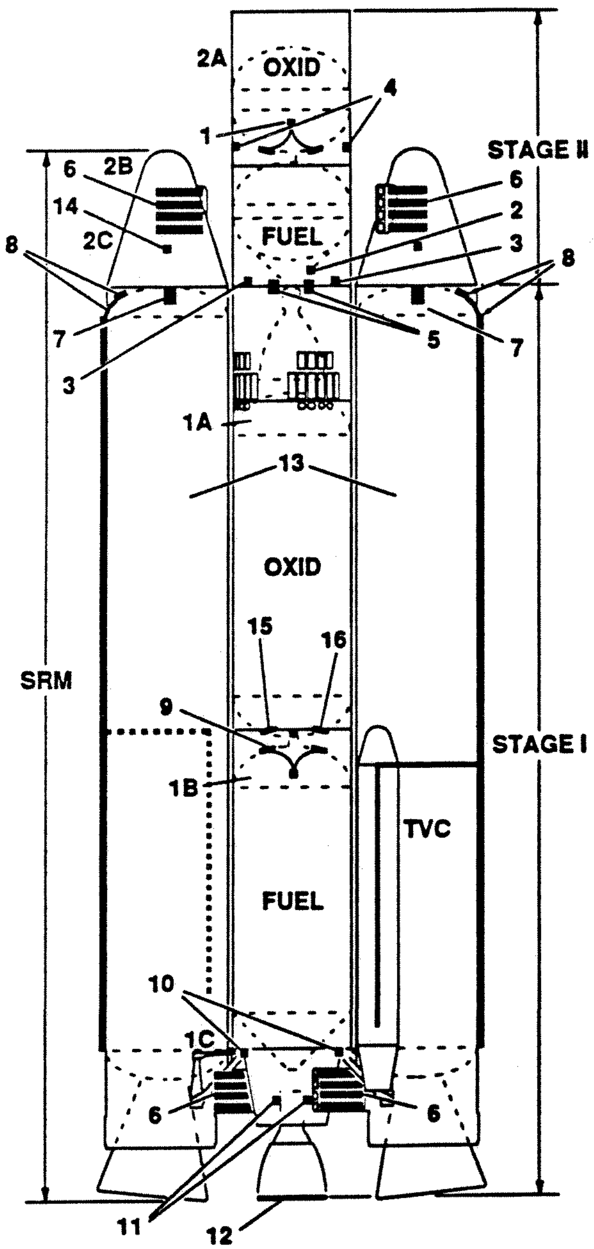
The Titan IV Flight Safety System assures compliance with Range Safety requirements for flight malfunctioning occurrences. The Flight Safety System functions consist of a commanded engine shutdown and/or a commanded vehicle destruct and an Inadvertent Separation Destruct System (ISDS) vehicle destruct action, reference Figure 3.2.3.5.1-1.

The command system includes two Command Receiver/Decoders (CRDs), two antennas for the IUS/NUS configurations, a four port junction, initiators and ordnance to provide the capability for the Range Safety Officer (RSO) to either initiate engine shutdown or engine shutdown and vehicle destruct.

Both CRDs are located in a Titan IV Stage II truss for the IUS and NUS vehicles; and, on the Centaur configuration, two CRDs are located on the Centaur and two on the Titan IV Stage II, reference Figure 4.2.3-1.

Titan IV contains four ISDS circuits. An ISDS circuit is contained in SRM 1, SRM 2, Stage I and Stage II. The system automatically generates destruct signals in the event of vehicle breakup or premature stage separation. The system will destruct the prematurely separated stage, or stages, which cannot be destroyed with the CRDs.

The system is connected in such a way as to make maximum utilization of destruct components provided for command destruct.



*1. STAGE II DESTRUCT PACKAGE (ON OXIDIZER AFT DOME)

2. STAGE II START CARTRIDGE (1 EA)

3. STAGING NUT PRESSURE CARTRIDGES (36 EA)

4. STAGE II RETRO ROCKET MOTORS AND IGNITERS (4 EA)

5. SRM OUTRIGGER STAGING NUT PRESSURE CARTRIDGES (8 EA)

6. SRM STAGING ROCKETS AND IGNITERS (8 PER SRM)

7. SRM IGNITER (2 EA)

8. SRM DESTRUCT PACKAGE (2 EA)

*9. STAGE I DESTRUCT PACKAGE

10. SRM STAGING EXPLOSIVE BOLTS (4 EA)

11. STAGE I START CARTRIDGES (2 EA)

12. STAGE I THRUST CHAMBER EXIT CLOSURE SEPARATION PACKAGES (2 EA)

13. SRM BOOSTERS (2)

14. SRM ISDS DESTRUCT BATTERIES

15. STAGE DESTRUCT BATTERY

16. STAGE I ISDS LOGIC

* DESTRUCT PACKAGE
- INITIATOR (1)
- DESTRUCT CHARGES (2)
- DETONATOR CORD ASSEMBLIES (3)

FIGURE 3.2.3.5.1-1 ORDNANCE ITEM LOCATION

3.2.3.4.2 Range Safety Compliance (Continued)

The ISDS is a hot-wire system. The loss of the 28 VDC (breaking the hot wire) signal in conjunction with loss of ground connections to the ISDS in any stage, results in destruction of that stage. Furthermore, the breaking of the hot wire in Stage II results in a destruct signal, reaching the initiator in Stage I and both SRMs. The breaking of the hot-wire system in either Stage I or an SRM results in a destruct signal reaching the initiator in that particular stage only.

The ISDS in each stage contains a destruct battery, a destruct monitor circuit, an arm/safe switch and two squib firing circuits, reference Figure 3.6.4.3-1 and MCR-87-2563 Flight Termination System Report (TIV/IUS).

3.2.3.5 Electrical Subsystem

The Electrical Subsystem configuration is tailored to fulfill the needs of each of the specific Titan IV configurations and to meet mission unique requirements.

Core Vehicle power hardware includes batteries, cabling and power transfer switches. Seven batteries make up the airborne power sources. Two batteries, one 4 AH and one 40 AH battery, make up the source power for the VPS. This combination supplies the IPS, APS and power for the IGS. The TPS consists of two 40 AH batteries for TPS 1 and TPS 2 and are switched in a redundant manner. This system supplies power controlling the ordnance activities required excluding the ISDS. The Tracking & Flight Safety (T&FS) Power System consists of three 4 AH batteries supplying power for Stage I ISDS, Stage II ISDS and Command Control Power Systems (CCPS-1 and CCPS-2). Stage I and II ISDS use separate 4 AH batteries. The Stage II ISDS battery is co-utilized with the CCPS-2. CCPS-1 has its own 4 AH battery.

The electrical subsystem cable harness attaches to an interconnect harness located in compartment 2B of the Titan IV Core; to the TIV/SV interface panels located on the instrumentation truss; to the avionics components and to the 2A1E and 2A2E electrical umbilicals located on compartment 2A (CP2460).

3.2.3.5.1 Ordnance Events

Ordnance operated events during the vehicle flight, reference Figure 3.2.3.5.1-1.

- SRM Ignition (Ground Operated)
- PLF Forward Bearing Reactor Release (Centaur Only)
- Stage I Engine Start and Engine Nozzle Closure Release
- Stage 0/I Separation and Staging Rocket Ignition
- PLF Unlatch and Separation
- Stage I/II Separation and Stage II Engine Ignition
- Stage II Retrorocket Ignition
- Upper Stage Separation
- Payload Separation

3.2.3.6 Titan IV Launch and Flight Sequence

The countdown sequence is normally a four-phase event and is accomplished during a three-day time period. Phase I consists of LV and space vehicle testing. Phase II consists of ordnance connection with the exception of the destruct initiators, the space vehicle destruct system and the SRM igniter Safe and Arm (S&A) igniter devices and SRM staging motor interfaces.

Phase III consists of LV and space vehicle preparations. Phase IV consists of LV and SV readiness checks, destruct initiator connection, space vehicle destruct system connection and in Titan IV, the SRM igniter S&A device connection and SRM staging motor interface connection, MST removal, final verification and terminal count start.

At T–32 sec, ground power is transferred to vehicle power and the vehicle flight components begin functioning on vehicle power. At T–19 seconds the ordnance bus is energized in preparation to fire the SRM igniters.

At T–4 sec, the IMU Carousal goes inertial. At T–0 sec, the SRM igniters fire and the vehicle lifts off at T+0.25 sec. During the first minute of flight, the ordnance power bus, which is supplied by the TPS which provides power to the ordnance items, is de-energized.

After a little over a minute of flight the ordnance bus is again energized. Just prior to two minutes of flight the SRM ISDS is de-energized and on detection of a sensed longitudinal acceleration of 1.3 Gs (decreasing) the Titan IV Stage I engines are started. The ignition of the Stage I engines forces the expulsion of the engine thrust chamber exit closures from the engine nozzles.

About 10 sec after Stage I ignition, the SRMs aft explosive bolts securing the Core Vehicle to the SRM support trusses and the forward gas pressure cartridges securing the SRM outriggers to the forward Core/SRM attachment fittings are fired simultaneously with the 16 SRM staging motors. The Core Vehicle and the two SRMs become disengaged and the 16 staging motors (four forward and four aft on each SRM) propel the burned out SRM cases away from the flight path of the Core Vehicle and Payload.

For the next three minutes of flight, the vehicle is powered by Stage I of the Core Vehicle. During this time period, the PLF is jettisoned at the time that the free molecular heating constraint is satisfied.

Stage I engines shut down when either engine loses thrust because of propellant depletion. This is detected by either Thrust Chamber Pressure Switch (TCPS) in each engine assembly. On some Titan IV vehicles, a low-level sensor in the Stage I fuel tank will cause the guidance system to generate the engine shutdown command.

3.2.3.6 Titan IV Launch and Flight Sequence (Continued)

After 272 seconds of flight, Stage I is allowed to safe the ISDS. The safe command is sent when engine tail-off is detected. When the Stage I engines shut down, the Stage II Core engine is ignited and the Step I/Stage II separation gas pressure cartridges enable Step I/Stage II separation. Stage II accelerates away from Step I and Step I falls away from the flight path.

Once the vehicle endangers land mass (Europe or Africa), command destruct will no longer be issued. Since the time spent over land by the IUS and NUS configurations is very small, the Flight Termination System (FTS) is not safed. However, the Centaur upper stage is safed by a ground initiated command at the point where Range Safety will no longer take destruct action.

At approximately eight minutes of flight, the Stage II ISDS is safed in preparation for separation from the Upper Stage/SV; and then, on detection of the desired vehicle velocity, the vehicle velocimeter signals to command shutdown of the Stage II engine. The Stage II engine can be shutdown on detection of propellant depletion also.

After a short coast period, the Upper Stage or NUS separation signal is given which causes separation from the Titan IV Step 2 and fires the Step 2 retro-rockets on Step 2 for collision avoidance maneuvers.

3.3 Solid Rocket Motors (Stage 0) Description

Two types of Titan IV systems are defined based on the SRM configuration used for the Launch Vehicles' Stage 0.

The Titan IV Type I employs two standard seven segment SRMs by Chemical Systems Division (CSD) of United Technologies Corporation for the Stage 0.

The Titan IV Type II employs three segment SRMUs by Hercules, Inc. (HI) for the Stage 0. These may also be referred to as the upgraded Stage 0.

Reference Table 3.3-1 for comparison of the SRM/SRMU performance capability, reference Figures 3.3-1 SRM/SRMU Vacuum Thrust Comparison and 3.3-2.

3.3.1 Solid Rocket Motors – Seven Segment (SRM)

3.3.1.1 Characteristics

Titan IV Type I initial liftoff thrust is provided by two identical SRMs which are attached to the Core Vehicle Stages I and II by means of a forward support ring structure and Core attach support arms in the aft support skirt assembly. The two SRMs are mounted 180° apart in the Core Vehicle yaw plane.

TABLE 3.3-1 TITAN IV/7 SEGMENT SRMs AND TITAN IV/3 SEGMENT SRMUs PERFORMANCE CAPABILITY

Titan IV Type I with 7 Segment Solid Rocket Motors			Payload Capability (lbm)	
Contractual Reference	Vehicle Description	Final Mission (nmi x nmi x deg)	Minimum Guaranteed	Current Status (May 1990)
401 Spec (ESMC Launch)	TIV/Centaur	19,323x19,323x0 (Baseline GSO)	10,000	10,030
	TIV/Centaur	509x21,298x63.4 (12-hr Inclined Mission 270° ω_p)	11,500	15,270 (5) (6)
	TIV/Centaur	19,323x19,323x65 (24-hr Inclined Mission)	N/A	10,272
402 Spec (ESMC Launch)	TIV/IUS	19,323x19,323x0 (Baseline GSO)	38,780 Throw Wt to 80x95x28.6 Park Orbit	5,078
403 Spec (WSMC Launch	TIV/NUS	100x100x90	30,100	30,892
405 Spec (ESMC Launch)	TIV/NUS	80x95x28.6	39,100	38,803

Titan IV Type II with 3 Segment Solid Rocket Motor Upgrade

401 Spec (ESMC Launch)	TIV/SRMU/Cent	19323x19323x0 (Baseline GSO)	12,700 (3)	12,597 (4)
402 Spec (ESMC Launch)	TIV/SRMU/IUS	80x95x28.6	47,000 (1)	47,893
403 Spec (WSMC Launch)	TIV/SRMU/NUS	100x100x90	38,800 (1)	38,524 (4)
405 Spec (ESMC Launch)	TIV/SRMU/NUS	80x95x28.6	47,800 (1)	48,247 (4)

(1) Preliminary non-contractual values
(2) Payload weight above IUS
(3) Including Centaur structural modification required for payloads above 11,500 lbm
(4) Current best-estimates which will be revised as the design analyses mature
(5) TIV/Centaur 12 Hour inclined payload capability does not include launch window contingency
(6) Does not include Centaur structural modification for payloads above 11,500 lbm

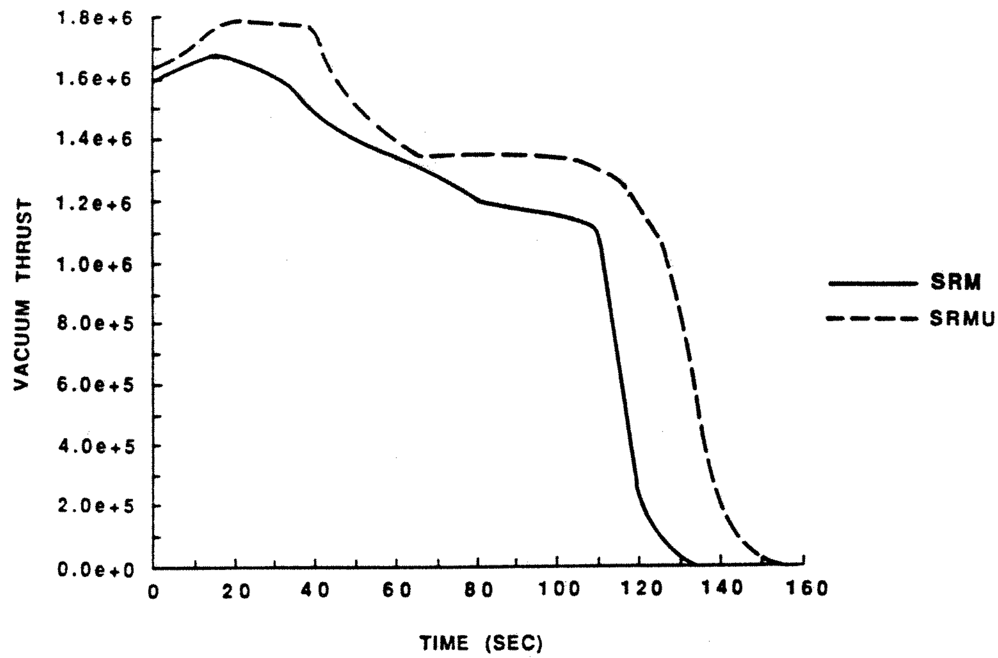


FIGURE 3.3-1 SRM/SRMU VACUUM THRUST COMPARISON

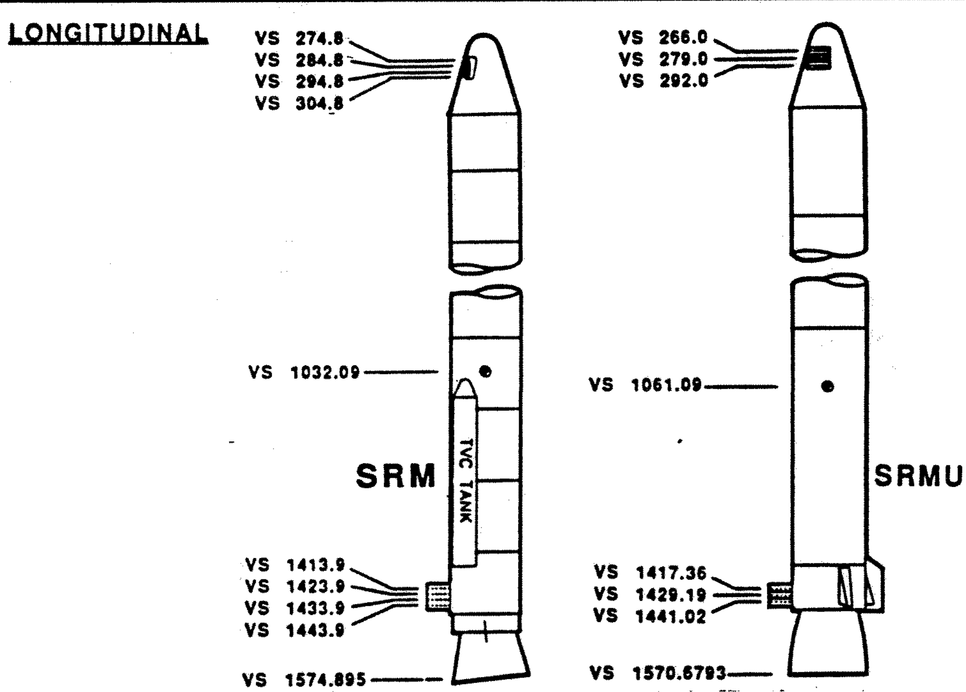


FIGURE 3.3-2 SRM/SRMU 0/I STAGING - STAGING MOTOR/CG RELATIONSHIP

3.3.1 Solid Rocket Motors – Seven Segment (SRM) (Continued)

Each SRM is 112.9 ft long in a static condition, the motor length increases less than one inch after ignition due to pressurization. In the dynamic condition, the length is about 114.8 ft. The SRM diameter is 10 ft. The dry weight of each SRM is 94,941 lb and the loaded weight is 687,052 lb. The propellant weight is 591,111 lb [Reference MCR-85-2518 (Centaur), MCR-86-2518 (IUS), MCR-86-2575 (403) and MCR-87-2617 (405) Mass Properties Report for each referenced configuration].

The average vacuum thrust is 1.394 million lb (web action time divided by web action time vacuum nozzle centerline impulse). The guaranteed vacuum specific impulse is 272.0, reference Figure 3.3-1.

3.3.1.2 SRM/Core Attachments

The Core Vehicle mates to the SRM aft skirt support arms on ball-type mounts supported by the tripod-type arms. The ball mount fits into the aft frame of the Core below each longeron and is held in place by an explosive-type frangible staging bolt. These explosive bolts are fired during the staging sequence, thus breaking the SRM/Core connection, reference Figure 3.3.1.2-1.

The forward Core/SRM support ring provides attachment points for the hardware that secures the Core Vehicle to the upper end of the SRM. The support ring accommodates the connections for the outrigger struts, thrusters and shear-tie locator fittings.

The forward outriggers are attached to the Core by a stud and two ordnance operated nuts. The thrusters mount between the SRM support ring and an outrigger. A thruster is comprised of a spring housed inside two telescoping tubes. The spring is compressed when the outrigger is connected to the Core and SRM. When the squib operated gas nuts are fired during the staging sequence, releasing the outriggers from the Core, the thrusters swing the outriggers out away from the Core.

3.3.1.3 Staging Motor System

The requirement for this system is to provide a lateral staging impulse to each SRM for movement away from the Core Vehicle. This is accomplished by eight staging motors, four in the nose section and four in the aft section.

The thrust of the staging motors is offset relative to the booster centerline in order to impart a rotation to the SRM. When fired simultaneously the staging motors provide an impulse to rotate and move the expended SRMs away from the Core Vehicle.

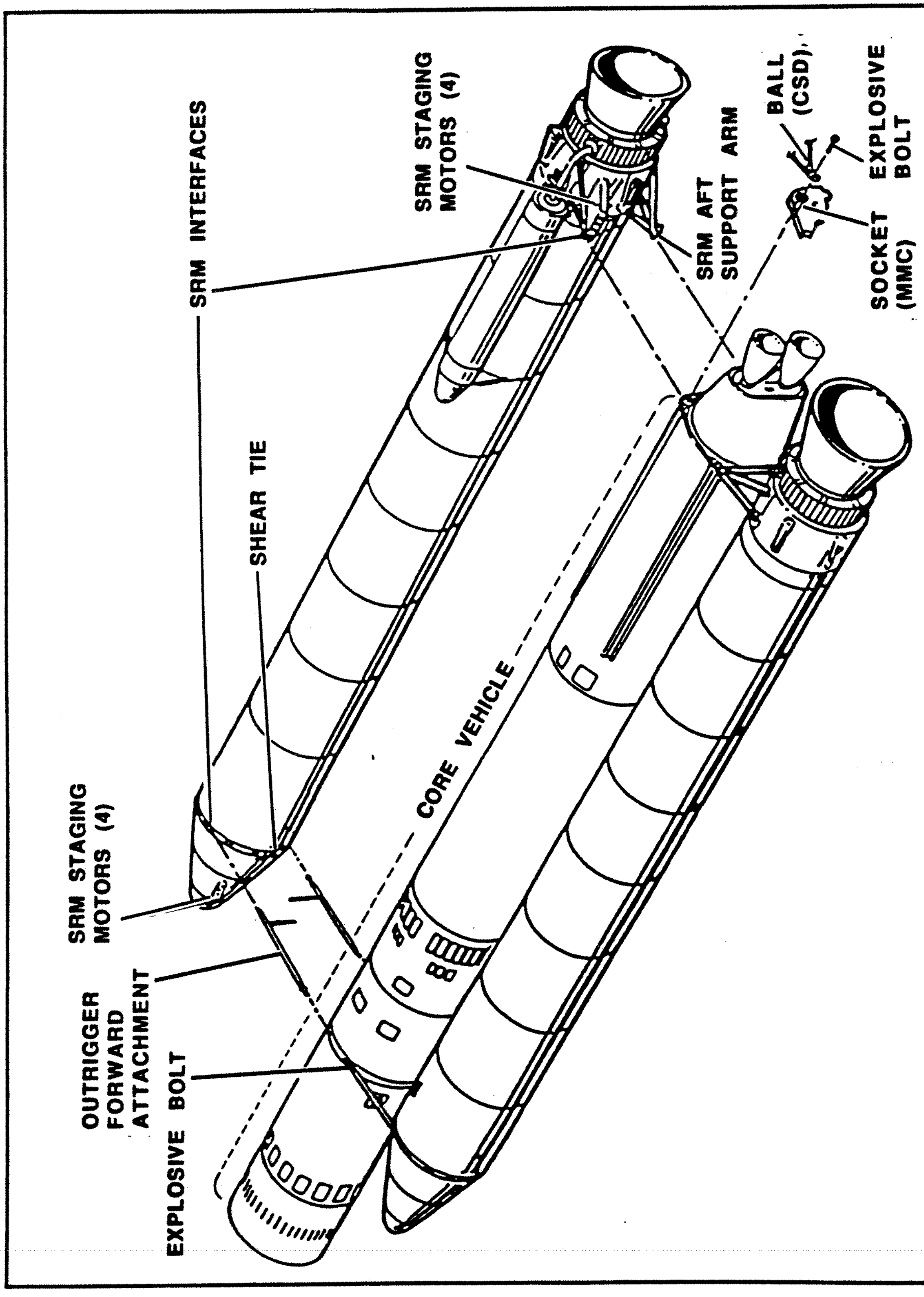


FIGURE 3.3.1.2-1 SRM INTERFACES WITH CORE VEHICLE

3.3.1.4 Thrust Vector Control System

The TVC is used to maintain a proper vehicle flight path and orientation during the Stage 0 portion of the mission trajectory via signals from the Core Vehicle guidance/computer.

The desired flight profile is achieved by exerting side forces on each SRM by injecting a flow of gas, at a specified angle, at a defined velocity and for a calculated time into the SRM gas flow stream through the nozzle. The SRM nozzles are at a fixed 6 deg cant angle and do not gimbal.

3.3.2 Solid Rocket Motors Upgrade (SRMU) - Three Segment

The Titan IV Type II SRMU assemblies are mounted to the Titan IV Core in an arrangement similar to the Titan IV Type I SRM configuration. The structural interface and envelope requirements are presented in ICD-TIV-16001 and ICD-TIV-16003.

3.3.2.1 Characteristics

The SRMU is 112.385 ft long and has a 126 in. nominal inside core diameter. The total nominal weight of each SRMU is 770,673 lb, of which 688,853 lb is propellant, reference Figure 3.3.2.1-1 and 3.3.2.1-2.

The vacuum thrust at 60° F and 26 sec after ignition fire signal is 1,783,080 lbf. The delivered minimum vacuum specific impulse is 284.33 lbf-sec/lbm. Rocket motor performance/operation has been calculated to be satisfactory up to 300,000 ft altitude.

3.3.2.2 Staging Motor System

The purpose of the staging rockets is the separation of the expended SRMUs from the Core Vehicle. They provide a lateral impulse to the SRMUs when a staging electrical command is received from the Core. The SRMUs separate with a "nose-out" motion relative to the Core Vehicle and in manner such that the staging rocket plume does not directly impinge on the Core structure or the PLF.

There are six staging rockets, three forward in the nose cone assembly and three in the aft skirt assembly (for each SRMU).

3.3.2.3 Thrust Vector Control System

The TVC system is of a redundant configuration that includes two turbocentrifugal pumps, two gas generators and two servoactuator assemblies.

The TVC system is functional within 2 sec after the TVC enable signal is issued. After the system has been initiated, the Titan IV Core Vehicle commands the nozzle to an off-null position that does not exceed ± 5 deg in pitch and 1 deg inboard or 6 deg outboard in yaw.

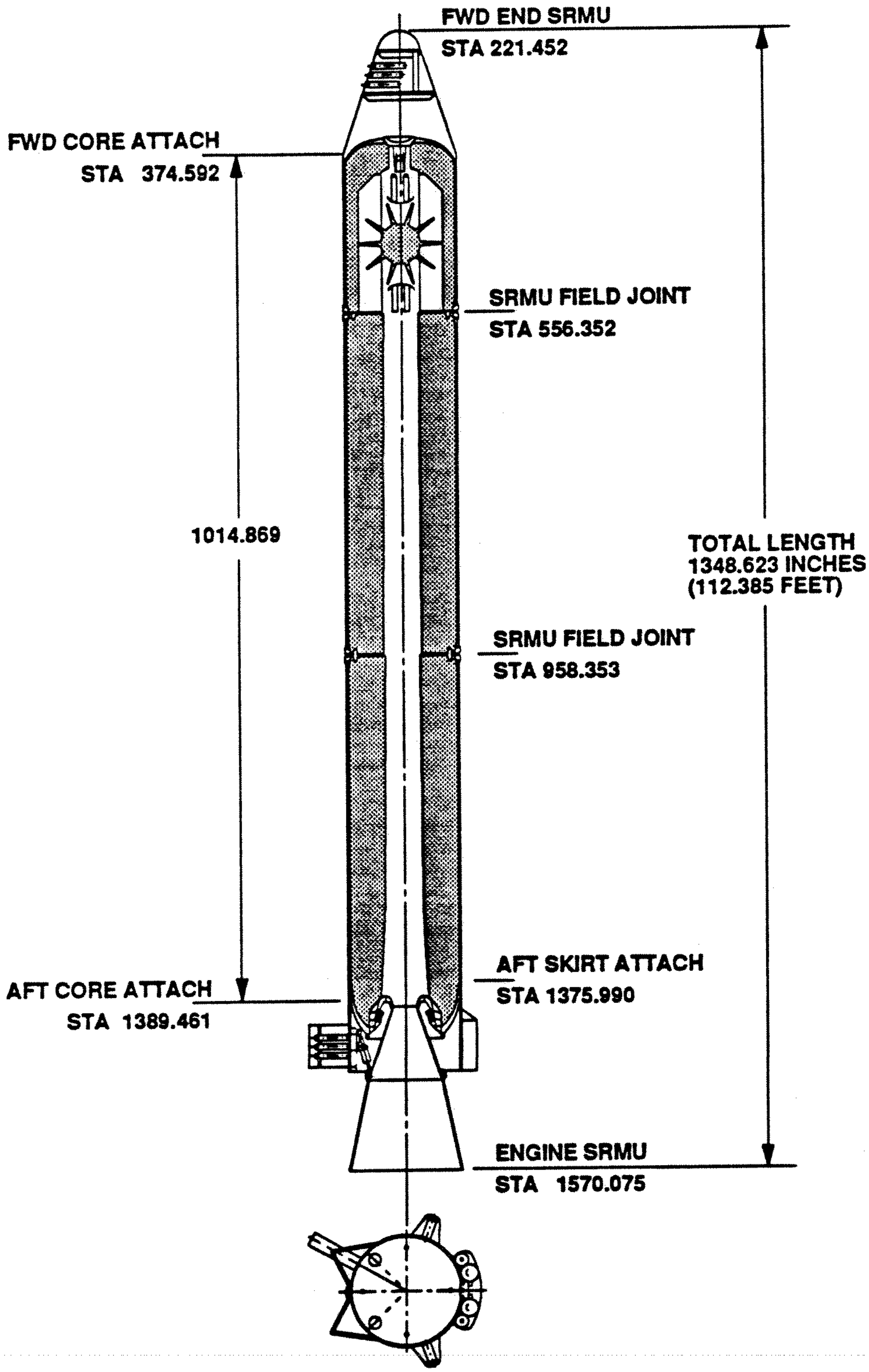


FIGURE 3.3.2.1-1 SRMU GENERAL CONFIGURATION

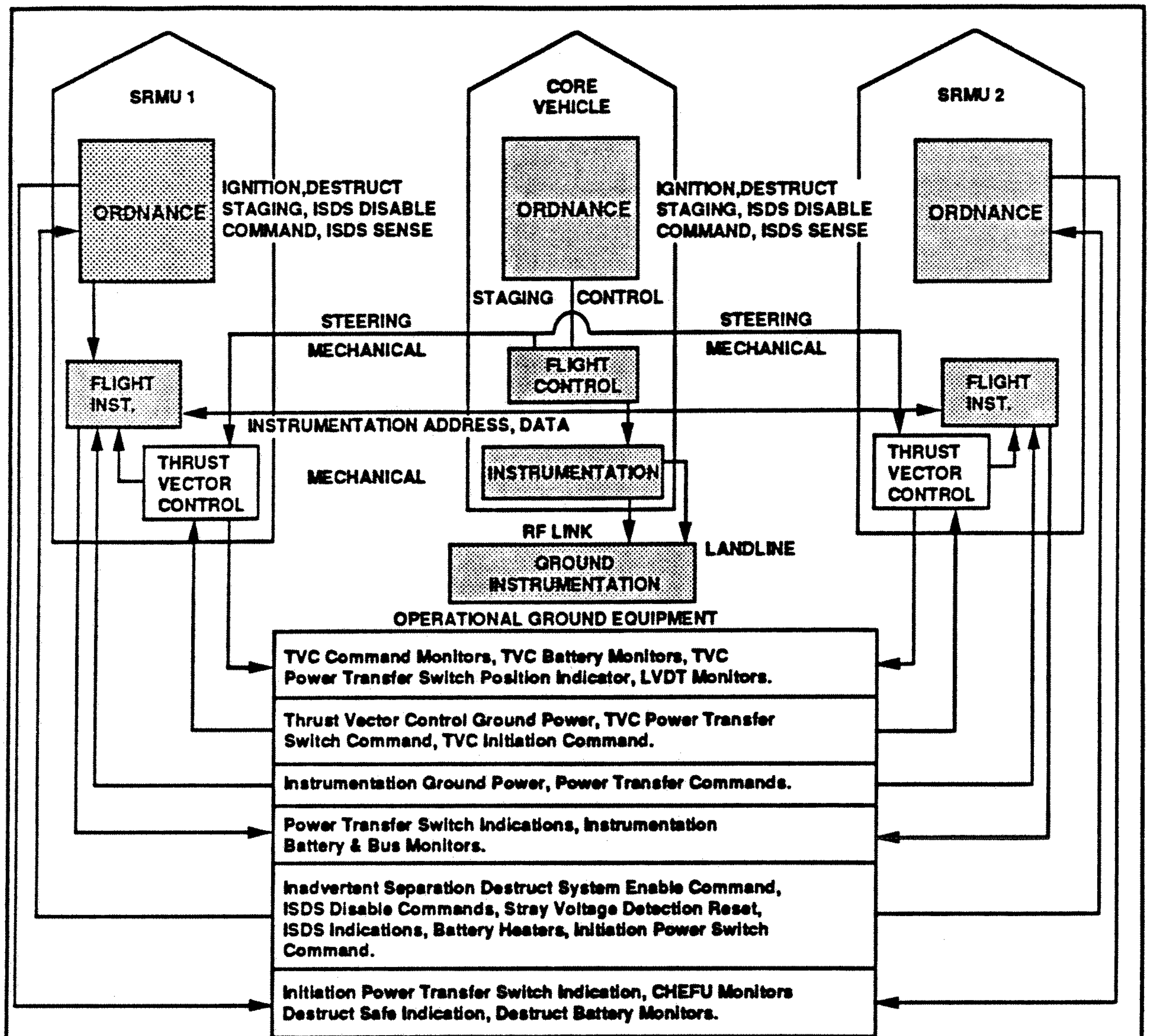


FIGURE 3.3.2.1-2 SRMU BLOCK DIAGRAM

3.3.2.3 Thrust Vector Control System (Continued)

From SRMU ignition for period up to 14 sec, the system provides an off-null nozzle position limited to 4 deg deflection.

After 14 sec from SRMU ignition, the system provides a vector angle of ± 6 deg when the booster vehicle aerodynamic pressures are below 400 psf. For higher aerodynamic pressure, the vector angles are ± 5 deg.

The slew rate is 10 deg/sec. The maximum differential between the commanded nozzle vector angle and the nozzle angle attained is within one-half a deg.

3.4 Payload Fairing

3.4.1 Payload Fairing Description

The PLF is an aerodynamic nose fairing for the Titan IV flight vehicle. It provides protection for a payload and upper stages from external environments during prelaunch, launch and ascent conditions.

Provisions for space vehicle access doors, acoustic attenuation, thermal insulation, antennas and conditioning air diffusers shall be provided as program–peculiar kits and as defined in the mission unique ICD.

The PLF is available in lengths from 56 ft to 86 ft in 10 ft increments and is 200 in. in outside diameter. Useable internal diameter for payloads is 180 in. Each payload fairing is built to accommodate specific satellite vehicle requirements.

PLF weight depends on the PLF requirements imposed by the mission criteria.

For additional detail information about the PLF refer to Figures 3.4.1-1, 3.4.1-2, 3.4.1-3, 3.4.1-4 and in Chapter 5 Figures 5.2-1, 5.2-2 and 5.2-3.

A PLF Mock-up, PLF Simulator Support Structure and PLF Test Code Articles are available for the payload user's requirements.

3.4.2 Payload Fairing Structures

The PLF consists of a Base Section which incorporates the PLF/TIV interface; a Barrel or Payload Section for variable payload lengths; and a Biconic Nose Section. Each section has three longitudinal separation joints.

The Base Section is 312.0 in. long and has a diameter of 200.0 in. at the outside skin line. The aft 52.0 inches is the Boattail Section. For PLF lengths greater than 56 ft the Boattail Section is stiffened. The boattail incorporates circumferential and longitudinal separation joints.

The Barrel Section has a maximum length of 480.0 in. (40 ft). It contains manufacturing joints for variable lengths of 10, 20 and 30 ft. The diameter is 200.0 in. This section has three longitudinal separation joints, an air conditioning inlet and a manufacturing joint at the top.

The Biconic Nose Section is 283.3 in. long and has three longitudinal separation joints. There are two conical sections, the section directly above the cylindrical section is a 15 deg cone and the top section is a 25 deg cone measured from the vertical. A spherical nose cap is attached to the 25 deg cone.

3.4.2.1 Venting and Leakage

The PLF is to withstand compartmental pressure loading and vehicle flight loads. Adjustable venting is provided. Leak areas do not exceed 50 $in.^2$

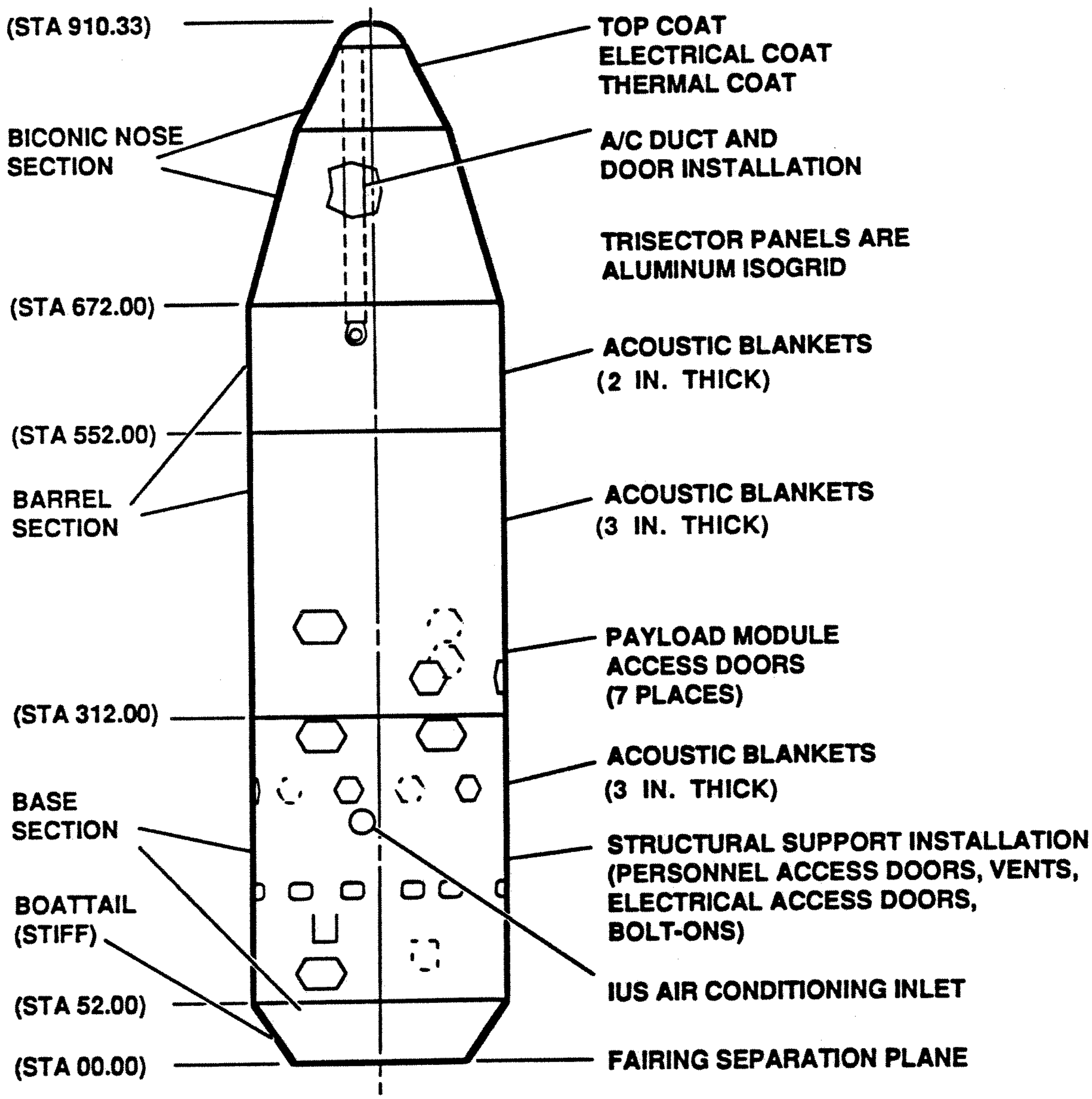


FIGURE 3.4.1-1 PAYLOAD FAIRING CONFIGURATION (DETAILS ARE MISSION VARIABLE) (SHEET 1 OF 3)

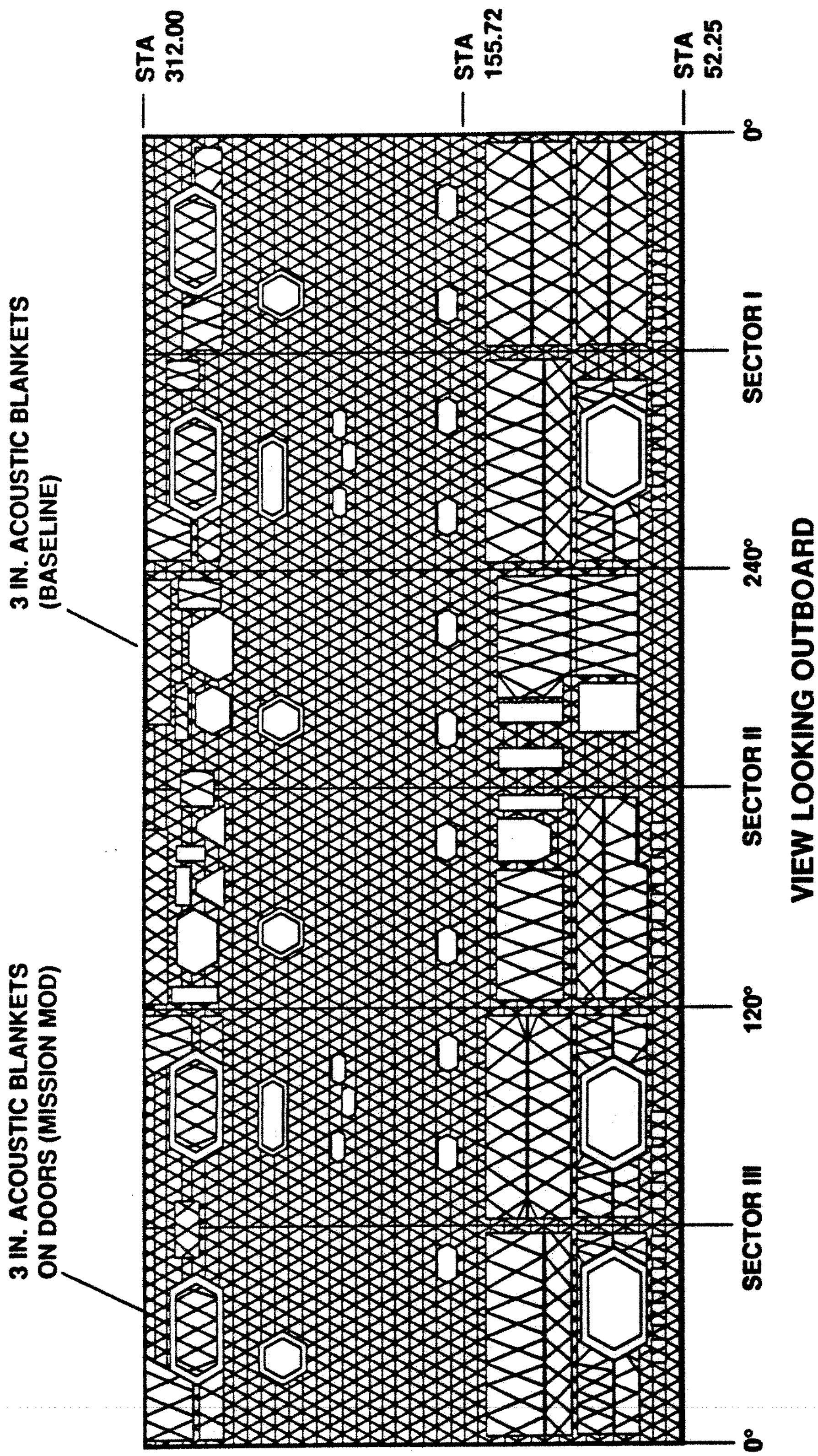
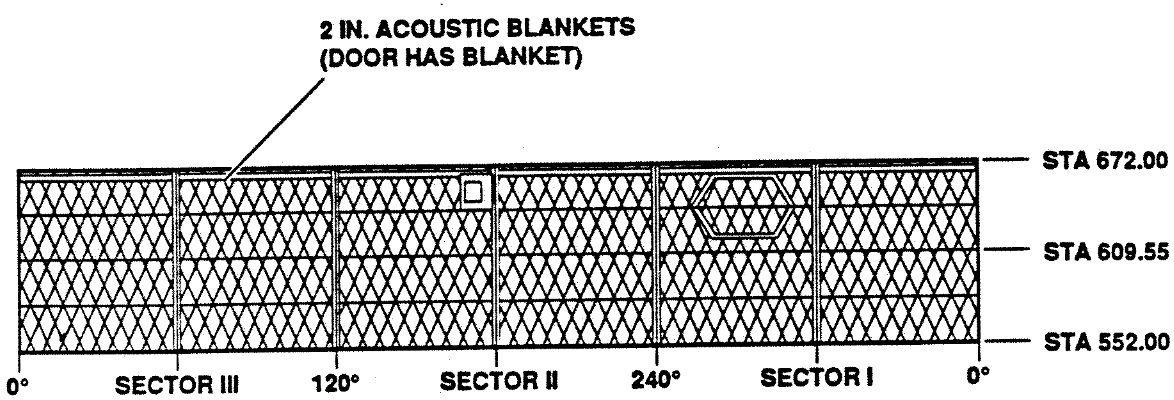
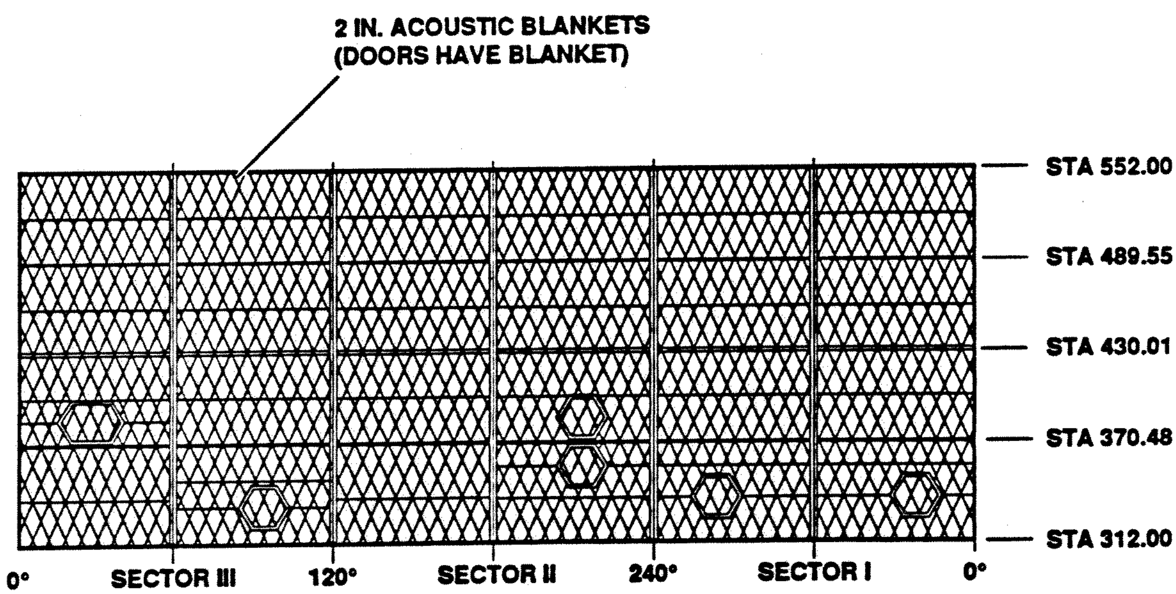


FIGURE 3.4.1-1 PAYLOAD FAIRING CONFIGURATION (DETAILS ARE MISSION VARIABLE)
(SHEET 2 OF 3)



ACOUSTIC BLANKETS FWD PAYLOAD MODULE
VIEW LOOKING OUTBOARD



ACOUSTIC BLANKETS AFT PAYLOAD MODULE
VIEW LOOKING OUTBOARD

FIGURE 3.4.1-1 PAYLOAD FAIRING CONFIGURATION (DETAILS ARE MISSION VARIABLE) (SHEET 3 OF 3)

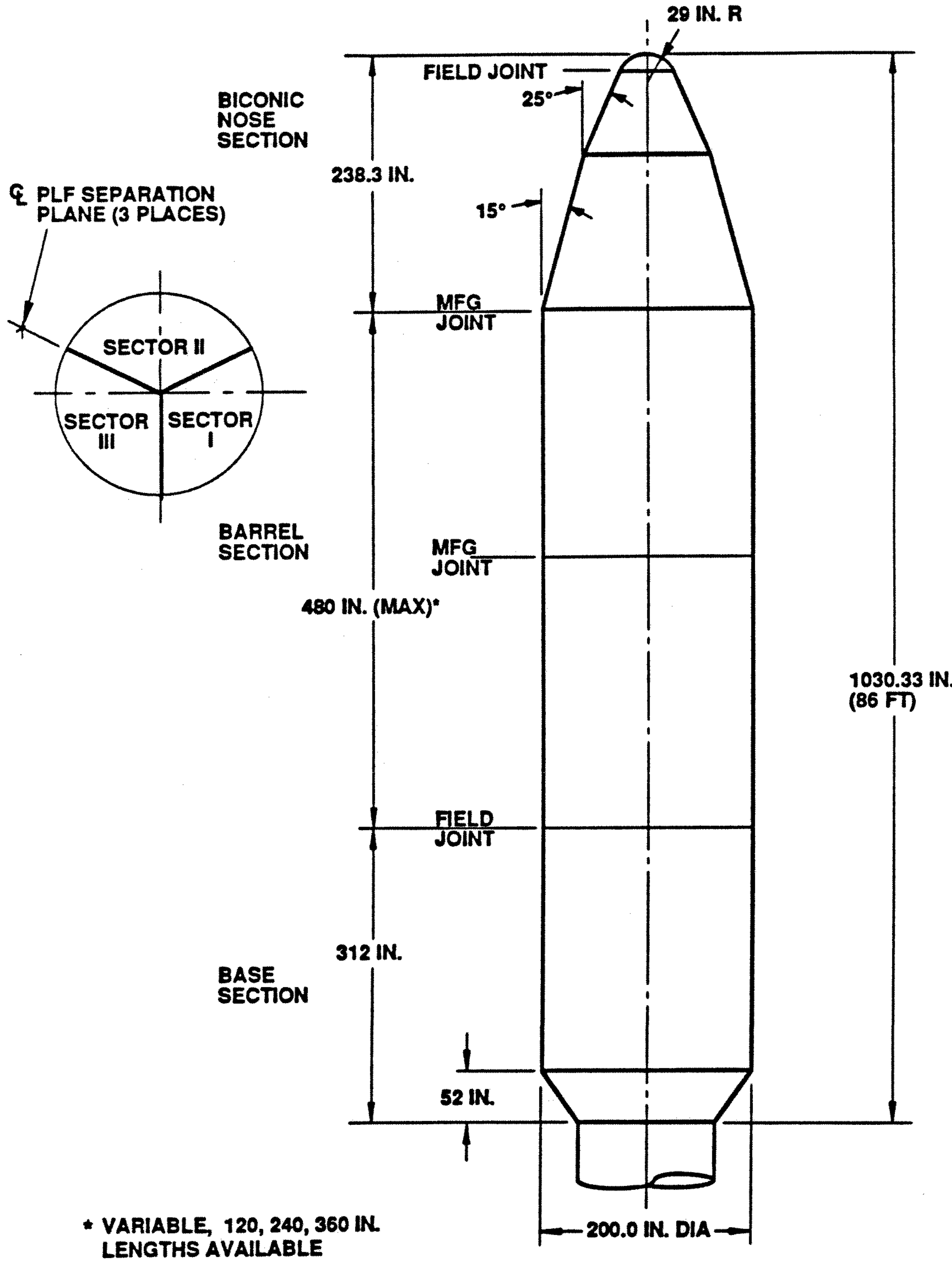


FIGURE 3.4.1-2 TITAN IV PLF SECTIONS

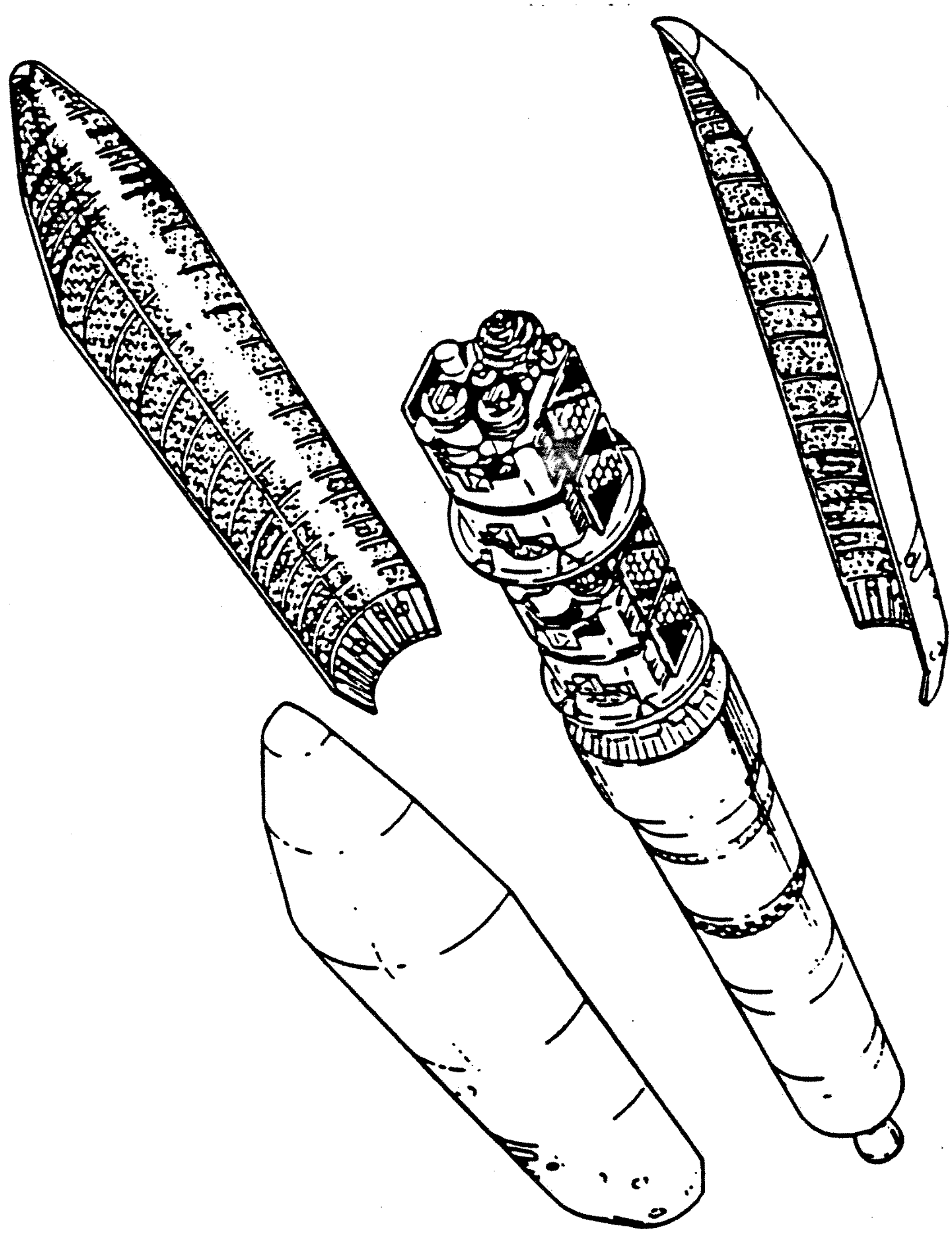


FIGURE 3.4.1-3 TITAN IV PLF TRI-SECTOR SEPARATION

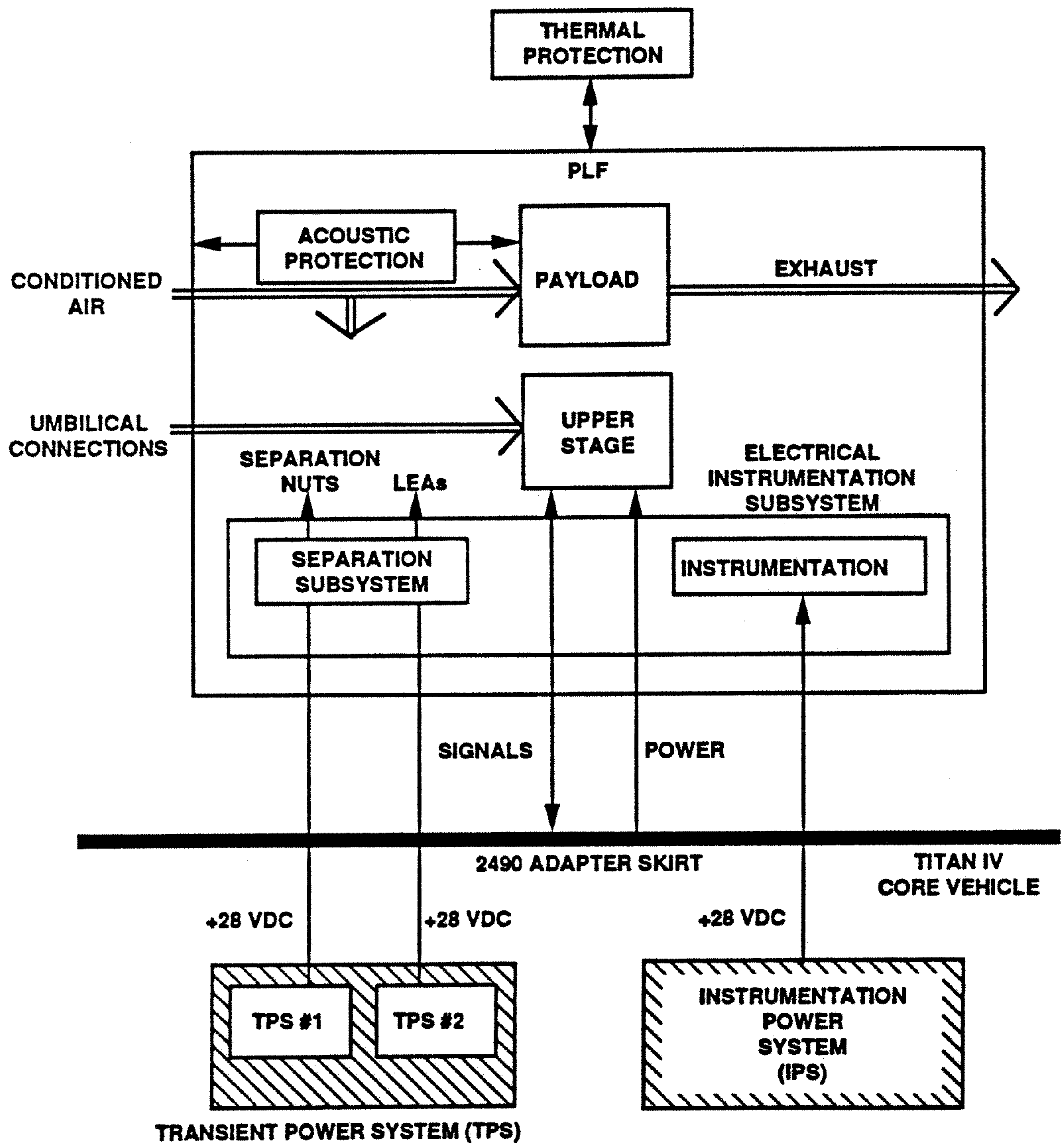


FIGURE 3.4.1-4 TITAN IV PLF INTERFACE DIAGRAM

3.4.3 Payload Fairing Attachment to Titan IV

The PLFs for the Titan IV SS ELV 401, 402, 403 and 405 configurations are all mounted on the Titan IV CP2490 Forward Extension Skirt tabs (36) with 36 longitudinal and radial cleat fittings and a minimum of 21 shear fittings, reference Figure 3.4.3-1. No tension joints are employed to fasten the PLF to the CP2490.

The three PLF sectors are tied together circumferentially at two PLF stations at the boattail and at one PLF station between the Biconic nose section and the top of the barrel section. Separation ordnance nut devices provide the circumferential ties at the sector separation joints, reference Figure 3.4.3-1.

The three PLF sectors are also held together via the Longitudinal Thrusting Joints which are continuous except at the Complex Tube Joints along each sector interface, reference Figures 3.4.3-2 and 3.4.3-3.

A Forward Bearing Reactor for the Centaur configuration is provided as a load sharing structural tie, reference Paragraph 4.2.2.3 and Figure 3.4.3-2.

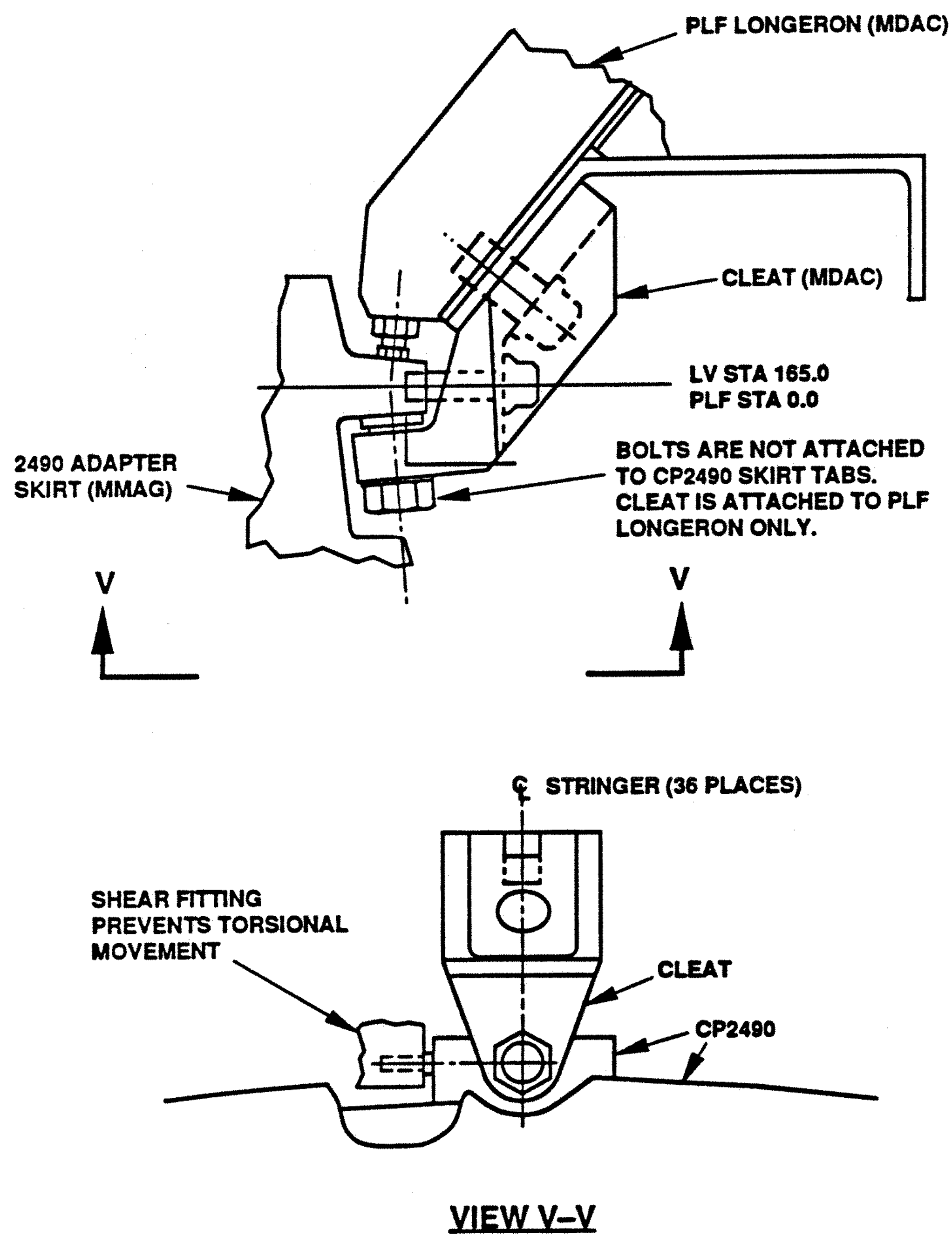


FIGURE 3.4.3-1 TITAN IV/PLF MOUNTING/POSITIONING MECHANISM ON CP2490 SKIRT TAB (36 PLACES)

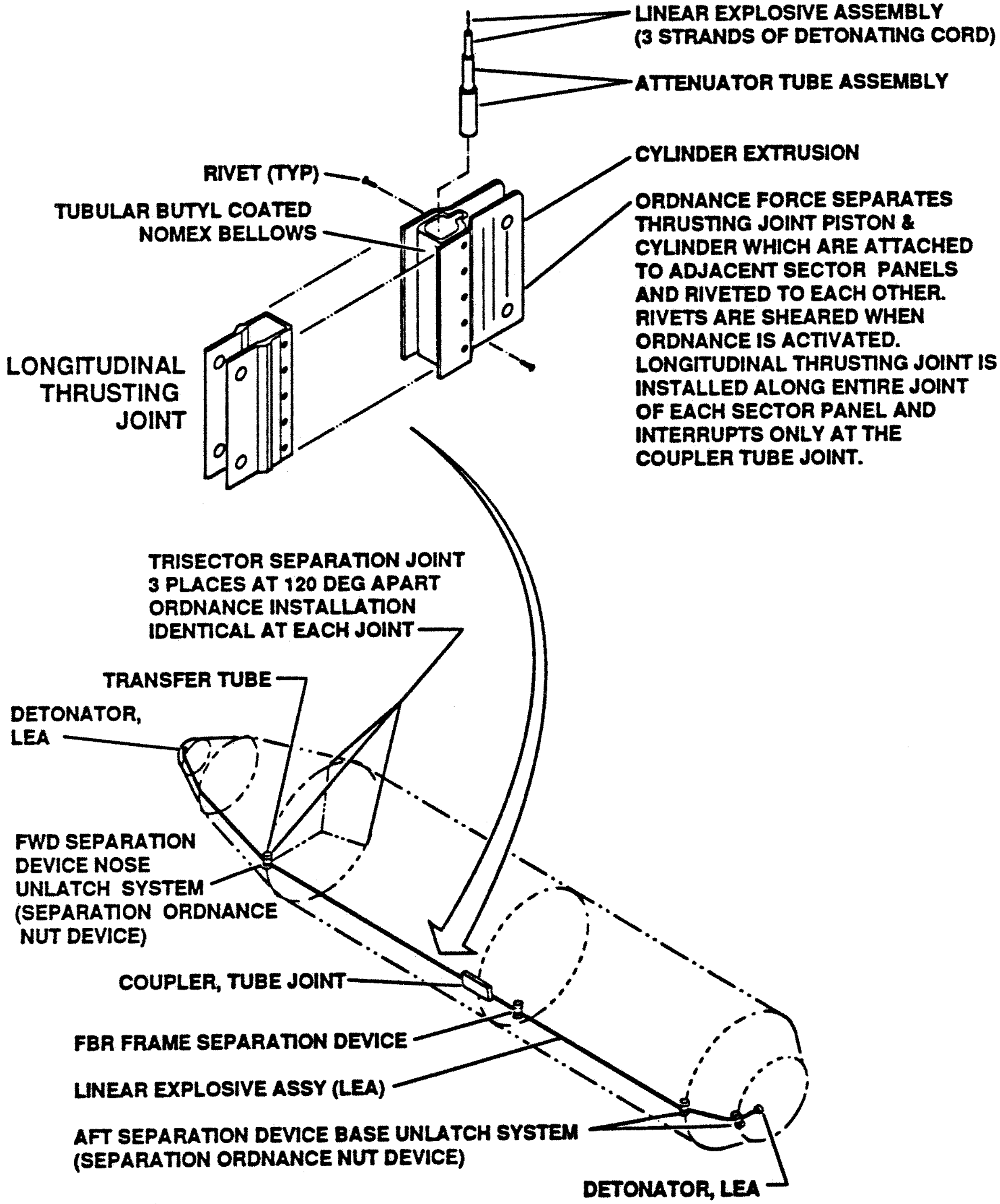


FIGURE 3.4.3-2 TITAN IV PLF SEPARATION SUBSYSTEM DIAGRAM

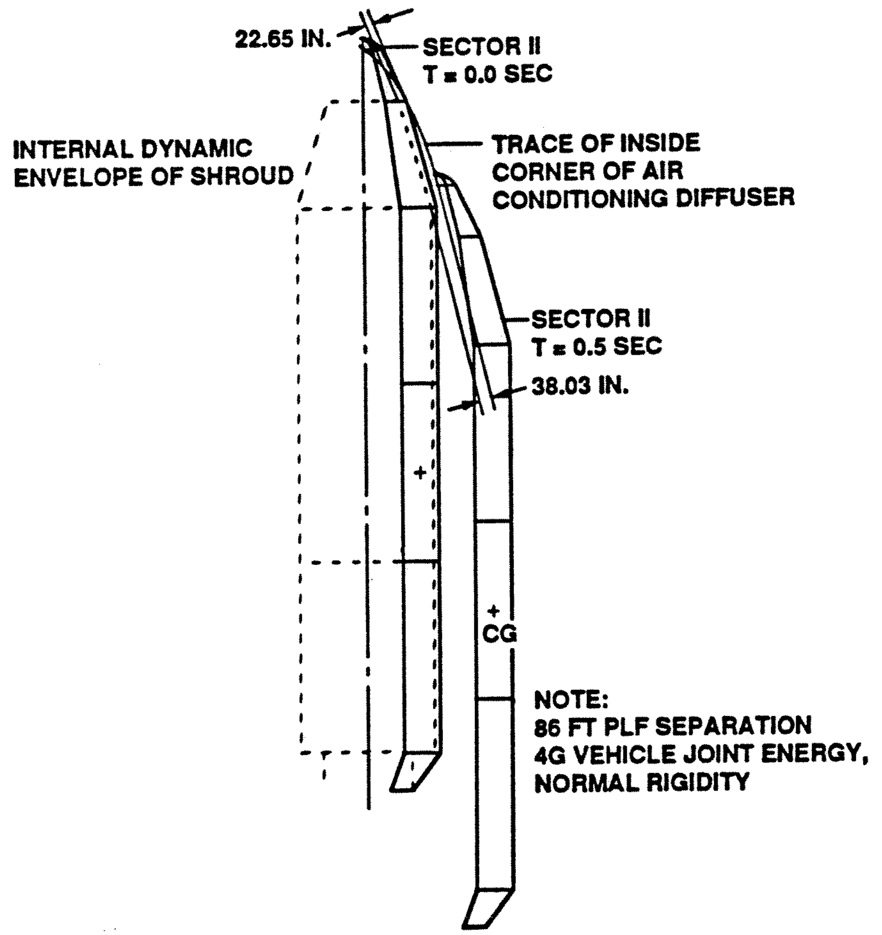


FIGURE 3.4.3-3 SEPARATION TRAJECTORY 86 FT FAIRING

3.5 Centaur Upper Stage SS-ELV-401 (ESMC)

3.5.1 Introduction

The delivery of the Centaur Upper Stage to the launch site and the processing necessary for the Centaur mate to Titan IV is accomplished by GDSS Division. The mating is accomplished by Martin Marietta. The Centaur processing and checkout equipment is provided, operated and maintained by GDSS.

The Centaur G Prime Vehicle overall length with the engines is 29.45 ft and its largest diameter is 14.17 ft excluding insulation pressure lines, etc. The propulsion is supplied from a single-stage, liquid-fueled, dual engine system. The Centaur interfaces with a forward adapter which contains Avionics for Guidance and Navigation, a RCS, an Electrical Power System, and a Telemetry Tracking and Command System, reference Figures 3.5.1-1, 3.5.1-2 and 3.5.1-3. The dry weight of the Centaur G Prime is approximately 6125 lb and the total fueled liftoff weight is over 52,650 lb.

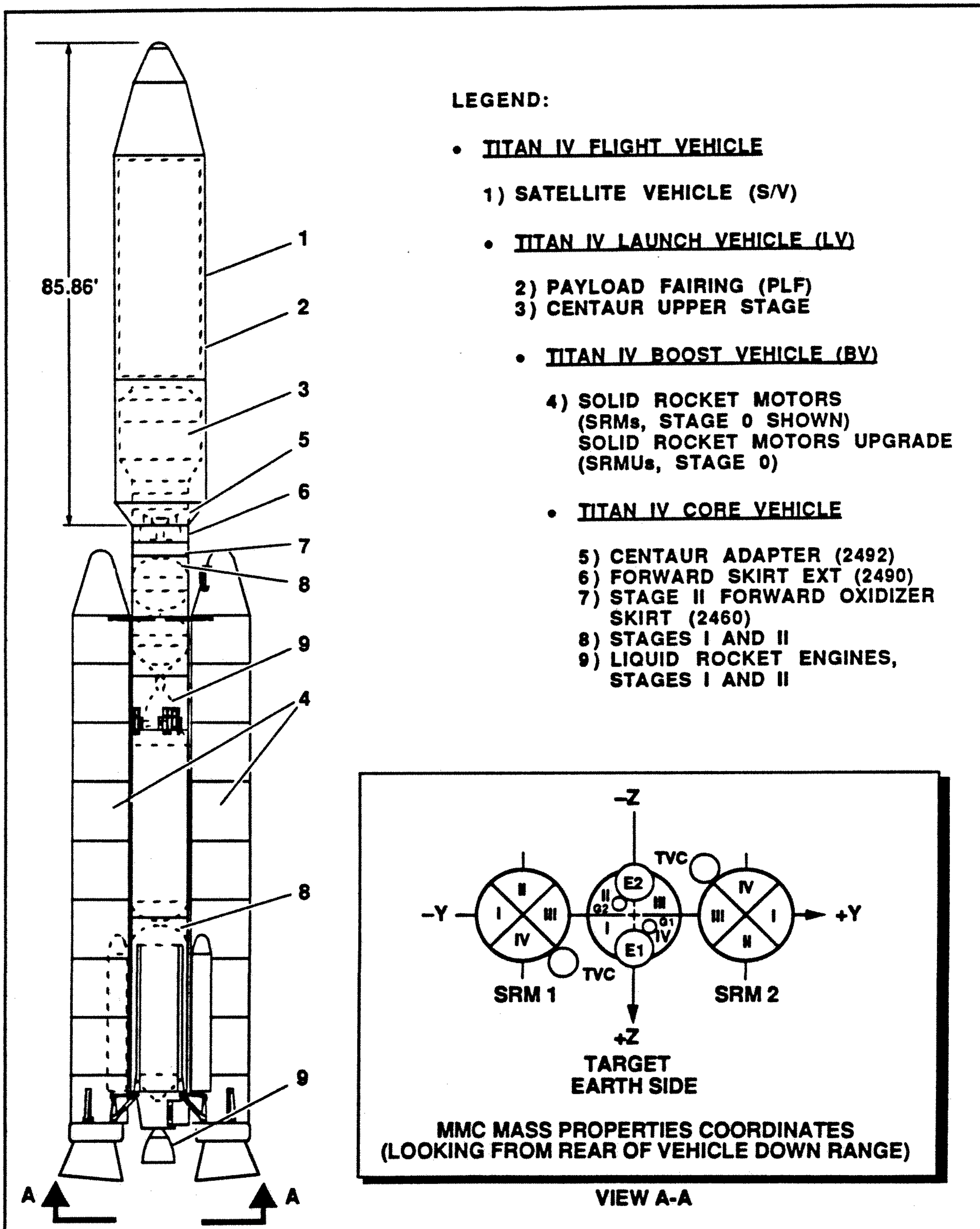


FIGURE 3.5.1-1 TITAN IV/CENTAUR FLIGHT VEHICLE CONFIGURATION (w/SRMs) (401)

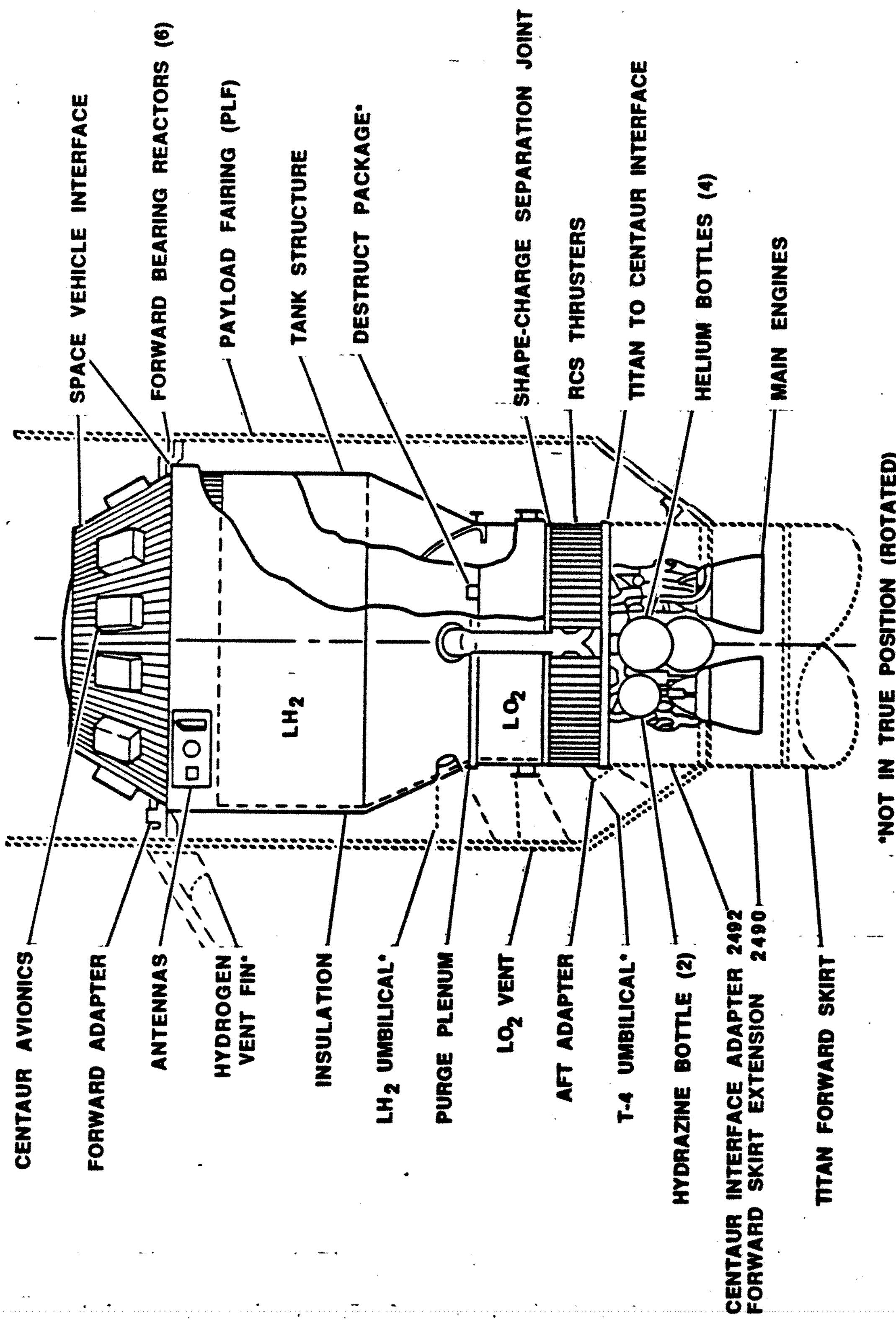
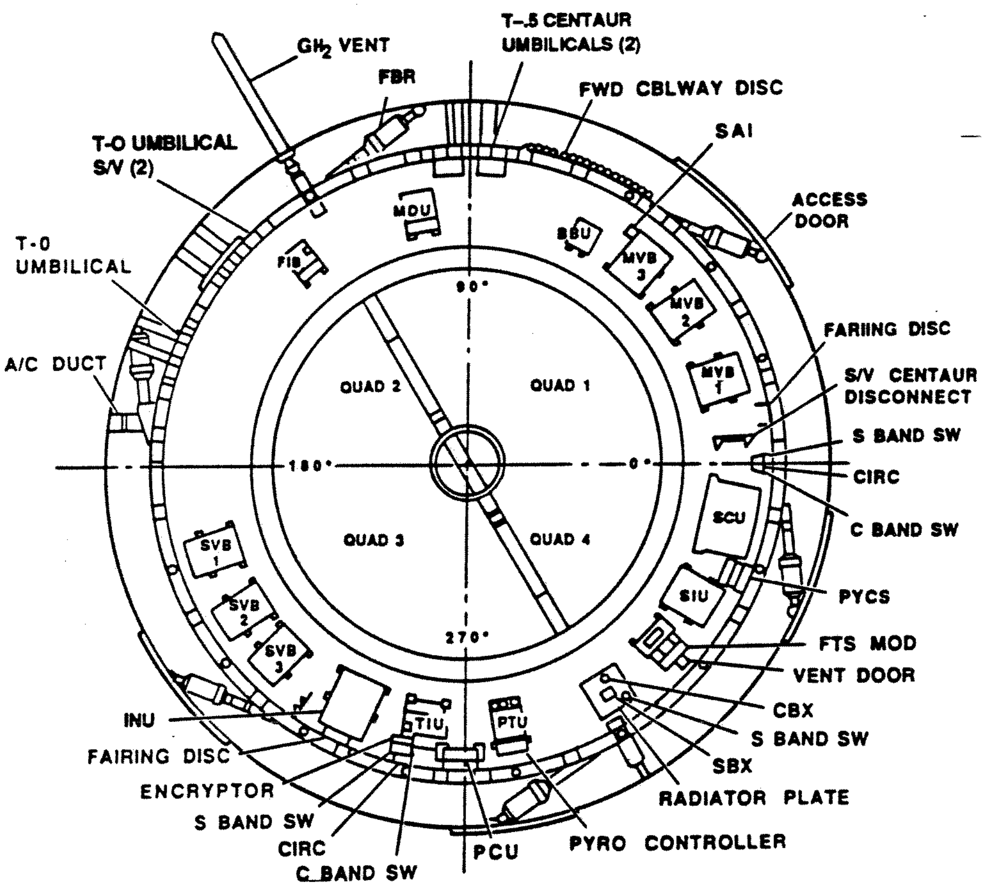


FIGURE 3.5.1-2 CENTAUR CONFIGURATION



MVB	MAIN VEHICLE BATTERY	PCU	POWER CONTROL UNIT
BBU	BATTERY BUSSING UNIT	PTU	POWER TRANSFER UNIT
SAI	SAFE/ARM INITIATOR	PYC	PYRO CONTROLLER
MDU	MASTER DATA UNIT	CBX	C-BAND TRANSPONDER
A/C	AIR CONDITIONING	SBX	S-BAND TRANSMITTER
FIB	FORWARD INSTRUMENTATION BOX	FTS MOD	FLIGHT TERMINATION SYSTEM MODULE
INU	INERTIAL NAVIGATION UNIT	SW	SWITCH
PLB	PAYLOAD BATTERY	S/V	SATELLITE VEHICLE
SCU	SEQUENCE CONTROL UNIT	CIRC	CIRCULATOR
TIU	TELEMETRY INTERFACE UNIT	FBR	FORWARD BEARING REACTOR
SIU	SERVO INVERTER UNIT		

FIGURE 3.5.1-3 COMPONENT LOCATIONS ON FORWARD ADAPTER (BLOCK II) (FLTS 5 AND UP)

3.5.2 Structural

The Centaur structural elements include the LO_2 and LH_2 tanks, forward adapter, forward bearing reactors and aft adapter. The structural system serves as integral/main propellant tanks, supports the S/V, and supports components of the propulsion, fluids and avionics systems.

The Forward Adapter consists of a conic section composed of frames and a corrugated cylindrical section. The conic section provides the mounting supports for the avionics packages and provides S/V attachment interfaces at the ring frames at each end of the conic section. Eight hardpoint interfaces are provided on the forward adapter face and 22 hardpoint truss attachment interfaces are loaded at the base of the conical adapter for attachment of the S/V, reference Figure 4.2.1.1-3.

The aft adapter attaches to the LO_2 tank aft ring and to the Titan IV CP2492 skirt, reference Figures 3.5.1-2, 3.5.2-2 and Table 3.2.1.2-1.

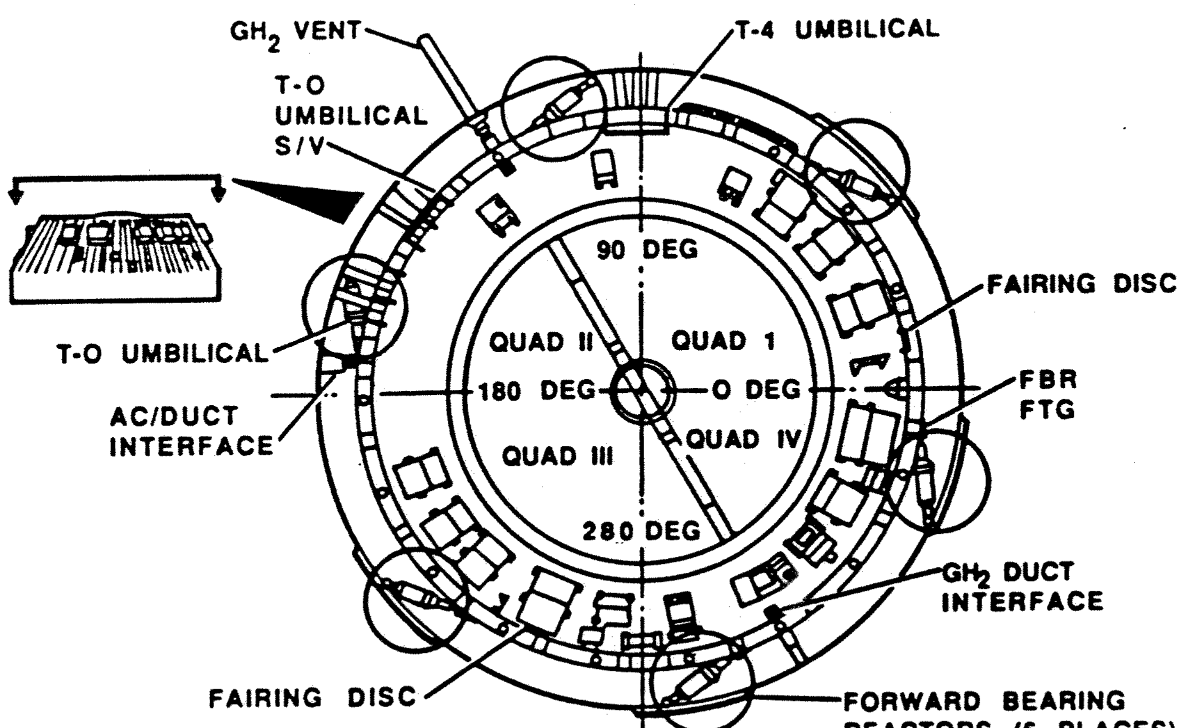


FIGURE 3.5.2-1 FBR ORIENTATION

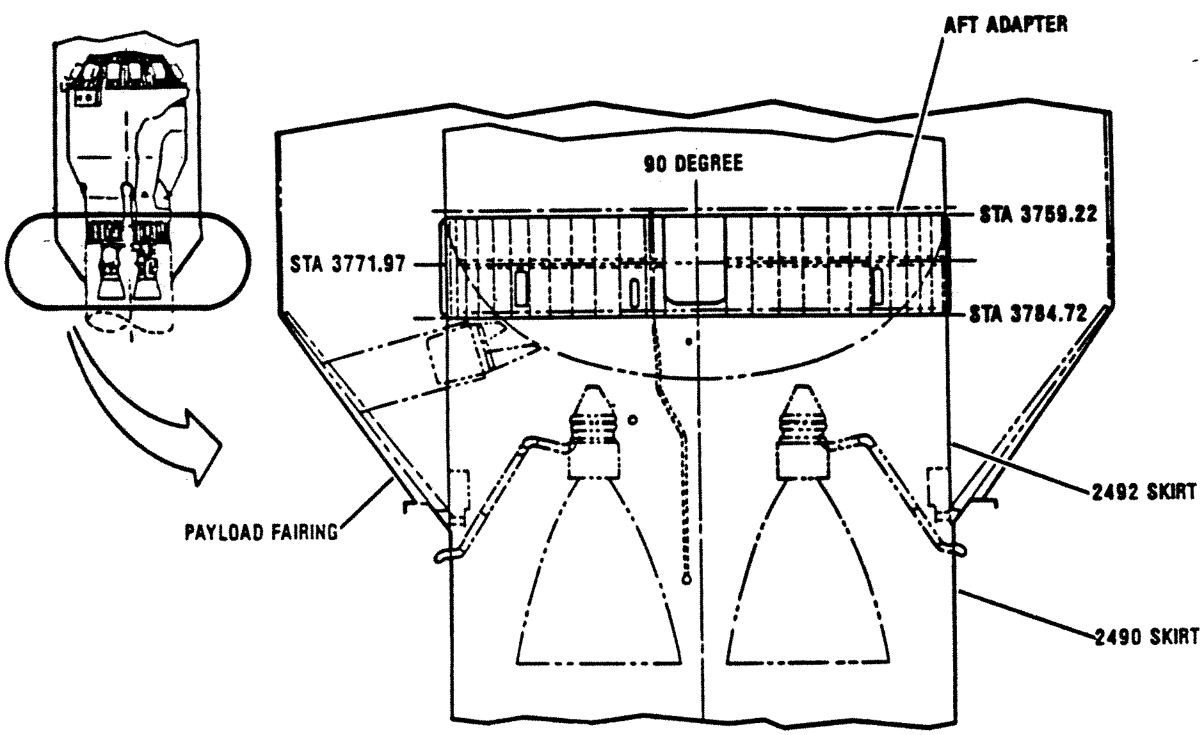


FIGURE 3.5.2-2 AFT ADAPTER

3.5.3 Payload Fairing

The standard PLF used on Titan IV is 16.7 ft in diameter and is available in various lengths (in 10 ft increments between 56 and 86 ft) to satisfy individual user requirements. Performance capabilities presented in this section are based on the 86-ft PLF for Titan IV/Centaur.

The Centaur peculiar Forward Bearing Reactor (FBR) system is located between the forward adapter and the PLF. This system allows for a controlled amount of load sharing which provides a reduction of Centaur deflections at Titan liftoff and ascent, reference Figure 3.5.2-1 and Paragraph 4.2.2.3.

3.5.4 Propulsion System

The Centaur is powered by two Pratt and Whitney RL 10A-3-3A engines. The propellant feed and main engine system provides the Centaur vehicle with the thrust generated by the combustion of LO_2 and LH_2 propellants. Activation of the system does not occur until the Centaur is safely separated from the Titan Core.

The LHe prelaunch chilldown system cools the main engine turbopumps by introducing cold helium gas obtained by vaporization of liquid helium. Prechilling of the turbopumps allows the in-flight chilldown time to be minimized for first burn prestart. The system is activated 45 minutes prior to liftoff.

3.5.4 Propulsion System (continued)

During Centaur flight, GH_2 is bled from each engine at a rate of 0.12 lb/sec for vehicle tank pressurization. The engines are capable of multiple restarts and are capable of fulfilling the requirements of a variety of missions.

The Propellant Tank Pressurization System provides net positive suction head pressure for Centaur main engine start and during engine burn by providing helium for both tanks. It maintains, in conjunction with the vent system, propellant tank pressures and intermediate bulkhead structural integrity throughout the entire mission.

The main engines provide the linear velocity and vehicle 3-axis rotational (roll, pitch, yaw) control during the main engine powered flight. The required vehicle control during coast periods is accomplished by the Reaction Control System (RCS).

The system in-flight performance allocation is:

a) Class Nominal Engine Performance (Prime Item Development Specification, 57-00210G)
   Thrust – 33,099 lb
   Specific Impulse (ISP) – 444.16 sec

b) Class Nominal Engine Performance (Data Book Nominal)
   Thrust – 33,030 lb
   Specific Impulse – 443.80 sec

c) Three-Sigma Dispersions (Two Engines)
   Thrust – ±467 lb
   Specific Impulse – ±2.10 sec

NOTE: 1) Current performance based on b).
2) Current performance includes –0.5 ISP bias based on engine acceptance data

3.5.4.1 H_2 and O_2 Vent System

The gases (hydrogen and oxygen) are vented overboard through the PLF via disconnects. The gaseous oxygen is vented to the atmosphere. The gaseous hydrogen is vented through one leg (the other is capped until PLF jettison) and routed to the AGE GH_2 vent burn stack prior to liftoff. After liftoff, venting occurs at the vent fin vehicle exit about 4 ft from the surface of the PLF.

3.5.4.2 Propellant Utilization

The Propellant Utilization (PU) System measures propellant quantities during Centaur operation and controls the engine operating mixture ratio to achieve near simultaneous depletion of the propellants.

3.5.4.3 Reaction Control System

The RCS consists of twelve 6 lb thrust units, two positive expulsion storage bottles with 170 lb hydrazine capacity each, two pyrotechnic isolation valves, four service valves, a filter, three pressure transducers and both ground and inflight heaters.

Vehicle roll, pitch and yaw control is accomplished using eight monopropellant hydrazine (N_2H_2) thrusters. The thrusters are fired in short bursts for control, to hold the vehicle in a rate-displacement limit cycle. Propellant setting in preparation for main engine burns is provided by an additional four thrusters.

3.5.4.4 Thrust Vector Control

The autopilot software in the DCU accepts attitude errors, differentiates them to obtain attitude error rates, and then computes the desired engine actuator commands. The DCU outputs these analog signals to the Servo-Inverter Unit (SIU) representing the desired engine position (pitch, yaw and roll). An electronic output from the SIU to the servo valve causes hydraulic fluid to flow in the hydraulic actuator producing engine movement.

To perform collision avoidance, the Centaur is maneuvered away from the spacecraft after spacecraft separation through the use of RCS thrust.

3.5.5 Electrical Systems

The Core Vehicle proves the electrical power, switching, instrumentation, guidance, navigation and control required to boost the Titan IV/Centaur system to the point of Upper Stage separation from the core vehicle, independent of the Upper Stage Avionics System.

Centaur flight 5 and subsequent flights incorporate Block II Centaur Avionics where the Digital Computer Unit (DCU) and the Inertial Measurement Group (IMG) used on flights 1 thru 5 are replaced with an Inertial Navigation Unit (INU). The INU consists of a ring laser gyro Inertial Measurement Unit (IMU) and a 1750A Flight Control Processor (FCP).

All Centaur flights will incorporate a Data Acquisition System (DAS) which replaces Remote Multiplexer Units (RMUs) and Signal Conditioners. The DAS consists of a Master Data Unit (MDU) and a Remote Data Unit (RDU).

The replacement of the IMU (which consists of an Inertial Reference Unit (IRU) and a System Electronics Unit (SEU)) and the DCU with the INU along with the use of the DAS results in a Centaur Vehicle weight savings of approximately 163 lb, reference Figures 3.5.5-1, 3.5.5-2, 3.5.5-3 and Figure 3.5.1-3.

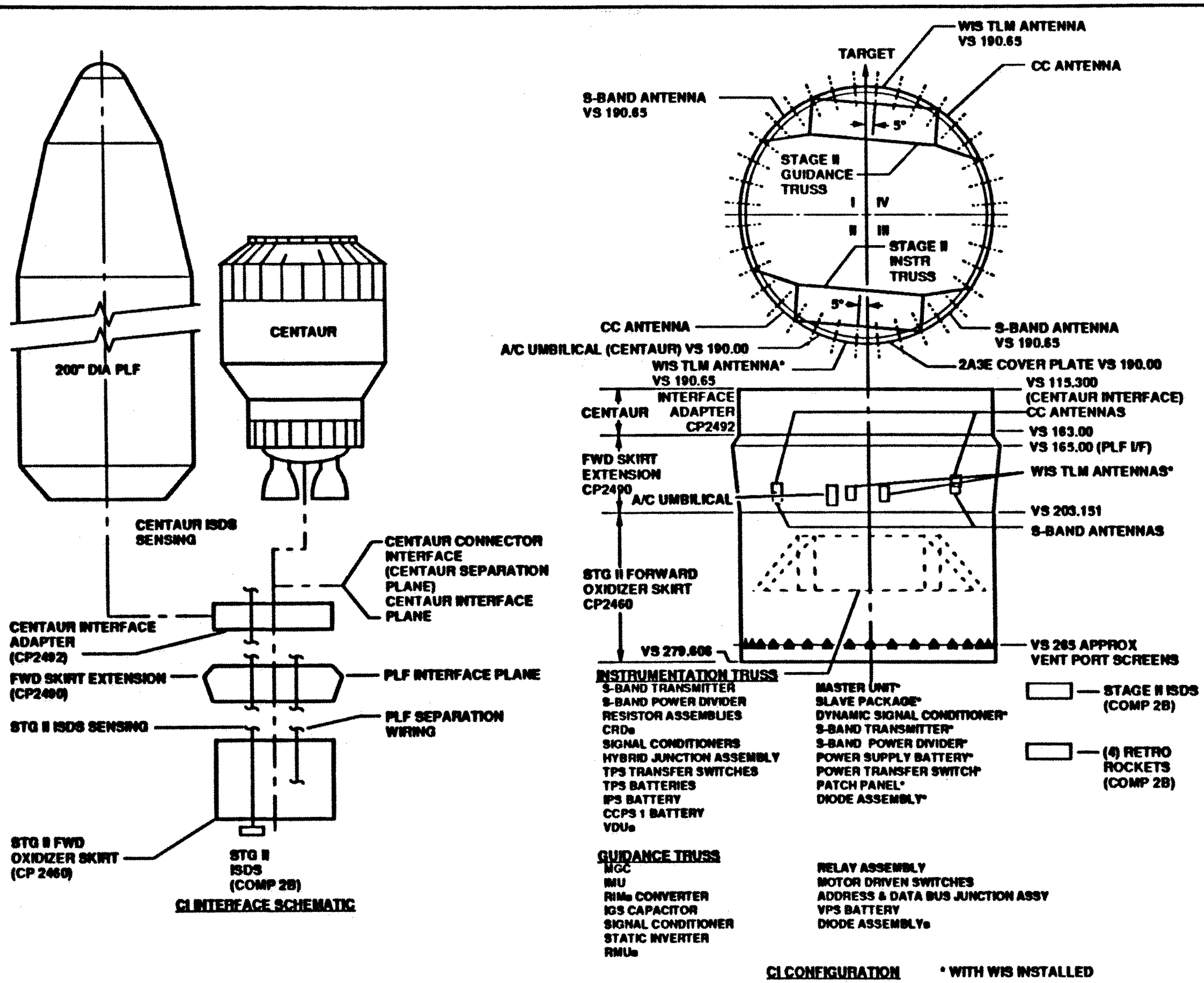


FIGURE 3.5.5-1 CENTAUR UNIQUE AIRBORNE EQUIPMENT (401)

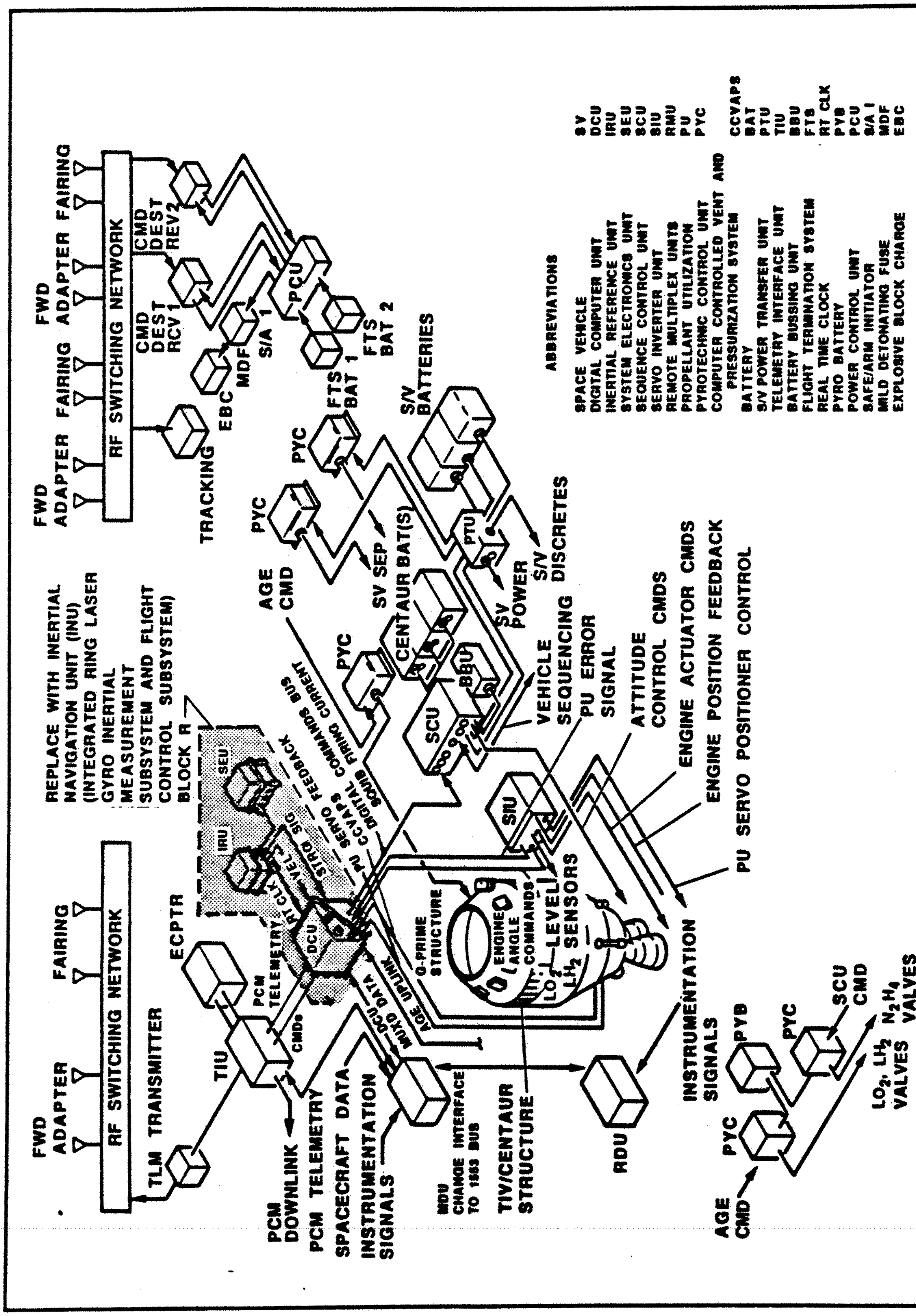


FIGURE 3.5.5-2 AVIONICS BLOCK DIAGRAM BLOCK II
AVIONICS CHANGES FROM BLOCK I

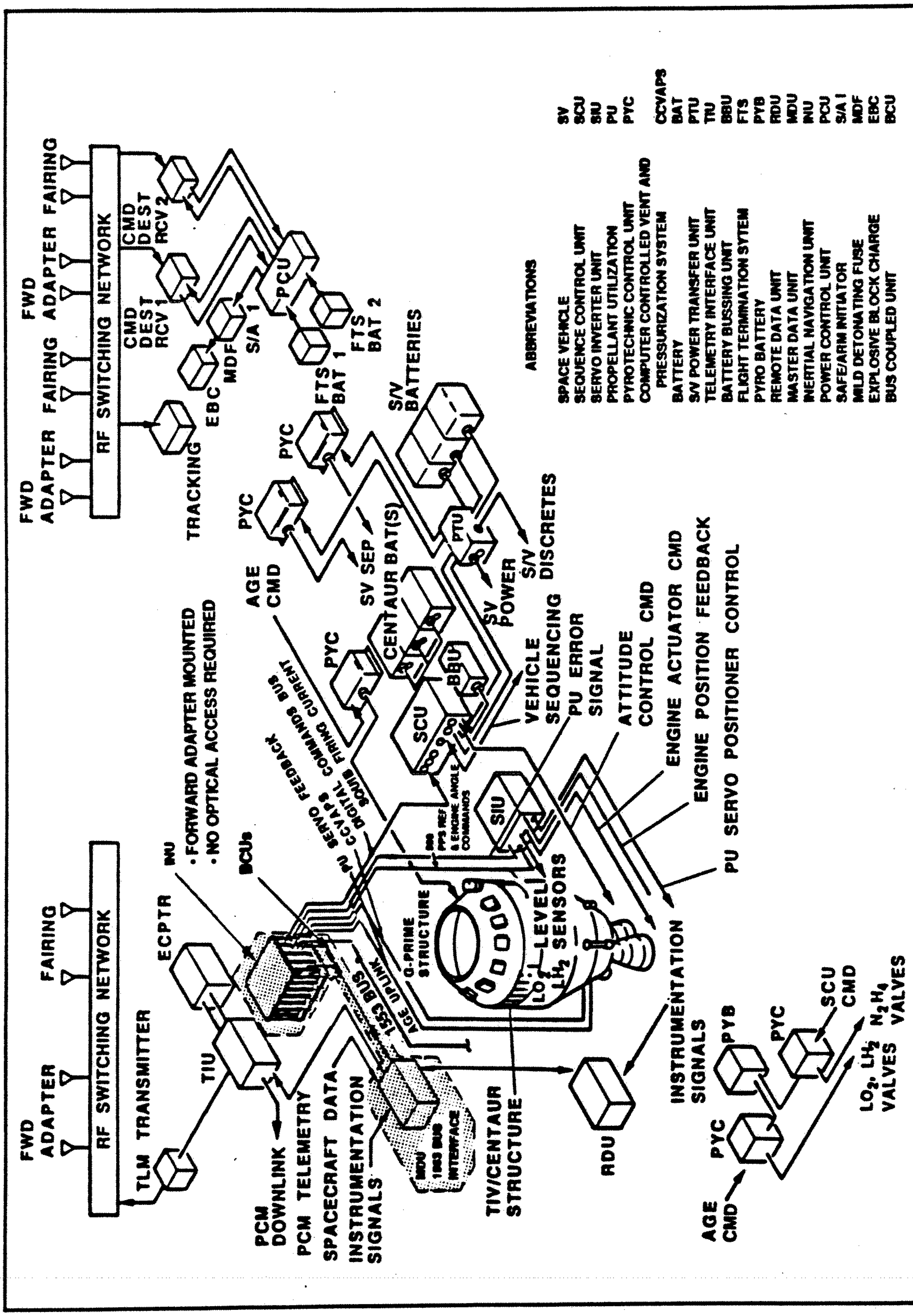


FIGURE 3.5.5-3 BLOCK II AVIONICS BLOCK DIAGRAM TITAN IV/CENTAUR PROGRAM

3.5.5.1 Guidance Navigation and Control

The Centaur Inertial Guidance and Navigation Subsystem provides autonomous flight guidance for the Centaur Vehicle. Beginning at launch and continuing throughout Titan Booster and Centaur flight, vehicle thrust and drag accelerations are measured by the IRU and sent to the DCU where Centaur position and velocity are determined.

The Centaur can store the current on-board State Vector at any time during the mission and transmit the data automatically to a ground station.

The DCU guidance software determines Main Engine Start (MES), the proper burn attitude and the Main Engine Cutoff (MECO) time. The DCU software generates error differences between DCU generated steering vectors and IRU attitude signals and uses these differences to correct the Centaur Vehicle attitude.

Attitude control of the Centaur Vehicle during main engine powered flight is accomplished through TVC of the main engine; and during coast, control is by means of the RCS.

Long coast times in orbit cause the build-up of significant errors in the navigation data. Since a major source of error is accelerometer bias uncertainty, a significant navigation improvement is achieved by in flight bias calibration of the accelerometers during a quiescent coast period in the mission.

The DCU is a stored program, general purpose, binary, twos complement, digital computer with a self-contained Random Access Memory (RAM), arithmetic section, timing and control section, input-output (I/O) section and a power supply. Additionally, the DCU provides a pulse code modulated Central Controller for control of DCU data flow to the Master Data Unit (MDU). The Central Control Unit (CCU) operates autonomously within the DCU and utilizes direct memory access to stored PCM formats located in up to 4096 words of Core Memory. The I/0 section provides the capability to handle the signal conversion necessary between the DCU and other system interfaces.

The Core Memory consists of 16,384 directly addressable words of 24-bits each. A complete read/restore of read/write is automatic and the cycle time is 3.13 microseconds.

3.5.5.2 Telemetry, Tracking and Command System

3.5.5.2.1 Telemetry

The Telemetry System is responsible for encrypting the Centaur Pulse Code Modulation (PCM) telemetry signal, formatting and encoding the encrypted signal, and modulating the signal for transmission via an RF link to an ESMC telemetry ground station or to an Air Force Satellite Control Facility Remote Tracking Station (AFSCF-RTS).

3.5.5.2.1 Telemetry (Continued)

All Centaur flights will incorporate a Data Acquisition System (DAS) which replaces Remote Multiplexer Units (RMUs) and Signal Conditioners. The DAS consists of a Master Data Unit (MDU) and a Remote Data Unit (RDU), reference Figure 4.2.3.2-1.

The transmitter is a 10 W S-band transmitter which accepts convolutionally encoded telemetry data from the Telemetry Interface Unit (TIU) and transmits it to cooperating ground stations from prelaunch to end of mission. It provides pre-modulation filtering and PCM-PM modulation with suppressed carrier of the Non-Return-to-Zero-Mark (NRZ-M) 256 ksps.

There are four antennas; two mounted approximately 180 deg apart on the PLF and two mounted approximately 180 deg apart on the Centaur's Forward Adapter. The interrogation signal is received at the vehicle by one or both antennas mounted on the PLF exterior. The two PLF antennas operate until PLF jettison at which time the two forward adapter mounted antennas provide spherical antenna coverage.

Preflight uplink communications from the Ground Computer System (GCS) are provided by a hardline serial data link to allow the transmission of programs, data, or commands to the DCU. A hardline downlink via the MDU to the TIU provides DCU and vehicle telemetry to the ground during preflight. The data is decrypted and decommutated in the Ground Telemetry Station (GTS) and sent to the GCS where it is processed in real time. Handshaking and check summing of data between the DCU and GCS provide for verification of uplinked messages with retransmission possible for erroneously received data. DCU preflight telemetry frame checksums provide for ground verification of downlink data. During flight, no uplink capability is provided and the downlink will utilize the Airborne RF Telemetry System. All downlink data will be encrypted on the vehicle.

3.5.5.2.2 Tracking

The Centaur has a C-band Tracking System for tracking capability from liftoff through park orbit insertion. This tracking system provides the necessary data to determine Titan IV SLV position in support of the Titan IV and Centaur Flight Termination Systems (FTS) (vehicle position is also derived from vehicle guidance data and by radar skin track of the vehicle), reference Figure 4.2.3.2.2-1.

The system receives and transmits C-band frequency signals from and to several ground-based interrogation radars. On the ground, these signals provide data to determine vehicle position, velocity and performance.

3.5.5.2.2 Tracking (Continued)

The non-coherent transponder is designed to extend the tracking range of the precision-tracking C-band instrumentation radar. The transponder accepts an RF coded pulse input signal and replies with an RF coded pulse output signal. The transponder operates from prime input voltage between 24.6 and 32 Vdc. It transmits a minimum power of 400 W, when measured at the transponder output port.

3.5.5.2.3 Commands

Uplink commands are limited to FTS signals, reference Figures 4.2.3.2.2-1 and 3.5.5-1.

3.5.5.3 Flight Termination System

The Titan IV Centaur configuration has two independent Range Safety Commanded encrypted, UHF, FTSs; one in Titan IV Stage II and one on the Centaur Forward Adapter. These independent systems each function upon receipt of the same ground station destruct command signal.

Before Centaur PLF jettison, the RF Flight Termination Signal from the ground is received by the CRDs via one or both of the Centaur antennas which are mounted on opposite sides of the metal PLF. After PLF jettison, the RF carrier frequency is received by one or both of the antennas which are mounted on opposite sides of the Centaur Forward Adapter.

The Titan IV Launch Vehicle FTS also utilizes an ISDS which operates upon the occurrence of a premature stage separation or a structural breakup of the vehicle. The Centaur FTS is not equipped with an ISDS.

The FTSs provide a reliable method of imposing a zero-thrust condition on the Titan Stage I and Stage II Liquid Rocket Engines, the SRMs and the Centaur Liquid Engine. The Range Safety Officer (RSO) has the option to command Titan IV core engine shutdown or Titan IV core, SRMs/SRMUs and Centaur propellant vessel destruct.

The RSO is provided vehicle flight path information from Centaur transponder tracking, vehicle skin tracking radar and vehicle guidance data. If the TIV/Centaur Vehicle deviates from the planned flight path corridor, the RSO can employ the FTS.

The basic requirements for the FTS are determined by the range (ESMC), the customer (USAF-SD), the contractor (MMC) and the subcontractor (GDSS). Requirements relating to mission peculiarities, hardware availability, vehicle-borne system interfaces, and launch equipment interfaces are to be determined jointly by the contractor, subcontractor and the customer, reference Figure 4.2.3.2.2-1.

3.5.5.4 Electrical Subsystems

The Centaur Airborne Electrical Power System provides electrical power to most Centaur components from the time ground power is removed during countdown until the end of mission. The Electrical Power System provides independent power sources for the main vehicle, the FTS, ordnance devices and the S/V; if required.

The Titan IV Centaur Umbilical connections consist of two Centaur T–.5 umbilicals, one Centaur T-O umbilical and two payload T-O umbilicals. These umbilicals pass through the PLF access doors and all terminate on the Centaur Forward Adapter, reference Figures 4.2.3.3-1 and 3.5.1-3.

3.5.5.5 Flight Software

The purpose of the Centaur flight software is to support in-flight operations of the Centaur from transition from the preflight mode, just prior to Titan IV/Centaur liftoff, until the end of Centaur mission operations, some time after Centaur/SC separation. Major functions of the flight software include sequencing, guidance, navigation, control, propellant utilization, tank pressure and venting control, telemetry formatting and self testing.

The IV&V for all DCU flight software will be accomplished by an independent contractor (Analex). This effort will be accomplished at the IV&V contractor facility and must be satisfactorily completed prior to final acceptance of the software.

3.5.6 Titan IV Type I /Centaur Performance Capability

Although the Titan IV Type I/Centaur Launch Vehicle can deliver payloads to a wide range of missions, only three reference missions are presented in this User's Handbook. The three missions are the Geosynchronous Equatorial Orbit (GSO), the 24-hour period Circular Orbit Inclined 65.0 degrees and the 12-hour period Elliptical Orbit Inclined 63.4 degrees launched from ESMC.Trajectory Simulation Ground Rules, Constraints and Typical Sequence of Events for the 24-hour Inclined Mission, the 12-hour Inclined Mission and the Geosynchronous Equatorial Orbit are presented in Tables 3.5.6-1, 3.5.6-2 and 3.5.6-3. Table 3.5.6-4 is a Performance Summary for all three missions showing the final payload capability as well as other significant performance related parameters.

TABLE 3.5.6-1 TITAN IV TYPE I/CENTAUR TYPICAL TRAJECTORY SIMULATION GROUND RULES

<u>VEHICLE CHARACTERISTICS</u>	
Stage 0 SRM Temperature	71.5° F
SRM Nozzle Exit Area	12,491 $in.^2$/SRM
Stage I Propellant Temperature (Ox and Fuel)	72.5°F
Stage II Propellant Temperature (Ox and Fuel) (Reduced ullage with optimum fuel bias and propellant depletion shutdown apply to both Stage I & II)	70.0° F/72.5° F
Average Stage I Nozzle Centerline Thrust	544,414 lbf
Average Stage I Nozzle Centerline ISP	301.23 sec
Average Stage II Vacuum Thrust (Including Roll Nozzle)	106,224 lbf
Average Stage II Vacuum ISP (Including Roll Nozzle)	317.7 sec
Payload Fairing:	
Dimensions (diameter and length)	16.7 x 86 ft
Weight	14,115 lbm
Average Centaur Vacuum Thrust	33,030 lbf
Average Centaur Vacuum ISP	443.80 sec
<u>MISSION PROFILE</u>	
<u>Launch Azimuth</u>	
GSO Mission	93.0 deg
12-hr & 24-hr Inclined Missions	37.9 deg
Stage 0 separation sequence initiated when axial acceleration decreases to 1.3 Gs	
PLF jettisoned in Stage I when FMH ≤ 100 (BTU/ft^2)/hr	

TABLE 3.5.6-2 TITAN IV TYPE I/CENTAUR TYPICAL THREE DEGREE-OF-FREEDOM TRAJECTORY SIMULATION CONSTRAINTS
• Dynamic pressure (Q): not to exceed 926 lbf/ft^2 • Aerodynamic Heating Indicator (AHI): not to exceed 95.0×10^6 ft-lbf/ft^2 • Stage 0 separation: Dynamic pressure shall not exceed 60 lbf/ft^2 Pitch Angle-of-Attack shall not exceed ± 4.088 deg Yaw Angle-of-Attack shall not exceed ± 4.847 deg Stage I separation: Dynamic pressure shall not exceed 30 lbf/ft^2 Total Angle-of-Attack shall not exceed ± 15 deg • Nominal FMH rate: Not to exceed 100 (BTU/ft^2)/hr at or following PLF jettison • Magnitude of inertial velocity at park orbit perigee injection GSO mission: 25,887.3 ft/sec (Consistent with 85 x 250 nmi park orbit)

TABLE 3.5.6-3 TITAN IV TYPE I/CENTAUR TYPICAL SEQUENCE OF EVENTS FOR REFERENCE MISSIONS

	GSO (sec)	12-Hour Elliptical Inclined Orbit (sec)	24-Hour Circular Inclined Orbit (sec)
SRM Ignition	0.0	0.0	0.0
Begin Roll to Flight Azimuth	6.0	6.0	6.0
End Roll Maneuver	9.0	12.0	12.0
Begin Pitch Maneuver	10.0	14.0	14.0
Maximum Dynamic Pressure	57.0	57.0	57.0
Stage I Ignition (87FS-1)	115.9	115.9	115.9
SRM Separation	126.0	126.0	126.0
Jettison Payload Fairing	233.3	237.7	232.7
Step 1/Stage II Separation	304.0	304.1	304.0
Step 2/Centaur Separation	541.8	541.8	541.8
Centaur First Burn Ignition (MES1)	553.8	554.8	553.8
End Centaur First Burn (MECO1)	767.9	788.4	789.5
Park Orbit Injection	1017.9	793.4	794.5
Centaur Second Burn Ignition (MES2)	1,412.3	3,582.2	2,218.2
End Centaur Second Burn (MECO2)	1,678.8	3,839.6	2,477.8
Centaur Third Burn Ignition (MES3)	20,376.9	7,824.9	20,999.6
End Centaur Third Burn (MECO3)	20,498.1	7,874.0	21,100.8
Final Orbit Injection	20,528.1	8,118.0	21,130.8

NOTE: MES is Main Engine Start, MECO is Main Engine Cutoff.

TABLE 3.5.6-4 TITAN IV TYPE I/CENTAUR TYPICAL PERFORMANCE SUMMARY

		MISSION	
	GSO	12-Hour Elliptical Inclined	24-Hour Circular Inclined
Payload Weight, lbm	10,030.0	15,270.0 (1)	10,272.0
Mission Required Propellant Margin, lbm	602.0	750.0 (2)	643.0
Launch Azimuth, deg	93.0	37.9	37.9
Park Orbit			
Perigee, nmi	85.0	98.0	86.0
Apogee, nmi	250.0	101.0	156.0
Inclination, deg	28.6	55.0	55.0
Transfer Orbit			
Perigee, nmi	96.0	112.0	110.0
Apogee, nmi	19,413.0	17,488.0	19,396.0
Inclination, deg	26.6	55.0	55.6
Final Orbit			
Perigee, nmi	19,323.0	509.0	19,323.0
Apogee, nmi	19,323.0	21,298.0	19,323.0
Inclination, deg	0.0	63.4	65.0
Argument of Perigee, deg	N/A	270.0	N/A

(1) Payload weights above 11,500 lbm require structural modification which is not reflected in this payload weight.

(2) Does not include launch window contingency.

Note: These payload weights are for the 16.7 x 86 ft (14,115 lbm) PLF. If other PLF weights are required, the sensitivity is approximately –0.04 lbm Payload/lbm PLF for the GSO class of missions.

(3) Orbital altitudes relative to oblate earth.

3.5.6.1 ESMC Titan IV/Centaur Missions

3.5.6.1.1 Geosynchronous Orbits Launched from ESMC

Titan IV/Centaur can inject spacecraft into the Geosynchronous Orbit (GSO) or near geostationary positions in either the eastern or western hemispheres. This is accomplished by injecting the Upper Stage-S/V into a low-altitude parking orbit after a Centaur "first" burn of approximately 15,750 lbm of propellant. The Centaur engine is then restarted at either the first equatorial crossing (for satellite positioning in the eastern hemisphere), or the second equatorial crossing (for satellite positioning in the western hemisphere). This transfer orbit burn ("second" Centaur burn, approximately 19,600 lbm of propellant) produces an elliptical transfer orbit with an apogee altitude corresponding to geostationary altitude and reduces the orbital inclination by some optimum value (approximately 2.0 deg). The final Centaur burn (Centaur "third" burn, approximately 17,400 lbm of propellant) circularizes the orbit at geostationary altitude and executes a plane change that reduces orbital inclination to 0 deg.

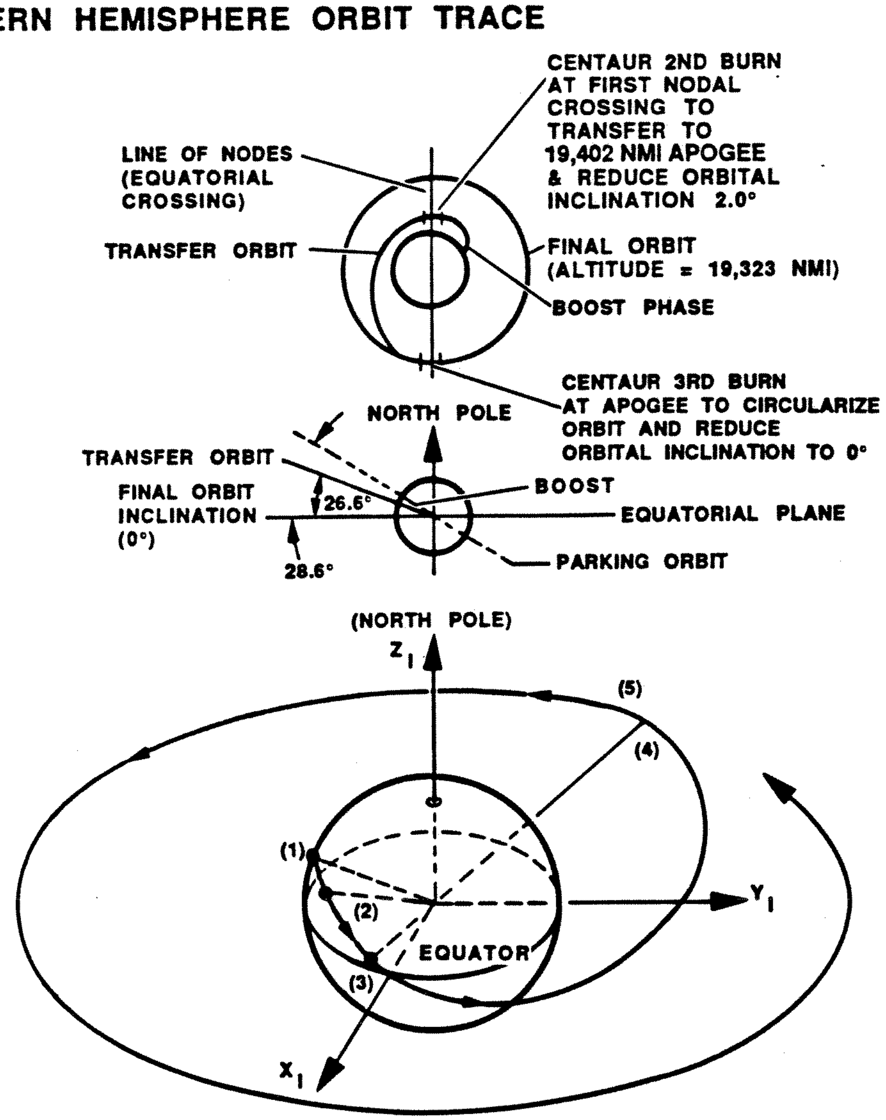
A typical GSO mission profile is presented in Figure 3.5.6.1.1-1 to facilitate understanding the relationships between the orbital parameters. This figure is for a first equatorial crossing transfer orbit burn (satellite positioned in the eastern hemisphere).

The Typical Ascent Profile from liftoff to park orbit injection is presented in Figure 3.5.6.1.1-2 for the GSO mission.

3.5.6.1.2 24-Hour Period Circular Orbit Inclined 65 Degrees

Because of Range Safety considerations, it is impossible to insert the Spacecraft into the required 65 deg orbital plane without either a yaw maneuver during the boost portion of flight or by changing the orbital inclination during the transfer and final Centaur burns. The nominal launch azimuth cannot be less than 37.9 deg (measured clockwise from north) because of Range Safety and this results in a park orbit with an inclination of 55 deg. The chosen method for increasing the final orbit inclination to 65 deg is to increase the inclination 0.6 deg during the Centaur transfer orbit burn and 9.4 deg during the Centaur final orbit burn.

The park orbit for this mission is achieved with a Centaur "first" burn of approximately 17,400 lbm of propellant. When the vehicle is in the vicinity of the equator (on either the first or second crossing) the Centaur transfer orbit burn ("second" Centaur burn, approximately 19,100 lbm propellant) injects the vehicle into an elliptical orbit and increases the orbital inclination approximately 0.6 deg. Centaur final burn ("third" Centaur burn, approximately 17,650 lbm propellant) occurs near apogee and places the vehicle in a circular orbit and increases the orbital inclination to 65 deg.



NOTES: (NOT DRAWN TO SCALE)

1. LAUNCH FROM ESMC (TIME = 0 SEC)
2. PARK ORBIT INJECTION (TIME = 1,018 SEC; ALT = 85.0 NMI PERIGEE, 250 NMI APOGEE ; INC = 28.6 DEG; ECC = 0.01)
3. FIRST EQUATORIAL CROSSING, CENTAUR 2ND BURN, TRANSFER ORBIT INJECTION (TIME = 1703.8 SEC; ALT = 96 NMI PERIGEE X 19,413 NMI APOGEE, INC = 26.6 DEG, ECC = 0.7318)
4. CENTAUR 3RD BURN, FINAL ORBIT INJECTION (TIME = 20528 SEC; ALT = 19,323 NMI; INC = 0.0 DEG, ECC = 0.0)
5. FINAL ORBIT INJECTION OCCURS AT END OF THE 30 SEC SETTLING PHASE

REFERENCE - MCR-86-2515 ISSUE 6 "TITAN IV/CENTAUR 3-D OF REFERENCE TRAJECTORY REPORT," MAY 1990

FIGURE 3.5.6.1.1-1 TITAN IV TYPE I/CENTAUR TYPICAL GSO MISSION ORBIT PROFILE (FINAL SATELLITE POSITIONING IN EASTERN HEMISPHERE)

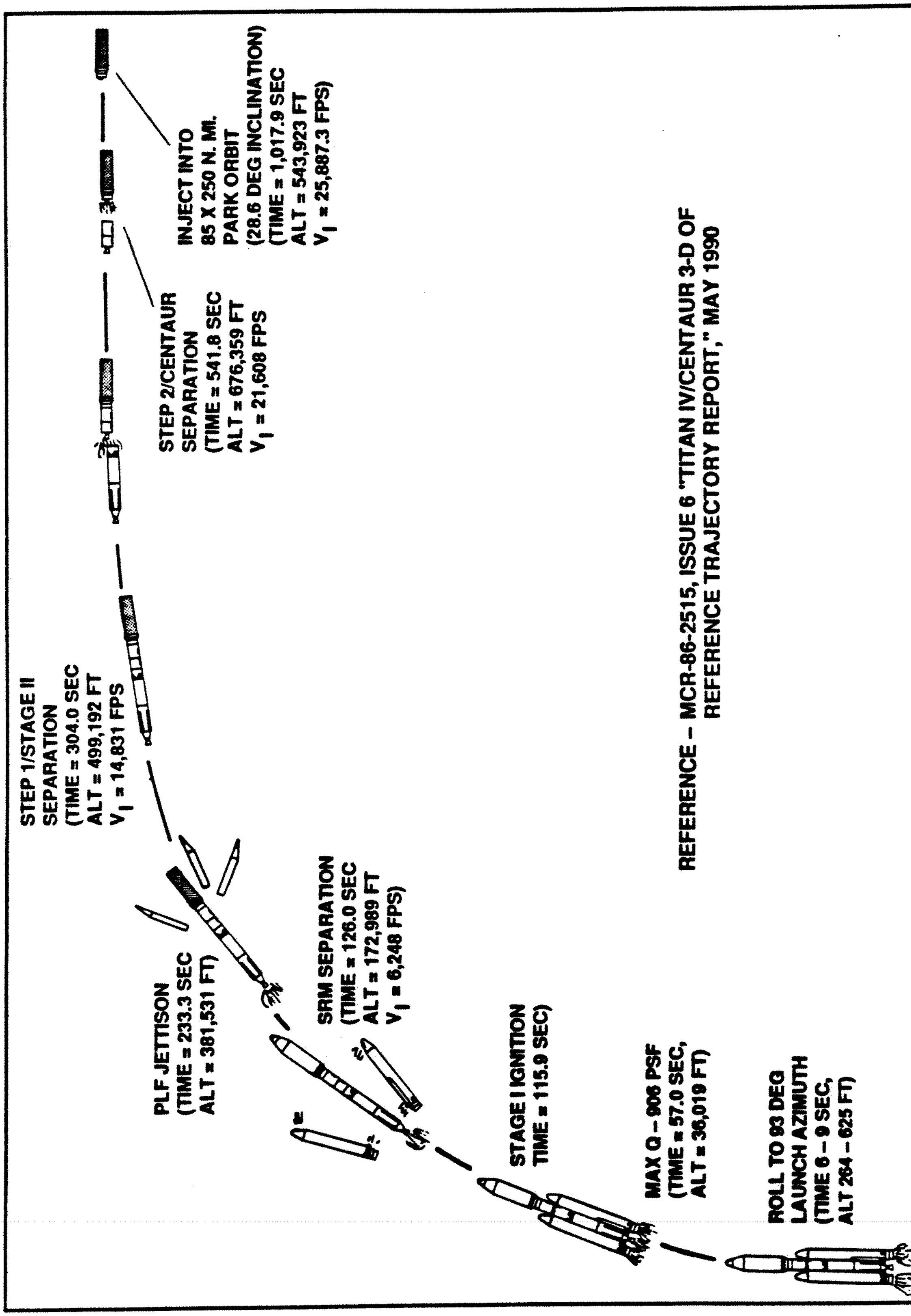
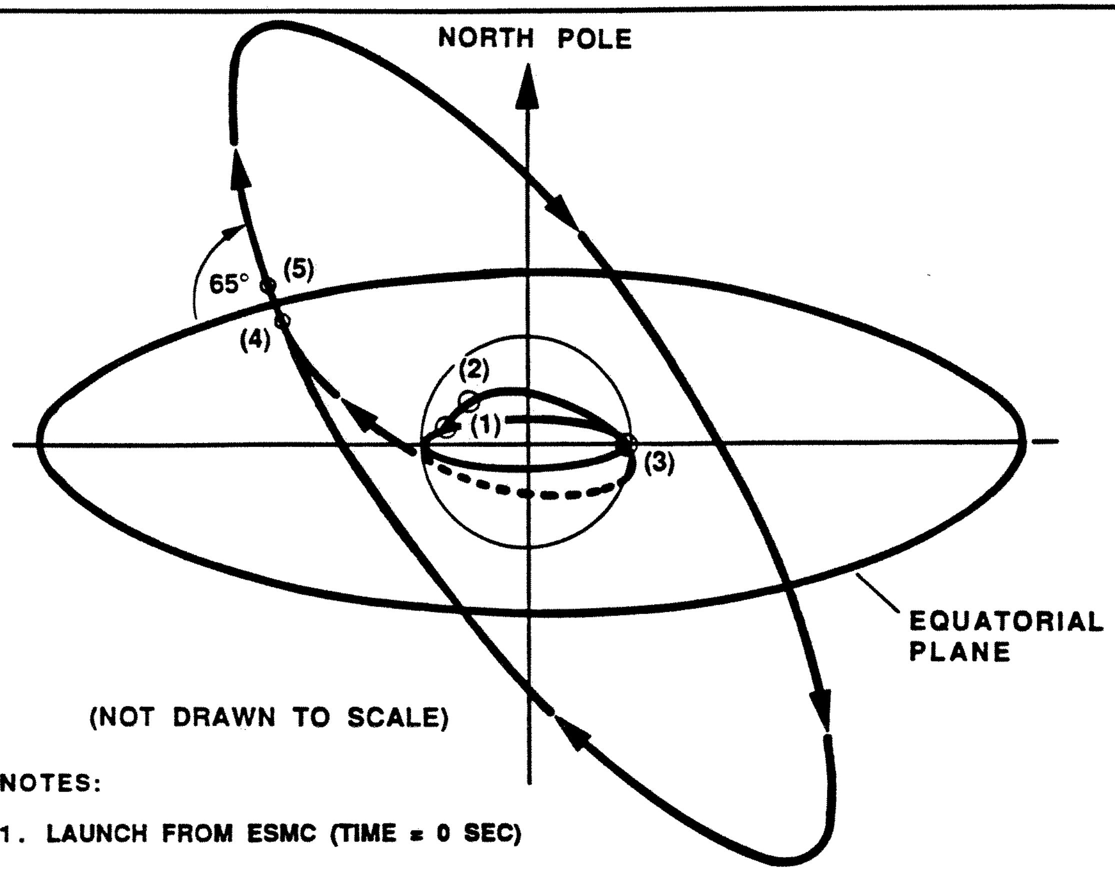


FIGURE 3.5.6.1.1-2 TITAN IV/CENTAUR TYPICAL ASCENT PROFILE FOR GSO MISSION

3.5.6.1.2 24 Hour Period Circular Orbit Inclined 65 Degrees (Continued)

Figure 3.5.6.1.2-1 presents the Orbital Relationships for the 24-Hour Inclined Mission. Typical Ascent Profile from liftoff to park orbit injection for the 24-Hour Inclined Mission is presented in Figure 3.5.6.1.2-2.



NOTES:

1. LAUNCH FROM ESMC (TIME = 0 SEC)
2. PARK ORBIT INJECTION (TIME = 794.5 SEC; ALT = 86 NMI PERIGEE, 156 NMI APOGEE; INC = 55.0 DEG, ECC = 0.0098)
3. FIRST EQUATORIAL CROSSING
   CENTAUR 2ND BURN, TRANSFER ORBIT INJECTION
   (TIME = 2,503 SEC; ALT = 110 NMI PERIGEE, 19,396 NMI APOGEE; INC = 55.65 DEG; ECC = 0.731)
4. CENTAUR 3RD BURN, FINAL ORBIT INJECTION (TIME = 21,131 SEC; ALT = 19,323 NMI; INC =65 DEG; ECC = 0.0)
5. FINAL ORBIT INJECTION OCCURS AT THE END OF THE 30 SEC SETTLING PHASE

REFERENCE – MCR-86-2517 ISSUE 6 "TITAN IV/CENTAUR 3-D OF REFERENCE TRAJECTORY REPORT," MAY 1990

FIGURE 3.5.6.1.2-1 TITAN IV TYPE I/CENTAUR TYPICAL ORBIT PROFILE FOR 24-HOUR PERIOD INCLINED MISSION

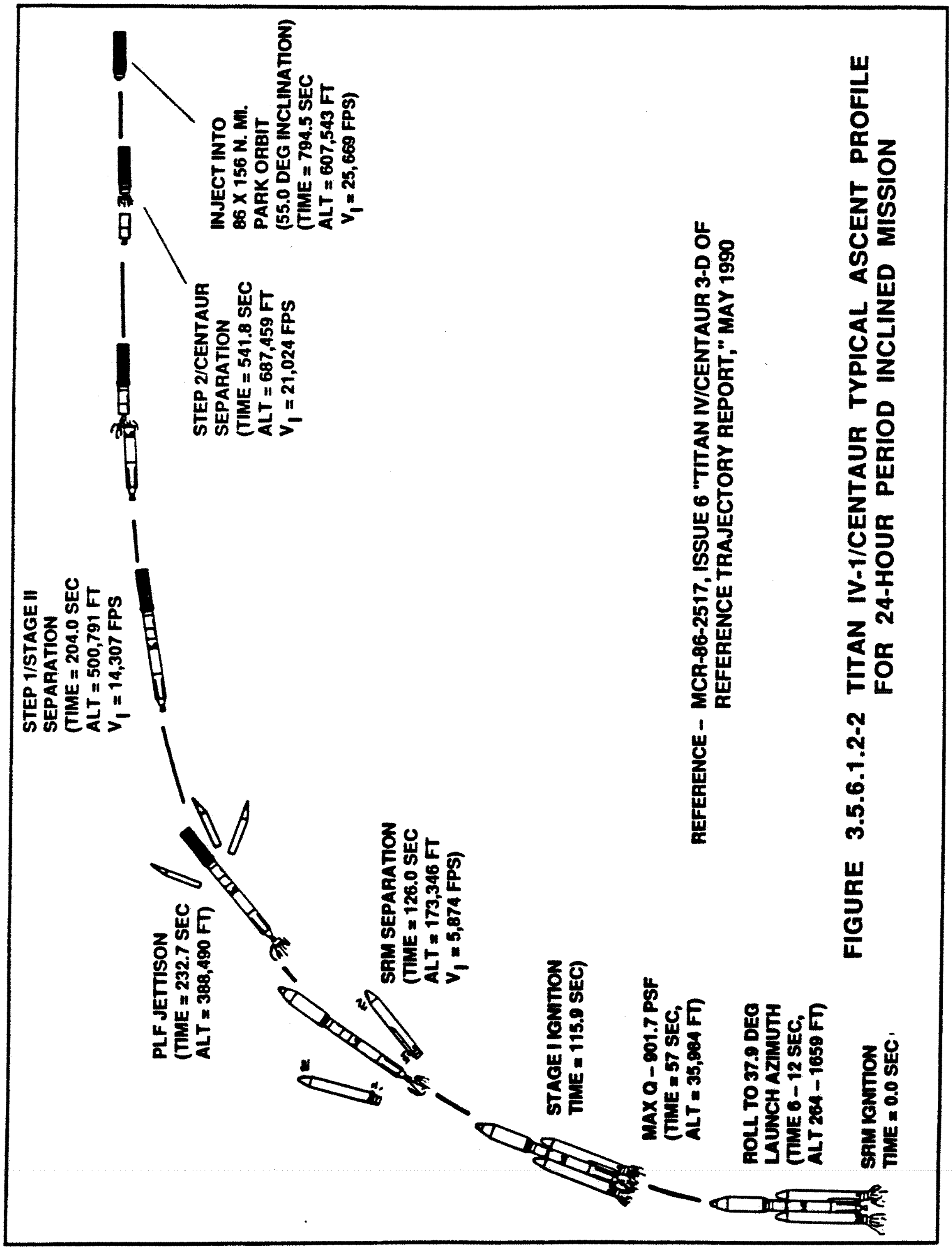


FIGURE 3.5.6.1.2-2 TITAN IV-1/CENTAUR TYPICAL ASCENT PROFILE FOR 24-HOUR PERIOD INCLINED MISSION

3.5.6.1.3 12-Hour Period Elliptical Orbit Inclined 63.4 Degrees

This class of trajectories (sometimes referred to as "Molniya" missions) have similar range safety considerations as the 24-hour Inclined Mission. Range safety concerns limit the park orbit inclination to 55 deg (launch azimuth 37.9 deg) without a yaw dogleg. To achieve the required 63.4 deg final orbit inclination, the inclination is increased 0.04 deg during the Centaur transfer orbit burn with the final inclination increase of 8.36 deg accomplished with the Centaur final burn.

A brief flight description for the Molniya class of missions is as follows: The spacecraft is initially placed into a low-altitude parking orbit after a Centaur "first" burn (approximately 17,160 lbm of propellant). The Centaur engine is then restarted near the most southerly latitude of the orbit (latitude approximately 55 deg south). After this "second" burn (approximately 18,900 lbm of propellant) the perigee is approximately 112 nmi, apogee is approximately 17,488 nmi, orbital inclination is 55.0 deg, and the argument of perigee is approximately 266 deg.

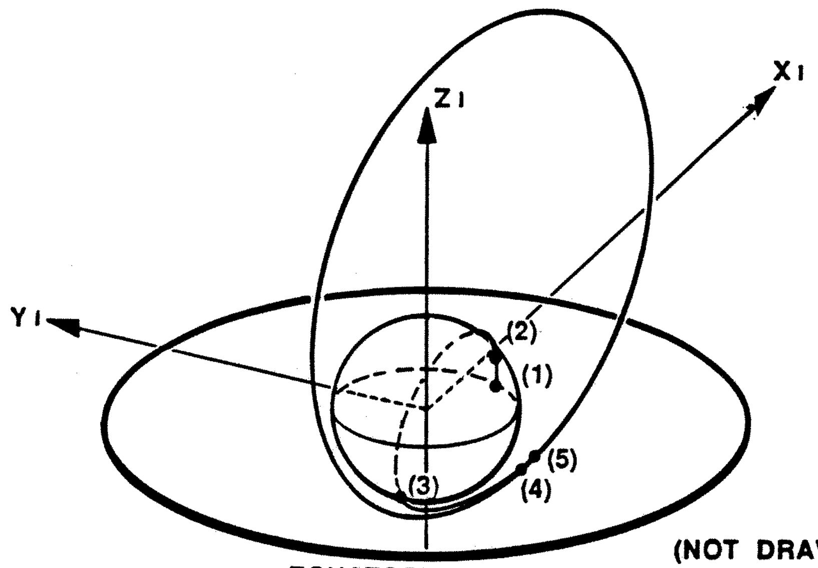
The Centaur final burn (approximately 22,000 lbm propellant) occurs at a latitude of approximately 30 deg north and results in the following final orbit conditions: perigee is 509 nmi, apogee is 21,298 nmi, inclination is 63.4 deg, argument of perigee is 270 deg. The orbital relationships between the park, transfer, and final orbits are presented in Figure 3.5.6.1.3-1.

A typical Ascent Profile from liftoff to park orbit insertion is presented in Figure 3.5.6.1.3-2 for this mission.

3.5.6.1.4 Titan IV Type I/Centaur Launched from ESMC Near Earth Missions for Various Orbital Inclinations

The Titan IV/Centaur is capable of placing SC into various orbital inclinations when launched from ESMC. Figure 3.5.6.1.4-1 presents the Titan IV/Centaur throw weight capability, as a function of Centaur propellant consumed into the park orbit, for various orbital inclinations. This figure presents the launch azimuths and their corresponding orbital inclinations that encompasses the range of nominal launch azimuths which vary from minimum launch azimuths of 37.9 deg measured clockwise from north (north-easterly launch) to the maximum launch azimuth of 112.0 deg (south-easterly launch).

Figure 3.5.6.1.4-1 can be used for preliminary feasibility analysis when the final orbital parameters are defined. Once the delta velocity above the park orbit is defined, for a given inclination, it is possible to calculate the approximate payload capability by using the ideal velocity equation with the remaining Centaur usable propellant and jettison weight.



NOTES:

1. LAUNCH FROM ESMC (TIME = 0 SEC)
2. PARK ORBIT INJECTION (TIME = 793.4 SEC; ALT = 98 NMI PERIGEE, 101 NMI APOGEE, INC = 55.0 DEG)
3. TRANSFER ORBIT INJECTION, CENTAUR 2ND BURN (TIME = 4,140 SEC; ALT = 112 NMI PERIGEE 17,488 NMI APOGEE; INC = 55.0 DEG (ECC = 0.71)
4. CENTAUR 3RD BURN, (TIME = 8,118 SEC; ALT = 509 NMI PERIGEE, 21,298 NMI APOGEE; INC = 63.4 DEG; ECC = 0.72; ARG OF PERIGEE = 270 DEG)
5. FINAL ORBIT INJECTION OCCURS AT THE END OF THE 244 SEC SETTLING PHASE

REFERENCE - MCR-87-2620, ISSUE 6 "TITAN IV/CENTAUR 3-D REFERENCE TRAJECTORY REPORT (12-HR INCLINED MISSION)," MAY 1990

FIGURE 3.5.6.1.3-1 TITAN IV TYPE I/CENTAUR TYPICAL ORBITAL PROFILE FOR 12-HOUR PERIOD ELLIPTICAL ORBIT INCLINED MISSION (ARGUMENT OF PERIGEE = 270 DEG)

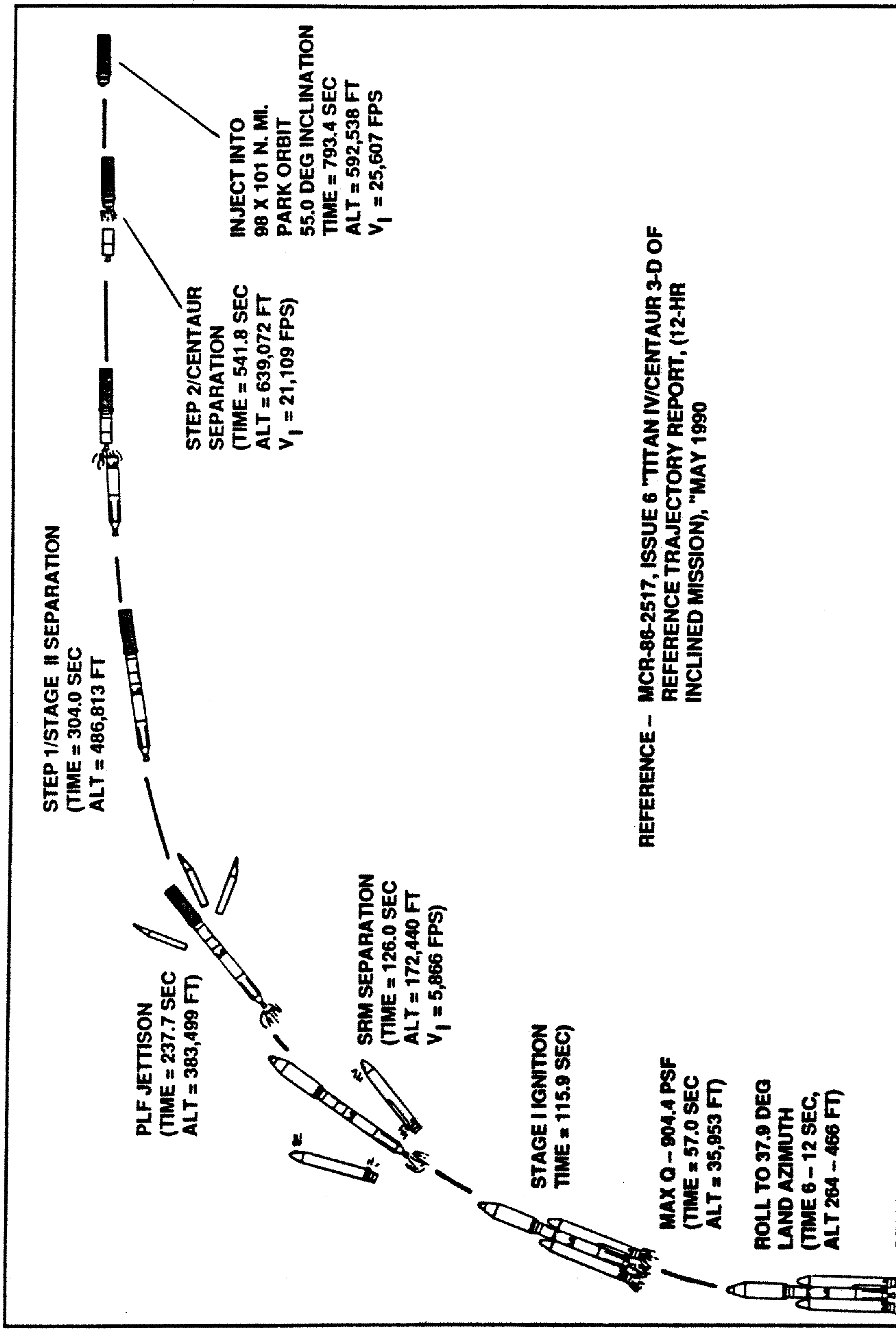


FIGURE 3.5.6.1.3-2 TITAN IV TYPE I/CENTAUR TYPICAL ASCENT PROFILE FOR 12-HOUR PERIOD ELLIPTICAL ORBIT INCLINED MISSION (ESMC)

(TO BE REVISED)

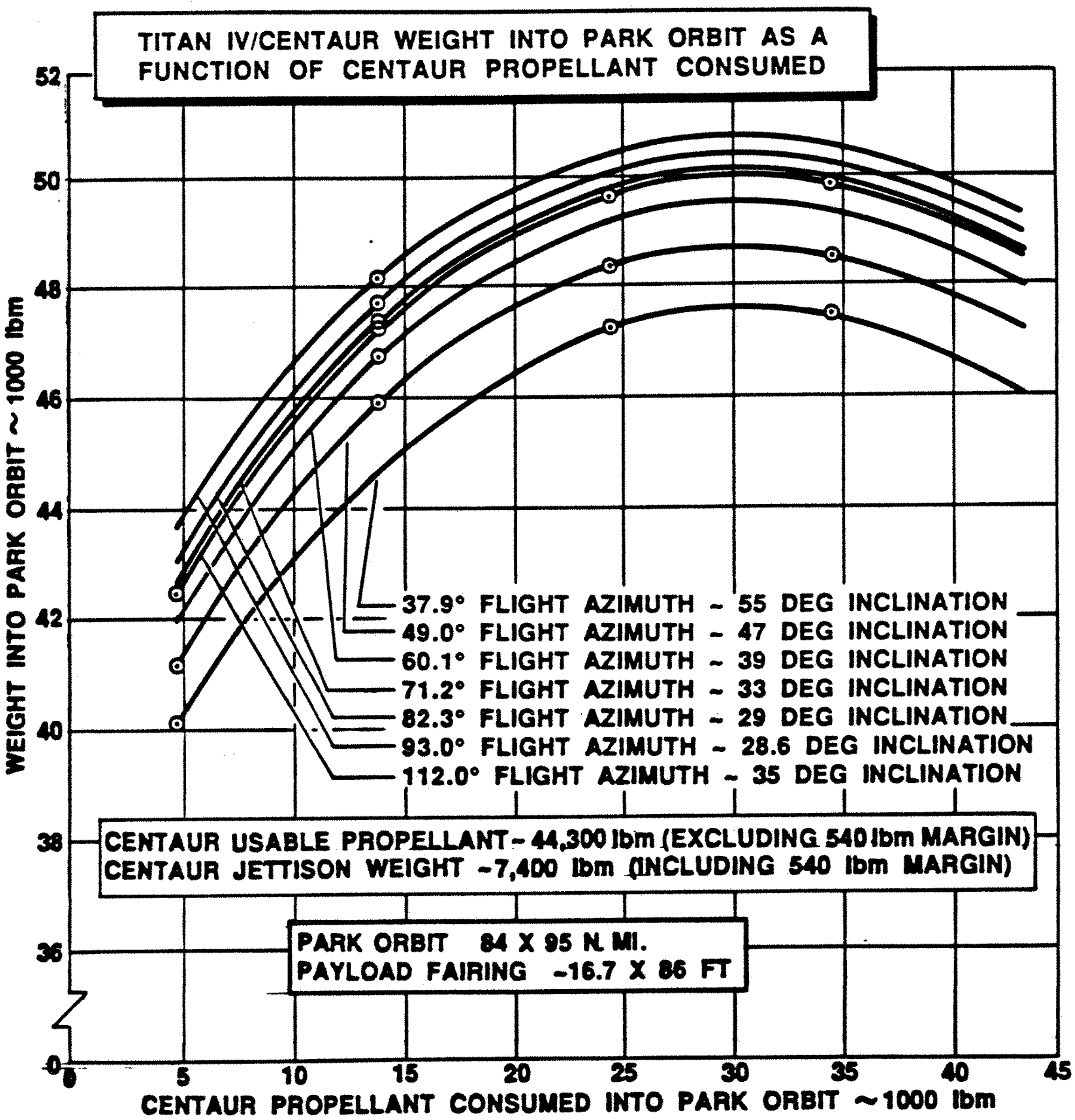


FIGURE 3.5.6.1.4-1 TITAN IV TYPE I/CENTAUR TYPICAL THROW WEIGHT CAPABILITY FOR VARIOUS ORBITAL INCLINATIONS

3.5.6.1.5 Titan IV/Centaur Launched from ESMC Flight Description

A typical mission for the Titan IV/Centaur, for the 24-hour Inclined mission, the 12-hour Inclined mission, or the Geostationary Orbit, begins with SRM ignition and vehicle liftoff. After clearing the launch pad, the vehicle rolls to the desired launch azimuth and begins to pitch over in the trajectory plane. Maximum dynamic pressure occurs approximately 57 seconds after SRM ignition. Stage 0/Stage I overlap portion of flight is initiated when the axial acceleration decreases to 1.3 gs (approximately 116 sec after SRM ignition).

During this overlap phase the SRMs are tailing off while the Stage I engine is ignited and achieves steady-state conditions. This phase lasts approximately 10 seconds and then the SRMs are jettisoned. The next major event is the separation of the PLF which occurs when the Free Molecular Heating rate (FMH) on the vehicle has decreased to an acceptable value (nominally 100 BTU/ft^2-hr). This nominally occurs during Stage I flight between 220 and 245 seconds after SRM ignition with the altitude approximately 380,000 feet. PLF jettison constraints are dependent upon the particular user's requirements and can be adjusted to satisfy individual needs. The nominal payload fairing is 16.7 feet in diameter, 86 feet long and weighs approximately 14,115 lbm. Stage I and Stage II portions of flight are terminated when propellant depletion is detected. Stage II tailoff lasts 11 seconds and then the Centaur vehicle is separated. After a coast phase of several seconds the Centaur prestart (chilldown) phase is initiated. The Centaur "first" burn is terminated when the vehicle achieves the orbital velocity for the required park orbit. This orbit is designed to insure that the vehicle does not exceed the FMH constraint on the exposed payload.

After MECO there is a propellant retention phase which uses the Centaur Reaction Control System (RCS) to accumulate the propellant in the bottom of the tanks and minimize losses due to propellant vaporization.

For the GSO mission, the Centaur transfer orbit burn occurs in the vicinity of the equator at either the first or second crossing (depending on whether the final position of the spacecraft is in the eastern or western hemisphere). This transfer orbit burn is preceded by a propellant settling phase which consists of an RCS burn. Centaur prestart (chilldown), start, steady state and tailoff consume approximately 20,100 lbm of propellant. After Centaur MECO there is another RCS propellant retention phase. The final Centaur burn is initiated at apogee of the transfer orbit. This burn is preceded by a propellant settling phase immediately prior to the Centaur prestart (chilldown). The prestart, start, steady state and tailoff consume approximately 9,200 lbm of propellant. After the payload is separated the Centaur uses its RCS to perform a Contamination and Collision Avoidance Maneuver (CCAM).

3.6 Inertial Upper Stage SS-ELV-402 (ESMC)

3.6.1 Introduction

The IUS is designed and produced by the Boeing Aerospace Company (BAC) and is furnished as Government Furnished Property (GFP) to the Titan IV Launch Vehicle Program. The delivery of the IUS and associated equipment to Cape Canaveral Air Force Station (CCAFS) and the IUS field processing necessary for Titan IV/IUS mate is accomplished by BAC. The mating tasks are accomplished by Martin Marietta. IUS processing and checkout equipment is supplied, operated and maintained by BAC.

The IUS vehicle is 17.0 feet long, 7.6 feet in diameter in the cylindrical section, flaring to 9.5 feet at the forward end. It consists of an aft skirt, a relatively large First Stage Solid Rocket Motor (SRM-1), an interstage, a smaller second stage Solid Rocket Motor (SRM-2) and an Equipment Support Section (ESS) that interfaces with the Spacecraft. The ESS contains avionics for guidance and navigation, a RCS, an Electrical Power System, and a Telemetry, Tracking and Command System, reference Figure 3.6.1-1 Titan IV/IUS Vehicle Configuration.

3.6.2 Structural

The IUS Vehicle with an attached Spacecraft, is itself structurally attached to the Titan IV Booster Vehicle via its Aft Skirt and the Titan IV IUS Adapter CP2491. The Aft Skirt and the CP2491 adapter interface contains a circumferential linear charge (Super Zip) which is detonated to structurally separate the Booster Vehicle from the IUS.

The Spacecraft is bolted to the IUS Equipment Support Section (ESS) Spacecraft interface ring located at the top of the ESS. The Spacecraft separation joint is part of the Spacecraft structure and is located forward of the Spacecraft interface ring.

The plane between the IUS first stage and the second stage is at the interstage interface ring where the Aft side of the ESS is attached to the interstage with redundant back-to-back pyrotechnic separation nuts which are activated by NASA standard initiators, reference Figures 3.6.2-1 and 4.3.1.1-1.

3.6.3 Propulsion Systems

The IUS propulsion system consists of one large and one small United Technologies SRM and an RCS.

LEGEND:

- TITAN IV FLIGHT VEHICLE

  1) SATELLITE VEHICLE (S/V)
  2) IUS VEHICLE (GFP)
  3) IUS ADAPTER (GFP)

  - TITAN IV LAUNCH VEHICLE (LV)

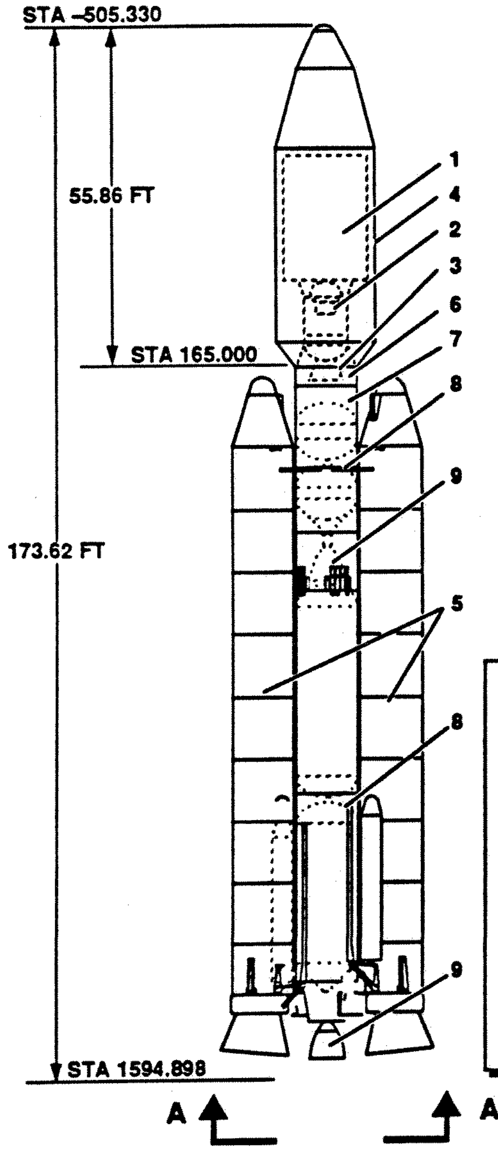
    4) PAYLOAD FAIRING (PLF)

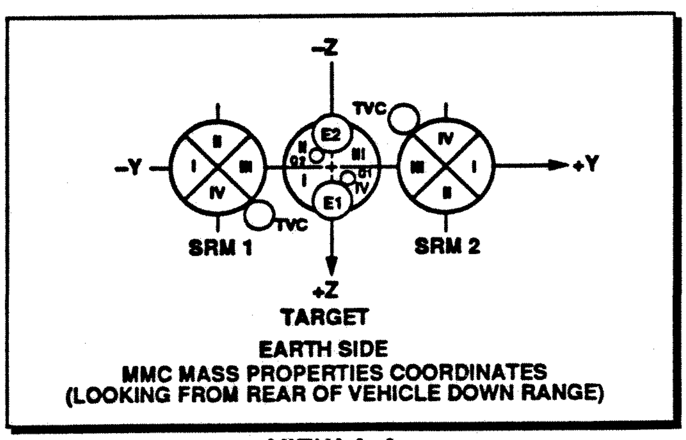
    - TITAN BOOST VEHICLE (BV)

      5) SOLID ROCKET MOTORS (SRMs, STAGE 0, SHOWN) OR SOLID ROCKET MOTORS UPGRADE (SRMUs, STAGE 0)

      - TITAN IV CORE VEHICLE (CV)

        6) STAGE II FORWARD SKIRT EXT (2490)
        7) STAGE II FORWARD OXIDIZER SKIRT (2460)
        8) STAGES I AND II
        9) LIQUID ROCKET ENGINES, STAGES I AND II





VIEW A-A

FIGURE 3.6.1-1 TITAN IV/IUS FLIGHT VEHICLE CONFIGURATION (402)

PARAMETER	VALUE
THRUST (LB)	
STAGE I	41,576
STAGE II	16,742
SPECIFIC IMPULSE (SEC)	
STAGE I	293.30
STAGE II	301.22
PROPELLANT WT (LB)	
STAGE I (NOMINAL ONLOADS)	21,584
STAGE II (NOMINAL ONLOADS)	6,060
DRY WT (LB)	
STAGE I	2,462
STAGE II	2,216
RCS PRO WT (LB)	245

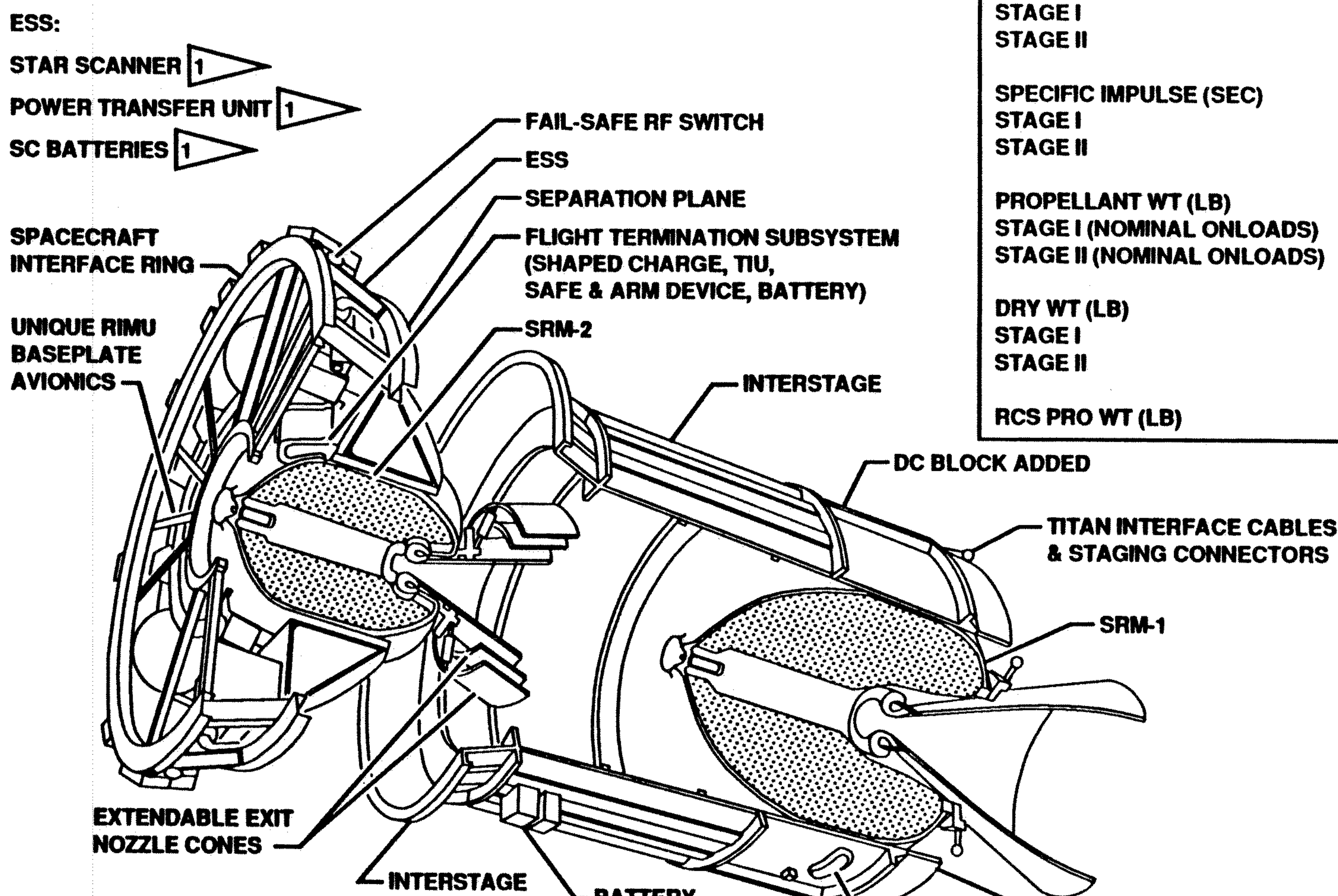


FIGURE 3.6.2–1 IUS CUTAWAY VIEW

3.6.3.1 Thrust Vector Control

An electrically operated vectoring nozzle on each SRM motor provides for directional control during motor burning. For greater adaptability to various missions and P/L weights, the SRM system is designed for a variable propellant loading range of 50% to 100% of maximum capacity.

The rocket nozzle is capable of angular deflection in any direction to an angle of 4 deg for SRM-1 and 7 deg for SRM-2, over the full range of motor chamber pressure to produce the required vehicle pitch and yaw control torques. Roll torque is supplied by the RCS roll-axis thrusters.

A Motor Case Cutter (MCC) is installed on SRM-1 and SRM-2. The MCC is part of the FTS and renders the SRMs nonpropulsive in the event of IUS or Titan IV Flight Vehicle breakup or on receipt of a Destruct Command from the Titan IV Booster Avionics, reference Figure 3.6.2-1.

3.6.3.2 Reaction Control System

The RCS is a mono-propellant hydrazine system which uses blowdown pressurization. It is housed in the avionics bay of the IUS, surrounding the upper dome of SRM-2.

RCS activation occurs approximately 30 sec after IUS separation from Titan. The RCS operational requirements include: orienting the vehicle prior to SRM firings, roll control during the SRM firings, vernier corrections for motor impulse variations, attitude control and maneuvering during transfer orbit coast period, P/L spinup if required and attitude control and maneuvering after P/L separation.

Six Rocket Engine Modules (REMs) each with a pair of thrusters are utilized. Pitch-and-yaw thrusters face aft to preclude SC plume impingement. Roll thrusters operate in pure-couple pairs and are located at the same tangential location as the yaw thrusters.

3.6.4 Electrical Systems

The Titan IV core vehicle provides the electrical power, switching, instrumentation, guidance, navigation and control required to boost the Titan IV/IUS to the point of IUS separation from the core vehicle, independent of the IUS avionics system, reference Figures 3.6.4-1, 3.6.4-2 and 3.6.2-1.

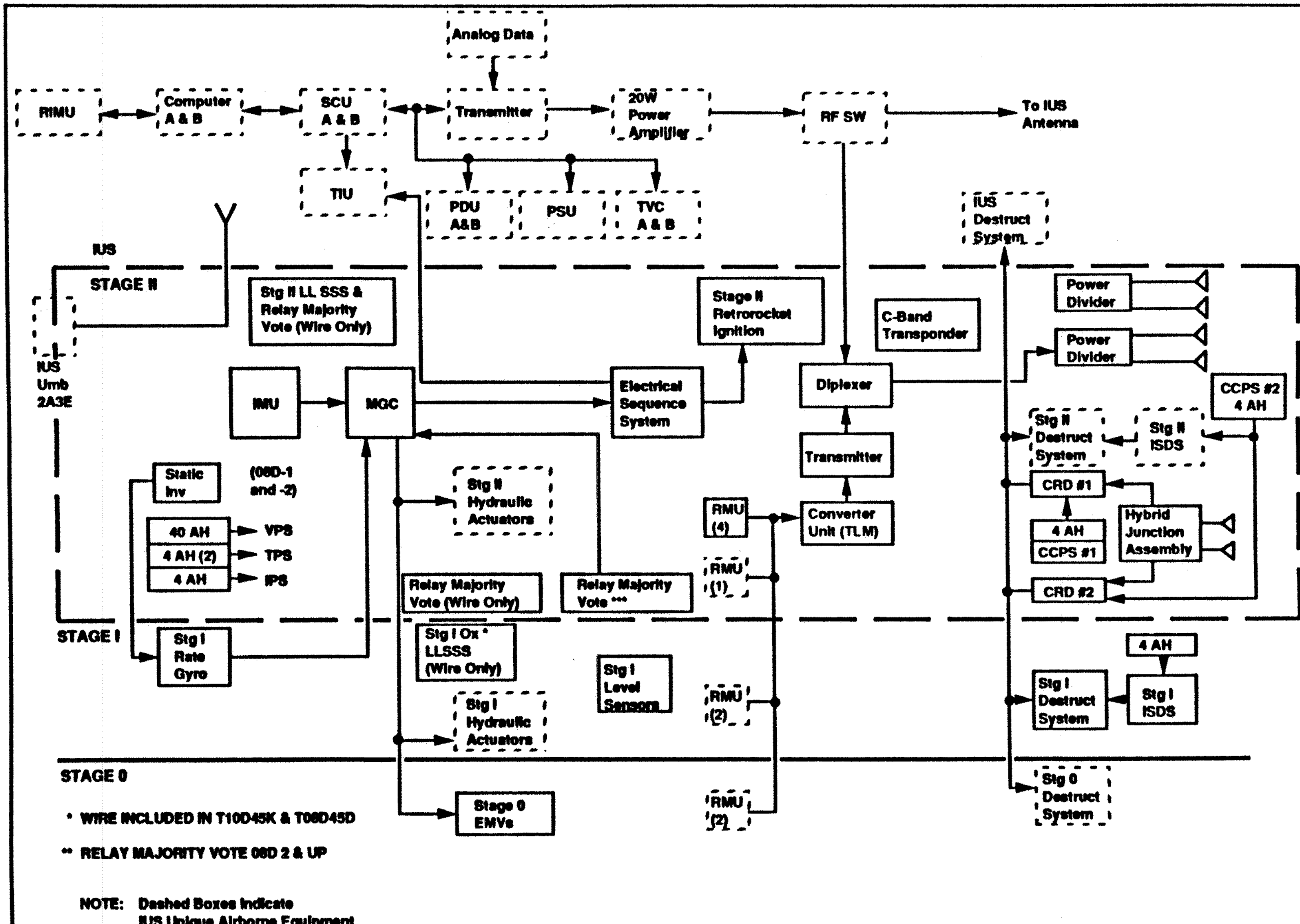


FIGURE 3.6.4-1 TITAN IV/IUS AVIONICS INTERFACES

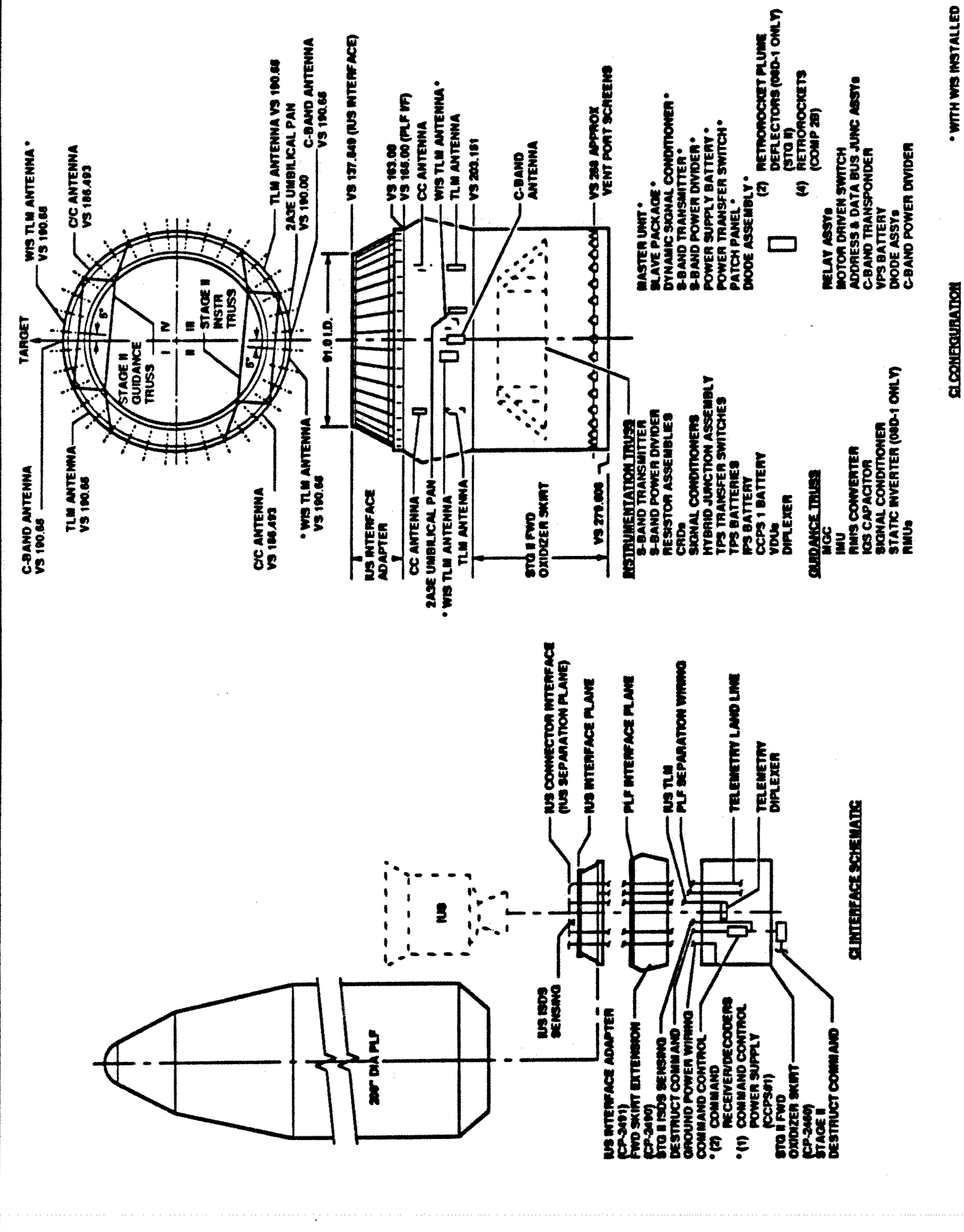


FIGURE 3.6.4-2 IUS UNIQUE AIRBORNE EQUIPMENT

3.6.4.1 Guidance, Navigation and Control

The IUS Inertial Guidance and Navigation Subsystem provides autonomous flight guidance for the IUS vehicle. It measures angular rates and linear accelerations and provides this information to the Data Management Subsystem (DMS). The system uses a strapped-down Redundant Inertial Measurement Unit (RIMU) that consists of five rate-integrating gyros and five accelerometers in a skewed conical array. Data from all five gyroaccelerometer sets are simultaneously sent to both computers of the DMS.

The IUS can store the current on board State Vector at any time during the mission and subsequently downlink the data automatically.

The DMS consists of two computers, two SCUs and an SIU. The DMS performs calculations, data processing and signal conditioning associated with guidance, navigation and control, safing, arming and firing of the IUS SRMs and Electro-Explosive Devices (EEDs); command decoding; telemetry formatting; redundancy management; and discrete issuance, reference Figure 4.3.3-1. Each computer has a 65,536 16-bit-word memory and an operational capability of at least 550,000 operations per second.

Attitude control in response to guidance commands is provided by TVC during powered flight and by RCS thrusters during coast.

Measured attitude from the navigation system is compared with guidance commands to generate error signals. During SRM burn, these error signals drive the motor nozzle actuators. The resulting nozzle deflections produce the desired attitude control torques in pitch and yaw. Roll control is maintained by the RCS roll-axis thrusters. During coast flight, the error signals are processed in the computer to generate RCS thrusters commands to maintain vehicle attitude or to maneuver the vehicle.

The TVC subsystem provides the interface between the IUS Guidance, Navigation and Control System and the SRMs gimbaled nozzles to accomplish thrust vector controlled flight.

Electric power is supplied from the Power Distribution Unit (PDU) to the TVC controller which controls the electrically driven actuators. The controller receives analog pitch and yaw commands, proportional to desired nozzle angles, and converts them to pulse-width-modulated voltages to power the actuator motors.

3.6.4.2 Telemetry, Tracking and Command

For a schematic description of the IUS Telemetry, Tracking and Command (TT&C) subsystem refer to Figures 3.6.4.2-1 and 4.3.3-1.

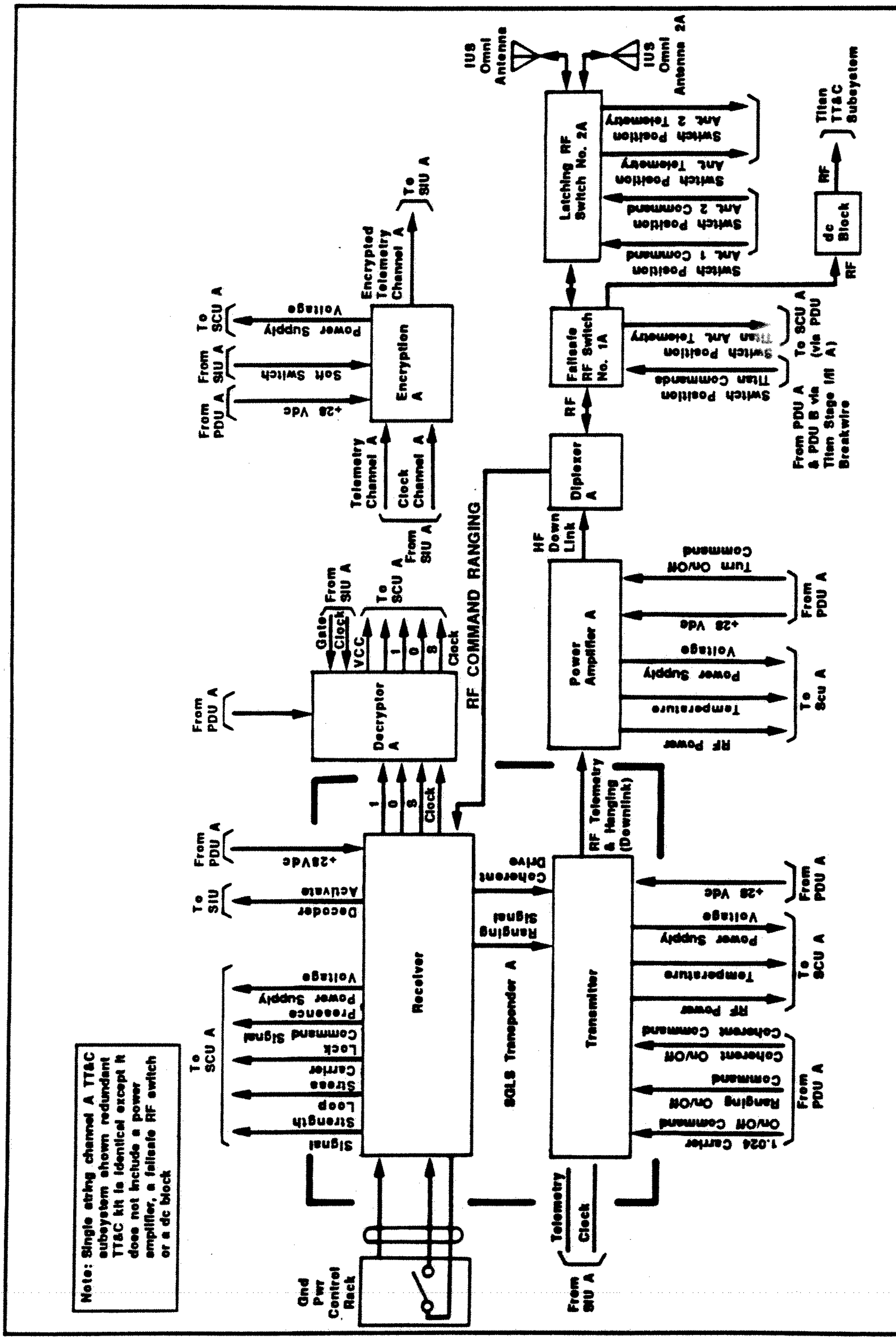


FIGURE 3.6.4.2-1 TT&C SUBSYSTEM FUNCTIONAL FLOW DIAGRAM

3.6.4.2.1 General

The IUS TT&C subsystem is compatible with the Space Ground Link Subsystem (SGLS). During orbital flight, the TT&C subsystem operates in conjunction with the Air Force Satellite Control Facility - Remote Tracking Station (AFSCF-RTS). The IUS TT&C subsystem receives uplink command and ranging signals, demodulates the command data and sends it to the DMS. It transmits downlink signals consisting of encrypted telemetry data and the turnaround ranging signal. The AFSCF-RTSs determine IUS orbital trajectory, using the two-way ranging signal and tracking antenna angle information. Range-rate is determined through coherent carrier Doppler shift.

3.6.4.2.2 Telemetry

During the early part of the Titan boost phase, while the Titan PLF still encloses the IUS, the IUS TT&C subsystem utilizes the Titan S-band antenna for telemetry transmission. The Titan S-band antenna and diplexer are not designed to pass uplink commands or uplink ranging signals through to the IUS. Shortly after PLF separation, Titan Stage I/II staging occurs causing the IUS failsafe RF switch to transfer the IUS to its own omni antennas. Thereafter, full uplink command reception, downlink telemetry and turnaround ranging capability is provided. The IUS RF path is switched to the IUS antennas prior to IUS/Titan separation to assure that the separation event can be monitored without telemetry interrupt.

3.6.4.2.3 Tracking

The TT&C subsystems uses the RTS generated uplink carrier and ranging signal for tracking functions. The IUS can receive uplink RF signals only after the RF link is switched to the IUS antennas after PLF separation. The transponder tracking controls are set to enable both the coherent mode and the ranging mode prior to launch.

3.6.4.2.4 Commands

Prior to launch, GSE baseband commands are sent through the Titan umbilical to the IUS. For umbilical commands, the transponder auxiliary command enable line is enabled by the GSE. The enable line is broken at launch to interrupt the auxiliary command enable signal which disables landline command and enables RF commands at the IUS receiver. After Launch, the TT&C subsystem can receive RF uplink commands only after PLF separation and the RF link is switched from the Titan antenna to the IUS antenna. On Titan/IUS missions, only the 0.3 modulation index of the SGLS RF carrier signal is applicable for uplink commands.

Uplink commands to the IUS are used to change inflight sequencing, or to request data dumps from the IUS.

3.6.4.3 Flight Termination System

The Titan IV/IUS FTS is used to destroy the pressure integrity of the IUS SRMs in the event of premature IUS stage separation, premature IUS/Titan IV separation or if a command destruct signal is transmitted to the Titan IV command shutdown and destruct system via the Titan IV CRDs.

The command destruct signal is transmitted when the Titan IV/Flight Vehicle departs from the planned trajectory as determined by the flight RSO.

The RSO also has the option to shutdown the Titan IV core engines to achieve a zero thrust condition. The flight path is monitored with skin track radar, C-band transponder and vehicle guidance data.

The IUS DMS provides command/control of the IUS FTS during ground test and prelaunch countdown. After liftoff, FTS safing commands and IUS/Titan IV separation commands are initiated by the Titan vehicle guidance computers.

The IUS FTS is armed one minute before launch by uplink command from the IUS Checkout Station (COS). Once the FTS is armed, loss of both IUS utility power buses while destruct power remains on would result in destruction of the IUS SRMs. Inadvertent turnoff of utility bus power during countdown is prevented by a GSE hardware inhibit that precludes sending ground commands that turn off the utility buses. For launch abort, the COS safes the IUS FTS, removes the GSE inhibit and safely powers down the IUS.

After completion of a successful boost phase and Titan IV Stage II engine cutoff, the Titan IV MGC safes both the Titan IV and IUS FTSs, initiates the Titan IV Retrorockets and fires the IUS Super-Zip to separate the IUS-S/V from Titan Step 2. The destruct safing commands are not issued to the IUS if the Titan calculated park orbit radius of perigee and angular momentum do not ensure at least one park orbit revolution, reference MCR-87-2563 Flight Termination Report and Figure 3.6.4.3-1.

3.6.4.4 Electrical Subsystem

The IUS receives +28 Vdc power from the GSE prior to launch and from the internal stage batteries from 3 min before launch through battery depletion. Stage I batteries include: Avionics A, Avionics B and a SC battery (option). The Avionics A battery consists of one set of 170 Ampere Hour (AH) batteries. The Avionics B battery consists of one set of 140 AH. The SC Batteries are an option. Stage II batteries include: Avionics A, Avionics B, Utility A, Utility B and an SC battery (option). These batteries are rated at 13 AH and packaged as a single unit. Stage I and II battery AH capacities can vary from these values for specific mission requirements. The 2A3E electrical umbilical provides cabling for the ground electric power to the IUS.

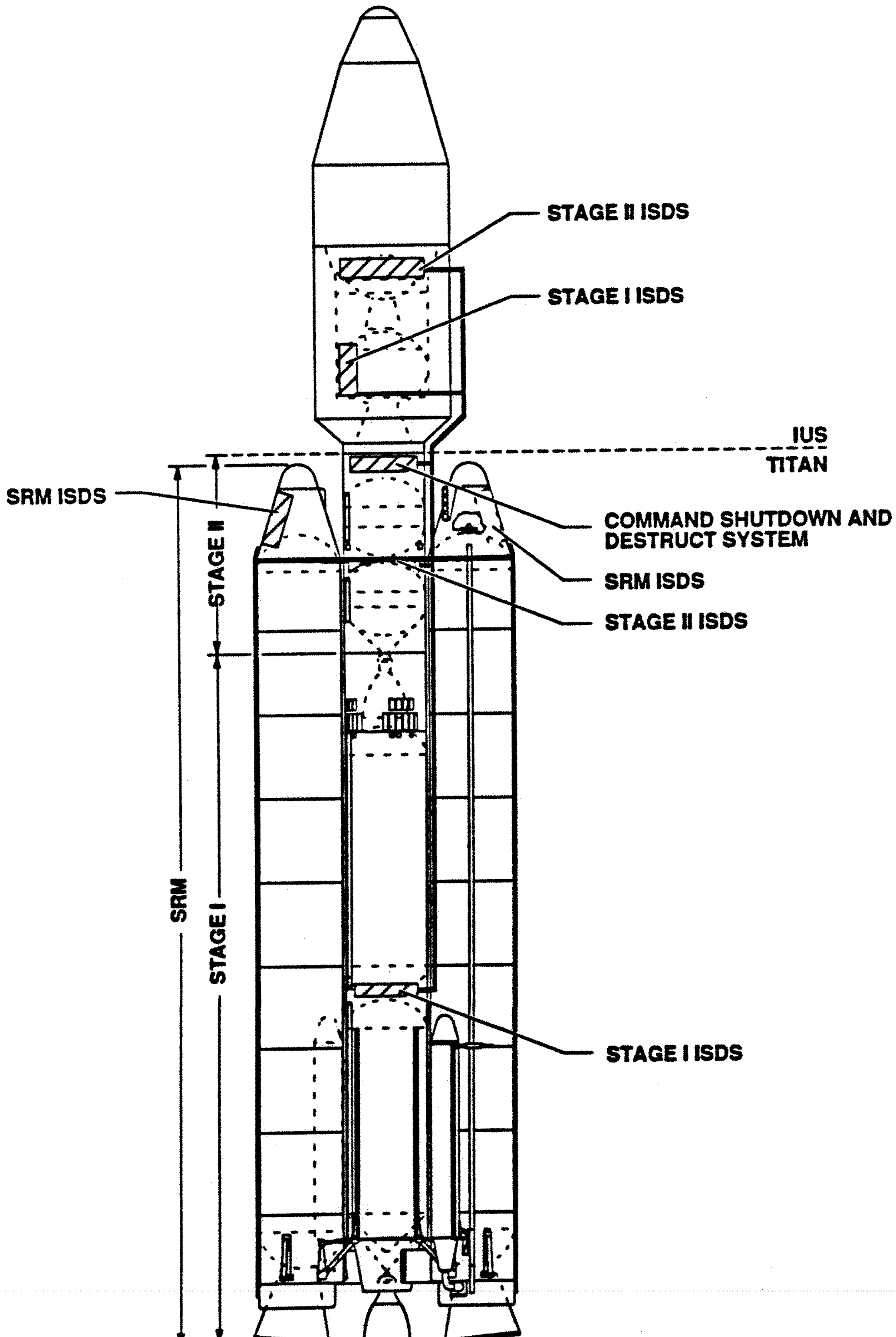


FIGURE 3.6.4.3-1 FLIGHT TERMINATION SYSTEM

3.6.5 Flight Software

The IUS operational Flight Software Subsystem is organized into eight functional areas: executive, mission sequencing, navigation, guidance, attitude control, communications, redundancy management and checkout. The executive function controls timing synchronization, memory load, input-output and scheduling of all flight operational software tasks. Mission sequencing controls timing and sequencing of events related to the IUS vehicle. Navigation consists of determining position, velocity and inertial attitude from vehicle linear acceleration and angular velocity (as sensed by the RIMU). This software function includes initialization, alignment and calibration. The guidance function determines the inertial attitude required to maintain the trajectory that will attain the desired set of end conditions. The guidance function also controls the ignition time of each SRM burn and the start time and duration of RCS vernier burns. Attitude control determines pitch and yaw commands to be sent to the TVC subsystem during SRM burn and roll, pitch and yaw commands to the RCS during coast periods. Attitude control also generates RCS roll commands to compensate for swirl torques during SRM burn. The communications function processes commands received by the vehicle, processes telemetry data and controls antenna switching. Redundancy management controls computer self-test and all fault detection, fault isolation and reconfiguration of the computers, RIMU, TVC, TT&C and the SCU. The checkout function performs prelaunch and predeployment checkout of the computers, RIMU, TT&C, RCS and SCU.

3.6.6 Titan IV Type I/IUS Performance Capability

The Titan IV/IUS can deliver payloads to the required positions for a number of missions among which are the Geosynchronous Equatorial Orbit (GSO) or the near GSO missions and Interplanetary Transfer Missions.

Trajectory simulations used to generate the Titan IV/IUS performance capabilities satisfy the ground rules and trajectory simulation constraints presented in Tables 3.6.6-1 and 3.6.6-2.

Trajectory simulations reserve enough propellant margin in Stage II to accommodate three-sigma booster vehicle dispersions and enough RSC propellant to cover the three-sigma low conditions during the IUS portion of flight.

A typical sequence of events for the GSO mission is presented in Table 3.6.6-3 for Titan IV/IUS. The relationship between the various orbits is presented in Figure 3.6.6-1 which presents the Typical Mission Profile for the second crossing GSO Mission.

A Typical Ascent Profile is presented in Figure 3.6.6-2. This figure shows selected trajectory parameters during the boost portion of flight. The performance capability for Titan IV/IUS is presented in Table 3.6.6-4

For specific mission performance data contact the USAF Space Division (SD/YXT) or Martin Marietta.

TABLE 3.6.6-1 TITAN IV TYPE I/IUS TYPICAL TRAJECTORY SIMULATION GROUND RULES

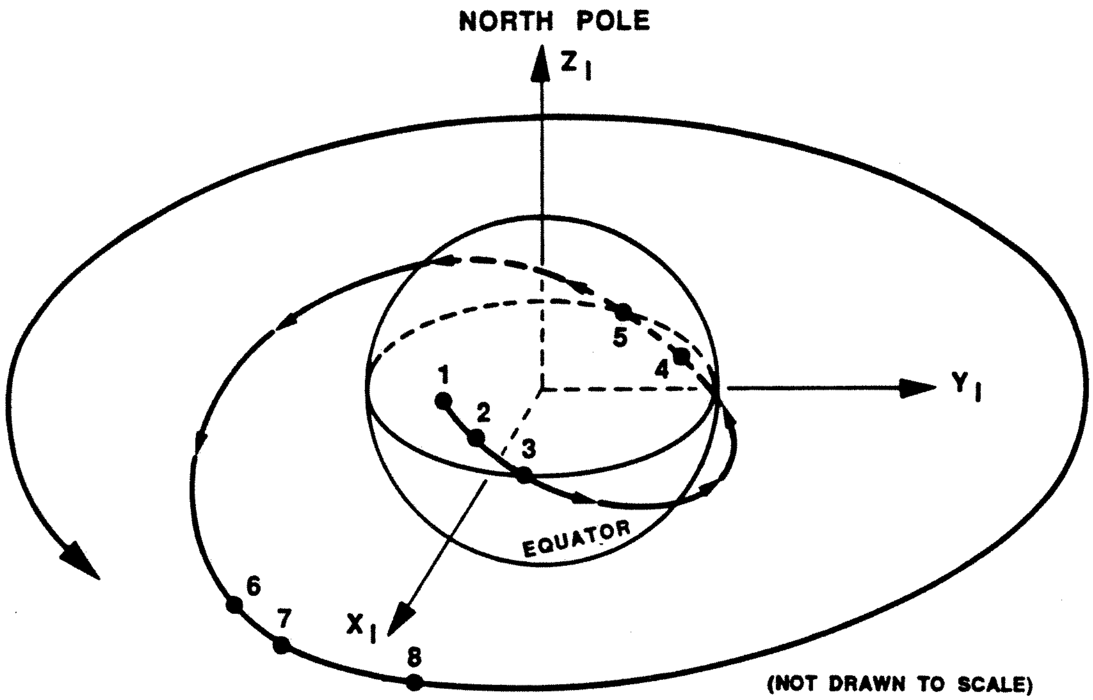
VEHICLE CHARACTERISTICS	
Stage 0 SRM Temperature	71.5° F
SRM Nozzle Exit Area	12,491 in.^{2}/SRM
Stage I Propellant Temperature (Ox & Fuel)	72.5° F
Stage II Propellant Temperature (Ox/Fuel)	70.0° F/72.5° F
Reduced Ullage with Optimum Fuel Bias on Stages I and II, Propellant depletion shutdown on Stage I	
Guidance Command Shutdown on Stage II	
Average Stage I Nozzle Centerline Thrust	546,267 lbf
Average Stage I Nozzle Centerline ISP	301.18 sec
Average Stage II Vacuum Thrust (Including Roll Nozzle)	106,392 lbf
Average Stage II Vacuum ISP (Including Roll Nozzle	317.66 sec
Payload Fairing:	
Dimensions (diameter & length)	16.7 x 56 ft
Weight	8,884 lbm
Inertial Upper Stage:	
Stage I Average Vacuum Thrust	41,780 lbf
Stage I Average Effective ISP	294.1 sec
Stage II Average Vacuum Thrust	17,222 lbf
Stage II Average Effective ISP	302.2 sec (with Extendible Exit Cone, EEC)
MISSION PROFILE	
Launch Azimuth	93.0 deg
Stage 0 Separation Sequence Initiated when Axial Acceleration Decreases to 1.3 Gs	
PLF Jettisoned in Stage I when FMH rate <150 (BTU-ft^2)/hr	

TABLE 3.6.6-2 TITAN IV TYPE I/IUS TYPICAL THREE DEGREE-OF-FREEDOM TRAJECTORY SHAPING CONSTRAINTS

- Dynamic pressure (Q): not to exceed 975 lbf/ft^2
- Aerodynamic Heating Indicator (AHI):
  not to exceed 95.0×10^6 $ft\text{-}lbf/ft^2$
- Stage 0 Separation:
  Dynamic Pressure shall not exceed 60 lbf/ft^2
  Pitch Angle-of-Attack shall not exceed ± 4.5 deg
  Yaw Angle-of-Attack shall not exceed ± 5.0 deg
- Stage I Separation:
  Dynamic Pressure shall not exceed 30 lbf/ft^2
  Total Angle-of-Attack shall not exceed 15.0 deg
- Nominal FMH rate on the exposed payload
  Shall not exceed 150 (BTU/ft^2)/hr at or following PLF jettison
- Perigee altitude of park orbit not less than 70 nmi
- Apogee altitude of park orbit not less than 90 nmi relative to equatorial radius of Earth.

TABLE 3.6.6-3 TITAN IV TYPE I/IUS TYPICAL SEQUENCE-OF-EVENTS FOR THE GSO MISSION (Transfer Orbit Initiated During Second Equatorial Crossing)

	GSO Mission (sec)
SRM Ignition	0.0
Begin Roll to Flight Azimuth	6.0
End Roll Maneuver	9.0
Begin Pitch Over	10.0
Maximum Dynamic Pressure	56.3
Stage I Ignition	116.3
SRM Separation	126.2
Jettison PLF	229.7
Step 1/Stage II Separation	302.4
Step 1/IUS Separation, Park Orbit Inject	534.9
IUS First Stage Ignition	4,137.5
IUS First Stage Shutdown, Begin Tailoff	4,290.2
End Tailoff, Transfer Orbit Inject	4,646.6
IUS Second Stage Ignition	23,068.7
IUS Second Stage Shutdown, Begin Tailoff	23,178.1
Final Orbit Inject	23,582.8
End of Mission	26,000.0



NOTES:

1. LAUNCH FROM ESMC (TIME = 0 SEC)
2. PARK ORBIT INJECTION (FOR OFF PERIGEE INJECTION, TIME = 535 SEC, ALT = 93 NMI, INC = 28.6 DEG, INERTIAL FLIGHT PATH ANGLE = +0.99 DEG, ECC = 0.021)
3. FIRST EQUATORIAL CROSSING, DESCENDING NODE (TIME = 1519 SEC, ALT = 171 NMI)
4. APOGEE OF PARK ORBIT (TIME = 2417 SEC, ALT = 216 NMI)
5. SECOND EQUATORIAL CROSSING, ASCENDING NODE, STAGE III BURN AND RCS 1ST VERNIER BURN, TRANSFER ORBIT INJECTION (TIME = 4646 SEC, ORBIT 96 x 19,390 NMI, INC = 26.6 DEG, ECC = 0.732)
6. STAGE IV BURN AND RCS 2ND VERNIER BURN, FINAL ORBIT INJECTION (TIME = 23583 SEC, ALT = 19,323 NMI, INC = 0.0 DEG, ECC = 0.0)
7. SPACECRAFT SEPARATION (TIME = 23757 SEC, ALT = 19,323 NMI)
8. END OF MISSION FOR IUS (TIME = 26000 SEC, ALT = 19,323 NMI)

FIGURE 3.6.6-1 TITAN IV TYPE I/IUS TYPICAL ORBIT PROFILE FOR SECOND CROSSING GSO MISSION

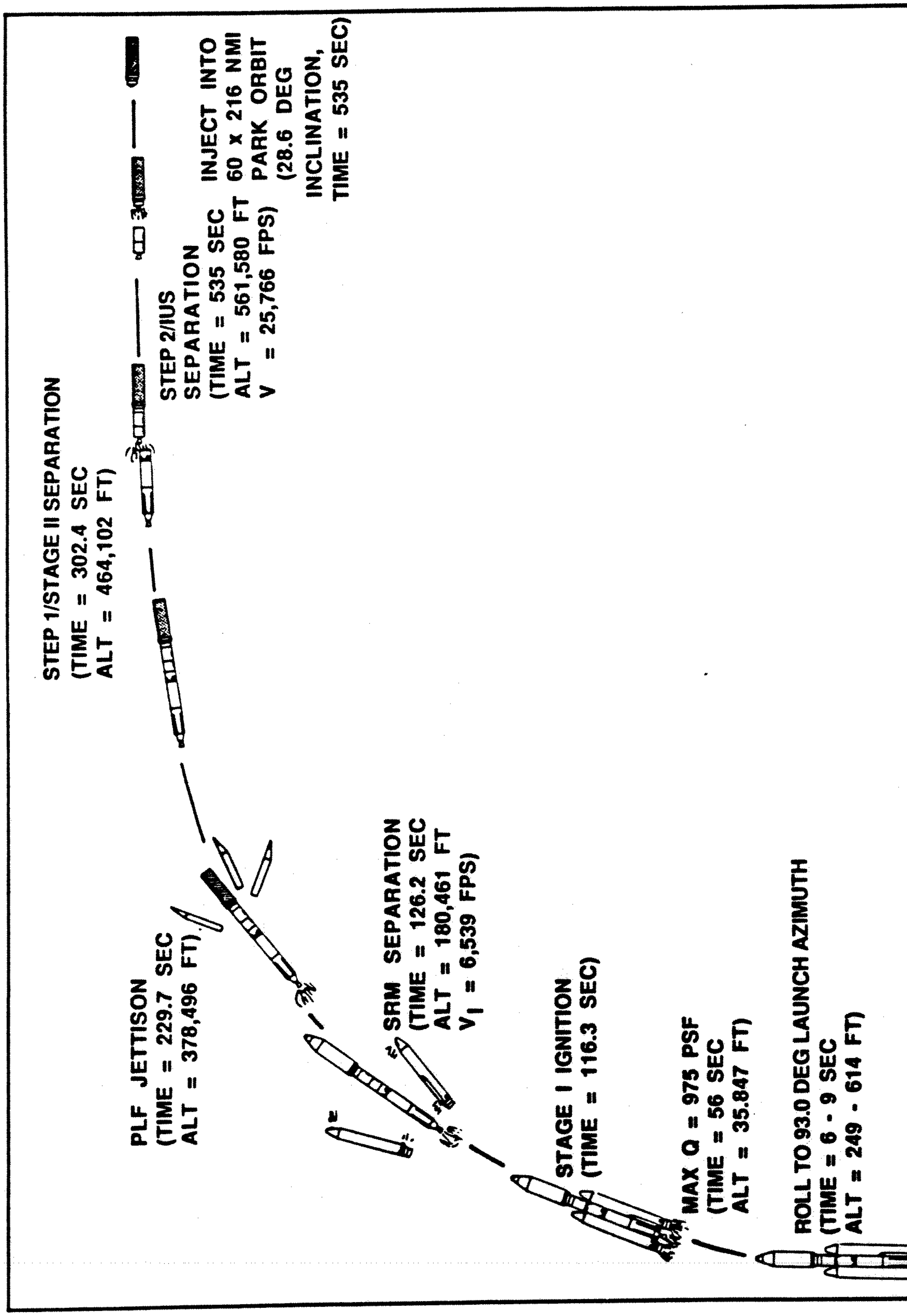


FIGURE 3.6.6-2 TITAN IV TYPE I/IUS TYPICAL ASCENT PROFILE FOR GSO MISSION

TABLE 3.6.6-4 TITAN IV TYPE I/IUS TYPICAL PERFORMANCE SUMMARY

	TYPICAL PAYLOAD CAPABILITY (lbm)	
(TO BE REVISED)	Park Orbit Inject at Perigee	Park Orbit Inject Off-Perigee
GSO Mission (First Crossing) 0.0 deg/day Drift 0.0 deg Inclination 0.0 Eccentricity	5,201	5,201
GSO Mission (Second Crossing) 0.0 deg/day Drift 0.0 deg Inclination 0.0 Eccentricity	5,077	5,152

NOTES – 1) Payload weight includes the spacecraft adapter.

2) This payload capabilities are for the 16.7 x 56 ft (8,884 lbm) PLF. If other fairing weights are required, the sensitivity is approximately –0.012 lbm Payload/lbm Fairing.

3.6.6.1 Geosynchronous Orbits (GSO)

The GSO mission is accomplished by initially placing the Upper Stage into a low altitude parking orbit. Two types of parking orbits are currently being investigated for this vehicle. For one case, the park orbit insertion occurs at perigee (80 nmi) and the orbit is slightly elliptical. If more payload capability is desired for the second crossing mission (approximately 75 lbm), the park orbit insertion can occur at "off perigee" (i.e., insertion occurs with a positive flight path angle and a 60 nmi perigee). This type of orbit does increase payload capability, however, it is at the expense of orbital lifetime of the park orbit. Off perigee injection is possible for the first equatorial crossing transfer orbit burn and this yields a gain of approximately 10 lbm of payload.

The IUS Stage I is ignited on either the first (descending) or sec (ascending) crossing of the equator. This burn produces an elliptical transfer orbit with an apogee altitude of approximately 19,500 nmi and reduces the orbital inclination by some optimum value (approximately one deg). The IUS Stage II burn circularizes the orbit at geosynchronous altitude and reduces the orbit inclination to zero deg.

3.6.6.2 Reference Mission Description

A typical mission begins at LC 41 of the ESMC with SRM ignition (time zero). The vehicle lifts off and rolls to the required launch azimuth (93.0 deg) between 6 and 9 sec with pitchover initiated at 10 sec. To alleviate aerodynamic loading on the vehicle, load relief is initiated by the autopilot between 35 and 85 sec and pitch/yaw polynomial steering is used for vehicle control during the 10 to 90 sec portion of Stage 0 flight. Maximum dynamic pressure occurs at approximately 56 sec and the nominal design trajectory is shaped to limit dynamic pressure to 975 psf.

The Stage 0/I overlap portion of flight is initiated when the axial acceleration of the vehicle has decreased to 1.3 gs. This initiates the Stage I ignition signal and both the SRMs and Stage I burn until the SRMs are jettisoned approximately 10 sec later. The payload fairing (16.7 ft diameter, 56 ft long) is retained until the FMH rate on the vehicle decreases to 150 (BTU/ft^2)/hr which nominally occurs at approximately 230 sec (altitude approximately 378,000 ft). Stage I flight is terminated upon propellant depletion initiating the Stage II ignition signal and Step 1/Stage II staging sequence.

Stage II flight is terminated with a command shutdown with the propellant margin adequate to guarantee a successful mission (margin approximately 1,955 lbm.). Park Orbit Injection (POI) is coincidental with Stage II jettison and occurs 10 sec after the commanded shutdown signal.

Depending upon the desired hemisphere for final payload insertion, the transfer orbit burn is initiated at either the first or second equatorial crossing. For the first crossing transfer orbit the POI occurs at perigee and the apogee altitude is maximized for the payload weight and mission for this particular vehicle configuration. This usually results in a park orbit with a 80 nmi perigee and an apogee altitude between 200 and 300 nmi For the second crossing case, POI can be either a perigee inject (similar to the one just described) or an off-perigee inject, which yields approximately a 75 lbm P/L increase. For off-perigee inject the perigee altitude is constrained to 60 nmi, apogee altitude is approximately 180 to 250 nmi, and inertial flight path angle is approximately one deg.

Injection into the transfer orbit is achieved by the IUS Stage I near one of the equatorial crossings. This transfer burn decreases the orbital inclination approximately 2.0 deg and increases the apogee altitude to geosynchronous altitude. This burn consumes approximately 21,500 lbm of solid propellant and uses approximately 41 lbm of RCS propellant for the vernier burn.

Final orbit inject is accomplished with the IUS Stage II and its associated RCS velocity vernier burn. This final burn uses approximately 6,060 lbm of solid propellant and 88 lbm of RCS propellant for the velocity vernier burn. Approximately 58 lbm of propellant is reserved in the RCS system to protect against IUS performance dispersions.

Final P/L inject, for the second crossing transfer orbit burn, occurs at approximately 24,000 sec with the IUS being disabled at 26,000 sec. The final P/L orbit is 19,323 nmi circular with zero deg inclination.

3.7 Titan IV/No Upper Stage (NUS) SS-ELV-403 (WSMC)

3.7.1 Introduction

The Titan IV/NUS 403 system utilizes the basic Titan IV/Core Vehicle, appropriate skirts, PLF, SRMs/SRMUs, associated facilities and GSE for the purpose of placing an SC into low earth orbit missions from VAFB Space Launch Complex SLC-4E. To the maximum extent practical, the Titan IV/NUS/WTR System shall use Titan IV subsystems and assemblies of the same design and fabrication as those used in the Titan IV/Centaur System defined by SS-ELV-401.

3.7.2 Structural

The Structural System consists of the basic Titan IV Centaur Booster Vehicle with a SC Interface at VS 163.00. The interface is a tension joint that attaches the SV NUS 403 GFP adapter to the Titan IV 2490 Forward Skirt Extension, reference Figures 3.7.2-1 and Figure 4.4.1.1-2.

3.7.3 Electrical

The Electrical System consists of the basic Titan IV Booster Vehicle arrangement with provisions for SV peculiar interfaces, reference Figure 3.7.3-1.

3.7.4 Titan IV/NUS Performance Capability (403-WSMC)

Titan IV/NUS LEO missions launched from WSMC can place payloads into LEO for orbital inclinations between 113.0 and 63.4 deg. Inclinations less than 63.4 deg are possible but usually require a yaw dog-leg during the ascent portion of the flight.

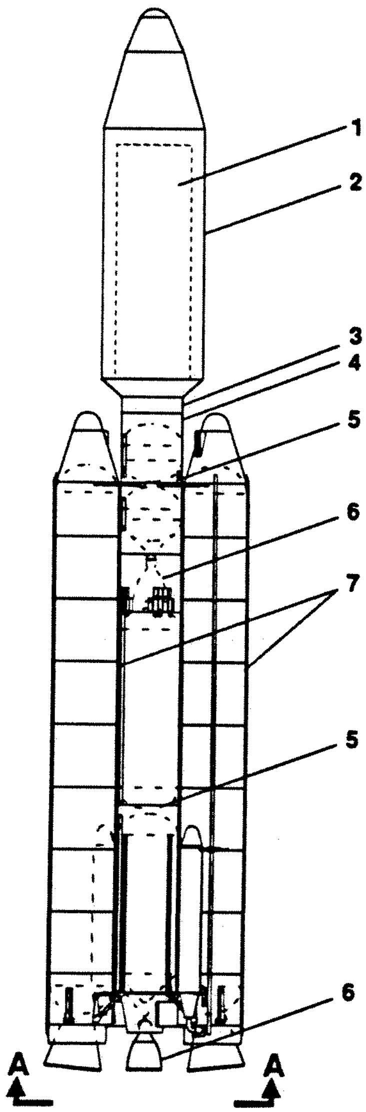
Circular orbit altitudes are limited to direct injection performance capabilities because Stage II does not currently have restart capability. Because of this restriction the vehicle has to be at altitude with a zero deg flight path angle at park orbit inject and this constrains the payload capability for higher circular orbit altitudes. Elliptical park orbits are also possible.

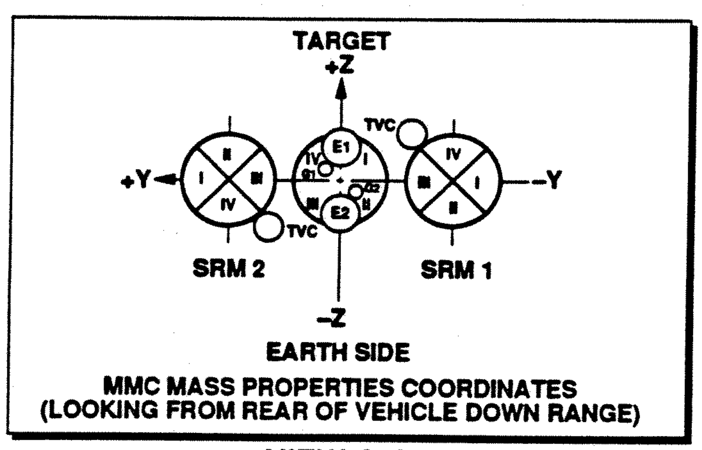
The performance capabilities for four different missions are presented:

1) Polar missions with orbital inclinations of approximately 90 deg,
2) Sun-synchronous missions which have orbital inclinations of approximately 96 to 98 deg (orbital inclination is a function of the perigee and apogee altitudes),
3) 12-Hour Elliptical Mission with a final orbital inclination of approximately 63.5 deg, and
4) 12-Hour Circular Mission with a final orbital inclination of 55 deg

LEGEND:

- **TITAN IV/NUS FLIGHT VEHICLE**
  1) SATELLITE VEHICLE (S/V)
  - **TITAN IV LAUNCH VEHICLE (LV)**
    2) PAYLOAD FAIRING (PLF)
    - **TITAN IV BOOST VEHICLE (BV)**
      7) SOLID ROCKET MOTORS (SRMs, STAGE 0 SHOWN) OR SOLID ROCKET MOTORS UPGRADE (SRMUs, STAGE 0)
      - **TITAN IV CORE VEHICLE**
        3) FORWARD SKIRT EXT (2490)
        4) STAGE II FORWARD OXIDIZER SKIRT
        5) STAGES I AND II
        6) LIQUID ROCKET ENGINES





VIEW A-A

FIGURE 3.7.2-1 TITAN IV/NUS FLIGHT VEHICLE CONFIGURATION (403)

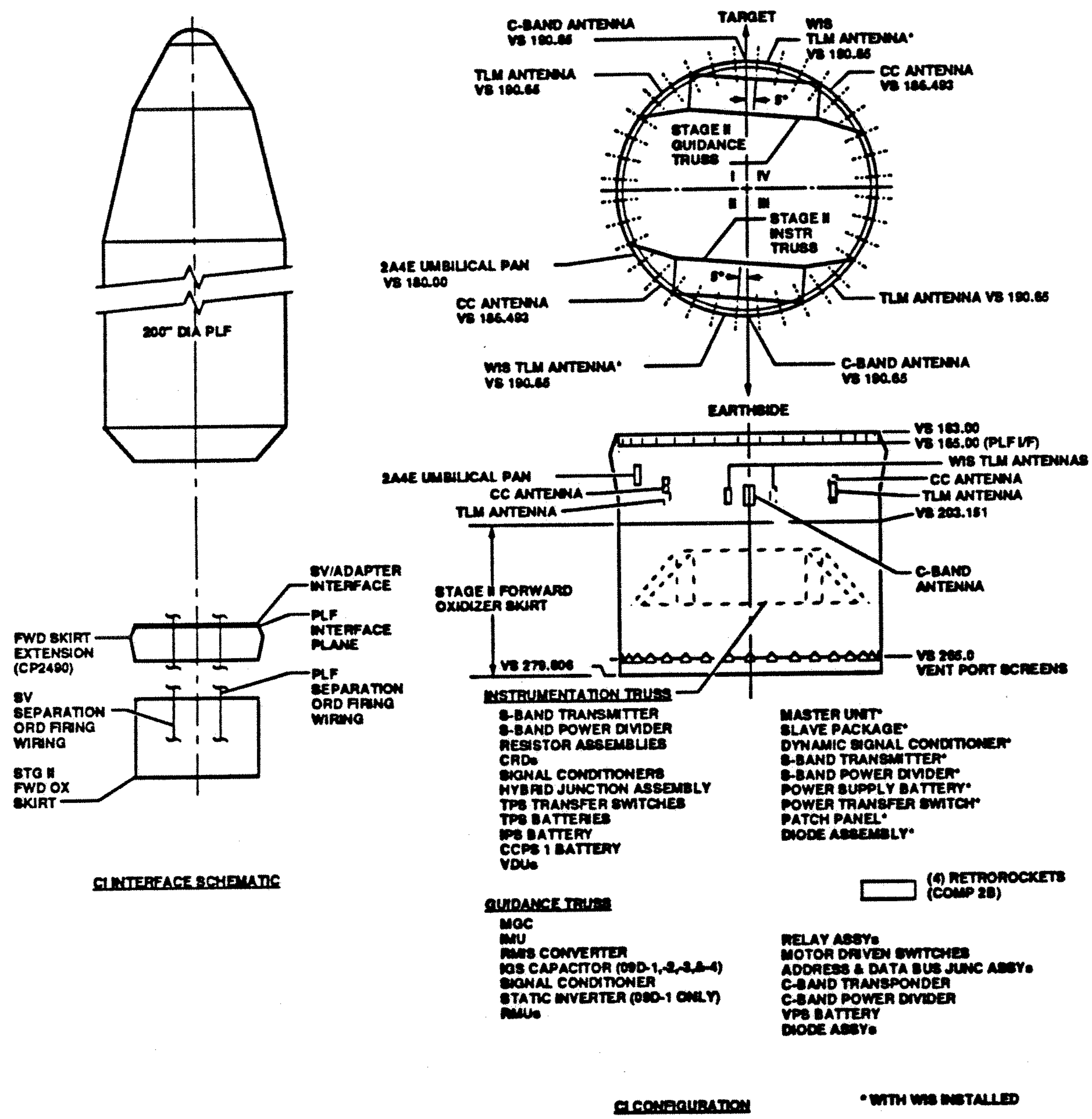


FIGURE 3.7.3-1 NUS/WTR UNIQUE AIRBORNE EQUIPMENT (403)

3.7.4 Titan IV/NUS Performance Capability (403-WSMC) (Continued)

Both the 12-Hour Elliptical and the 12-Hour Circular Mission require special range safety considerations when there is a direct inject into their respective orbital inclinations (63.5 deg corresponds to a launch azimuth of 149 deg, and 55 deg corresponds to a launch azimuth of 137 deg). If a yaw dogleg trajectory is required the nominal launch azimuth is approximately 172 deg with a yaw turn implemented in Stage I to alleviate range safety concerns. A yaw dogleg will, however, degrade the quoted payload capability.

Tables 3.7.4-1, 3.7.4-2 and 3.7.4-3 present the typical Trajectory Simulation Ground Rules, Three Degree-of-Freedom Trajectory Shaping Constraints and Sequence of Events respectively for the Titan IV Type I NUS class of vehicle. Typical performance summaries for various Titan IV Type I/No Upper Stage missions are presented in Table 3.7.4-4. Table 3.7.4-5 presents two special missions which require an additional Upper Stage for final orbit circularization. The two missions are both 12-hour orbital periods which are elliptical (63.5 deg orbital inclination) and circular (55.0 deg orbital inclination).

TABLE 3.7.4-1 TITAN IV TYPE I/NO UPPER STAGE LAUNCHED FROM WSMC TYPICAL TRAJECTORY SIMULATION GROUND RULES

<u>VEHICLE CHARACTERISTICS</u>	
Stage 0 SRM Temperature	56.0° F
SRM Nozzle Exit Area	12,491 in.^{2} SRM
Stage I Propellant Temperature	
Fuel	57.5° F
Ox	60.0° F
Stage II Propellant Temperature	
Fuel	60.0° F
Ox	62.5° F
Average Stage I Vehicle Centerline Thrust	545,894 lbf
Average Stage I Vehicle Centerline ISP	300.4 sec
Average Stage II Vacuum Thrust (Including Roll Nozzle)	106,312 lbf
Average Stage II Vacuum ISP (Including Roll Nozzle)	317.5 sec
<u>MISSION PROFILE</u>	
Launch Azimuth (90.0 deg Inclination)	182 deg
Stage 0 Separation Sequence Initiated when Axial Acceleration Decreases to 1.3 Gs	
PLF Jettisoned in Stage I when FMH rate <100 (BTU/ft^2)/hr	

TABLE 3.7.4-2 TITAN IV TYPE I/NO UPPER STAGE LAUNCHED FROM WSMC TYPICAL THREE DEGREE-OF-FREEDOM TRAJECTORY SHAPING CONSTRAINTS

- Dynamic pressure (Q): not to exceed 950 lbf/ft^2
- Aerodynamic Heating Indicator (AHI): not to exceed 95.0 x 10^6 $ft\text{-}lbf/ft^2$
- Stage 0 Separation:

  Dynamic Pressure shall not exceed 60 lbf/ft^2
  Pitch Angle-of-Attack shall not exceed ± 3.75 deg
  Yaw Angle-of-Attack shall not exceed ± 5.0 deg

- Stage I Separation:

  Dynamic Pressure shall not exceed 30 lbf/ft^2
  Pitch Angle + Yaw Angle shall not exceed 15.0 deg

- Nominal FMH rate on the exposed payload shall not exceed 100 (BTU/ft^2)/hr at or following PLF jettison

TABLE 3.7.4-3 TITAN IV TYPE I/NO UPPER STAGE LAUNCHED FROM WSMC TYPICAL SEQUENCE OF EVENTS FOR THE LOW EARTH ORBIT (LEO) MISSION

	LEO Mission (sec)
SRM Ignition	0.0
Begin Roll to Launch Azimuth	6.0
End Roll Maneuver	9.0
Begin Pitch Maneuver	10.0
Maximum Dynamic Pressure	55.2
Stage I Ignition	117.9
SRM Separation	127.8
Jettison PLF	234.0
Step I/Stage II Separation	307.2
Step 2/NUS Separation, Orbit Inject	543.1

TABLE 3.7.4-4 TITAN IV TYPE I/NO UPPER STAGE LAUNCHED FROM WSMC TYPICAL PERFORMANCE SUMMARY

TYPICAL PAYLOAD CAPABILITY (lbm)			
Orbit nmi x nmi	99 Degree Orbital Inclination	90 Degree Orbital Inclination	63.5 Degree Orbital Inclination
	(TO BE REVISED)		
100 x 100	30,000	31,200	34,900
200 x 200	24,000	25,000	27,500
300 x 300	16,200	16,600	19 ,000

TABLE 3.7.4-5 TITAN IV TYPE I/NUS (WITH KICK MOTOR FOR FINAL CIRCULARIZATION) TYPICAL PERFORMANCE SUMMARY

(TO BE REVISED)	TYPICAL PAYLOAD CAPABILITY (lbm)
12-Hour Period Elliptical Orbit Inclined 63.5 deg and with 270 deg Argument of Perigee (180 nmi Perigee and 21,616 nmi Apogee)	12,300
12-Hour Period Circular Orbit (10,898 nmi Apogee and Perigee Inclined 55.0 deg)	7,000

3.7.4 Titan IV/NUS Performance Capability (403-WSMC) (Continued)

Figure 3.7.4-1 shows the payload weight for various orbits out of WSMC. Elliptical and circular orbits are shown with inclinations of 63.5, 90 and 99 deg. As noted earlier, an inclination of 63.5 deg may have range safety concerns which need to be evaluated.

3.7.4.1 Polar Orbits

A Polar Mission has an inclination of approximately 90 deg which corresponds to a launch azimuth of 182 deg. The payload capability for various polar orbits is shown on Figure 3.7.4.1-1. A typical polar mission is a 100 nmi circular orbit inclined 90 deg. An ascent profile for this mission from liftoff to park orbit is given in Figure 3.7.4.1-2. During the ascent portion of flight, various tracking stations are used to track and monitor various vehicle systems.

(TO BE REVISED)

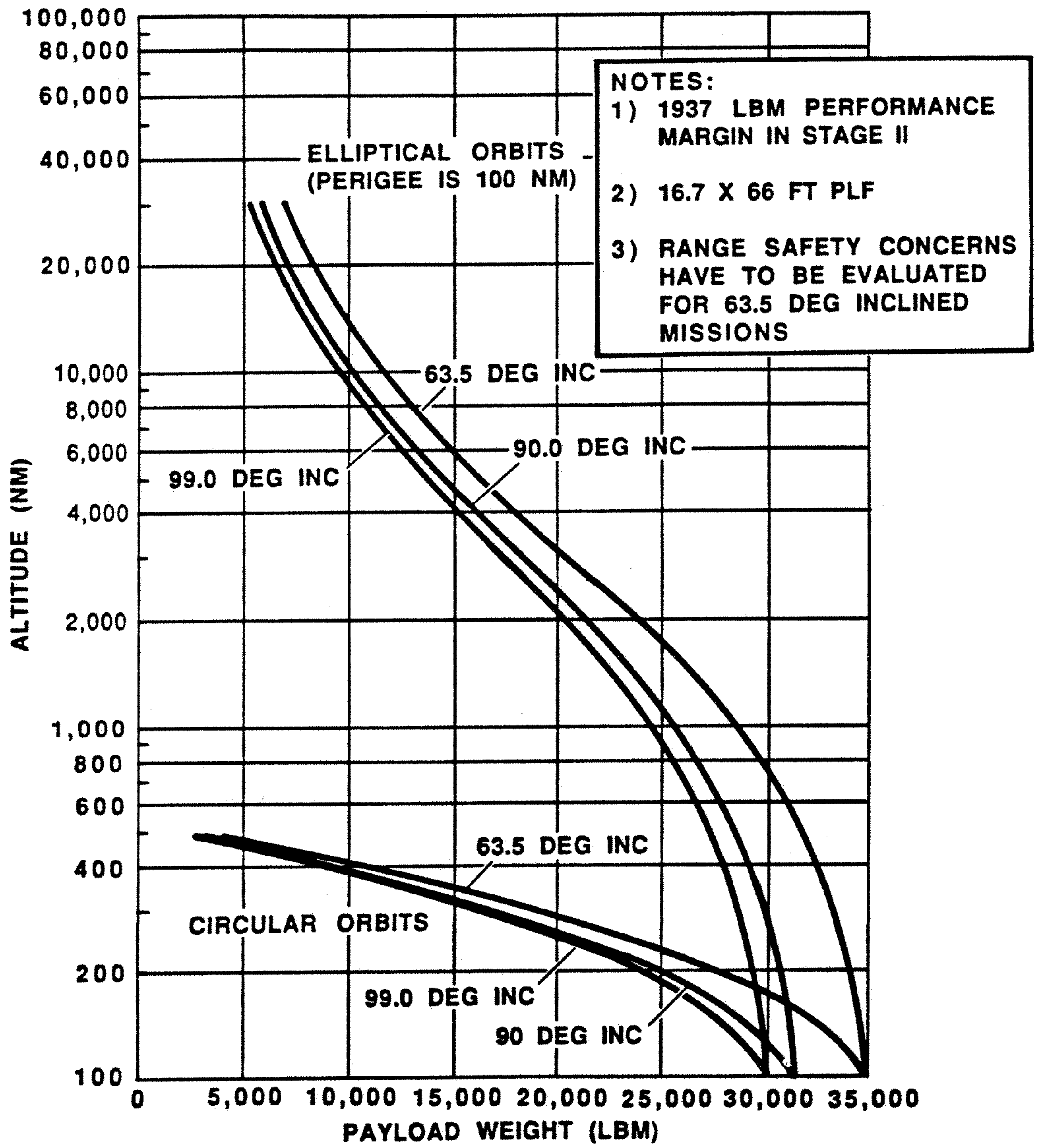
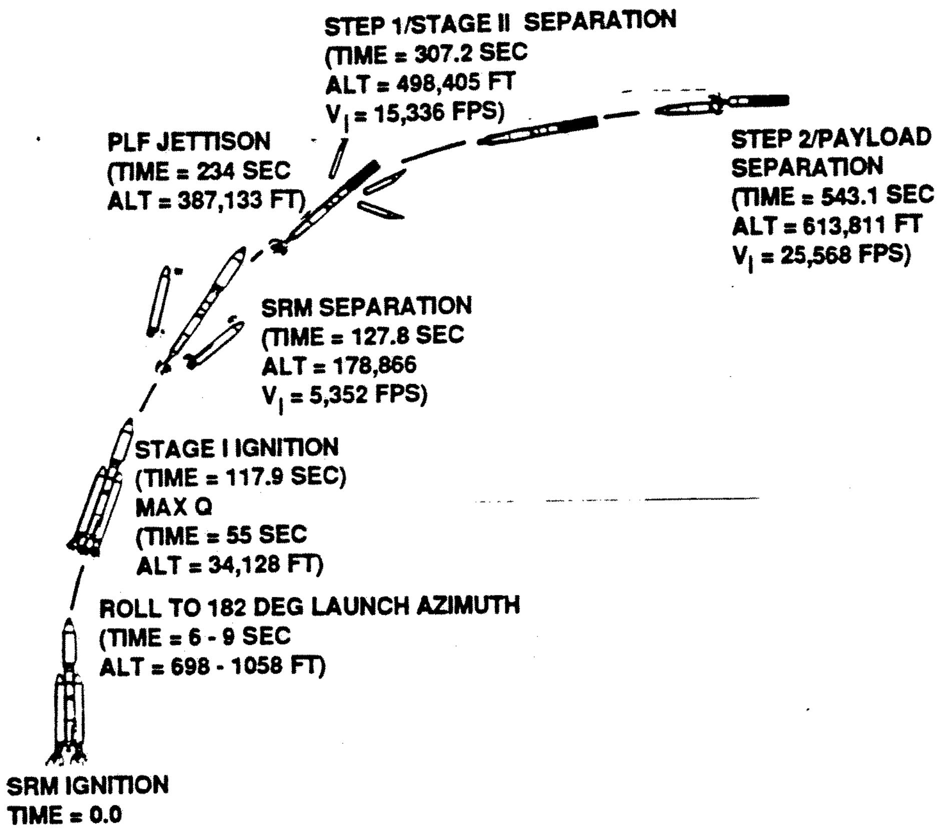


FIGURE 3.7.4.1-1 TITAN IV TYPE I/NUS TYPICAL PAYLOAD CAPABILITY FOR ORBITS INCLINED 63.5, 90.0 AND 99.0 DEGREES LAUNCHED FROM WSMC



REFERENCE: MCR - 90 - 2569, "TITAN IV/WTR NUS GENERIC 3 - D REFERENCE TRAJECTORY REPORT." MARCH 1990

FIGURE 3.7.4.1-2 TITAN IV TYPE I/NUS LAUNCHED FROM WSMC TYPICAL ASCENT PROFILE FOR POLAR MISSION

3.7.4.2 Sun Synchronous Orbits

Sun synchronous orbits satisfy specific user requirements. The orbital elements are defined such that the orientation of the orbit with respect to the sun remains constant with time. This requires an orbital inclination of greater than 90 deg. The exact inclination needed depends on the shape and altitude of the orbit. If the launch time is chosen such that the line of nodes (the intersection of the orbital plane with the earth's equatorial plane) is perpendicular to the earth sun line then the spacecraft is always in the sunlight.

Figure 3.7.4.2-1 presents an estimate of the payload weight to various sun synchronous orbits. For the orbits shown, the orbital inclinations range between 96 and 98 deg.

(TO BE REVISED)

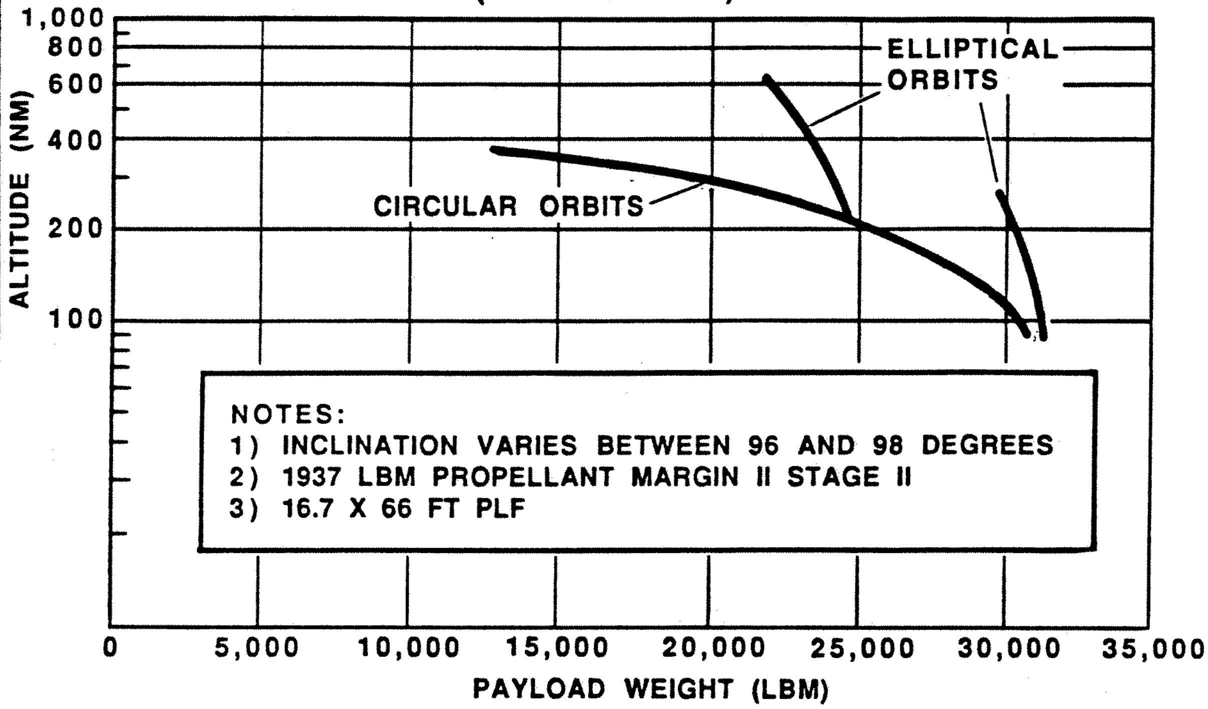


FIGURE 3.7.4.2-1 TITAN IV TYPE I/NUS LAUNCHED FROM WSMC TYPICAL PAYLOAD CAPABILITY TO VARIOUS SUN SYNCHRONOUS ORBITS

3.7.4.3 12-Hour Period Elliptical Mission

This mission is a highly elliptical orbit with an inclination of approximately 63 deg, an argument of perigee of 270 deg and a period of approximately 12 hr.

The 12-Hour Elliptical Mission requires a kick motor to perform the final maneuver which results in a final orbit of 180 x 21,616 nmi orbit inclined 63.5 deg with a 270 deg argument of perigee. This is based on a direct injection into the 63.5 deg inclination (i.e., no yaw dogleg). The Upper Stage kick motor required for this mission has a thrust of 15,000 lbf, a specific impulse of 290 sec and a mass fraction (propellant weight divided by total kick motor weight) equal to 0.9. The final orbit P/L weight is 12,300 lbm.

3.7.4.4 12-Hour Period Circular Mission

The 12-Hour Circular Mission has a final orbit of 10898 x 10898 nmi inclined 55 deg. The payload weight to this orbit is approximately 7000 lbm. This is a preliminary number based on a direct injection into the 55 deg inclination (i.e., no yaw dogleg). The Upper Stage kick motor required for this mission has a thrust of 15,000 lbf, a specific impulse of 290 sec and a mass fraction equal to 0.9 (propellant weight of 5415 lbm and a jettisonable case weight of 602 lbm).

3.8 Titan IV No Upper Stage SS-ELV-405 (ESMC)

3.8.1 Introduction

The Titan IV/NUS 405 System utilizes the basic Titan IV Core Vehicle, appropriate skirts, PLF, SRM/SRMU associated facilities and GSE for the purpose of placing SC into low-earth orbit missions from CCAFS, LC-41. To the maximum extent practical, the Titan IV/NUS/ETR system shall use Titan IV subsystems and assemblies of the same design and fabrication as those used in the Titan IV/Centaur System defined by SS-ELV-401.

3.8.2 Structural

The Structural System consists of the basic Titan IV Core Vehicle SLV with a Space Vehicle Interface at Vehicle Station 163.00. The interface is a tension joint that attaches the SV NUS 405 adapter to the Titan IV 2490 Forward Skirt Extension, reference Figures 3.8.2-1 and 4.5.1.1-2.

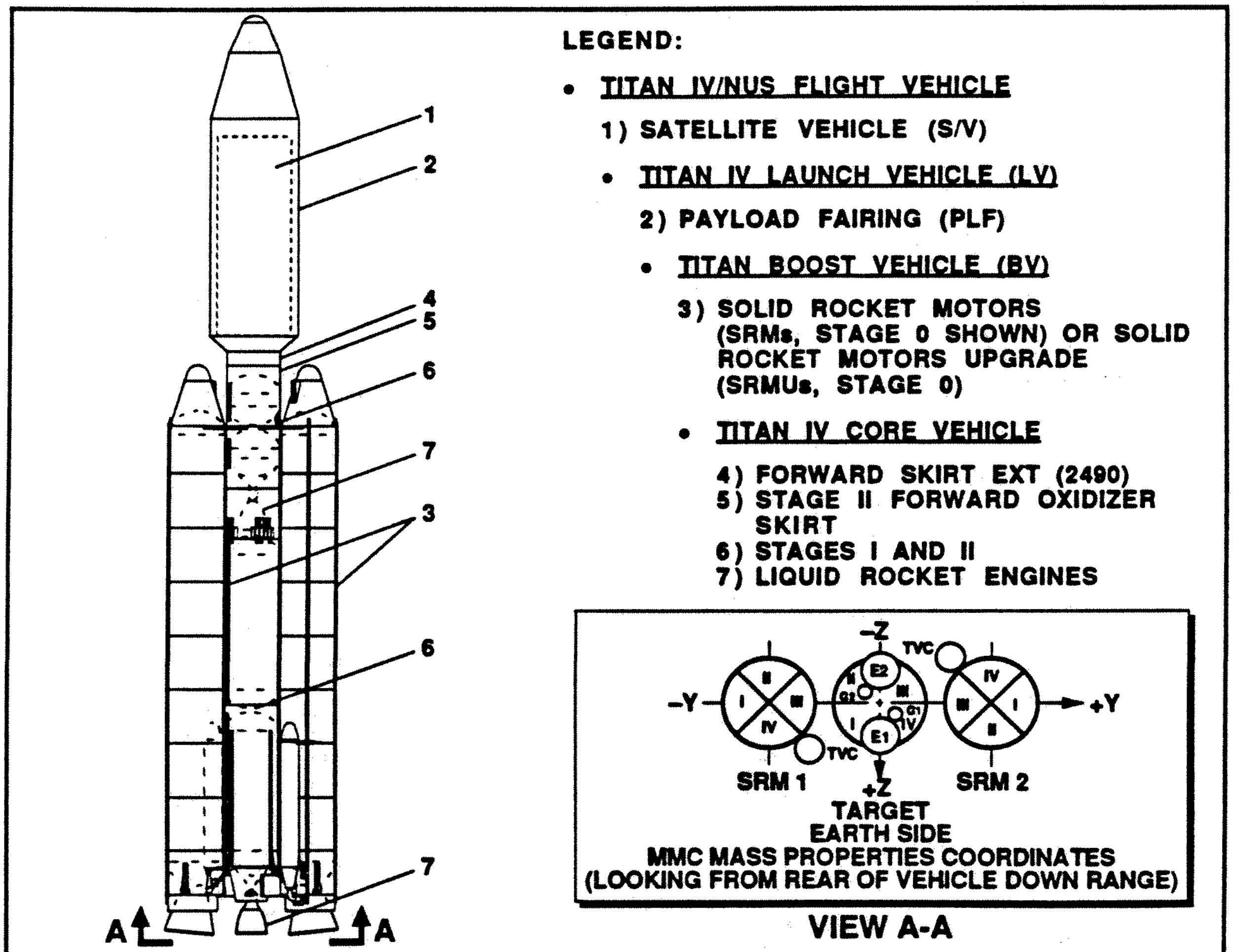


FIGURE 3.8.2-1 TITAN IV/NUS/ETR FLIGHT VEHICLE CONFIGURATION (405)

3.8.3 Electrical

The Electrical System consists of the basic Titan IV Booster Vehicle system with provisions for S/V peculiar interfaces, reference Figure 3.8.3-1 NUS/ETR Unique Airborne Equipment.

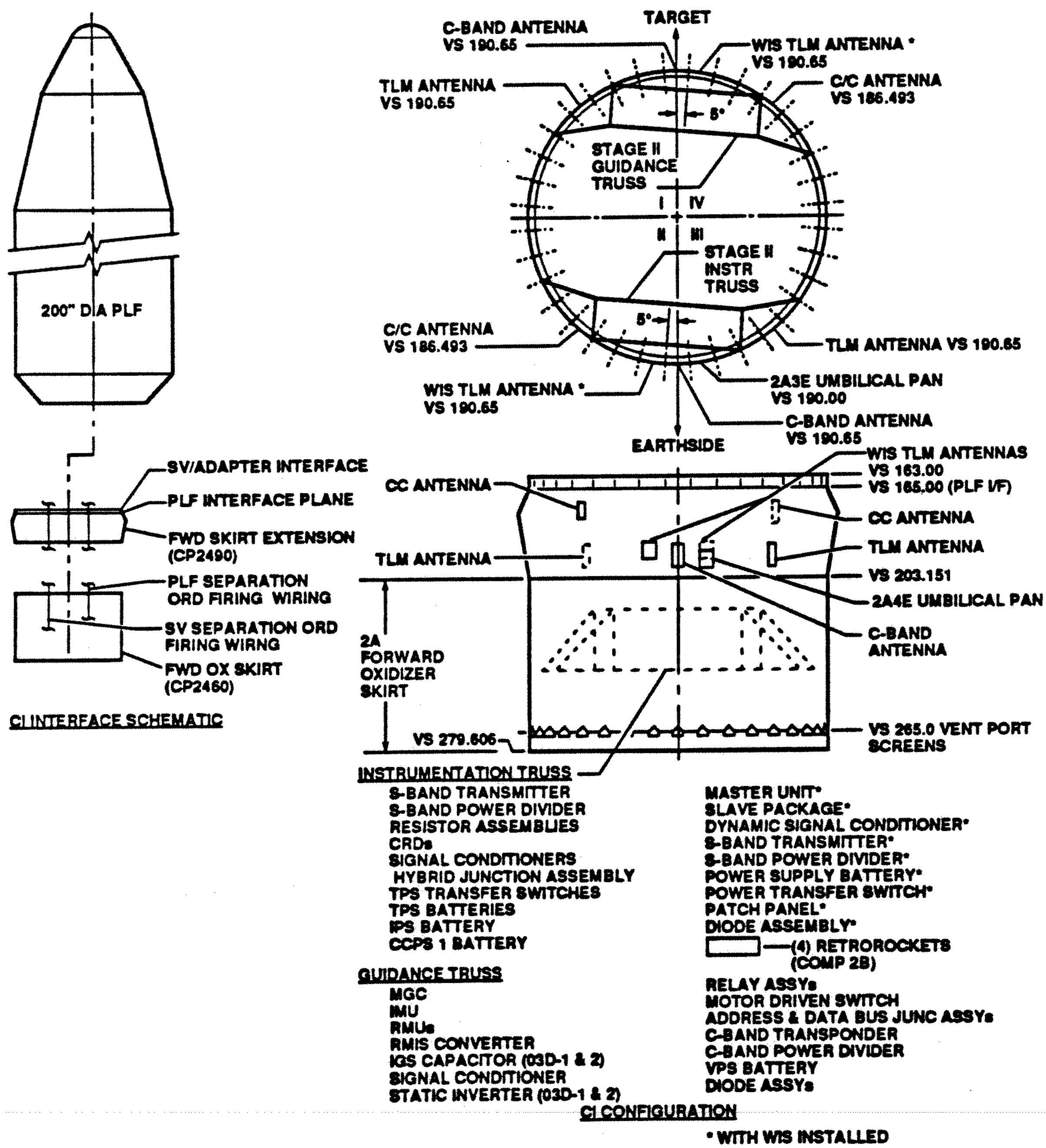


FIGURE 3.8.3-1 NUS/ETR UNIQUE AIRBORNE EQUIPMENT (405)

3.8.4 Titan IV Type I/NUS Performance Capability (405 ESMC)

Titan IV/NUS Low-Earth Orbit (LEO) missions launched from ESMC can place P/Ls into LEO for orbital inclinations between 28.6 and 55 deg. Inclinations greater than 55 deg are possible but require a yaw dog-leg during the ascent portion of the flight.

Circular orbit altitudes are limited to direct injection performance capabilities because Stage II does not currently have restart capability. Because of this restriction, the vehicle has to be at altitude with a zero deg flight path angle at park orbit insertion and this constrains the P/L capability for higher circular orbit altitudes. Elliptical orbits are also possible.

Tables 3.8.4-1, 3.8.4-2, 3.8.4-3 and 3.8.4-4 present the Typical Trajectory Simulation Ground Rules, Three Degree-of-Freedom Trajectory Shaping Constraints, Sequence of Events and Performance Summary for this class of vehicle.

TABLE 3.8.4-1 TITAN IV TYPE I/NO UPPER STAGE LAUNCHED FROM ESMC TYPICAL TRAJECTORY SIMULATION GROUND RULES

<u>VEHICLE CHARACTERISTICS</u>	
Stage 0 SRM Temperature	71.5° F
SRM Nozzle Exit Area	12,491 in.^{2}/SRM
Stage I Propellant Temperature (Ox & Fuel)	72.5° F
Stage II Propellant Temperature (Ox/Fuel)	70.0° F/72.5° F
Average Stage I Nozzle Centerline Thrust	547,005 lbf
Average Stage I Nozzle Centerline ISP	301.16 sec
Average Stage II Vacuum Thrust (Including Roll Nozzle)	106,460 lbf
Average Stage II Vacuum ISP (Including Roll Nozzle)	317.7 sec
Payload Fairing:	
Dimensions (diameter & length)	16.7 x 56 ft
Weight	8,950 lbm
<u>MISSION PROFILE</u>	
Launch Azimuth	93.0 deg
Stage 0 Separation Sequence Initiated when Axial Acceleration Decreases to 1.3 Gs	
PLF Jettisoned in Stage I when FMH rate <150 (BTU/ft^2)/hr	

TABLE 3.8.4-2 TITAN IV TYPE I/NO UPPER STAGE LAUNCHED FROM ESMC TYPICAL THREE DEGREE-OF-FREEDOM TRAJECTORY SHAPING CONSTRAINTS

- Dynamic pressure (Q): not to exceed 975 lbf/ft^2
- AHI: not to exceed 95.0×10^6 $ft\text{-}lbf/ft^2$
- Stage 0 Separation:
  Dynamic Pressure shall not exceed 60 lbf/ft^2
  Pitch Angle-of-Attack shall not exceed + 4.5 deg
  Yaw Angle-of-Attack shall not exceed + 5.0 deg
- Stage I Separation:
  Dynamic Pressure shall not exceed 30 lbf/ft^2
  Total Angle-of-Attack shall not exceed 15.0 deg
- Nominal FMH rate on the exposed payload shall not exceed 150 (BTU/ft^2)/hr at or following PLF jettison
- Perigee altitude of park orbit not less than 80 nmi
- Apogee altitude of park orbit not less than 95 nmi relative to equatorial radius of Earth

TABLE 3.8.4-3 TITAN IV TYPE I/NO UPPER STAGE LAUNCHED FROM ESMC TYPICAL SEQUENCE OF EVENTS FOR THE LEO, 28.6 DEG INCLINATION MISSION

	LEO Mission (sec)
SRM Ignition	0.0
Begin Roll to Launch Azimuth	6.0
End Roll Maneuver	9.0
Begin Pitch Maneuver	10.0
Maximum Dynamic Pressure	56.4
Stage I Ignition	116.5
SRM Separation	126.4
Jettison PLF	231.0
Step 1/Stage II Separation	302.0
Step 2/NUS Separation, Orbit Inject	533.7

TABLE 3.8.4-4 TITAN IV TYPE I/NO UPPER STAGE LAUNCHED FROM ESMC TYPICAL PERFORMANCE SUMMARY		
(TO BE REVISED) TYPICAL PAYLOAD CAPABILITY (LBM)		
Orbit nmi x nmi	55 Degree Orbital Inclination	28.6 Degree Orbital Inclination
100 x 100	36,000	38,500
200 x 200	28,500	29,500
300 x 300	19,500	19,600
Note – These payload capabilities are for the 16.7 x 56 ft (8,950 lbm) payload fairing. If other fairing weights are required, the sensitivity is approximately –0.14 lbm Payload/lbm Fairing.		

3.8.4 Titan IV Type I/NUS Performance Capability (405 ESMC) (Continued)

Figure 3.8.4-1 presents circular and elliptical orbit payload capability for orbital inclinations of 28.6 deg (93.0 deg launch azimuth). To determine circular orbit throw weight capability look up the required altitude and read the circular orbit capability directly. For elliptical orbits look up the required perigee altitude on the circular orbit curve and then follow the elliptical orbit family curve until it reaches the required apogee. The payload corresponding to this point is the payload for this elliptical orbit.

Titan IV launches are nominally constrained to launch azimuths between 37.9 deg (clockwise from north) and 112.0 deg which correspond to orbital inclinations of 55 and 35 deg respectively. Figure 3.8.4-2 presents the P/L as a function of orbital altitude for various circular and elliptical orbits for these two inclinations. The effect of orbital inclination on P/L into the park orbit for northerly launches is presented in Figure 3.8.4-3. This figure shows the changes in payload capability between 55 deg (37.9 deg launch azimuth), 45 deg (51.1 deg launch azimuth) and 35 deg (67.8 deg launch azimuth) orbital inclinations as functions of orbital altitude.

If orbital inclinations greater than 55 deg are required from ESMC it is possible to perform a yaw turn to the required inclination. This yaw turn is usually accomplished during the early portions of Stage II and yields payload capabilities as shown in Figure 3.8.4-4. This figure presents the performance capabilities up to 70 deg orbital inclinations. It should be noted that an in-depth range safety analysis should be conducted for orbital inclinations greater than 55 deg as part of any preliminary feasibility analyses.

Figure 3.8.4-5 depicts a typical boost phase sequence of events for the 28.6° inclination, 80 x 45 nmi park orbit.

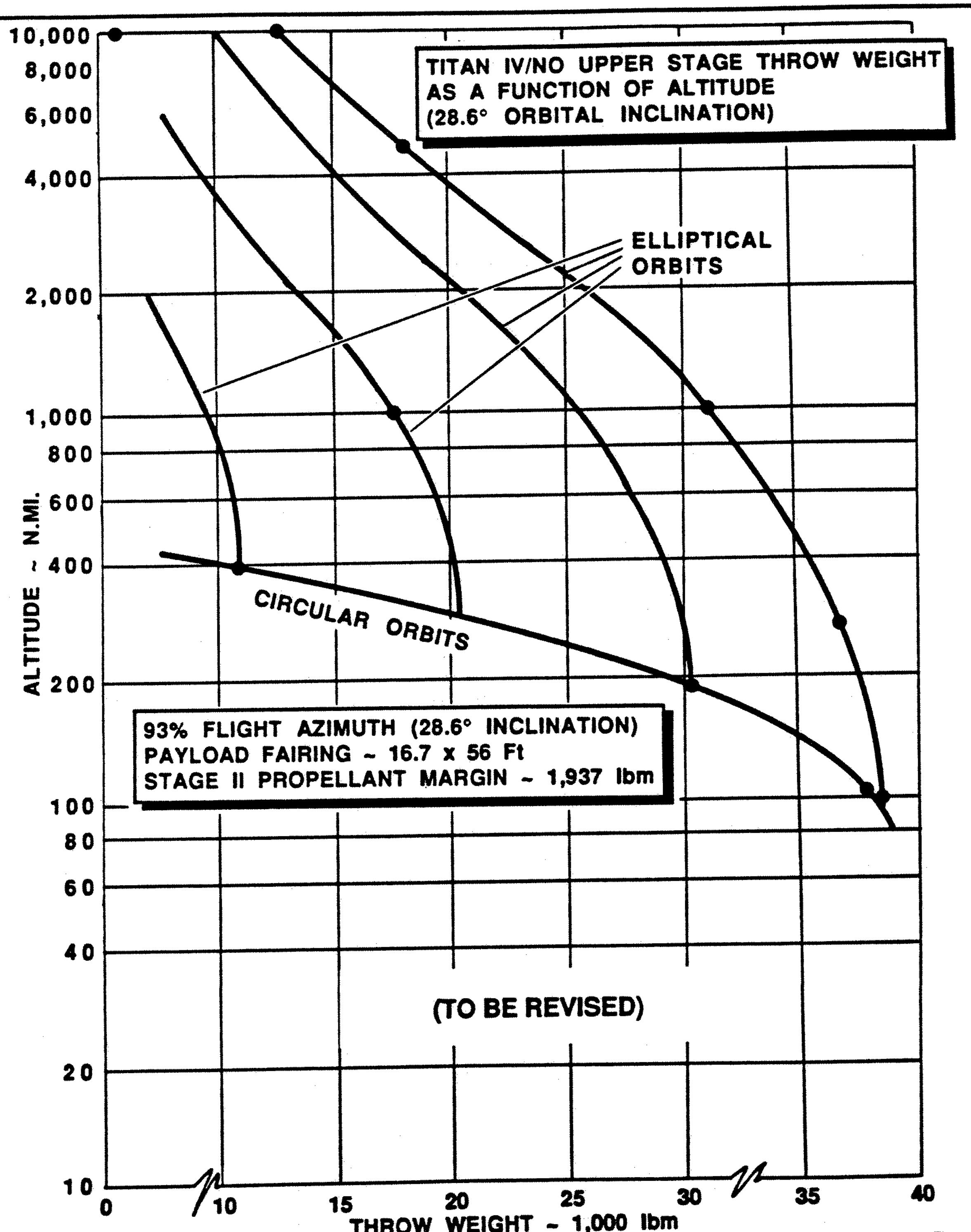


FIGURE 3.8.4-1 TIV TYPE I/NUS FROM ESMC TYPICAL CIRCULAR AND ELLIPTICAL ORBIT PAYLOAD CAPABILITY FOR 28.6 DEG INCLINED

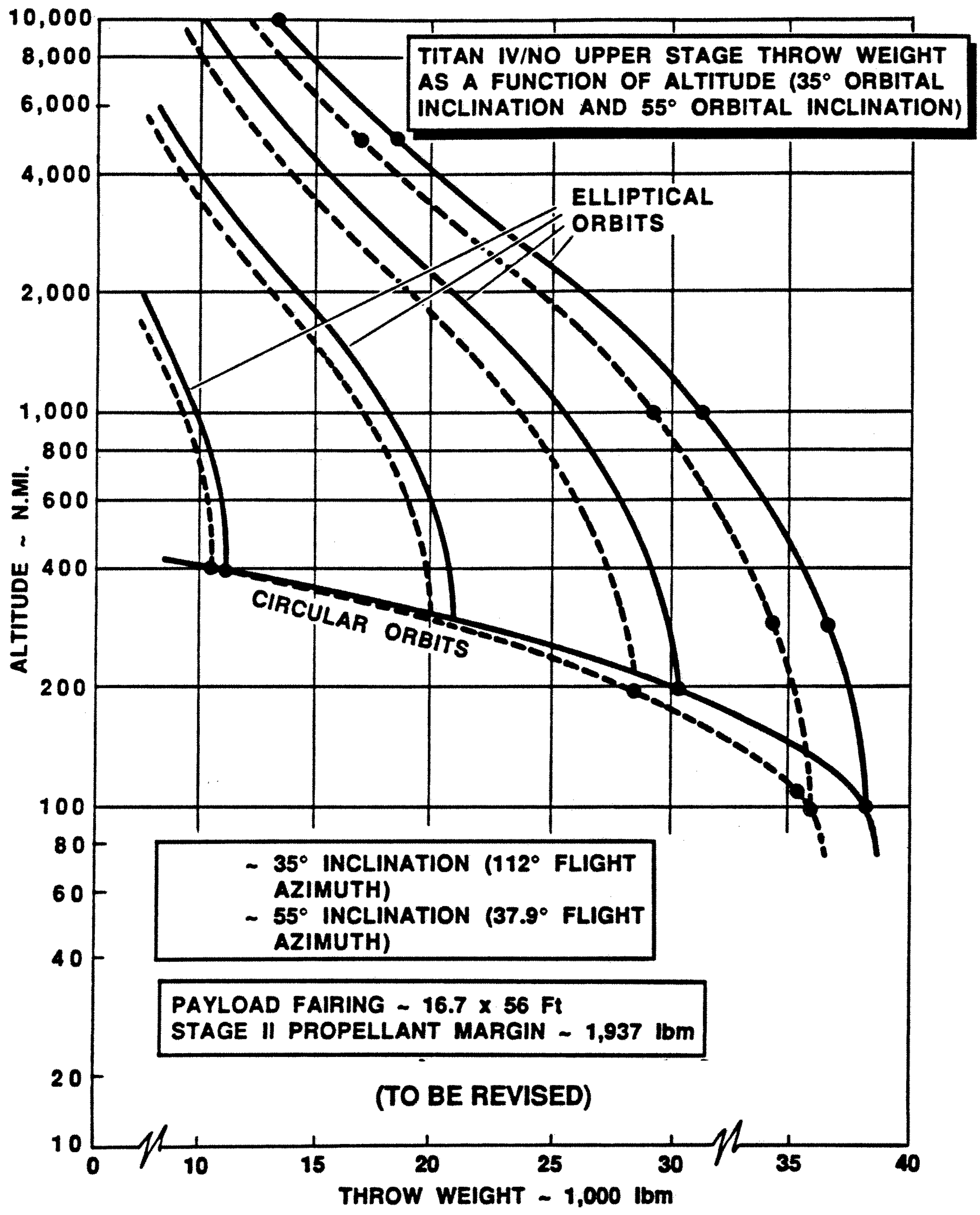


FIGURE 3.8.4.-2 TIV TYPE I/NUS LAUNCHED FROM ESMC TYPICAL PAYLOAD CAPABILITY AS A FUNCTION OF ORBITAL ALTITUDE FOR LOW EARTH ORBITS AT 35° AND 55° ORBITAL INCLINATIONS

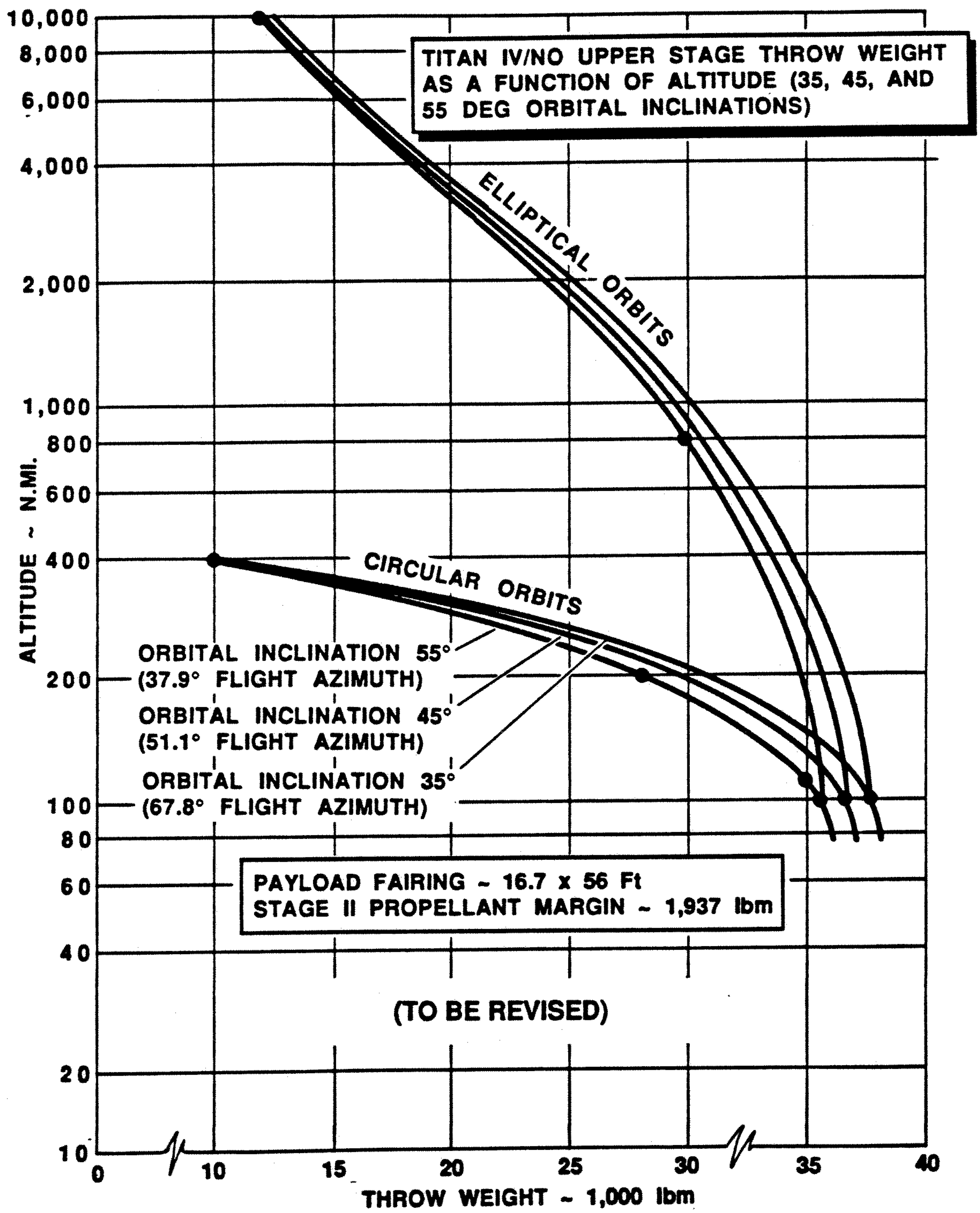
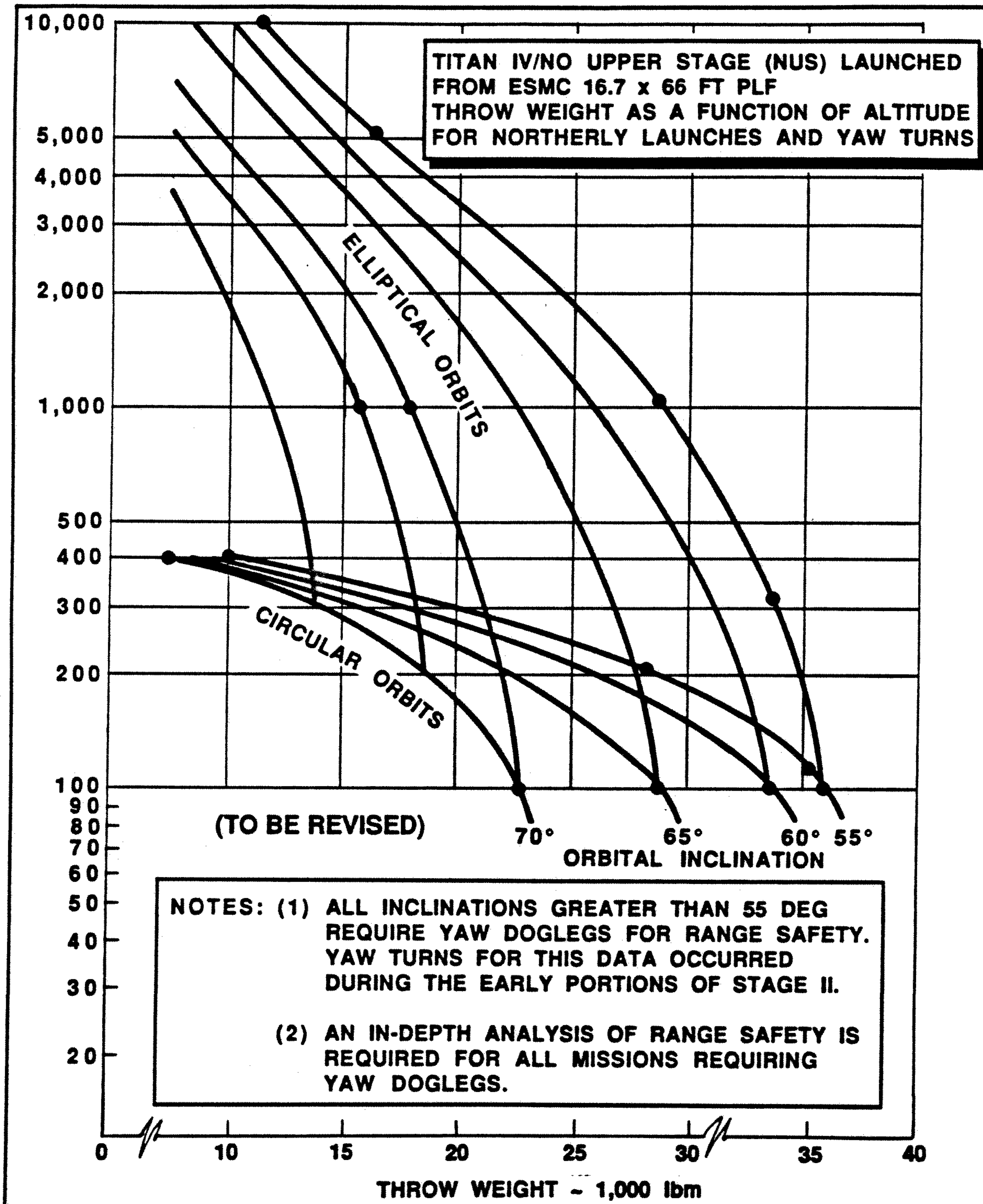


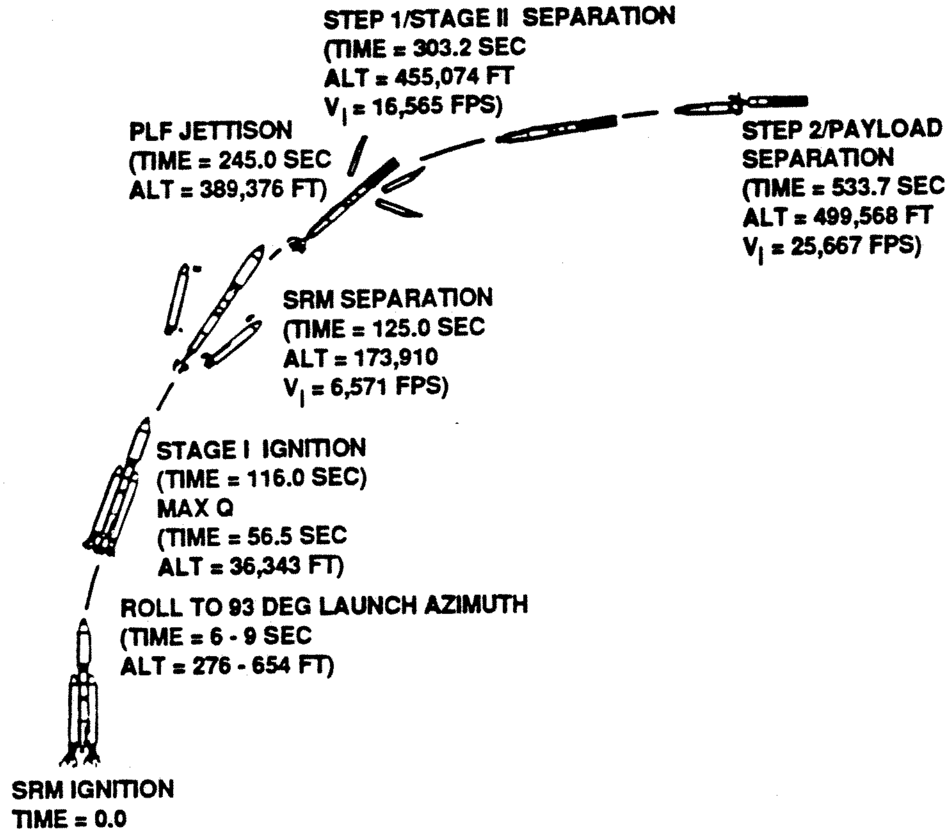
FIGURE 3.8.4-3 TIV TYPE I/NUS LAUNCHED FROM ESMC TYPICAL PAYLOAD AS A FUNCTION OF ORBITAL ALTITUDE FOR FLIGHT AZIMUTHS BETWEEN 37.9° AND 67.8°



NOTES: (1) ALL INCLINATIONS GREATER THAN 55 DEG REQUIRE YAW DOGLEGS FOR RANGE SAFETY. YAW TURNS FOR THIS DATA OCCURRED DURING THE EARLY PORTIONS OF STAGE II.

(2) AN IN-DEPTH ANALYSIS OF RANGE SAFETY IS REQUIRED FOR ALL MISSIONS REQUIRING YAW DOGLEGS.

FIGURE 3.8.4-4 TIV TYPE I/NUS LAUNCHED FROM ESMC TYPICAL PAYLOAD CAPABILITY FOR ORBITAL INCLINATIONS REQUIRING YAW TURNS



REFERENCE: MCR-90-2569, "TITAN IV/NUS ESMC 3 DEGREE-OF-FREEDOM REFERENCE TRAJECTORY REPORT," JUNE 1990

FIGURE 3.8.4-5 TITAN IV TYPE I/NUS LAUNCHED FROM ESMC TYPICAL ASCENT PROFILE FOR 80 x 95 NMI 28.6 DEG INCLINATION MISSION

Chapter 4

Spacecraft Interfaces

4.0 SPACECRAFT INTERFACES

4.1 Introduction

The material in this chapter is oriented to emphasize the structural and electrical interfaces for a Space Vehicle utilizing the Centaur Upper Stage (SS-ELV-401), the Inertial Upper Stage (SS-ELV-402), the No Upper Stage, (SS-ELV-403), or the No Upper Stage (SS-ELV-405) Titan IV Launch Vehicle configurations.

4.2 Centaur Upper Stage SS-ELV-401 (ESMC)

Reference Figure 4.2-1 Titan IV/Centaur System Specification Tree and Figure 4.2-2 Titan IV Centaur Profile.

4.2.1 Structural

4.2.1.1 Spacecraft Mechanical Interface

By means of a forward adapter, the physical interface between the Centaur and the Space Vehicle (SV) occurs at an eight hardpoint interface at Centaur Station 3519.22 and at a 22 hardpoint interface at Centaur Station 3563.90. Refer to Figures 4.2.1.1-1, 4.2.1.1-2, and 4.2.1.1-3

The forward interface consists of eight attach hardpoints on a 111.77 in. diameter bolt circle. Reference Figures 4.2.1.1-4 and 4.2.1.1-5.

The aft interface at Station 3563.90 is provided to accommodate a SV adapter and consists of a 171.92 in. diar bolt circle to which the SV truss fittings attach through a four-bolt pattern as shown in Figures 4.2.1.1-6 and 4.2.1.1.7. Structural analyses may show that a number less than 22 attachments may be sufficient, or this may be dictated by SV geometry.

In addition to the SV attachment, 180 attach points on a 120.80 in. dia bolt circle on the forward face of the adapter are provided for auxiliary equipment attachment. Reference Figure 4.2.1.1-8.

The combined effect of all Centaur manufacturing tolerances is not to be greater than the following misalignments at the SV interfaces with respect to TIV.

Angular tilt (any radial direction)	=	± .073 deg
Radial translation	=	± .52 in.
Rotation about center axis	=	± .135 deg

Additional details of the mechanical attachment configurations can be found in the Mission Peculiar Mechanical ICDs.

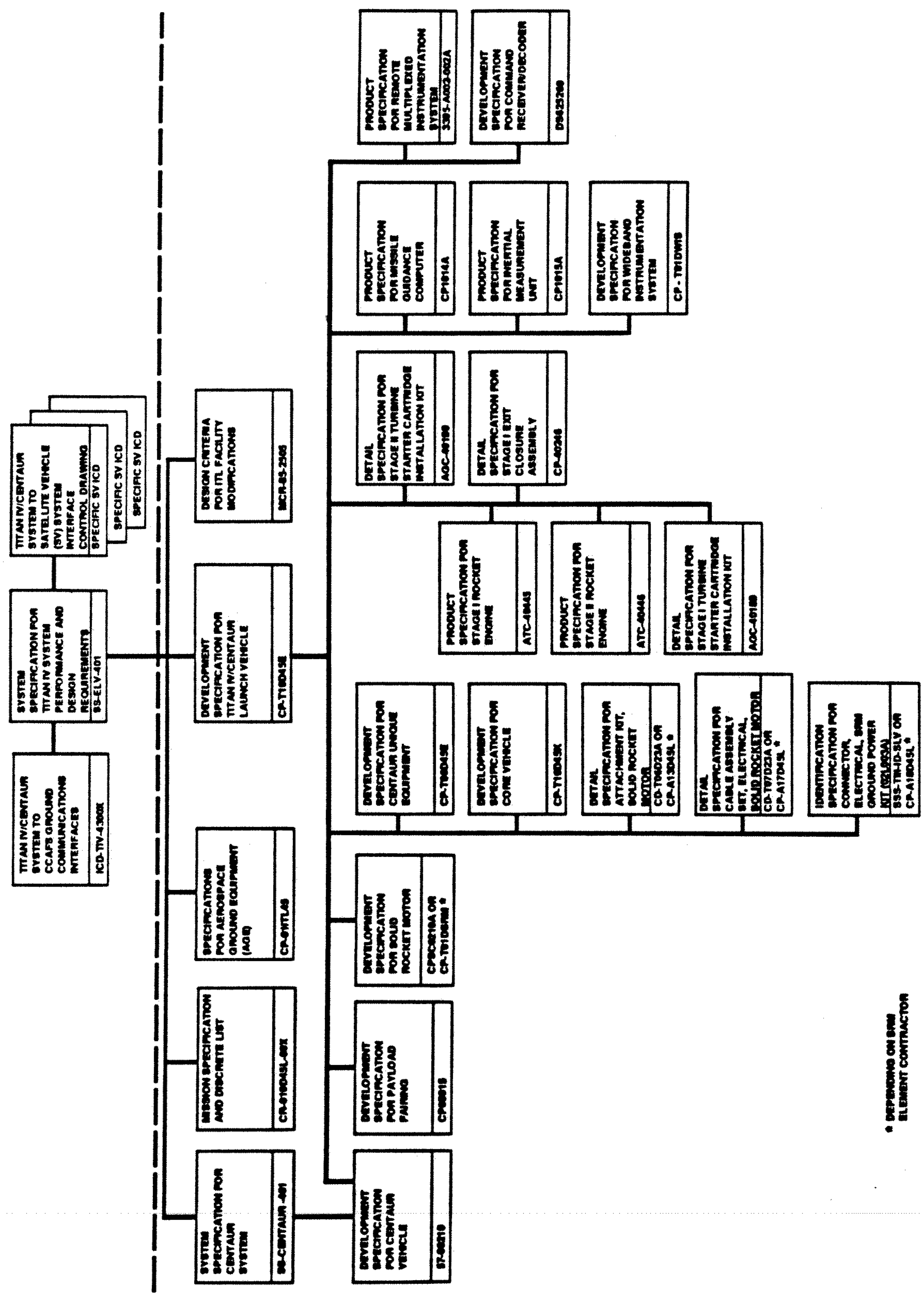


FIGURE 4.2-1 TITAN IV/CENTAUR SYSTEM SPECIFICATION TREE

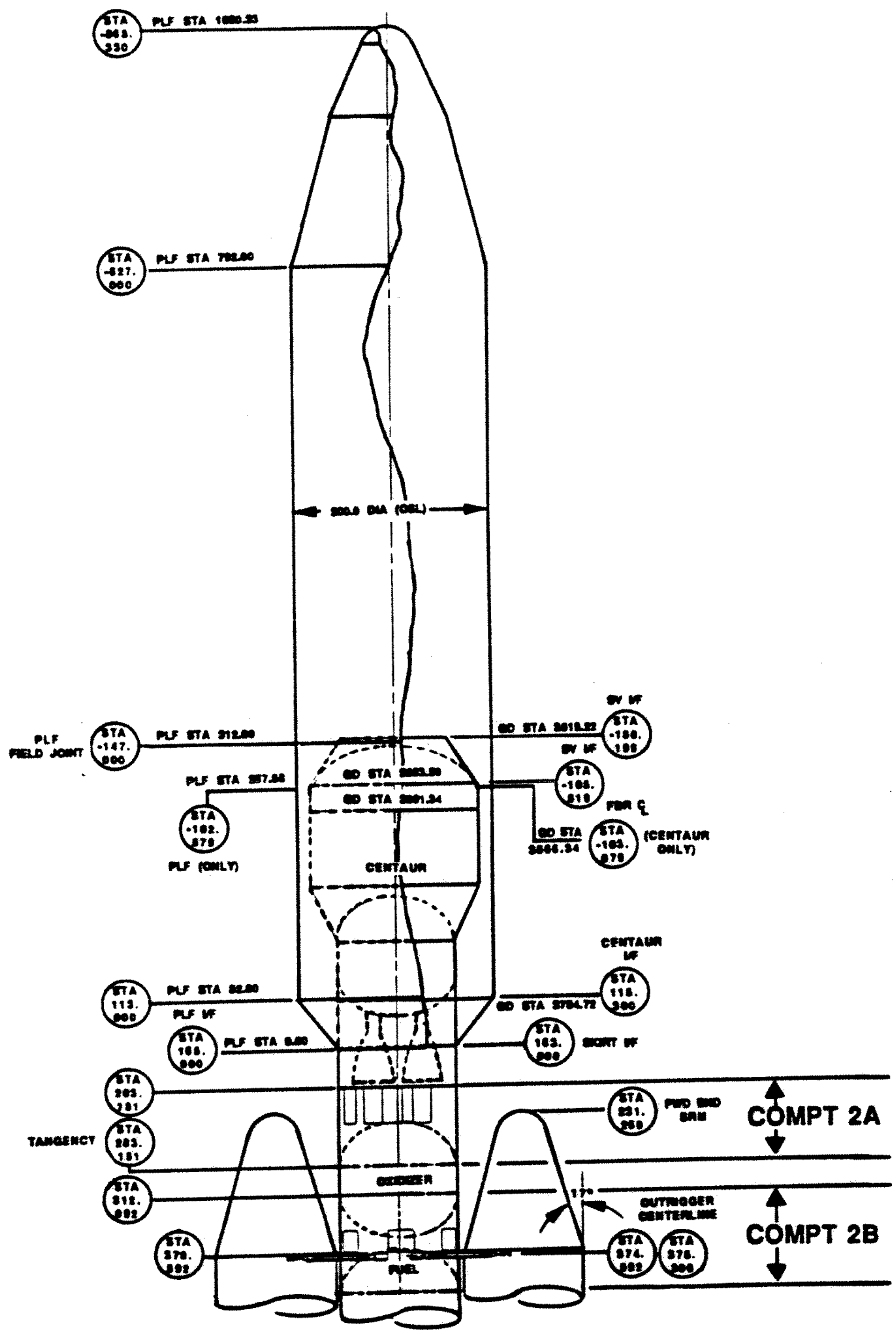


FIGURE 4.2-2 TITAN IV/CENTAUR PROFILE (401)

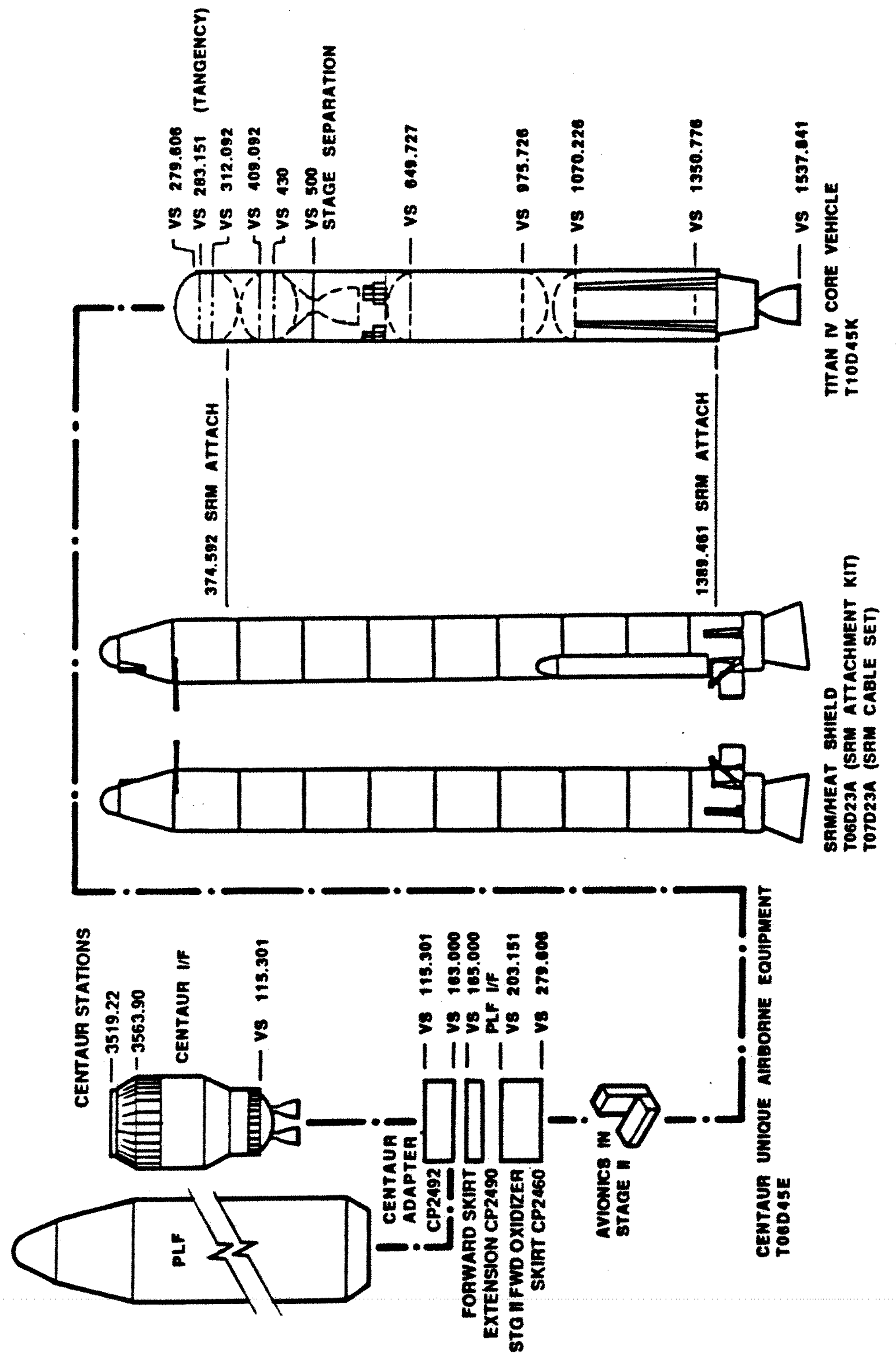


FIGURE 4.2.1.1-1 CI RELATIONSHIP WITH TITAN/CENTAUR LAUNCH VEHICLE

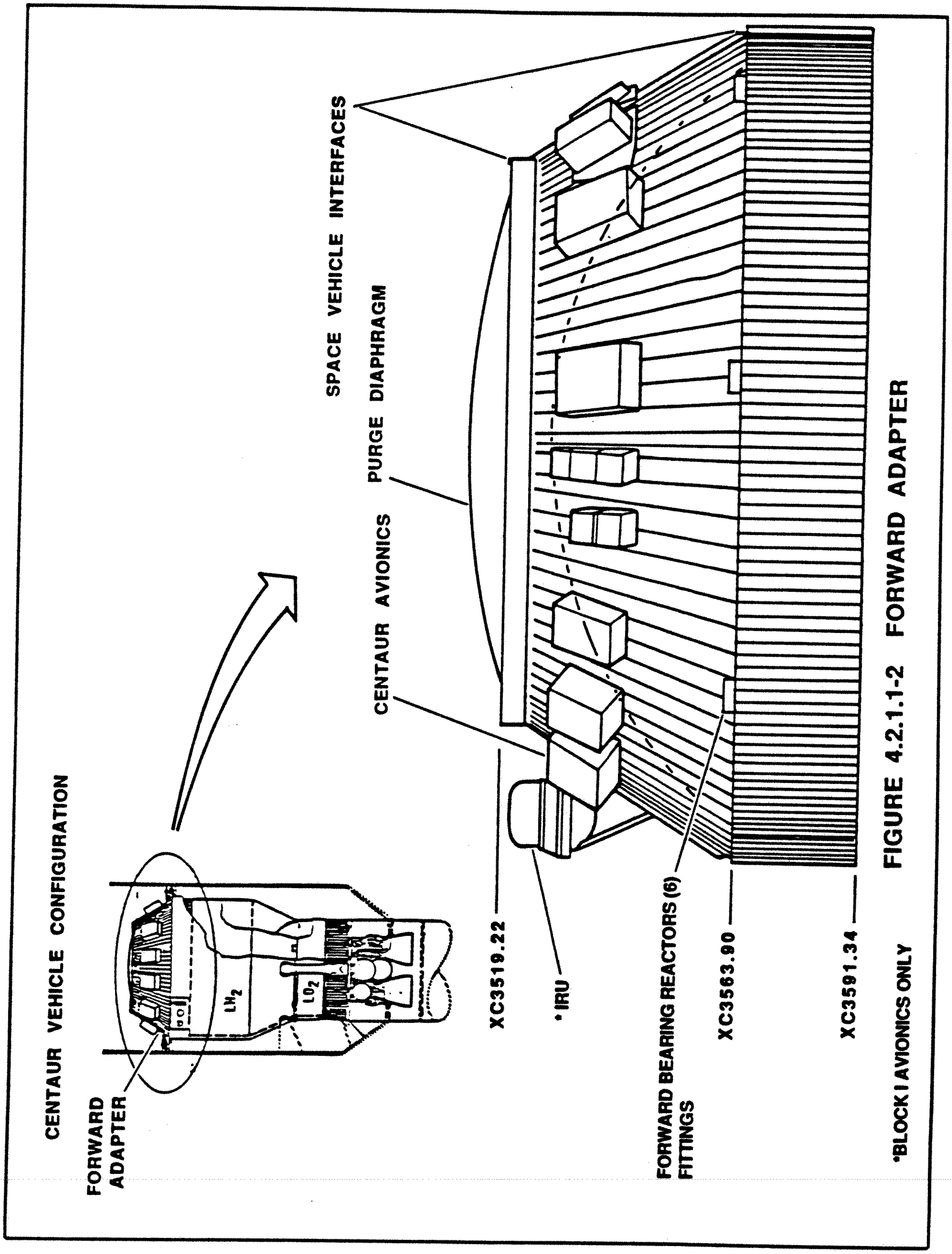
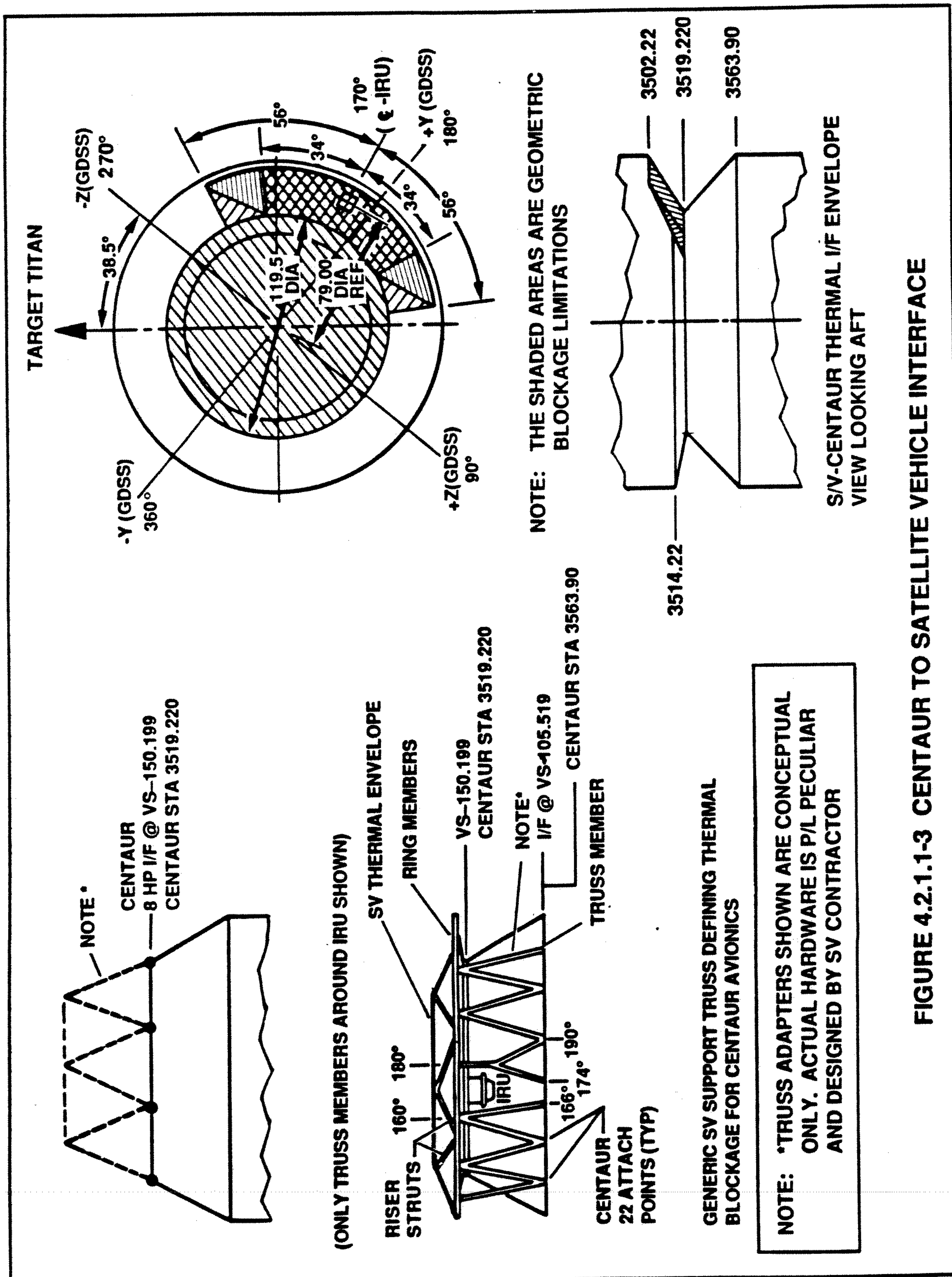
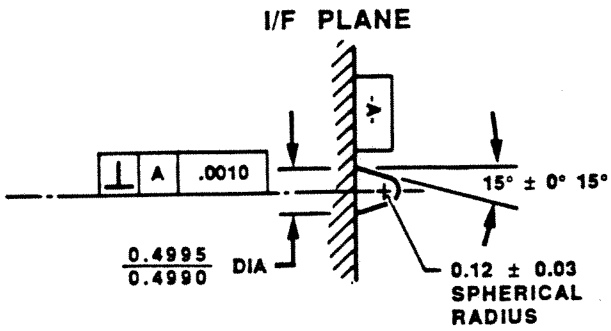


FIGURE 4.2.1.1-2 FORWARD ADAPTER



NOTE: *TRUSS ADAPTERS SHOWN ARE CONCEPTUAL ONLY. ACTUAL HARDWARE IS P/L PECULIAR AND DESIGNED BY SV CONTRACTOR

FIGURE 4.2.1.1-3 CENTAUR TO SATELLITE VEHICLE INTERFACE



SHEAR PIN DETAIL
TYP 16 PLACES

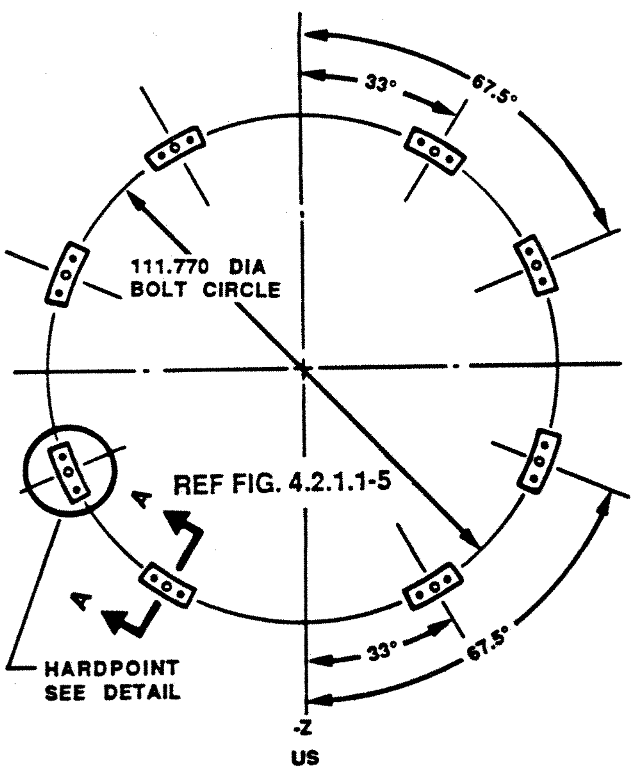
1. THE TRUE POSITION TOLERANCE INDICATED APPLY TO THE TRANSFER OF NOTED HOLES FROM THE MASTER GAGE TO THE FINAL HARDWARE

2. EACH US AND SPACECRAFT MOUNTING SURFACE SHALL BE FLAT WITHIN 0.006 INCHES PER INCH TOTAL NOT TO EXCEED 0.010 INCHES MEASURED IN ALL DIRECTIONS.
ALL ATTACH SURFACES SHALL LIE BETWEEN TWO PARALLEL PLANES WHICH ARE 0.010 INCHES APART AND PERPENDICULAR TO THE US X AXIS

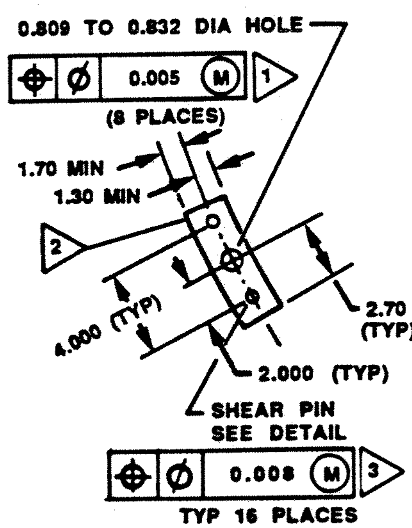
   THESE FLATNESS REQUIREMENTS SHALL BE MET FOR THE US CONFIGURATION THAT EXISTS JUST PRIOR TO SV/US MATE

3. THE TRUE POSITION TOLERANCE INDICATED INCLUDES 0.005 FOR TRANSFER OF NOTED HOLES FROM THE MASTER GAGE TO THE FINAL HARDWARE PLUS 0 .003 FOR CONCENTRICITY OF SHEAR PIN WITH RESPECT TO HOLE THEORETICAL CENTERLINE

TIV VEH STA – 150.199
GDSS – 3519.220



VIEW LOOKING AFT



HARDPOINT DETAIL
(TYP 8 PLACES)

FIGURE 4.2.1.1-4 CENTAUR 8 HP/SV ATTACHMENT INTERFACE DETAILED ARRANGEMENT

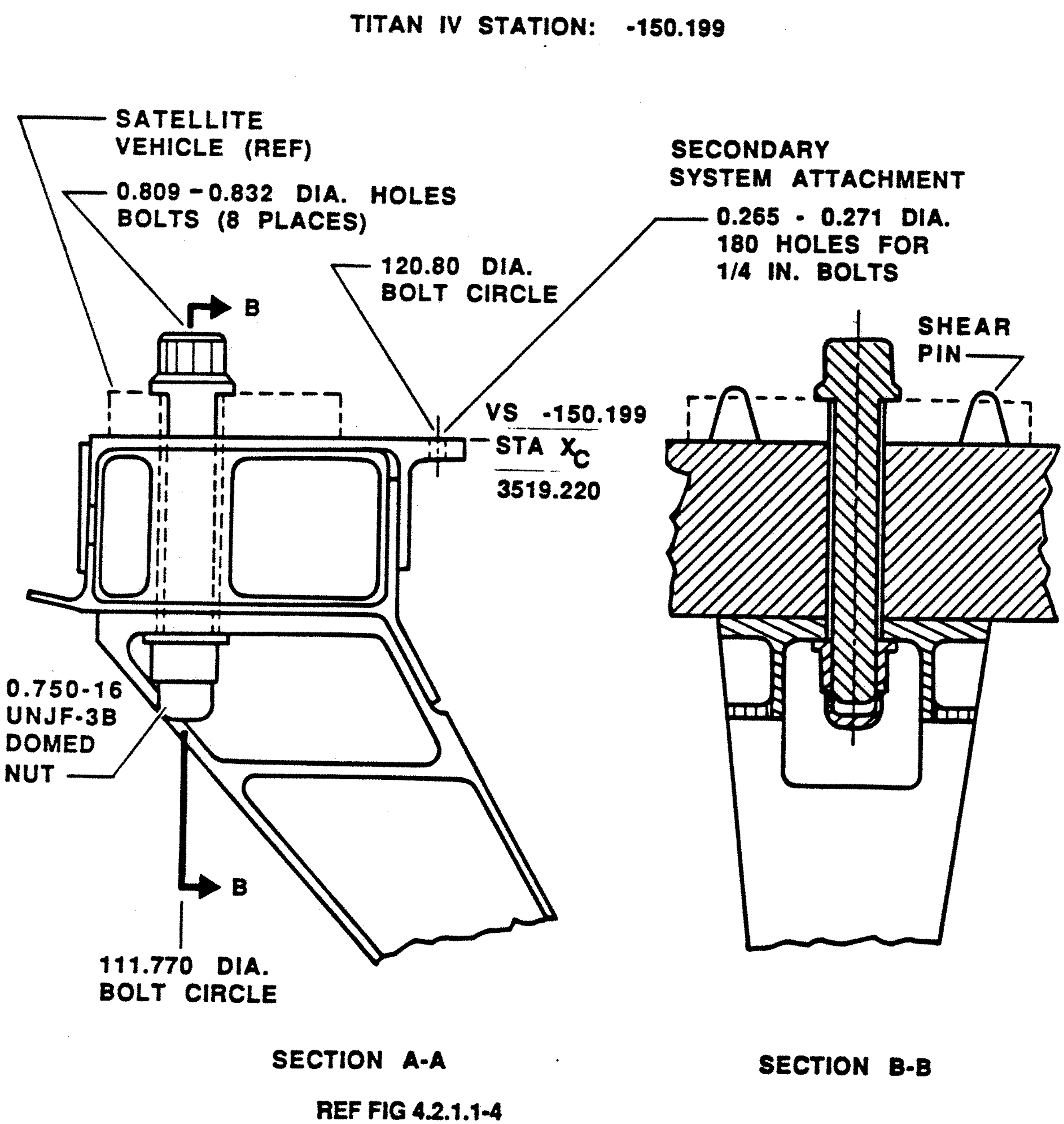


FIGURE 4.2.1.1-5 CENTAUR 8 HP/SV ATTACHMENT INTERFACE RING GEOMETRY

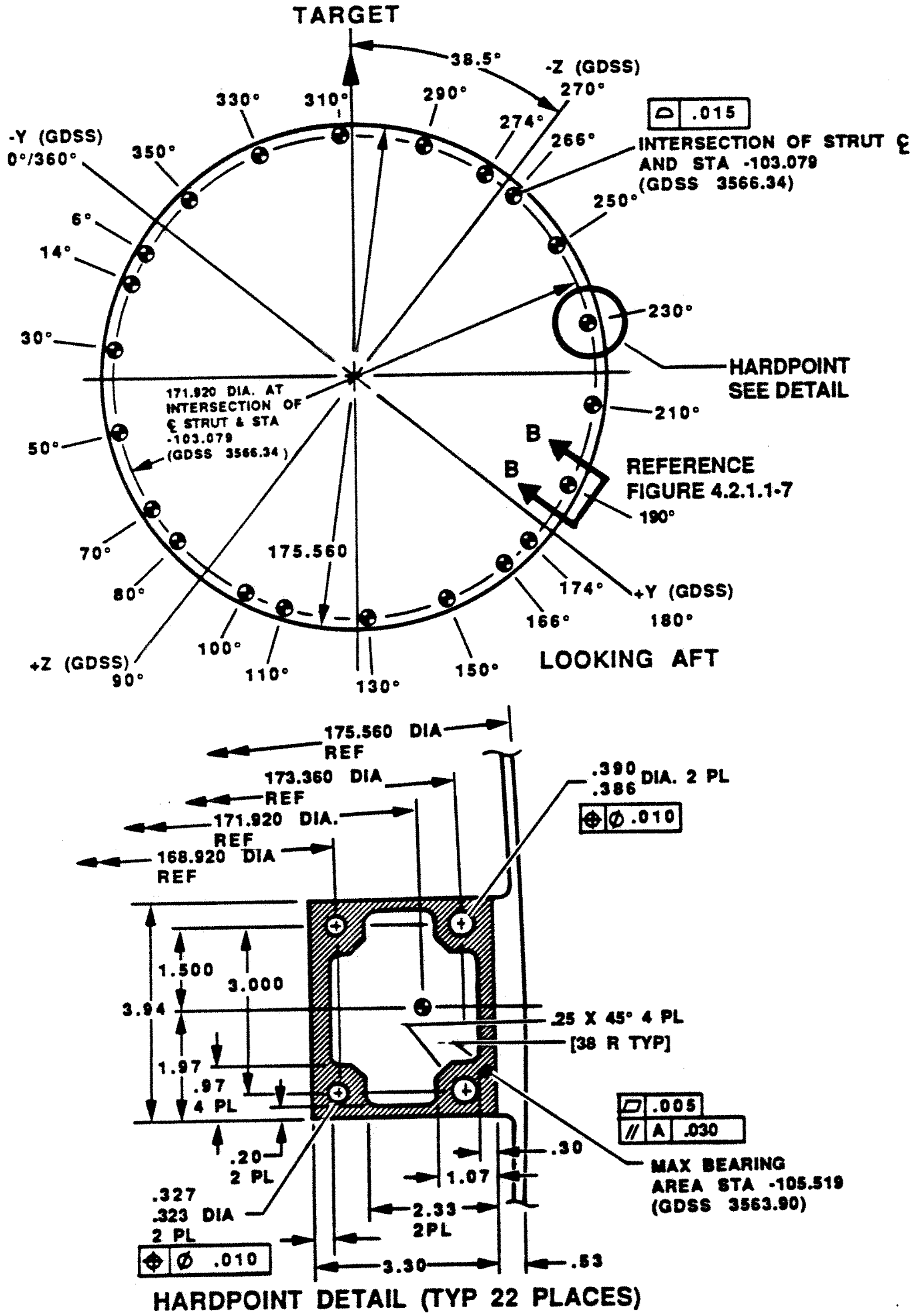


FIGURE 4.2.1.1-6 CENTAUR 22 HP/SV TRUSS ATTACHMENT INTERFACE

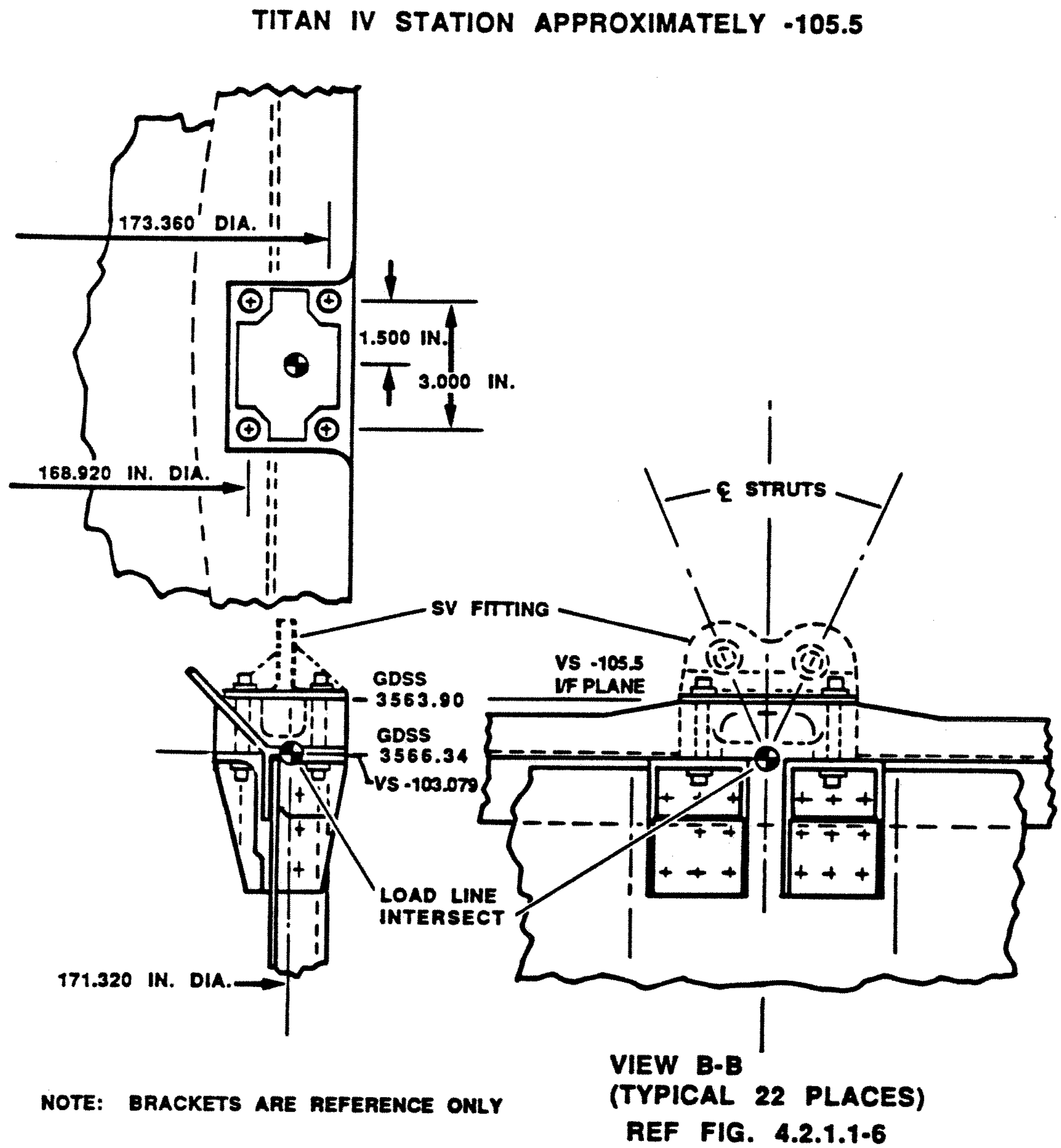


FIGURE 4.2.1.1-7 CENTAUR 22 HP/SV TRUSS ATTACHMENT INTERFACE DETAIL

TITAN IV STATION: -150.199

CENTAUR STATION: 3519.22

(UNFUELED DIMENSION)

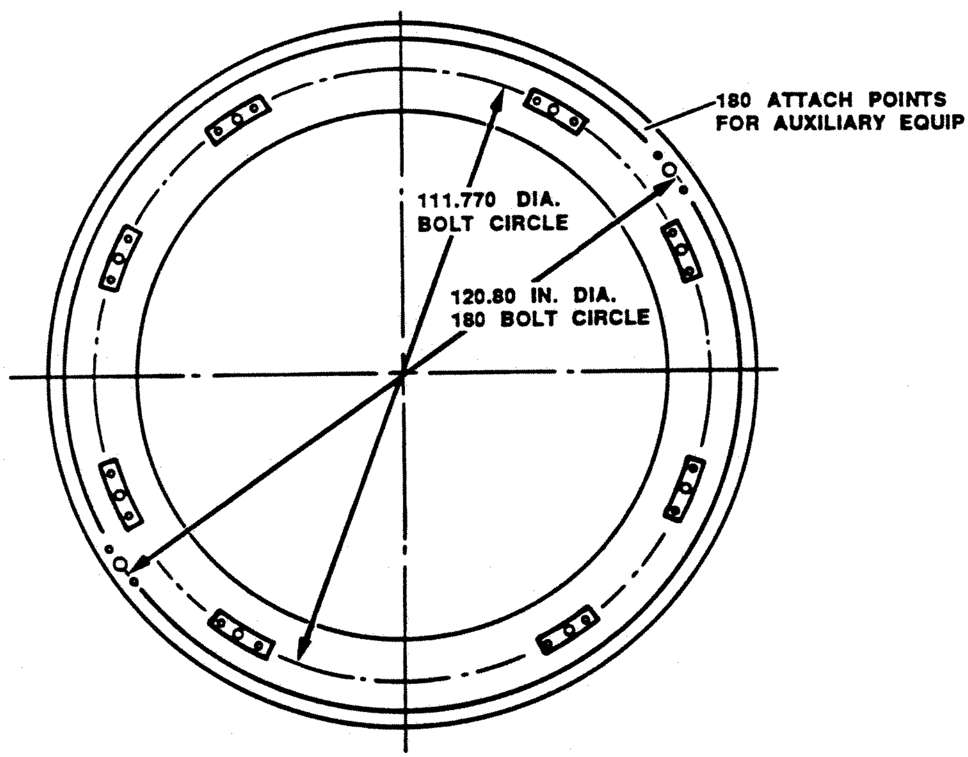


FIGURE 4.2.1.1-8 CENTAUR/SV AUXILIARY EQUIPMENT ATTACHMENT INTERFACE GENERAL ARRANGEMENT

4.2.1.2 Access

Access to the Centaur 171.92 in. diameter interface, for maintenance or replacement of Centaur forward adapter mounted components, is required at all times from installation of the PLF and/or SV, to the time of launch. To accommodate access requirements, individual struts of the SV truss adapter must be removable. Two adjacent struts are to be removable at any one time without requiring exterior support to the SV truss. Other SV adapter truss designs, such as hinged struts, are acceptable provided the access and structural requirements are satisfied. If installation of the truss adapter is required after the PLF Base Module is installed, access to some of the 22 hardpoints is limited in the area of PLF Sector II. Access across the SV/Centaur Interface within the 111.77 in. interface bolt circle at Centaur Station 3519.22 is not required.

The Centaur System design permits physical access to the exterior surfaces of the SV forward and aft of the Centaur/SV Interface plane.

4.2.1.3 Spacecraft Separation

The SV/Centaur Separation plane is on the SV side of the SV-to-Centaur Interface Adapter. The Interface Adapter remains attached to the Centaur after separation.

The Centaur is capable of withstanding, without damage to the avionics components, the separation force induced by the SV. A relative velocity of at least 0.5 ft/sec along the x-axis will be induced by the SV mechanism. Maximum tip-off rates will be specified by each user program.

The Centaur is capable of withstanding the SV/Centaur separation pyro shock environment without damage.

4.2.1.4 Satellite Vehicle Mass and Center of Gravity

The SC mass properties at SV/Centaur separation are defined by the SV contractor/LSIC. Centaur CG offset is to be within 5 in. of the longitudinal axis at SV/Centaur separation. SV CG offsets of up to the limits shown in Figure 4.2.1.4-1 from the Centaur x-axis can be accommodated. The zones of allowable SV CG offsets are based on a maximum 1.5 deg engine gimbal trim requirement. SV CGs that are outside the zone of allowable offsets may be acceptable but must be evaluated on a mission peculiar basis.

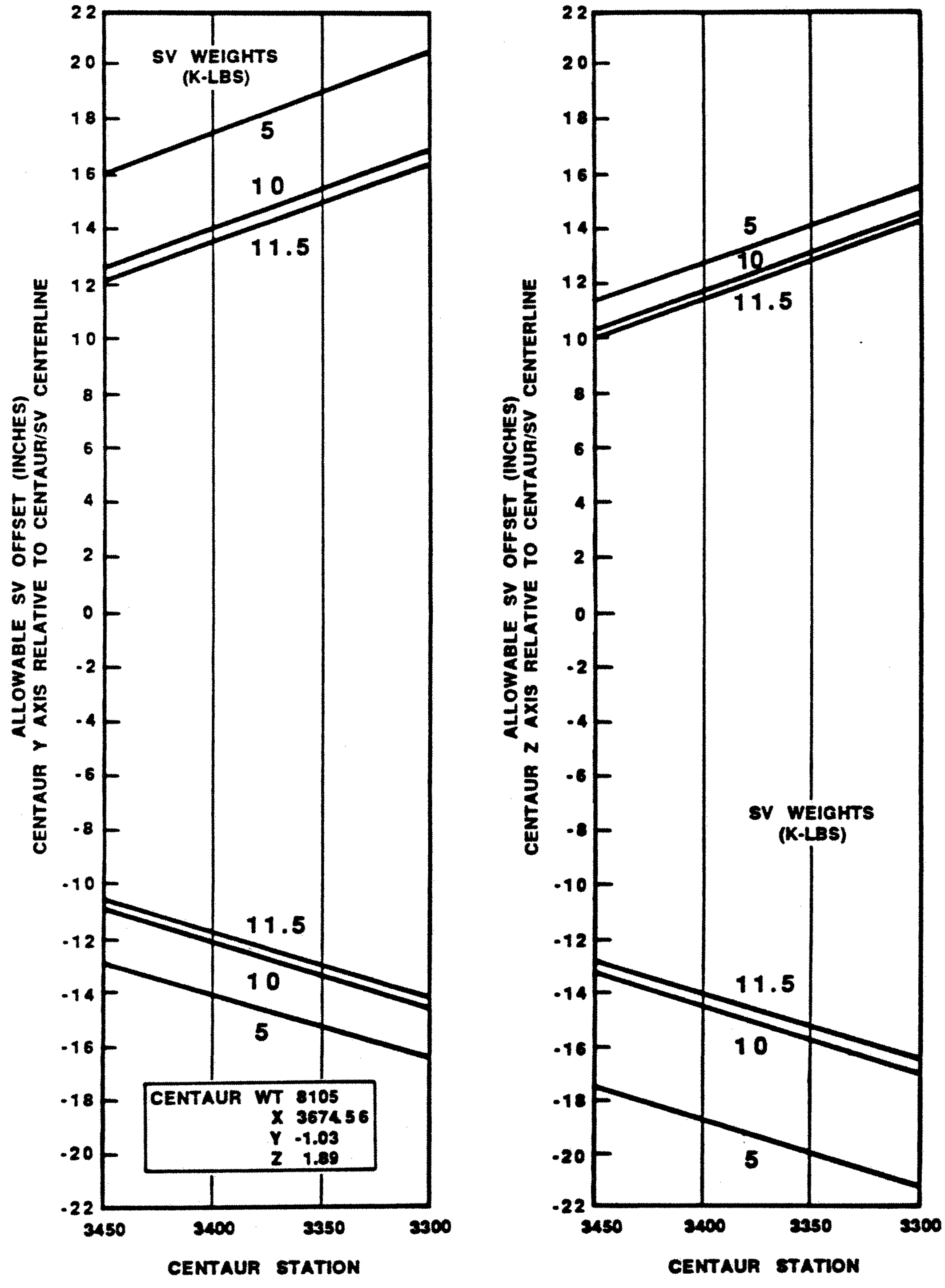


FIGURE 4.2.1.4-1 SV LATERAL CG ALLOWABLE OFFSETS (CENTAUR)

4.2.1.5 Coordinate System

Reference Figures 4.2.1.5-1, 4.2.1.5-2, 4.2.1.5-3 and 4.2.1.5-4.

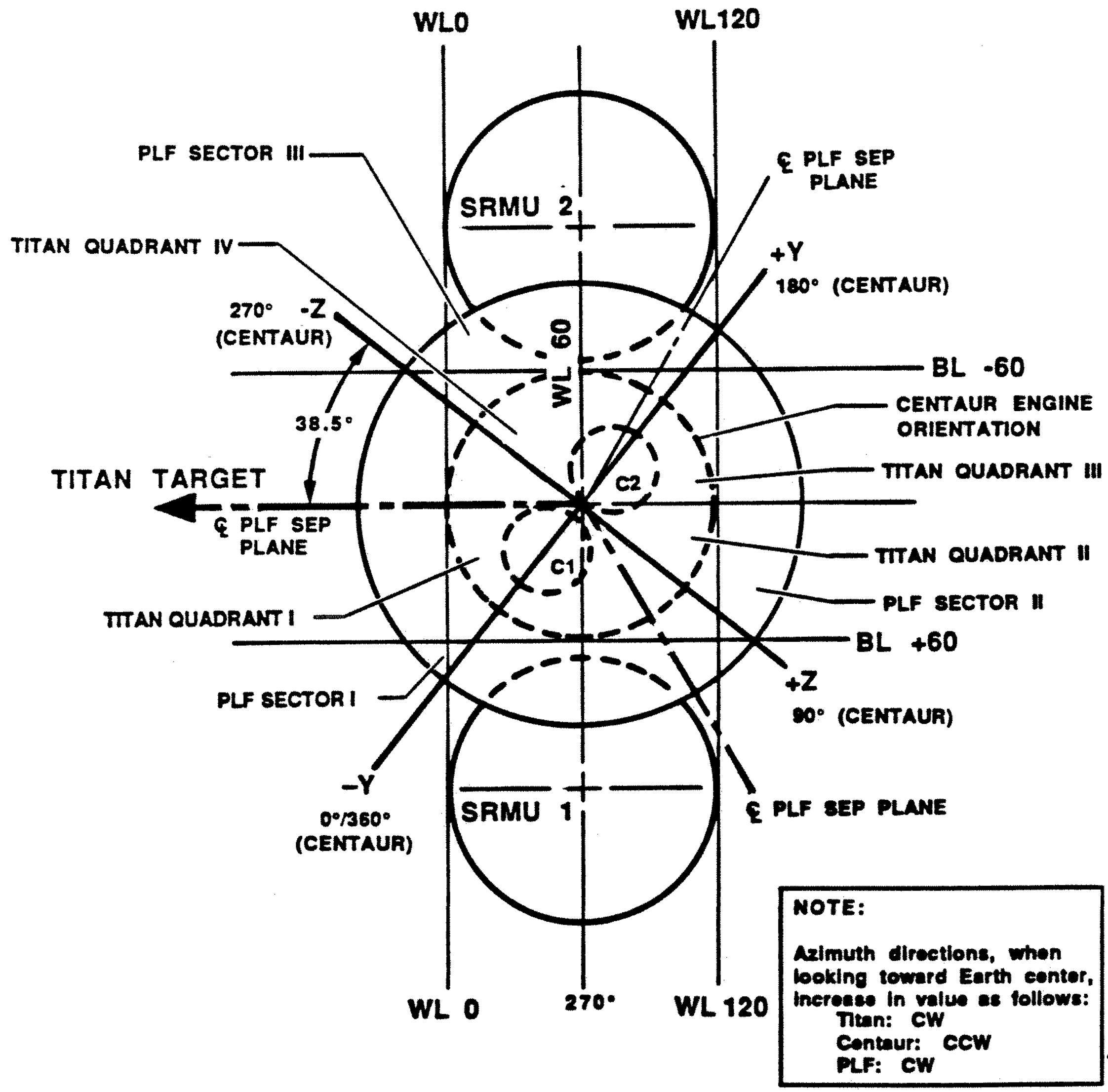
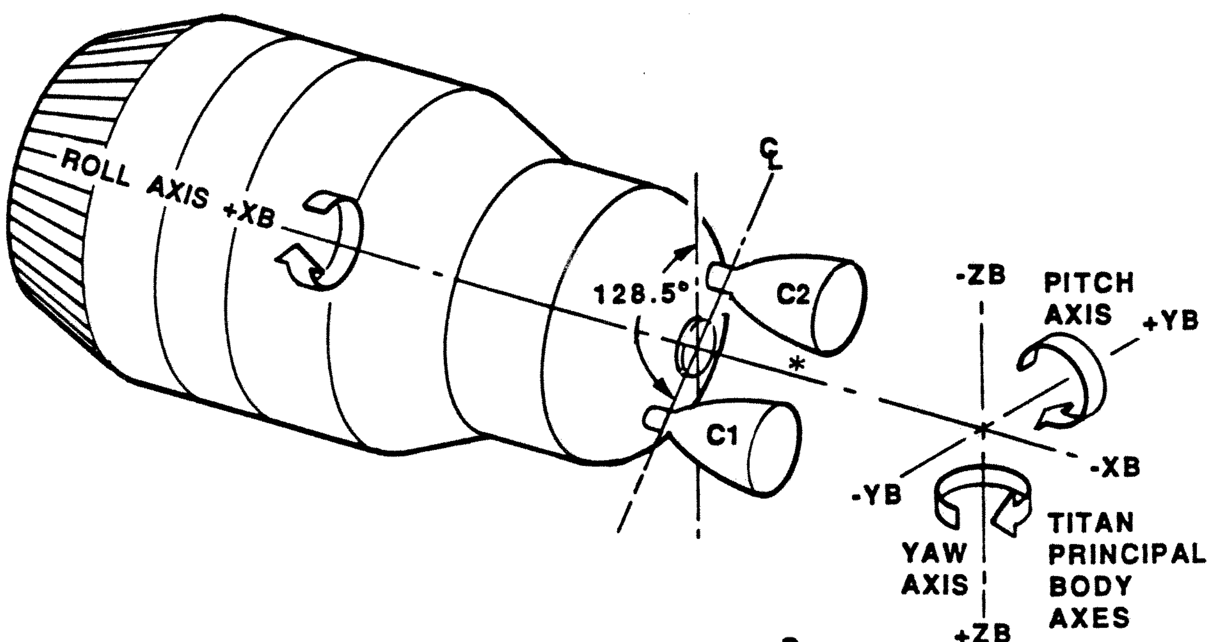
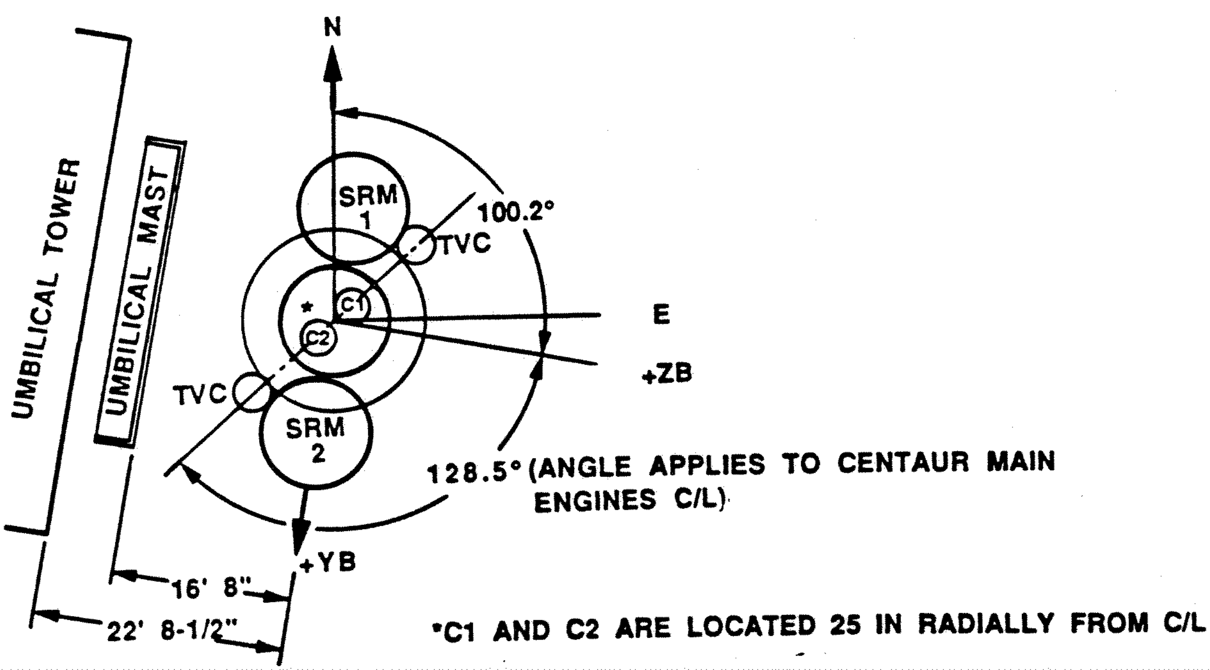


FIGURE 4.2.1.5-1 TITAN IV/CENTAUR PLAN VIEW (LOOKING TOWARD EARTH CENTER)



*C1 AND C2 ARE LOCATED 25 IN. RADIALLY FROM THE ℄ .

FIGURE 4.2.1.5-2 SIMULATION AXES AND NOTATION FOR TITAN IV/CENTAUR



*C1 AND C2 ARE LOCATED 25 IN RADIALLY FROM C/L

FIGURE 4.2.1.5-3 TITAN IV/CENTAUR ORIENTATION ON LC-41

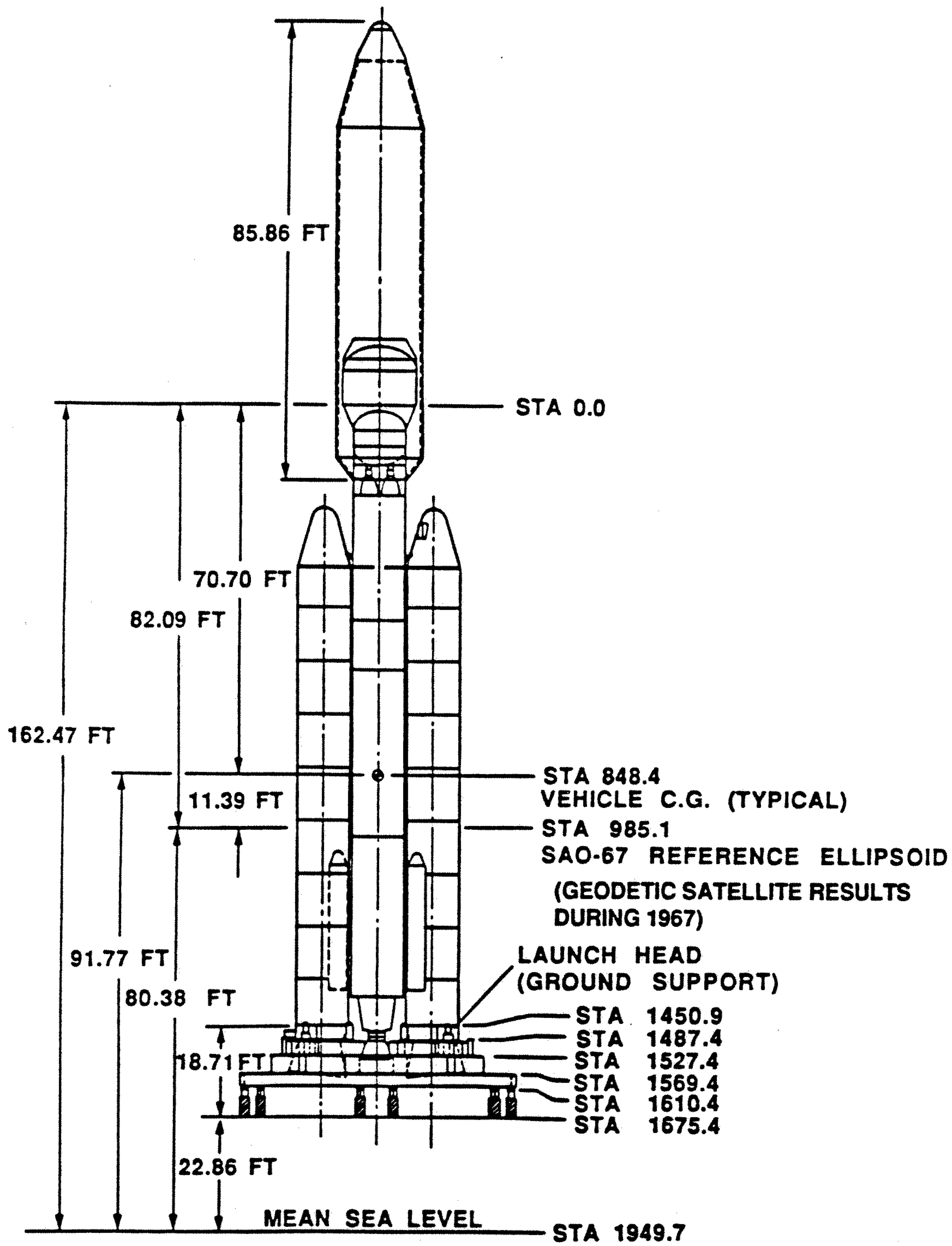


FIGURE 4.2.1.5-4 TITAN IV/CENTAUR LAUNCH PAD ELEVATIONS (401)

4.2.2 Payload Fairing

4.2.2.1 Clearances

The minimum hardware-to-hardware clearance between the structure of the PLF and the SC is specified by the Titan IV Systems Specification to be no less than 1.00 in. The clearance loss between the PLF and the structure inside the PLF will be determined for all critical dynamic loading events for each SC. The maximum dynamic clearance loss combined with manufacturing misalignment tolerances and loss due to temperature determine the clearance loss between the PLF and internal structure.

The SC dynamic envelope is 180 in. in diameter except for a local intrusion for an A/C duct at the forward end of the PLF, reference Figure 4.2.2.1-1.

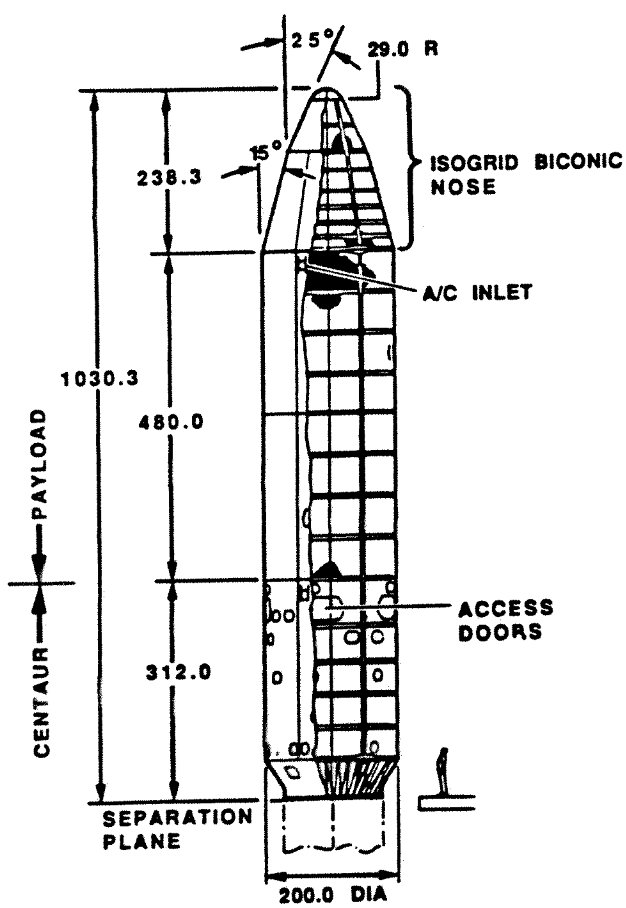


FIGURE 4.2.2.1-1 TITAN IV/CENTAUR PLF OVERVIEW

4.2.2.2 Compartment Venting

Venting of the SV, Upper Stage and PLF shall be through the vents located in the aft region cylindrical portion of the PLF. Any mass flow of gas forward from the Upper Stage/SV interface shall not contain harmful debris nor any molecular or particulate contamination above specified levels.

The 66, 76 and 86 ft PLF internal pressure decay during ascent shall not exceed 0.4 psi/sec, except for a perturbation not exceeding 3 sec in duration where the rate shall not exceed 0.6 psi/sec.

4.2.2.3 Forward Bearing Reactor

Provisions for a load sharing structural tie (Forward Bearing Reactor (FBR)) between the PLF and the Centaur is required for compliance with the specified load constraint and clearance requirement. The FBR system is located between the Forward Adapter and the PLF. This system allows for a controlled amount of load sharing when differential deflections occur between the Centaur Vehicle and the PLF. The amount of load sharing is controlled by selecting the spring rate of the load limiting spring in the reactor which provides a reduction of the Centaur deflections and loads at Titan IV liftoff and during ascent. The device is separated by explosive bolts and retracted into the PLF after maximum dynamic loading and prior to PLF separation. The explosive bolt firing signals are furnished by Titan IV; reference ICDs, and, Figures 4.2.1.1-1, 3.4.1-4, 3.5.1-2 and 3.5.1-3.

4.2.3 Electrical Systems

Reference Figure 4.2.3-1.

4.2.3.1 Attitude Control

The Centaur can stabilize attitude rates at 0 ± 0.1 deg/sec (3-sigma) about the pitch and yaw axes and 0 ± 0.3 deg/sec (3-sigma) about the roll axis (in the presence of no programmed roll) just prior to SV/Centaur separation command.

The Centaur can perform drift and orbital placement maneuvers for multiple SC. The SC can be separated in park, transfer, or final orbit. A separation velocity difference between multiple SC can be provided. The incremental velocities are achieved by using the aft-pointing, pitch and yaw attitude control engines.

The SV/Centaur orientation requirements for Park Orbit, Transfer Orbit and Final Orbit will be in accordance with the SV ICDs.

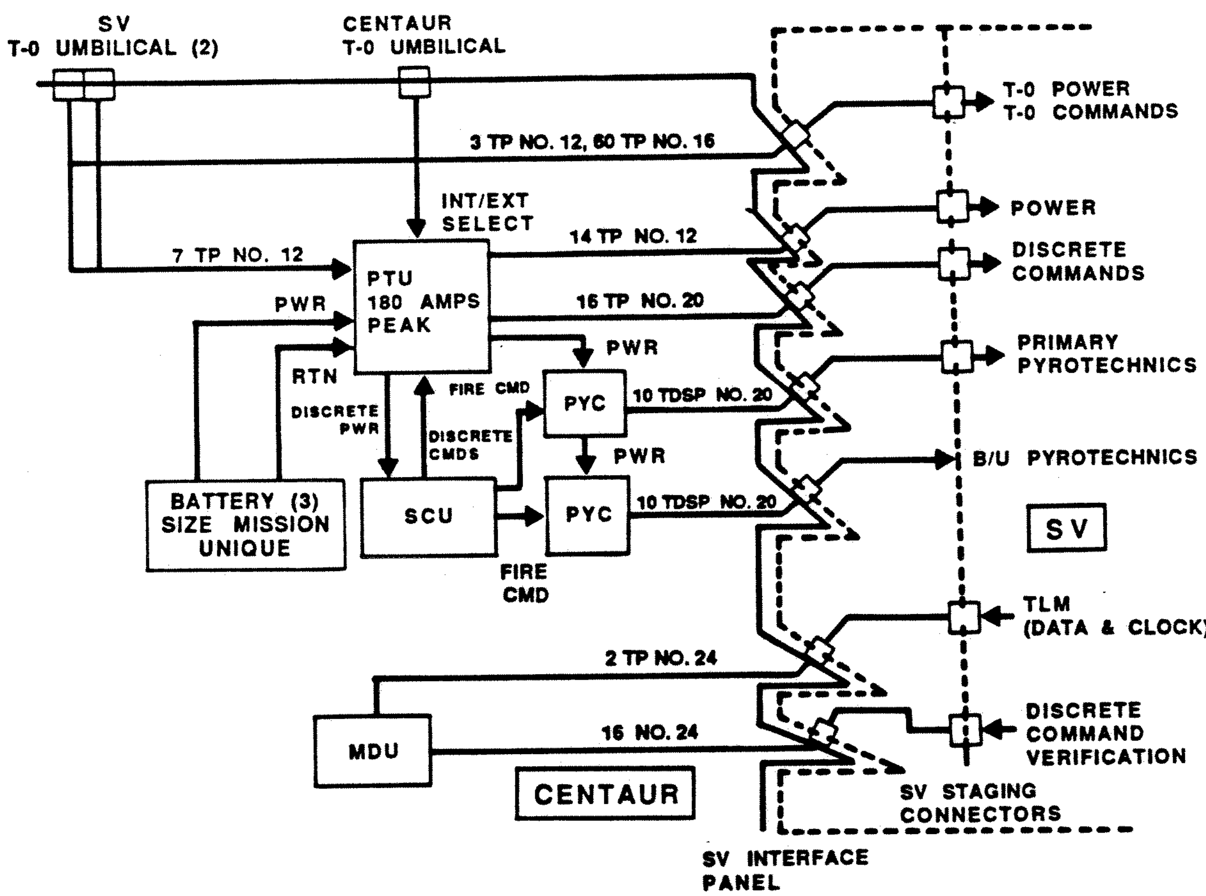


FIGURE 4.2.3-1 SV/CENTAUR ELECTRICAL INTERFACE

4.2.3.1.1 Collision Avoidance

To perform collision avoidance after SV separation, the RCS and/or Main Engine (ME) maneuvers place the Centaur in an orbit that precludes contact with the SC. ME thrust maneuvers are accomplished by expelling remaining engine propellants without their ignition (blow down). RCS/ME maneuvers are done in a manner such that the exhaust products do not contaminate any SC surfaces.

4.2.3.2 Signal Interfaces

4.2.3.2.1 Instrumentation

A one kHz bit stream of data from the SV, clear text or encrypted, can be interleaved with Centaur data at the Master Data Unit (MDU). The MDU can also receive and process 16 discretes which verify receipt of Centaur generated commands at the SV and six discrete measurements that monitor Centaur/SV separation.

The MDU provides acquisition of data from vehicle systems for use in operation/ performance evaluation and/or failure analysis. It provides signal conditioning of transducer outputs, transducer excitation, analog-to-digital encoding, Pulse Code Modulation (PCM) formatting and PCM outputting. It accepts serial digital data from external sources, as well as from the Remote Data Unit (RDU) and reformats the data to a form compatible with the required MDU output.

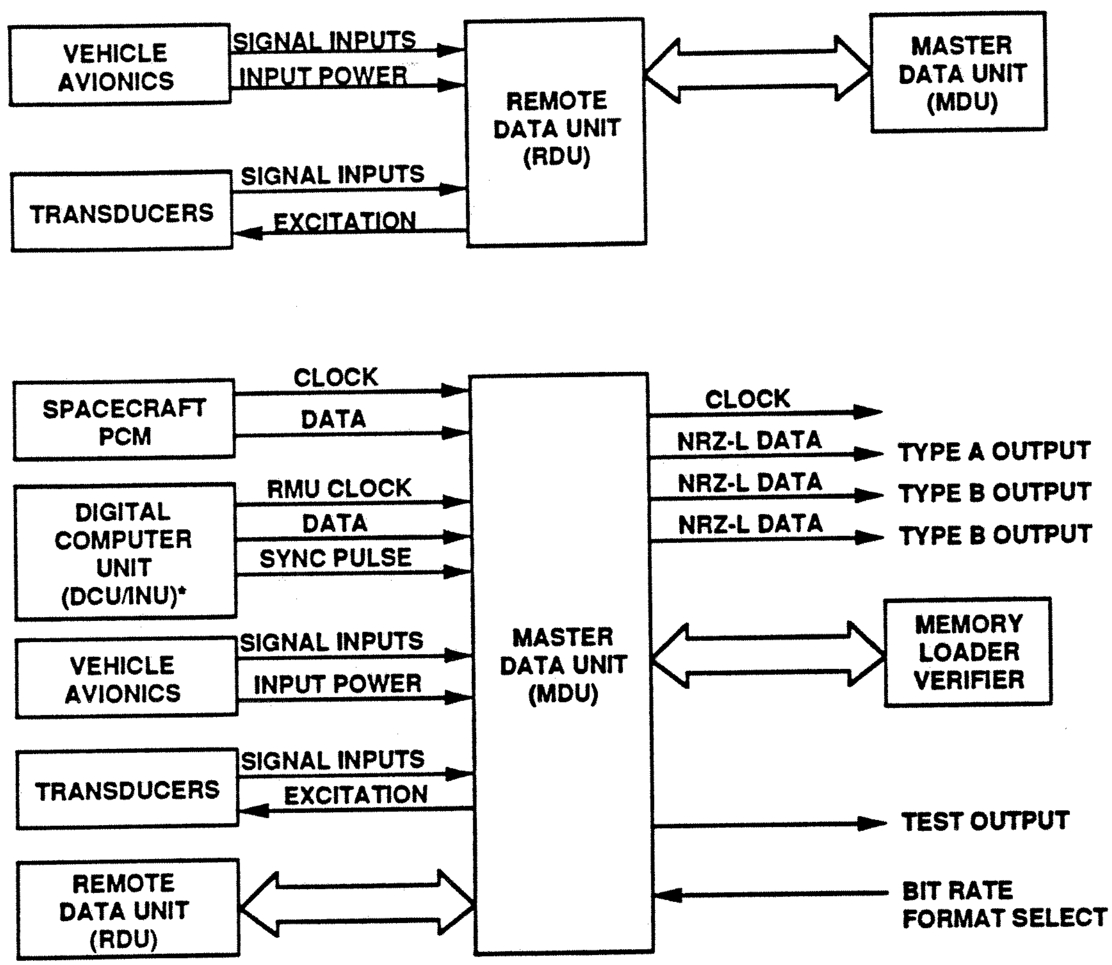
The MDU provides two independent serial data ports, each consisting of a Non-Return to Zero-Level (NRZ-L) input and a clock input. The serial data is buffered using input and output buffers, each capable of storing up to 256 eight-bit words. The input/ output buffers exchange assignments at the start of each major frame such that serial data received during one major frame may be output (under control of the PCM format) during the next major frame. Incoming data is grouped into eight-bit bytes prior to being input to the buffer. The first word in the buffer contains the number of bytes input to the buffer during that major frame, reference Figure 4.2.3.2-1.

4.2.3.2.2 Telemetry

SV instrumentation measurements originating on the SV can be telemetered through the Centaur Telemetry System. Centaur telemetry data is transmitted continuously throughout the flight through CCAM.

No special telemetry transmission maneuvers are required in park orbit. However, during the coast period in the elliptical transfer orbit to synchronous orbit, special maneuvers may be programmed to point the antennas to an appropriate tracking station.

The Centaur PCM bit stream is encrypted prior to transmission to the ground, reference Figure 4.2.3.2.2-1.



* DCU: CENTAUR FLIGHTS 1 THROUGH 4
* INU: CENTAUR FLIGHTS 5 AND UP

FIGURE 4.2.3.2-1 DAS FUNCTIONAL INTERFACE RELATIONSHIPS

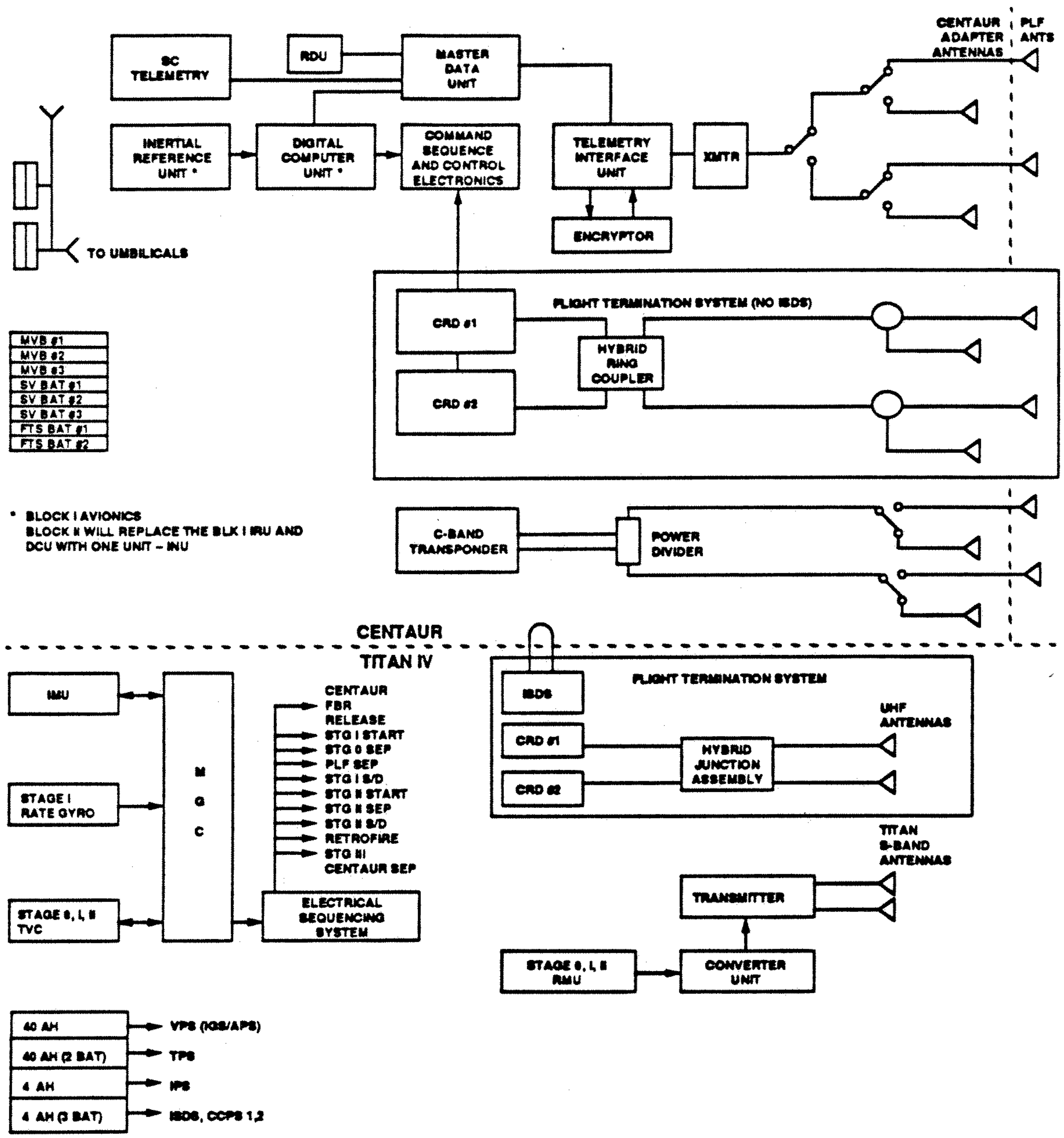


FIGURE 4.2.3.2.2-1 TITAN IV/CENTAUR AVIONICS BLOCK DIAGRAM

4.2.3.2.3 Commands

The Centaur Guidance, Navigation and Control (GN&C) System provides the capability to sequence 16 discrete SV commands and the SV pyrotechnics. Discretes are generated in the Centaur Avionics System by either an event scheduling function or by the normal on-board automatic sequencing.

4.2.3.2.3 Commands (Continued)

The capability exists to reference the initiation of a discrete command to Centaur guidance events, mission elapsed time, and/or selected mission scheduled events. Repeat discrete commands can be used.

A maximum of 10 primary and 10 backup firing circuits are provided by the Centaur for SC pyrotechnic ordnance devices. The circuits are designed to be compatible with NASA Standard Initiators (NSI) which meet the requirements of Specification JSC 08060.

4.2.3.2.4 Centaur Tracking

The Centaur Upper Stage and PLF house the range tracking system for the Titan IV/ Centaur configuration.

To preclude possible SV damage from C-Band radiation reflected from the PLF trisectors during the PLF separation two measures have been taken:

a. When the PLF is jettisoned, the transponder is connected to the Centaur forward adapter C-Band antennas. A 1.5 sec delay between the PLF separation event and antenna switching protects the payload from high level C-Band energy reflections from the PLF. During this period, this system will not respond to interrogations.

b. An RF switch/breakwire system provides for the DCU to inhibit transponder functions for a specified time (about 3 sec) during the PLF separation event.

4.2.3.3 Electrical Subsystem

The Centaur is designed to provide uninterrupted single-failure-tolerant 28 Vdc power to the SC Interface from liftoff through SC separation.

Voltage transients at the SC Interface are limited to excursions between 22.8 and 38 Vdc. Any excursions above 34 V will be limited to both a maximum of 38 V and a duration of less than 20 msec. Any excursions below 24.0 V will not exceed 10 msec.

There are between one and three dedicated SV silver oxide/zinc batteries provided for SV power needs. The number of batteries is dependent on the mission scenarios. Each 150 AH Battery is capable of a 60 day activated stand time and is rated at 26-30 Vdc at loads between 50 and 80 amp. These batteries are equipped with thermostats and heaters for temperature control. The capability for a 250 AH battery exists. The weight of these batteries are chargeable to the SV.

The Centaur also provides cabling to interconnect the SC with the required Aerospace Ground Equipment (AGE) interface through the T-0 umbilicals. A Power Transfer Unit (PTU) handles power changeover for the SV between ground power and the dedicated Centaur SV batteries, reference Figures 4.2.3.3-1, 4.2.3.3-2, 4.2.3.-1 and 3.5.1-3.

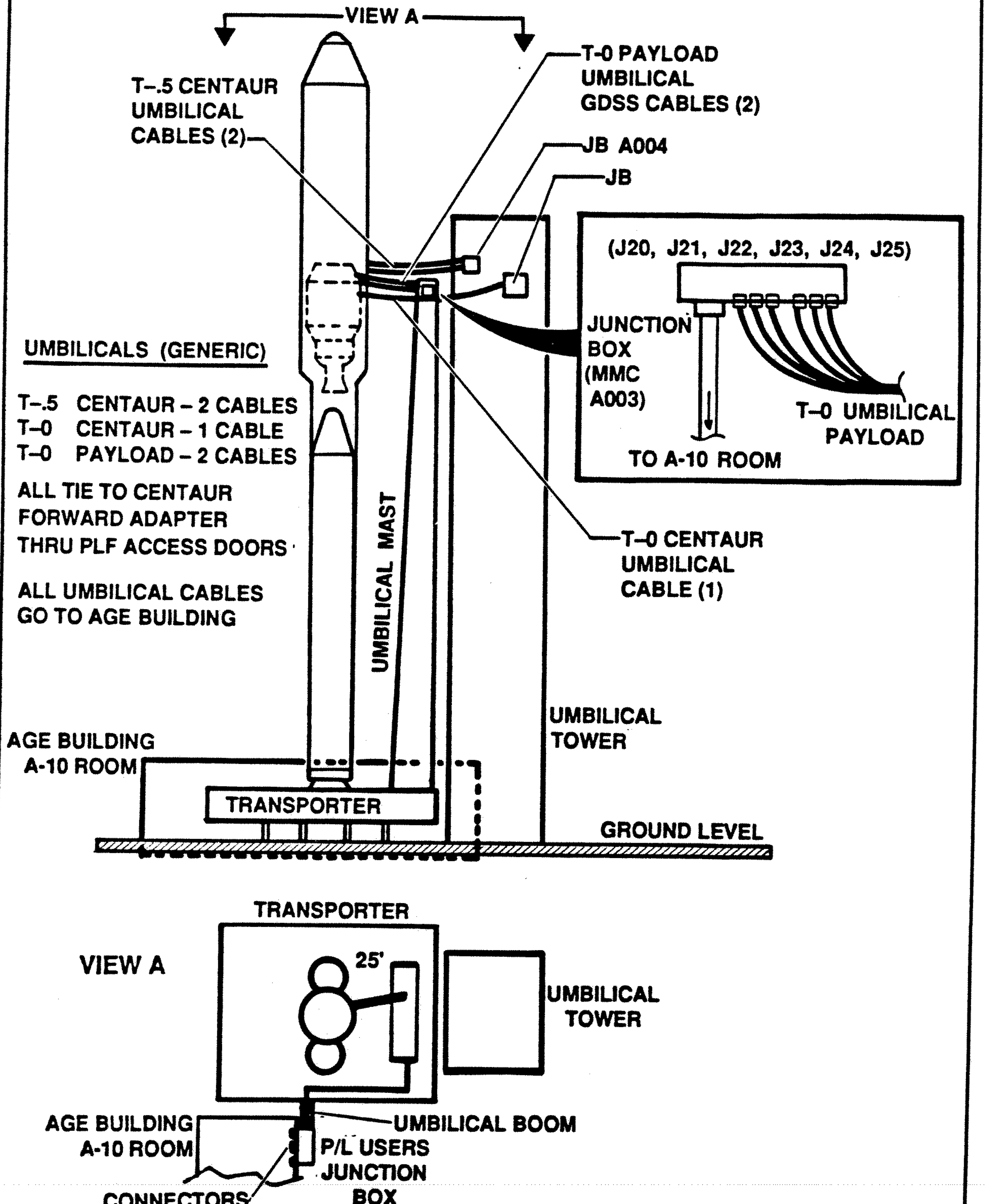


FIGURE 4.2.3.3-1 GROUND UMBILICAL INTERFACES - CENTAUR (CCAFS)

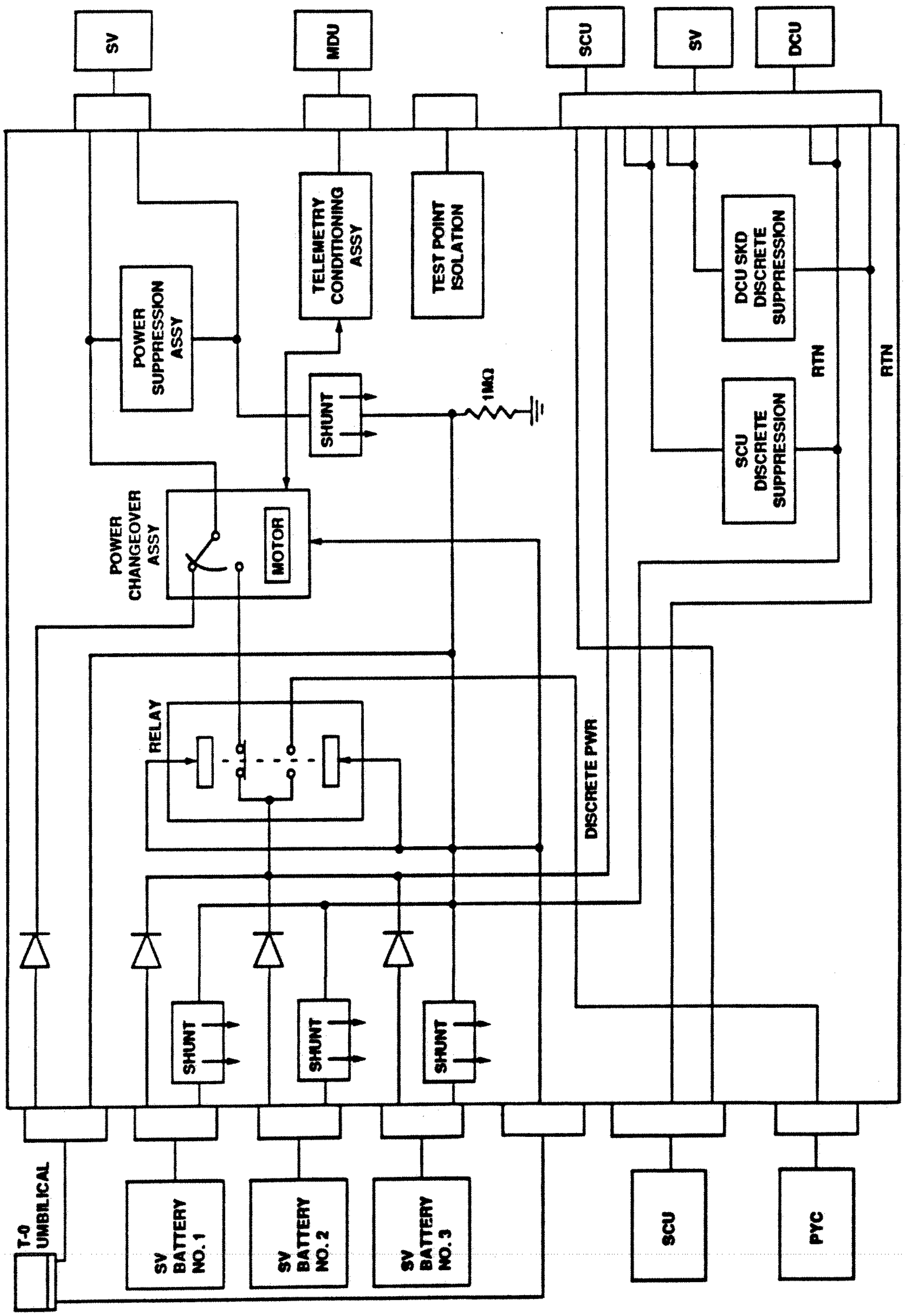


FIGURE 4.2.3.3-2 PTU BLOCK DIAGRAM

4.2.3.4 Software

Reference Figure 4.2.3.4-1 which illustrates the Centaur S/W Interfaces for data exchange and management direction.

4.2.3.4.1 Software Mission Functions

The Centaur DCU is sequenced to the autonomous inertial navigation mode (flight mode) approximately 60 sec prior to liftoff. During the Titan IV ascent, the DCU navigates and controls Centaur tank pressures but does not provide guidance for the Titan which navigates using its own equipment. At Titan/Centaur separation the DCU begins controlling the Centaur engines and attitude system, sequencing mission events until completion of the mission.

After separation from the Titan IV booster, the Centaur will make three main engine burns to place an SC into a near-geostationary orbit or into a 24-hour orbit with an inclination of up to 65 deg. The first engine burn is used to achieve a nominal parking orbit, the second injects the Centaur/SC into an Elliptical Transfer Orbit to geosynchronous attitude and the third burn injects the Centaur/SC into the nominal final orbit.

Whiie in the park orbit, the Centaur shall have the capability to be oriented with the roll axis normal to the ecliptic plane within ± 10 deg and to maintain a roll rate of +1 ± 0.3 deg/sec or –1 ± 0.3 deg/sec within 10 min after MECO1 to within 10 min prior to MES2. The Centaur shall also have the capability to be oriented with the forward-directed roll axis coincident with the velocity vector ± 10 deg. Velocity align is a thermally acceptable attitude pending mission peculiar analysis. Specific requirements shall be in accordance with the SV unique ICDs.

While in the transfer orbit, the Centaur shall have the capability to be oriented with the roll axis normal to the solar vector within ± 6 deg and to provide and maintain a continuous roll rate of +1 ± 0.3 deg/sec or –1 ± 0.3 deg/sec within 10 min after MECO2 to within 10 min prior to MES3. Following MECO3, any coasts in excess of 10 min will require a roll. Specific requirements shall be in accordance with the SV unique ICDs.

At the beginning of the second coast period an Accelerometer Bias Calibration will be performed to re-estimate and replace the preflight bias corrections. This will be followed at the mission mid-point by a 180 deg rotation of the inertial platform to reduce accumulated platform orientation errors due to gyro fixed drifts. These operations reduce navigational errors and are required to satisfy the mission accuracy specification.

After MECO3 the P/L will be separated from the Centaur. The orientation of the Centaur axes at the instant prior to separation may vary for each S/V. Tolerance on attitude orientation shall be ± 1.4 deg (3-sigma) about each axis. Attitude rates shall be stabilized at 0 ± 0.1 deg/sec (3-sigma) about the pitch and yaw axes and 0 ± 0.3 deg/sec (3-sigma) about the roll axis (in the presence of no programmed roll) just prior to the S/V-Centaur separation command.

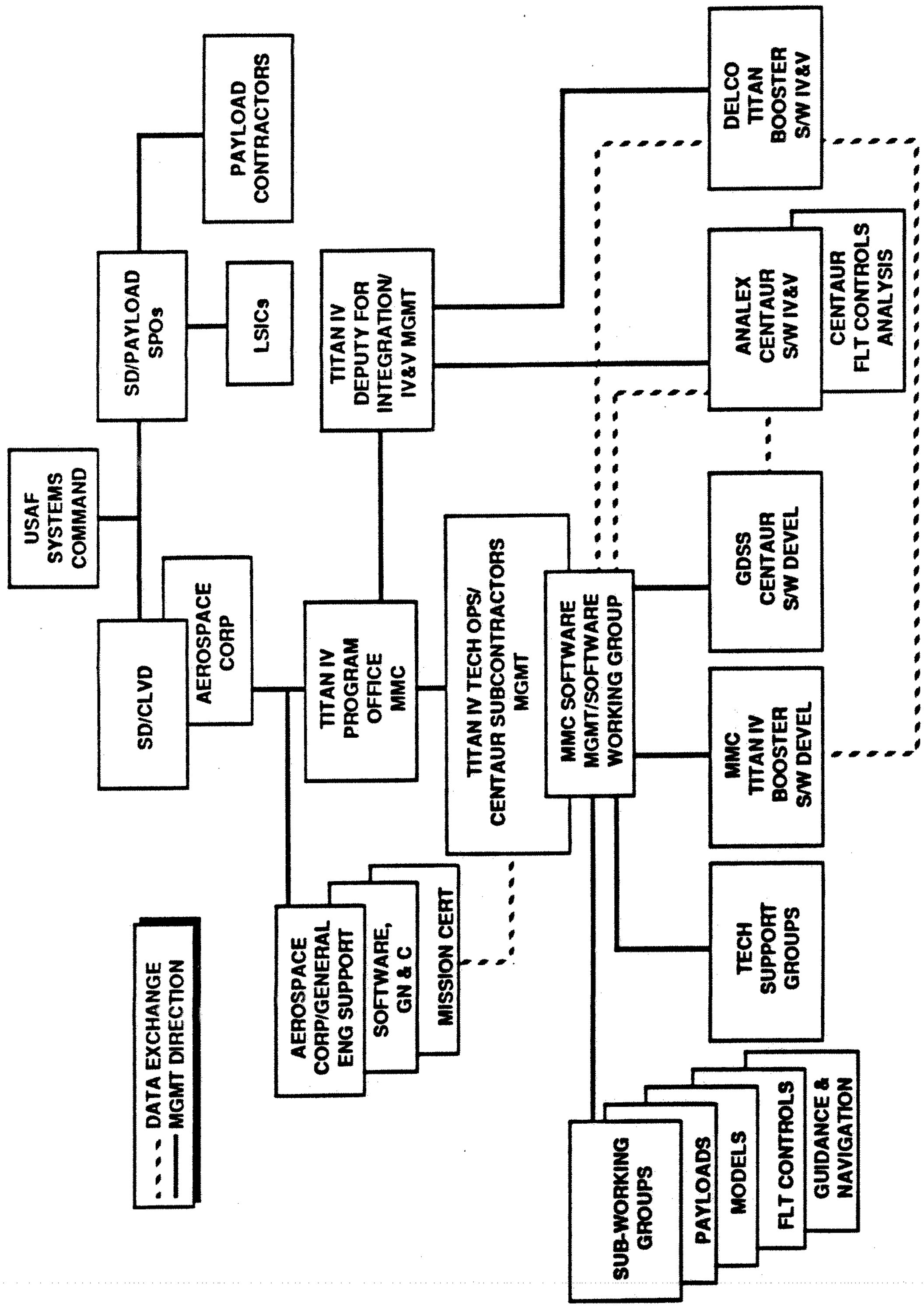


FIGURE 4.2.3.4-1 CENTAUR SOFTWARE INTERFACES (401)

4.2.3.4.1 Software Mission Functions (Continued)

After Centaur-P/L separation; the Centaur will maneuver 8 deg from the separation attitude in the direction of motion; turn 107 deg in the plane of the separation vector and the inertial velocity vector with a minimum angle to the ± velocity vector; and finally, turn normal to the flight plane and blowdown the remaining ME propellants.

4.3 Inertial Upper Stage (SS-ELV-402) (ESMC)

Reference Figure 4.3-1 Titan IV/IUS System Specification Tree and Figure 4.3-2. Titan IV/IUS Vehicle Profile.

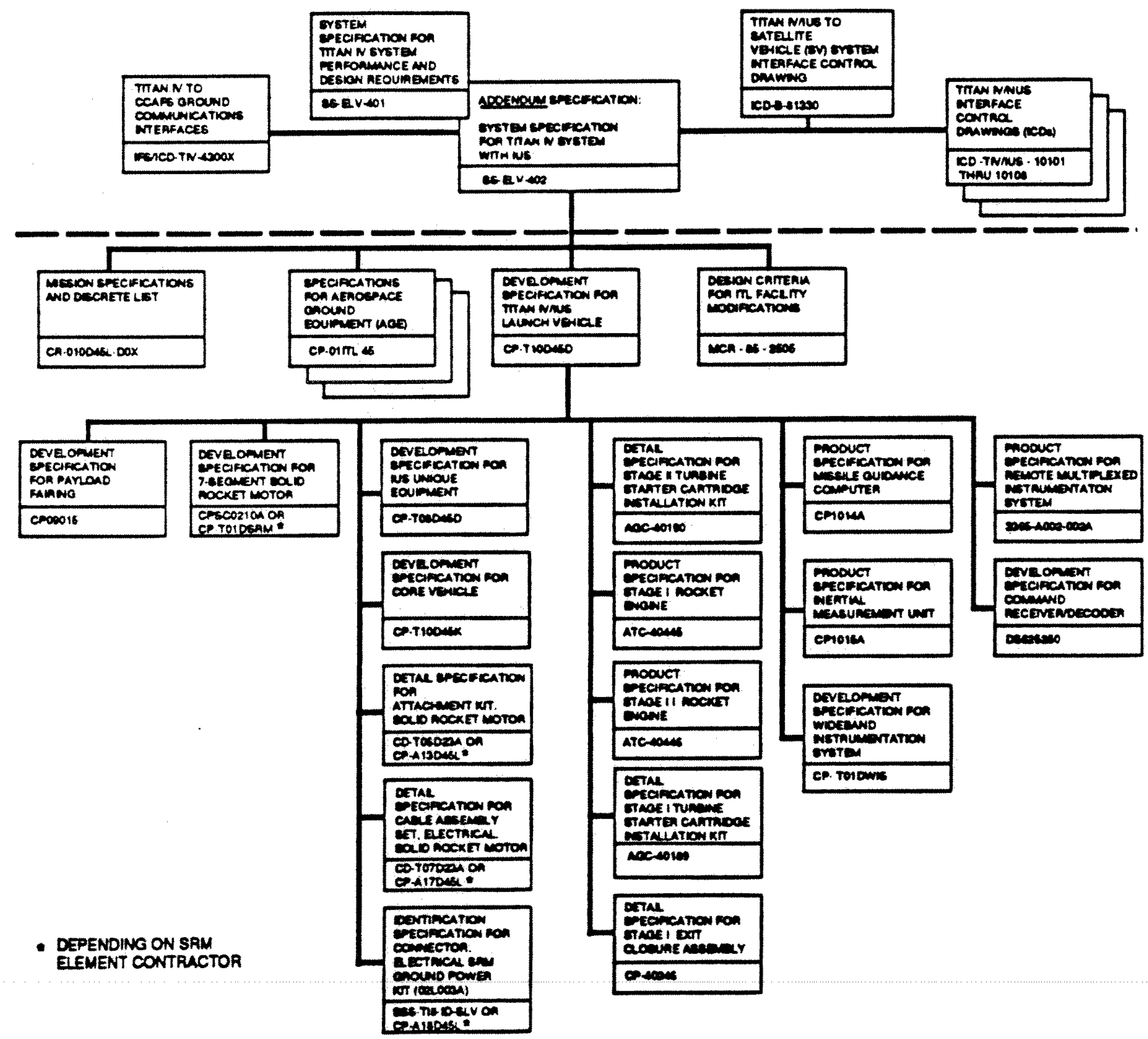


FIGURE 4.3-1 TITAN IV/IUS SYSTEM SPECIFICATION TREE

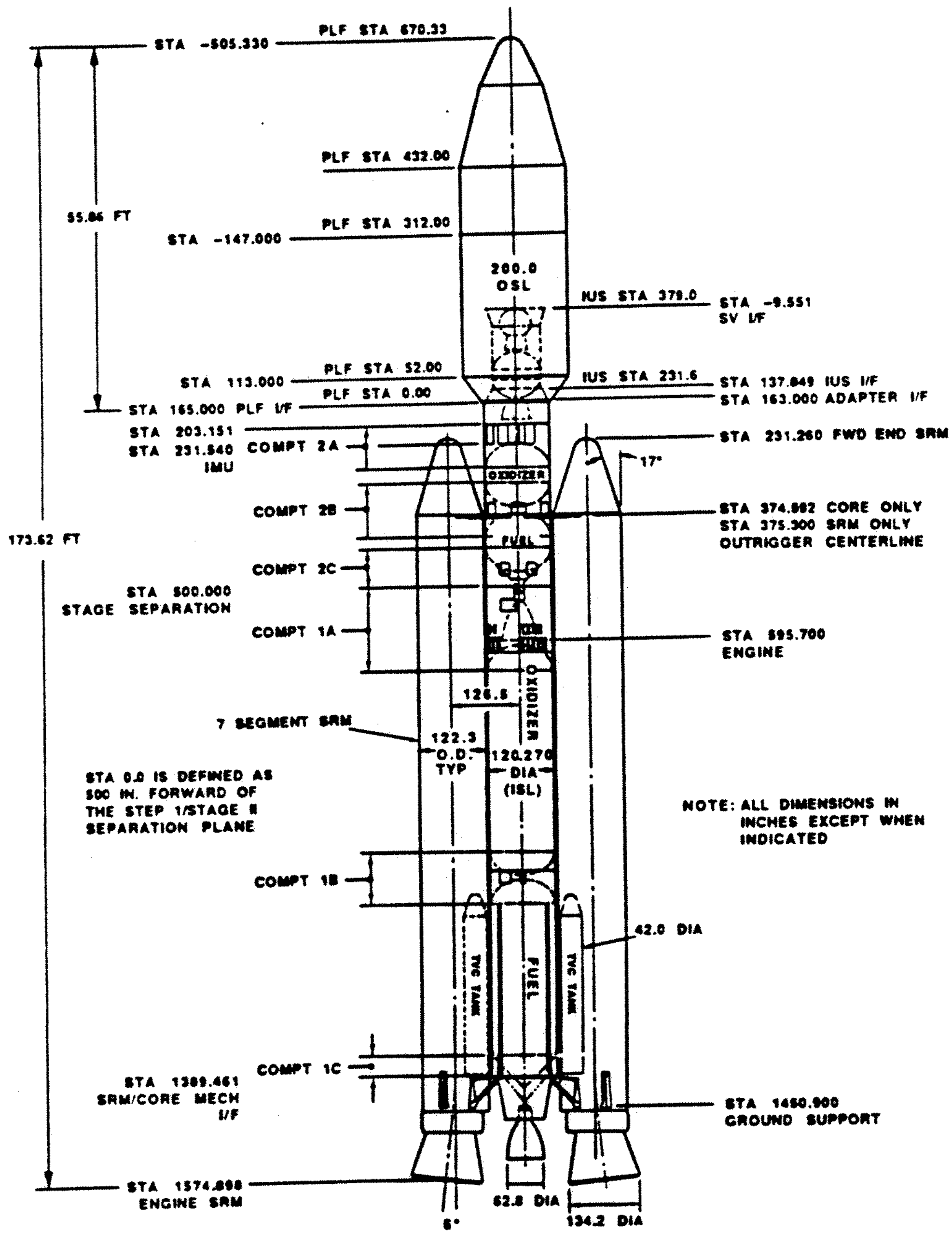


FIGURE 4.3-2 TITAN IV/IUS VEHICLE PROFILE (402)

4.3.1 Structural

4.3.1.1 Spacecraft Mechanical Interface

The IUS is structured with an Aft Skirt, an Interstage, and an Equipment Support Section (ESS). The IUS Aft Skirt is attached to the Titan IUS adapter (which attaches to the 2490 skirt). The Interstage attaches to the Aft Skirt and to an ESS. The top of the ESS provides a Spacecraft Interface Ring with eight spacecraft attach points. The Spacecraft Separation Joint is part of the Spacecraft Structure. Reference Figure 4.3.1.1-1 TIV/IUS Vehicle Assembly, Figure 4.3.1.1-2 CI Relationship with TIV/IUS, and Figure 4.3.1.2-1 DSP/IUS/TIV Coordinate Interface Orientation.

Structural/Mechanical interface definition between the IUS and the spacecraft is a primary responsibility of the IUS SPO and Contractor in conjunction with the spacecraft SPO and Contractor.

ICD-TIV/IUS-10107 is for Airborne Structural and Mechanical Interfaces.

ICD-TIV/IUS-10108 is for Ground Structural, Mechanical, and Access Interfaces.

ICD-TIV/IUS-10102 is for Structural, Mechanical, and Environmental Parametric Interfaces.

Figure 4.3.2-1 shows some TIV/IUS Mechanical Interfaces.
Boeing ICDs-B-81330, -81340 show IUS/SV Interface details.

4.3.1.2 Coordinate System

Reference Figure 4.3.1.2-1 DSP/IUS/TIV Coordinate Interface Orientation, and Figure 4.3.1.2-2 TIV/IUS/SV Interface Stations.

4.3.2 Payload Fairing

Titan IV/IUS configuration uses a 56-foot long fairing. Also the TIV/IUS has the capability of using a 10.5-foot diameter by 40-foot fairing if customer space requirements are satisfied. Reference Figure 4.3.2-1 IUS/PLF Mechanical Interfaces, and Figure 4.3.2-2 IUS PLF Dynamic Envelope.

4.3.2.1 Compartment Venting

Venting of the SV, IUS, and PLF shall be through the vents located in the aft region cylindrical portion of PLF. Payload Fairing internal pressure decay during ascent shall not exceed 0.4 psi/sec, except for a perturbation not exceeding 8 sec in duration where the rate shall not exceed 0.5 psi/sec.

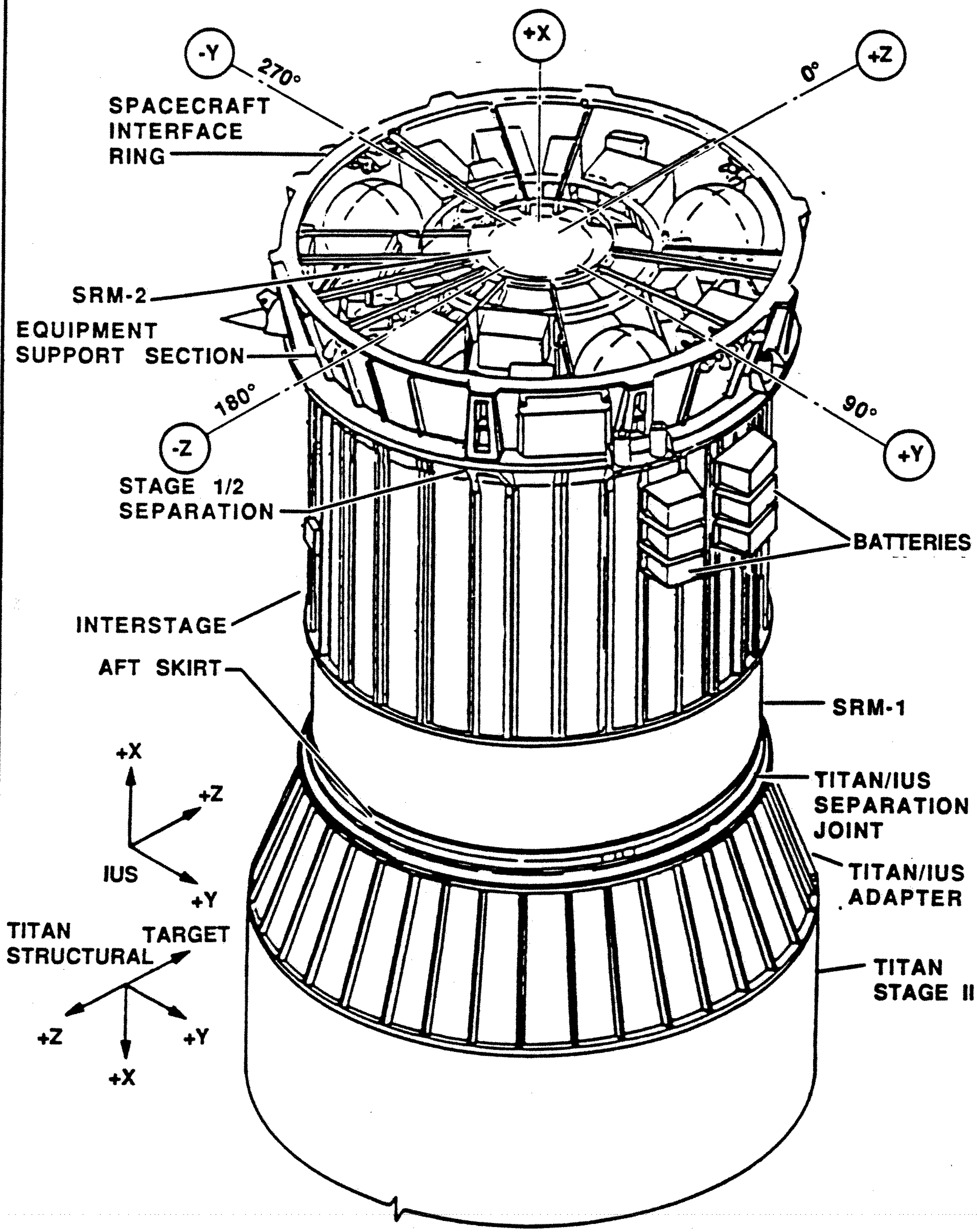


FIGURE 4.3.1.1-1 TITAN IV/IUS VEHICLE ASSEMBLY

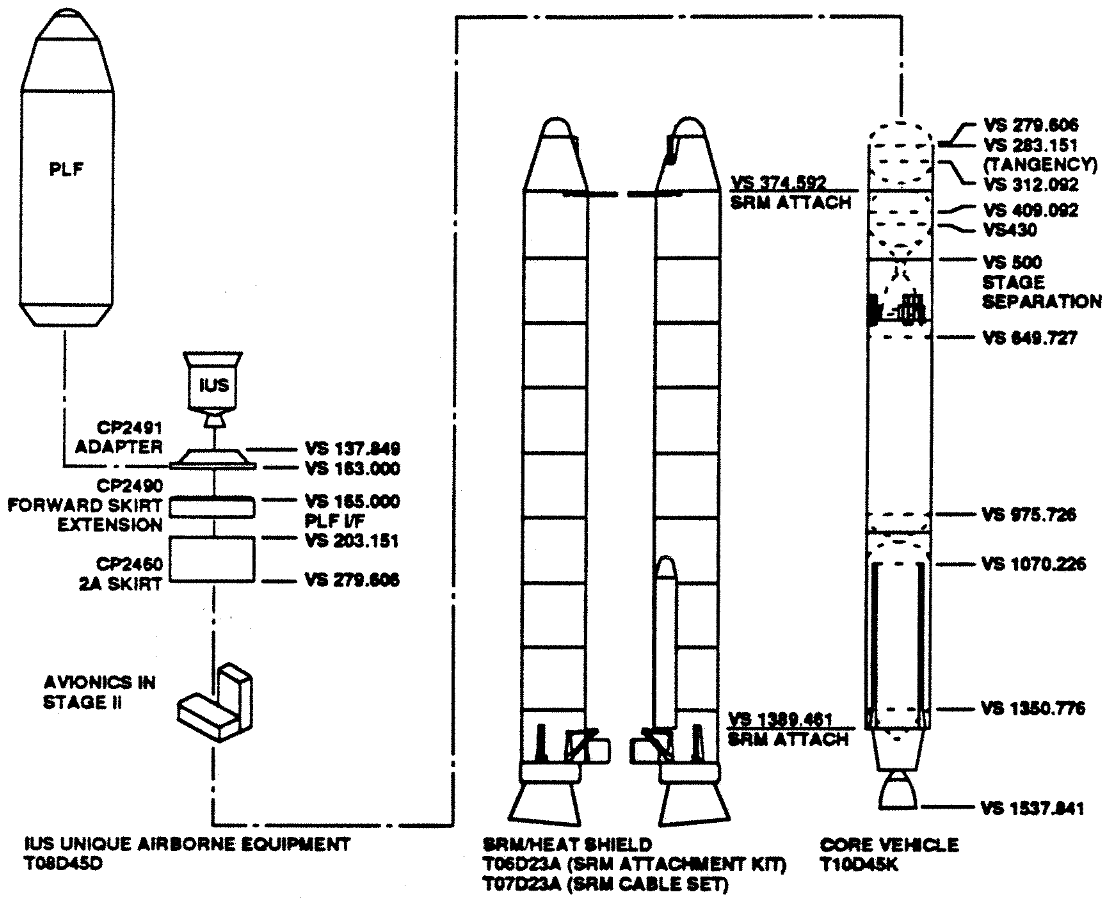
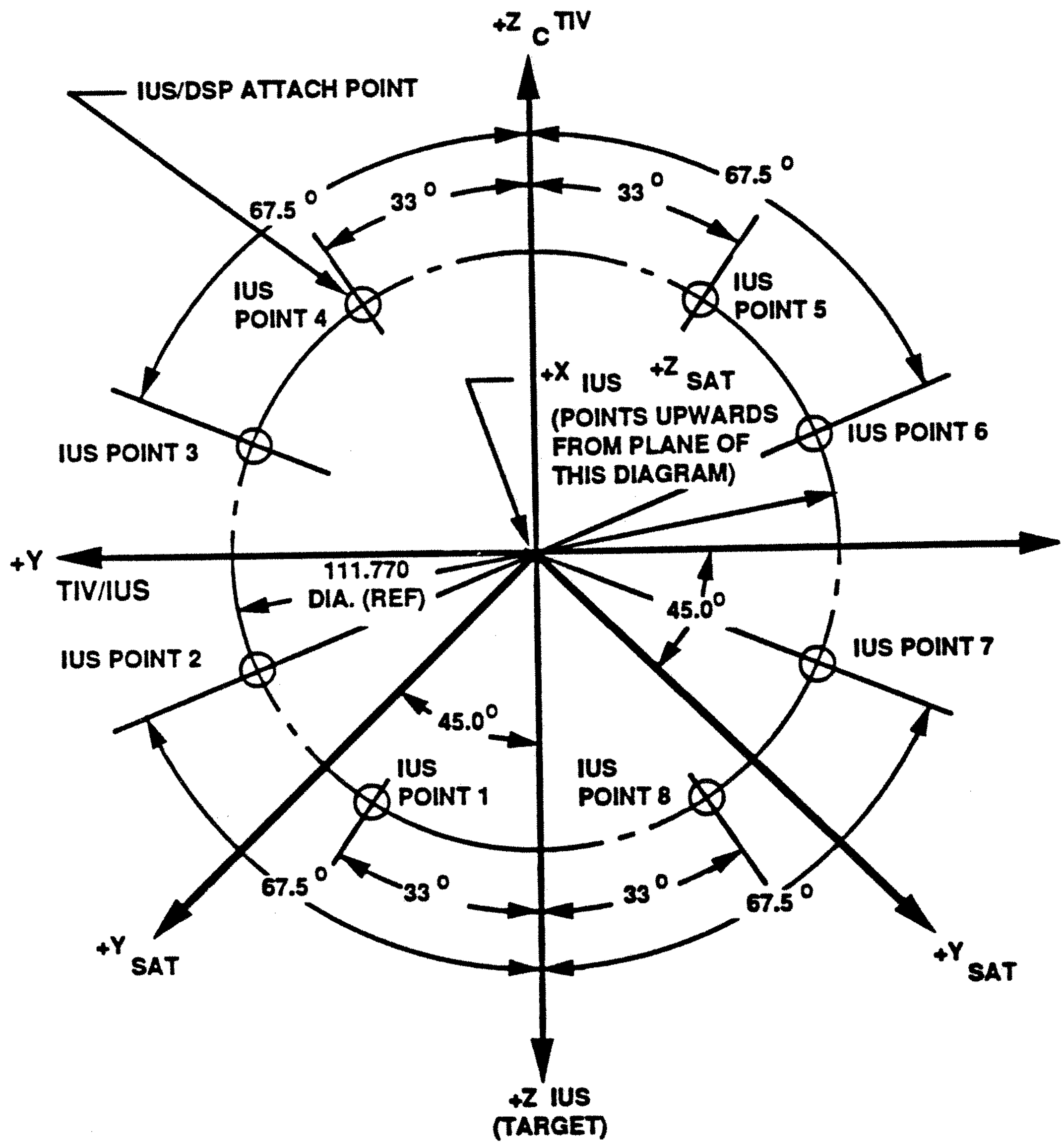


FIGURE 4.3.1.1-2 CI RELATIONSHIP WITH TITAN IV/IUS LAUNCH VEHICLE



VIEW LOOKING AFT TOWARDS IUS

FIGURE 4.3.1.2-1 DSP/IUS/TITAN IV COORDINATE INTERFACE ORIENTATION

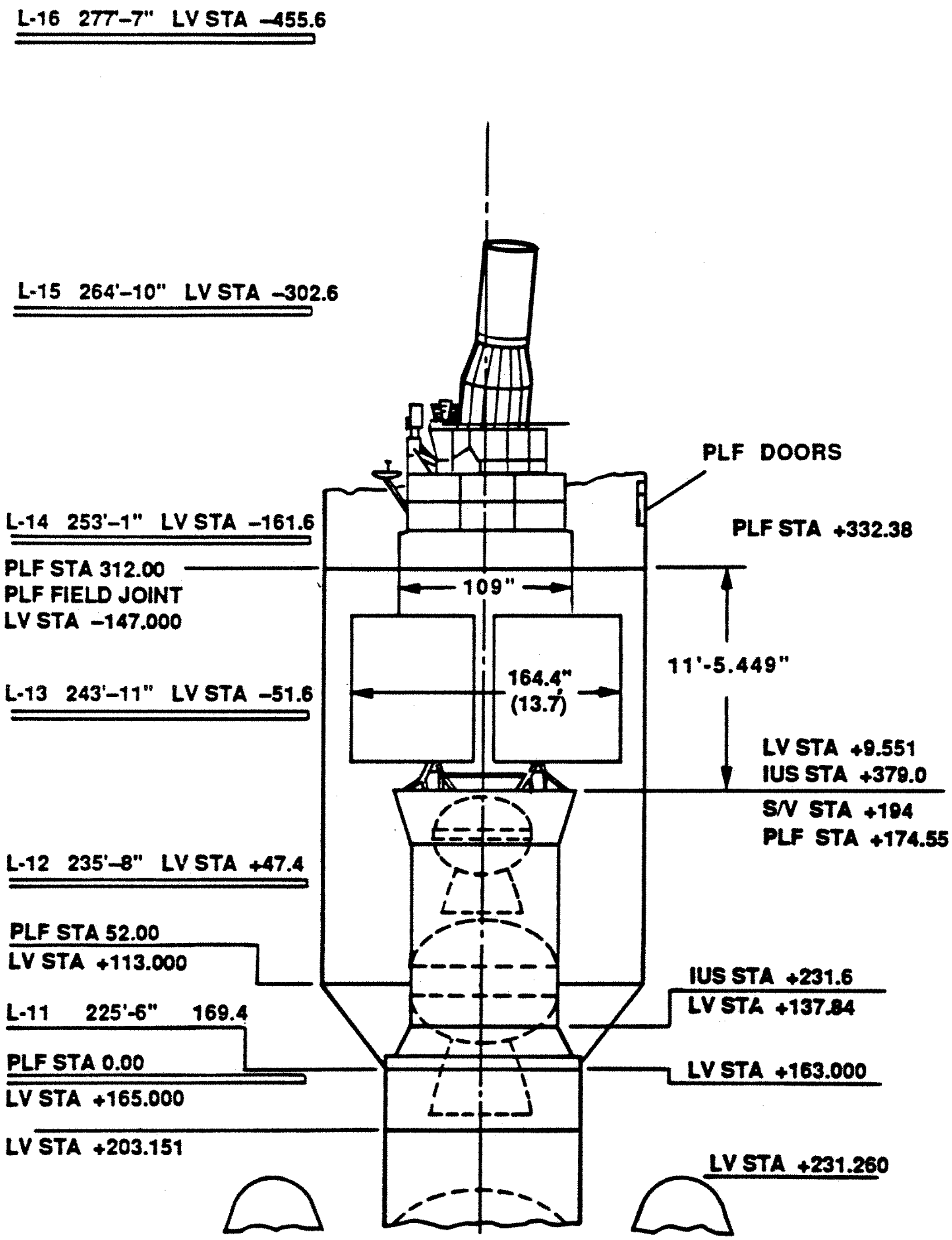


FIGURE 4.3.1.2-2 TITAN IV/IUS/SV INTERFACE STATIONS

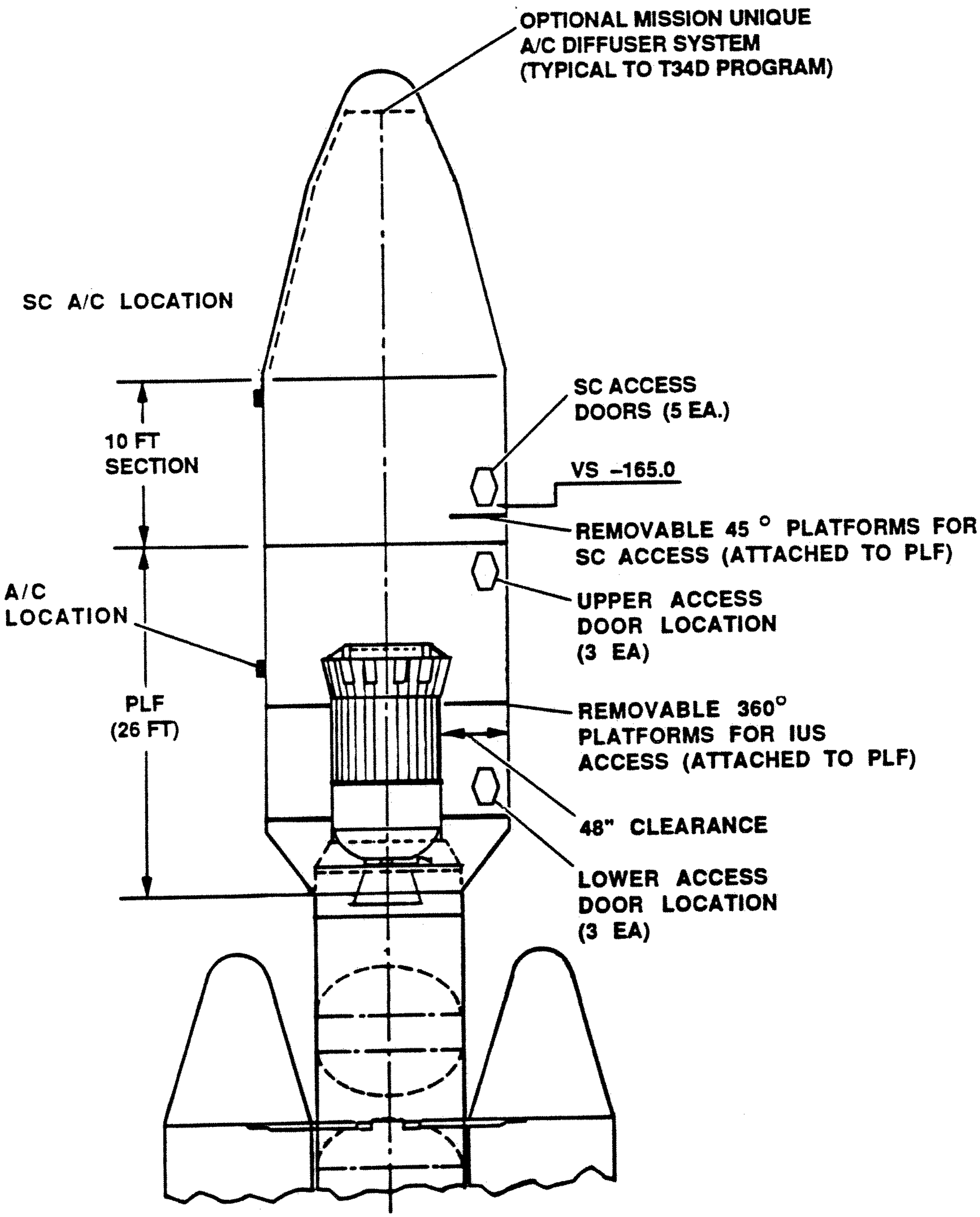


FIGURE 4.3.2-1 IUS/PLF CONFIGURATION

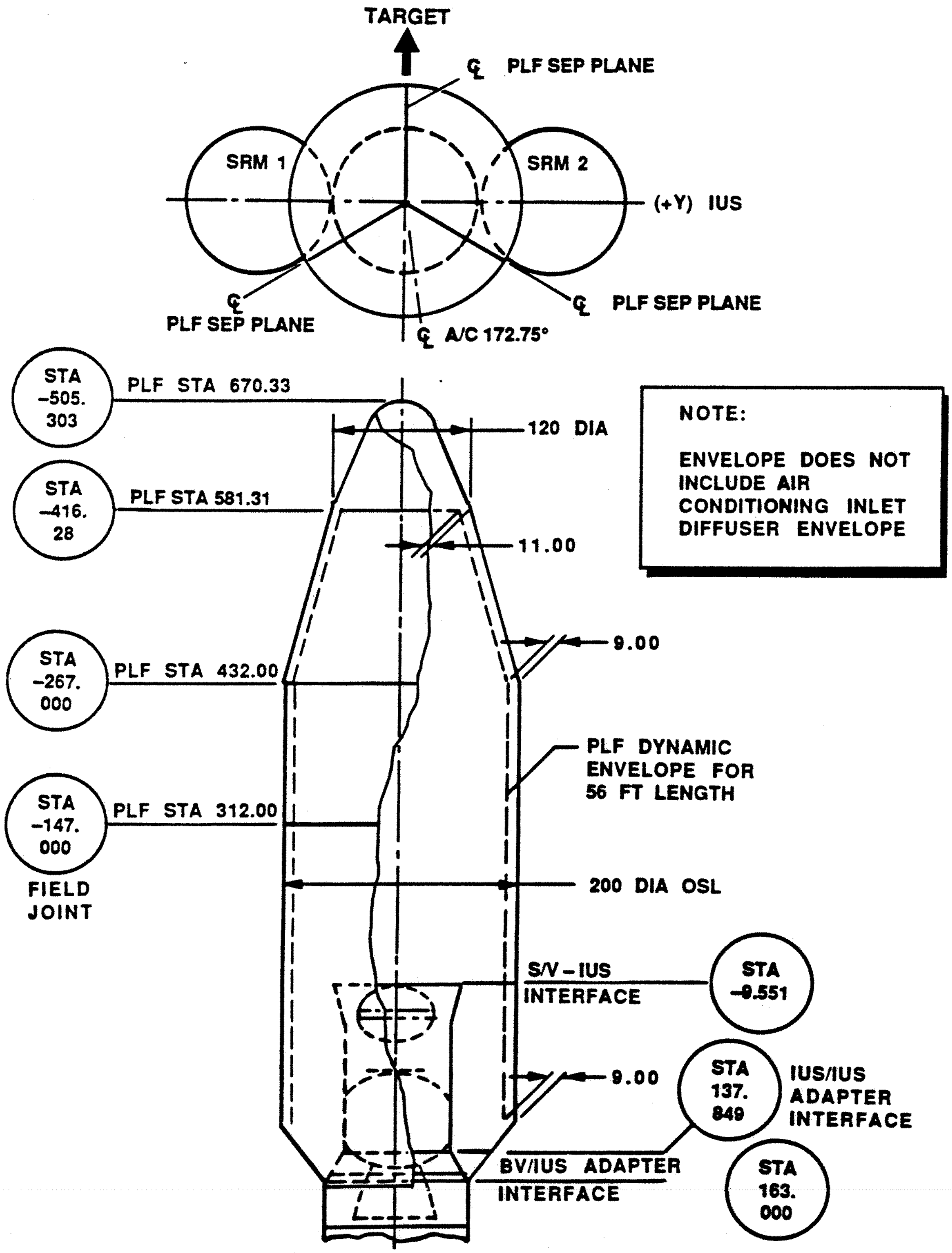


FIGURE 4.3.2-2 IUS PAYLOAD FAIRING DYNAMIC ENVELOPE

4.3.3 Electrical Systems

Specific payload electrical interface definitions with the IUS are established and documented between the IUS contractor and SPO and the Payload Integration Contractor.

- ICD-TIV/IUS-10105 – Airborne Electrical and Instrumentation I/Fs
- ICD-TIV/IUS-10106 – Ground Electrical and Instrumentation I/Fs
- ICD-TIV/IUS 10101 – TIV/IUS Avionics and Electrical Parametric I/Fs

Reference Figure 4.3.3-1 IUS Avionics System Block Diagram.

4.3.3.1 Attitude Control

Final pointing attitudes at payload separation are achieved by using the IUS Reaction Control/Attitude Control System. Mission-unique software controls the velocity increments and the reaction/attitude control switching logic to obtain the desired combination of velocity differential, pointing accuracies, attitude rates, and convergence times. The IUS typically uses a ± 0.5-degree attitude pointing deadband, and a ± 0.1-degree per second rate deadband in pitch and yaw, and a ± 0.3-deg/sec rate deadband in roll. These deadbands may be adjusted to satisfy spacecraft requirements.

The IUS can perform drift and orbital placement maneuvers for multiple spacecraft. The Spacecraft can be separated in park, transfer, or final orbit. A separation velocity difference between multiple spacecraft can be provided. The incremental velocities are achieved by using the aft-pointing, pitch and yaw, attitude control engines.

The IUS can be maneuvered away from the SC after SC separation by pre-programmed routines for the RCS. The collision avoidance maneuvers place the IUS in an orbit with an inclination and period that prevents recontact with the SC. These maneuvers are performed in a manner such that the RCS/ME exhaust products do not contaminate any SC surfaces.

4.3.3.2 Signal Interfaces

4.3.3.2.1 Telemetry

The IUS transmits telemetry data continuously from liftoff through loss of battery power following IUS/SC separation using either a 16 Kbps or a 64 Kbps transmission rate. Up to 4 Kbps of SC telemetry may be included in IUS telemetry at the 64 Kbps rate and up to 1 Kbps at the 16 Kbps rate. During a dipout, the IUS may be oriented in any direction for as long as required. No special telemetry maneuvers are required by the IUS, with the exception that the roll angle during the SRM2 burn should be chosen to point the Medium Gain Antenna at Earth center.

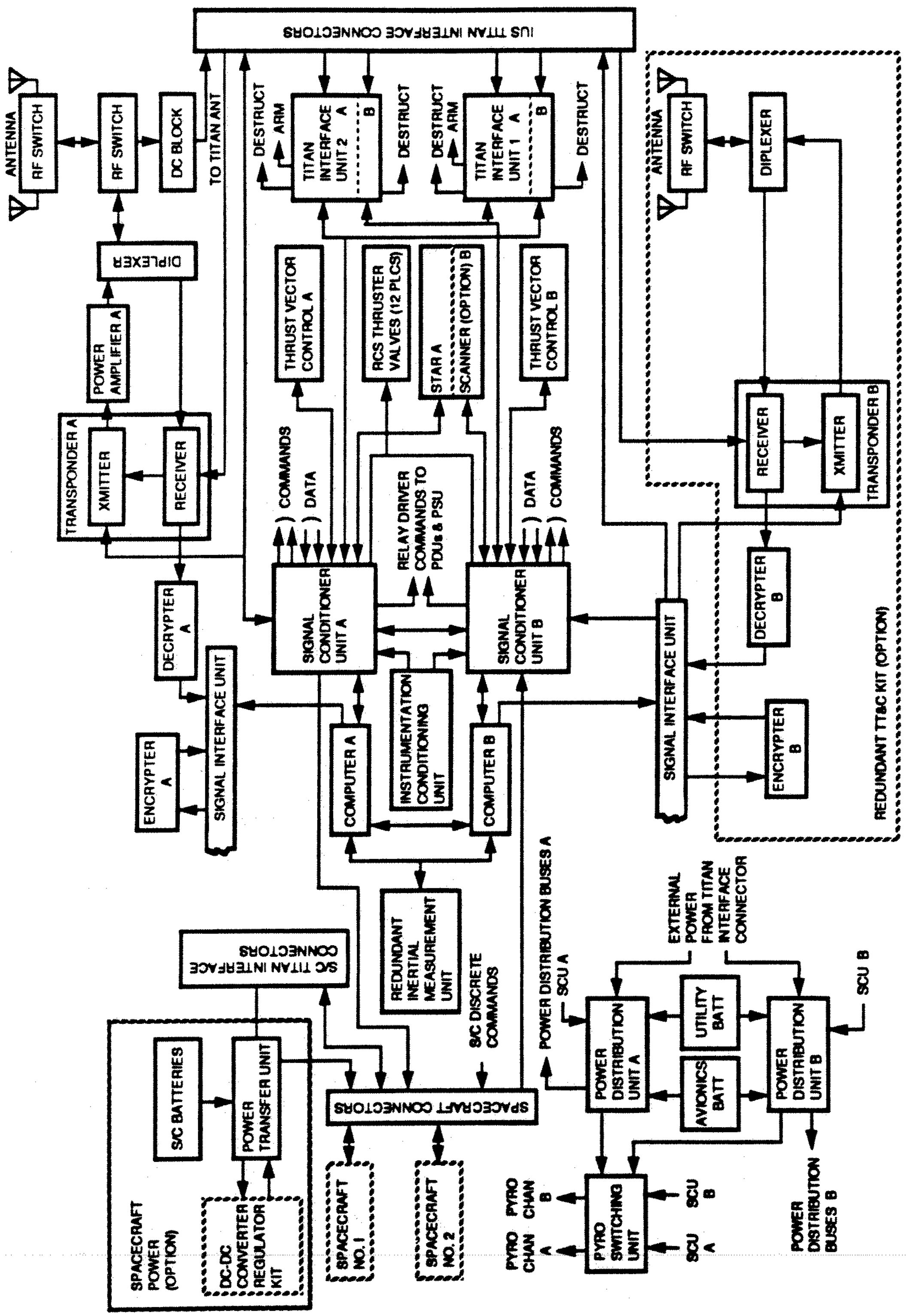


FIGURE 4.3.3-1 IUS AVIONICS SYSTEM BLOCK DIAGRAM

4.3.3.2.2 Commands

Although designed as an autonomous vehicle, the IUS is capable of being commanded in-flight via uplink commands through a remote tracking station. These commands can be used to alter the in-flight sequencing in order to respond to SC contingencies, to request SC discretes be issued at unplanned times, or to request data dumps from the IUS computers.

After completion of a successful boost phase and injection into park orbit, the Titan IV Missile Guidance Computer (MGC) issues the sequenced staging commands. After Sustaining Engine Cutoff (SECO), Titan IV safes both the Titan IV and the IUS Destruct Systems, initiates the Titan IV Retrorockets, and fires the IUS Super*Zip to separate the IUS/SC from the Titan IV Stage II. The destruct safing commands are not issued to the IUS if the Titan IV-calculated park orbit radius of perigee and angular momentum do not ensure at least one park orbit revolution.

4.3.3.2.3 Electrical Subsystem

The IUS includes provisions for a SV battery option from each of the IUS stages. In addition to batteries, the SV power options include a Power Transfer Unit (PTU) and a Converter/Regulator Unit (CRU). The DC-DC CRU provides up to 600 watts of regulated power for the SV. Reference 4.3.3.3-1 Electrical Power Busses Functional Diagram which includes the SV power path via umbilical 2A3E and Figure 4.3.3-1 IUS Avionics System Block Diagram.

The Titan IV/IUS umbilical connectors 2A3E and 2A4E have positions which are reserved to accommodate SV peculiar modifications. Also some SV peculiar wiring functions may be accommodated in the Titan IV core umbilical 2A2E as specified in the SV ICD.

Conducted interference is minimized by supplying power from the separate utility battery source to the TVC motors, RCS valves, motor-driven power transfer switches and all ordnance devices.

Reference Figure 4.5.3.3-2 IUS/NUS CCAFS 2A3E Umbilical Cables and Figure 3.6.4-2 IUS Unique Airborne Equipment.

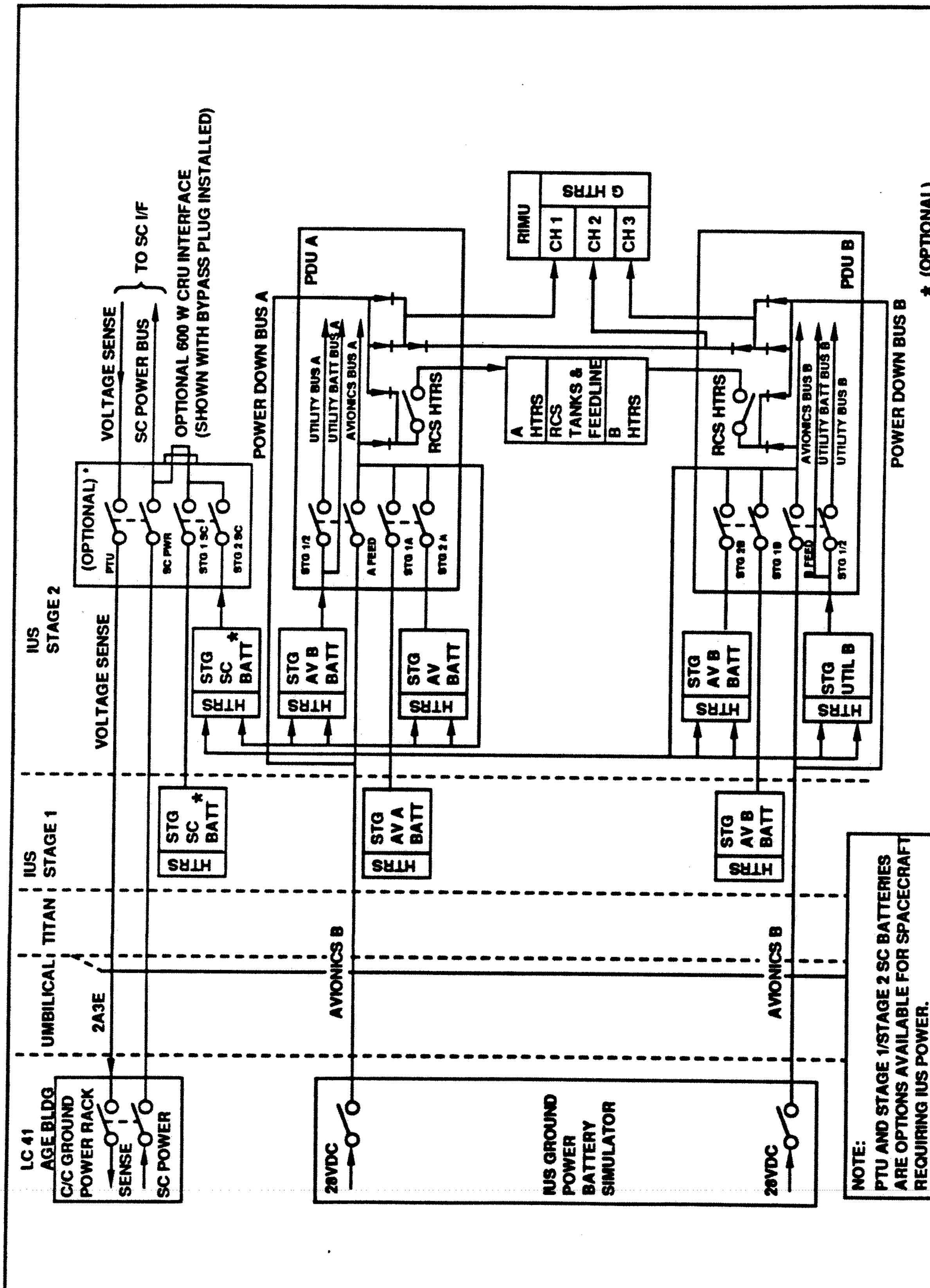


FIGURE 4.3.3.3-1 ELECTRICAL POWER BUSSES FUNCTIONAL DIAGRAM

4.4 No Upper Stage (NUS) SS-ELV-403 (WSMC)

Reference Figure 4.4-1 Titan IV/NUS System Specification Tree.and Figure 4.4-2 Titan IV NUS 403 Configuration.

4.4.1 Structural

4.4.1.1 Spacecraft Mechanical Interface

The Titan IV/NUS 403 is designed to be capable of mounting Satellite Vehicles/Satellite Vehicle adapters at the forward interface (VS 163.00). The interface is through a tension joint which uses 72-3/8-inch diameter bolts to attach the Forward Extension Skirt CP2490 to the NUS 403 Adapter.

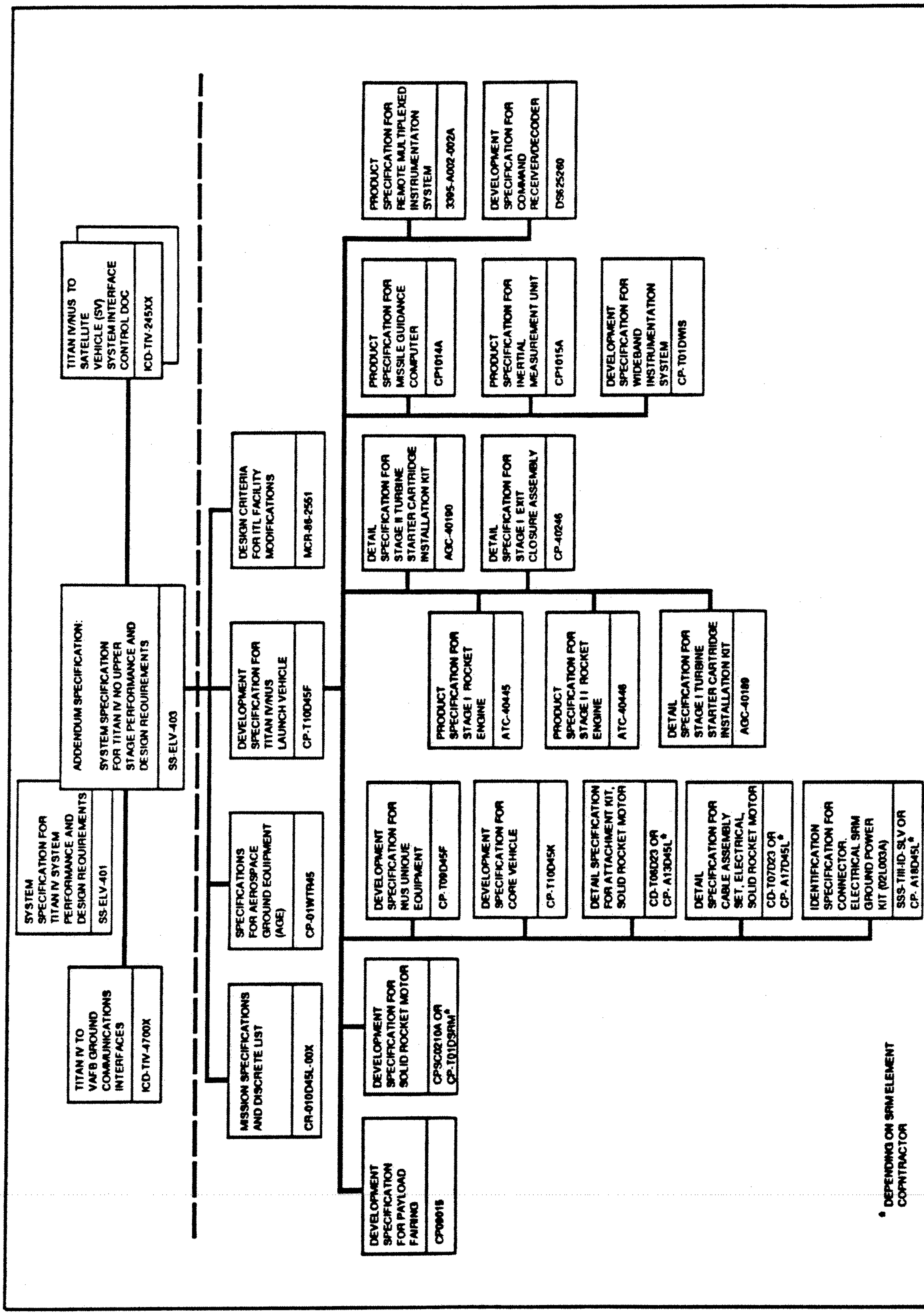
Spacecraft unique payload adapters are not part of the TIV/NUS System. The adapter between the Spacecraft and Titan IV is provided as Government Furnished Property. Reference Figure 4.4.1.1-1 CI Relationship with TIV/NUS 403, Figure 4.4.1.1-2 Launch Vehicle to Space Vehicle Interface, Figure 4.4.1.1-3 NUS 403 Adapter/PLF/2490 Details, Figure 4.4.1.1-4 Titan IV System 403 NUS Forward Skirt Extension CP2490, and Figure 4.4.1.1-5 SV/NUS Structural Interface.

The Spacecraft Contractor and Martin Marietta will exchange design data which is to include structural drawings and computer models in the form of MSC/NASTRAN bulk data card images so that both parties can develop models and analyze the interface. The Spacecraft Contractor will design his structure such that all margins of safety are positive on both sides of the interface without Martin Marietta hardware changes.

Specific Titan IV/NUS spacecraft interface agreements are contained in the mission unique Titan IV/NUS - Spacecraft ICDs.

4.4.1.2 Alignments

The design of the Stage II Forward Oxidizer Skirt and Forward Skirt Extension is such that when combined with the Common Core portion of Stage II, the actual centerline of Stage II at VS 163.0000 with reference to VS1389.461 does not deviate from the theoretical centerline by more than 0.820 inches. (The theoretical centerline is defined as a vertical line located at the geometric center of the SRM to common Core ball sockets.) This tolerance applies to a dry vehicle in the vertical position. Reference Figures 4.4.1.2-1 and 4.4.1.2-2.



* DEPENDING ON SRM ELEMENT COPNTRACTOR

FIGURE 4.4-1 TITAN IV/NUS SYSTEM SPECIFICATION TREE

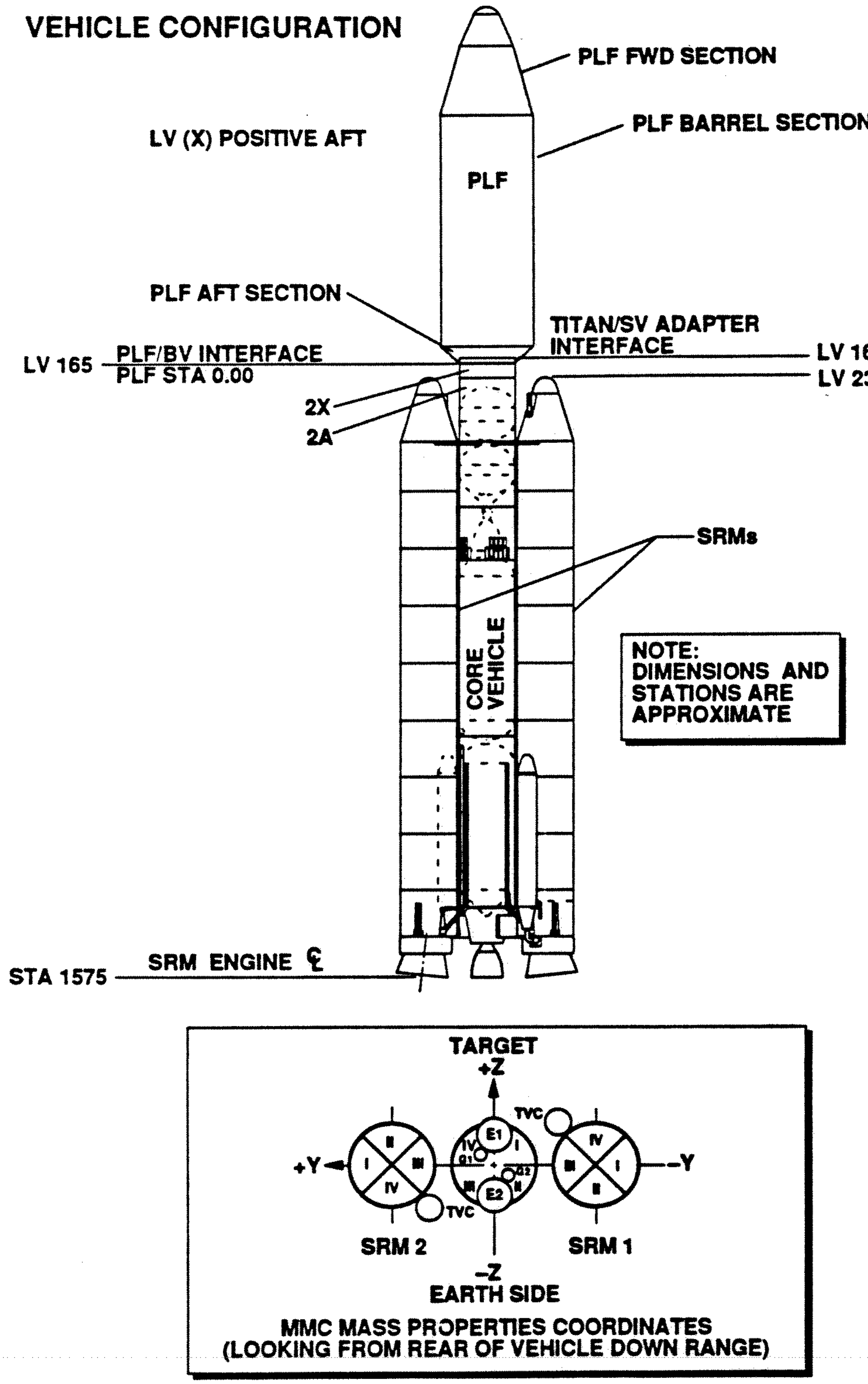


FIGURE 4.4-2 TITAN IV/NUS 403 CONFIGURATION

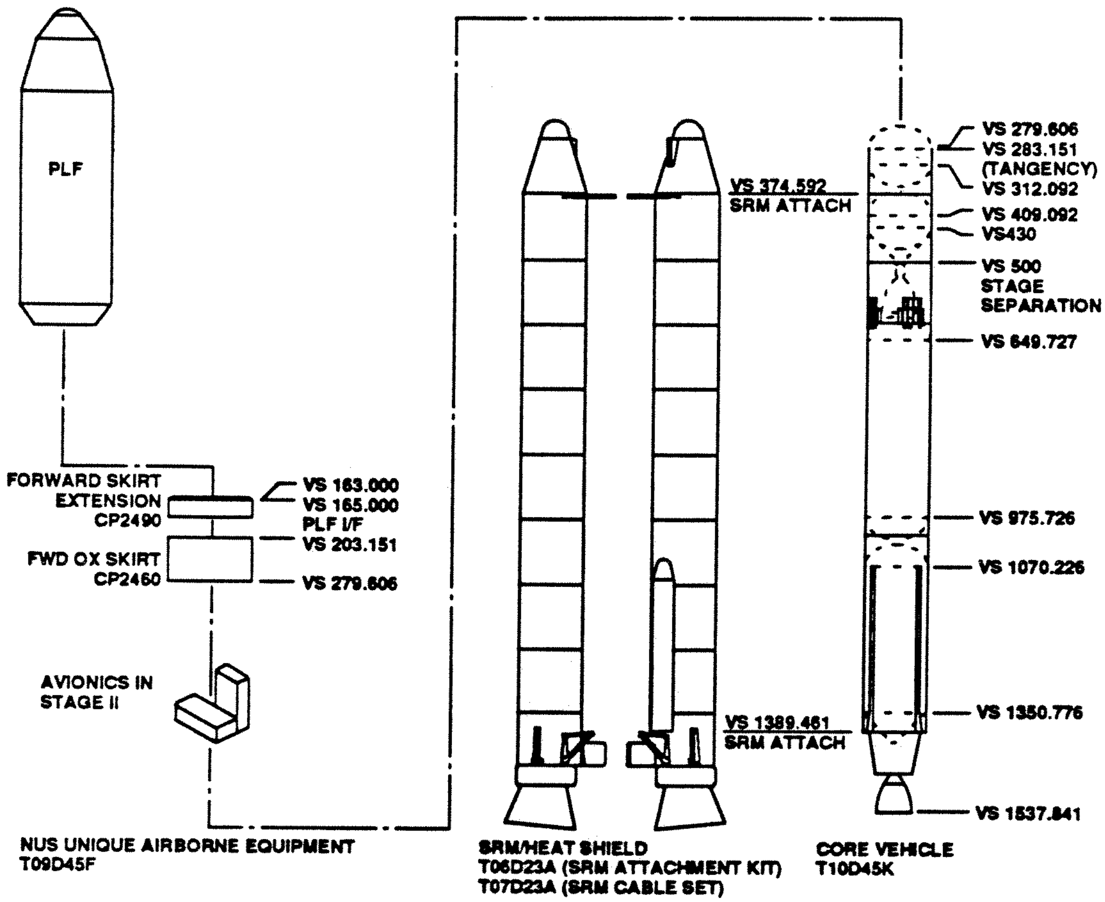


FIGURE 4.4.1.1-1 CI RELATIONSHIP WITH TITAN IV/NUS 403 LAUNCH VEHICLE

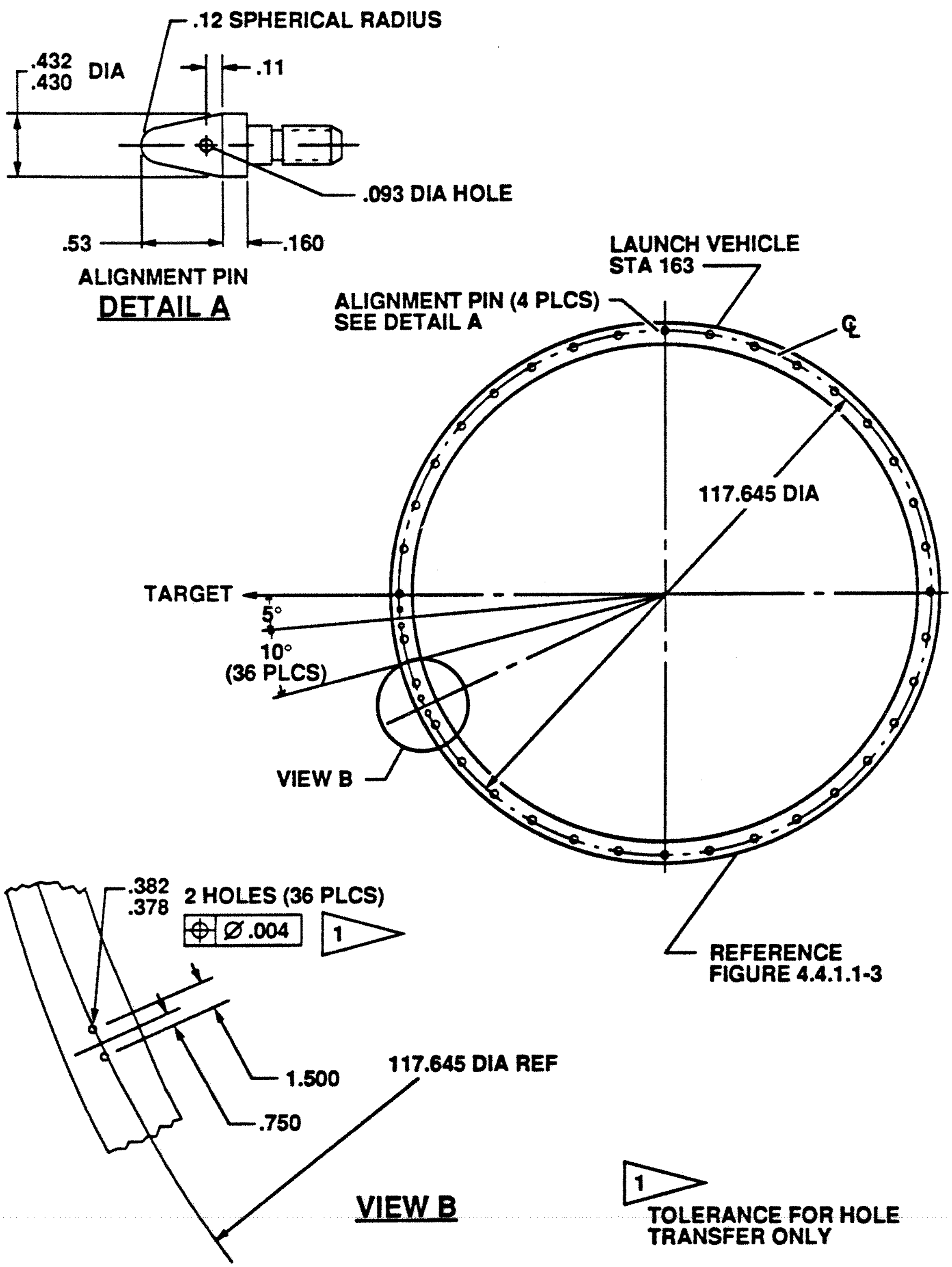
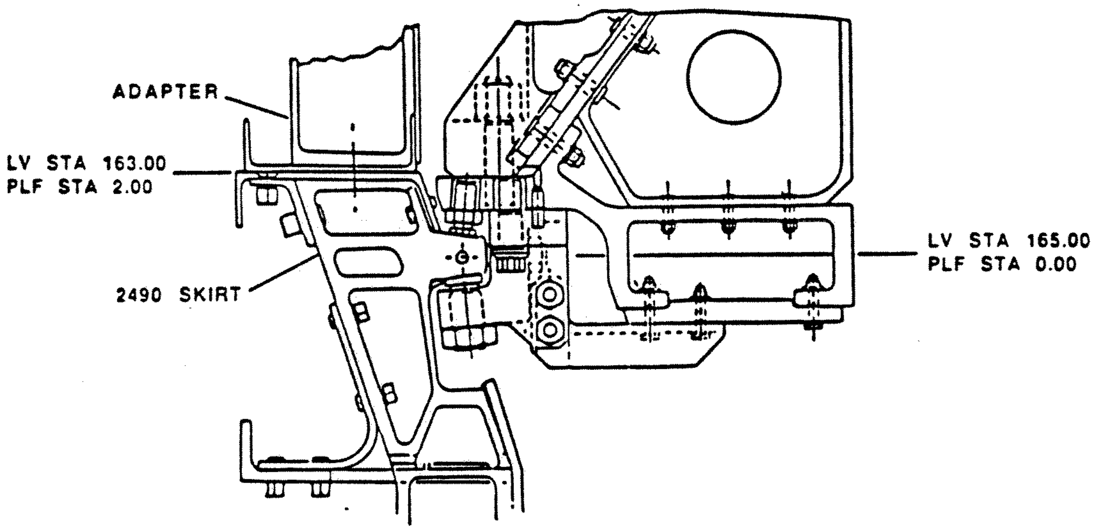
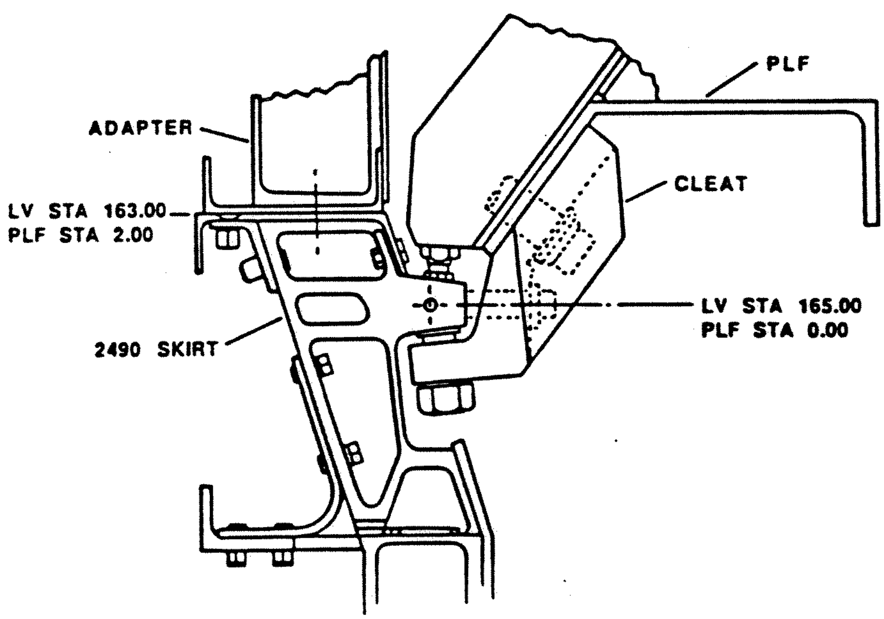


FIGURE 4.4.1.1-2 LAUNCH VEHICLE TO SV INTERFACE (403)

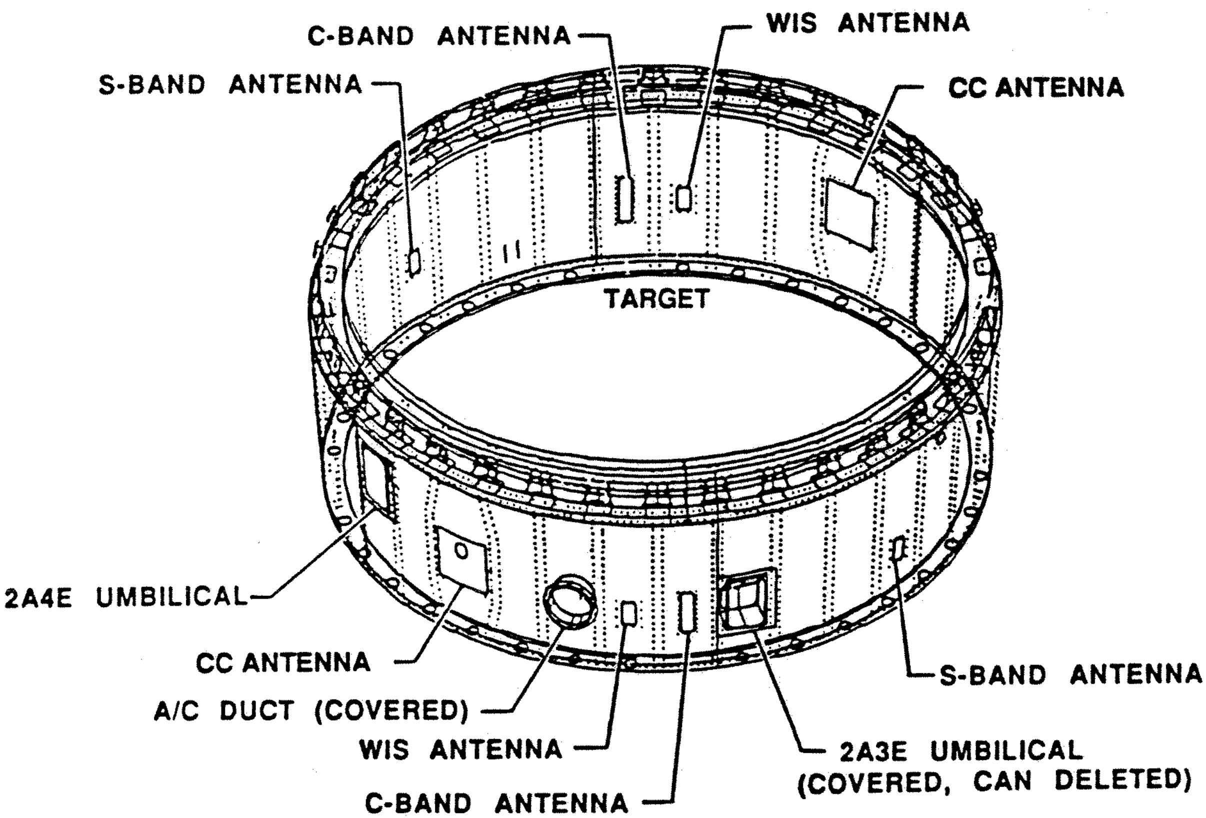


STIFF BOATTAIL CONFIGURATION



SOFT BOATTAIL CONFIGURATION

FIGURE 4.4.1.1-3 NUS 403 ADAPTER/PLF/2490 SKIRT INTERFACE DETAILS



NOTE: UMBILICALS
2A1E & 2A2E
ARE LOCATED
ON CP2460
(COMPARTMENT 2A)

FIGURE 4.4.1.1-4 TITAN IV SYSTEM 403 NUS FORWARD SKIRT EXTENSION–CP2490

COMPLIANCE STATUS

BV I/F DESIGN ASSUMES SV I/F SATISFIES THE FOLLOWING REQUIREMENTS:

- CROSS SECTIONAL AREA ≥ 1.18 IN.^{2}
- MOMENT OF INERTIA ≥ 2.59 IN.^{4}
- PEAKING FACTOR AT I/F ≤ 1.20

PURPOSE IS TO MORE EFFICIENTLY SHARE REACTION OF THE PLF KICK LOAD AT VS 165.0

PURPOSE IS TO ASSURE EFFICIENT STRUCTURAL INTERACTION BETWEEN SV AND TITAN IV/NUS STRUCTURE

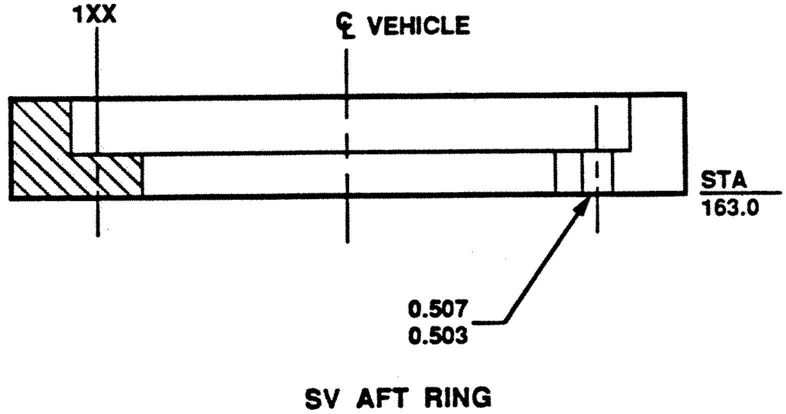


FIGURE 4.4.1.1-5 SV/NUS STRUCTURAL INTERFACE (403)

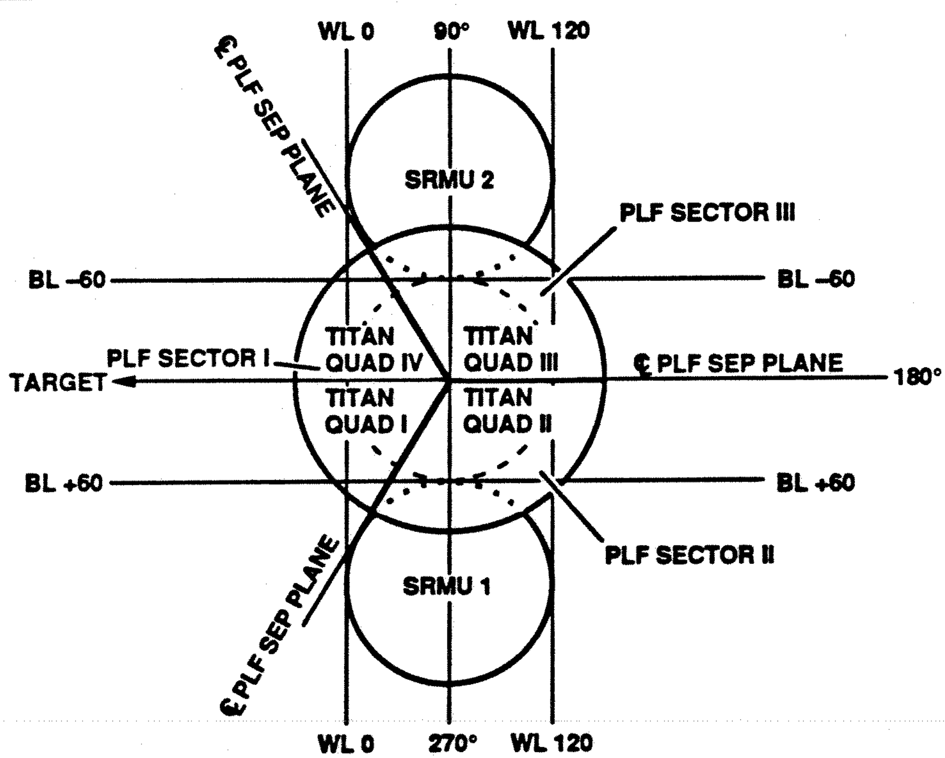


FIGURE 4.4.1.2-1 TITAN IV/NUS (403) PLAN VIEW (LOOKING TOWARD EARTH CENTER)

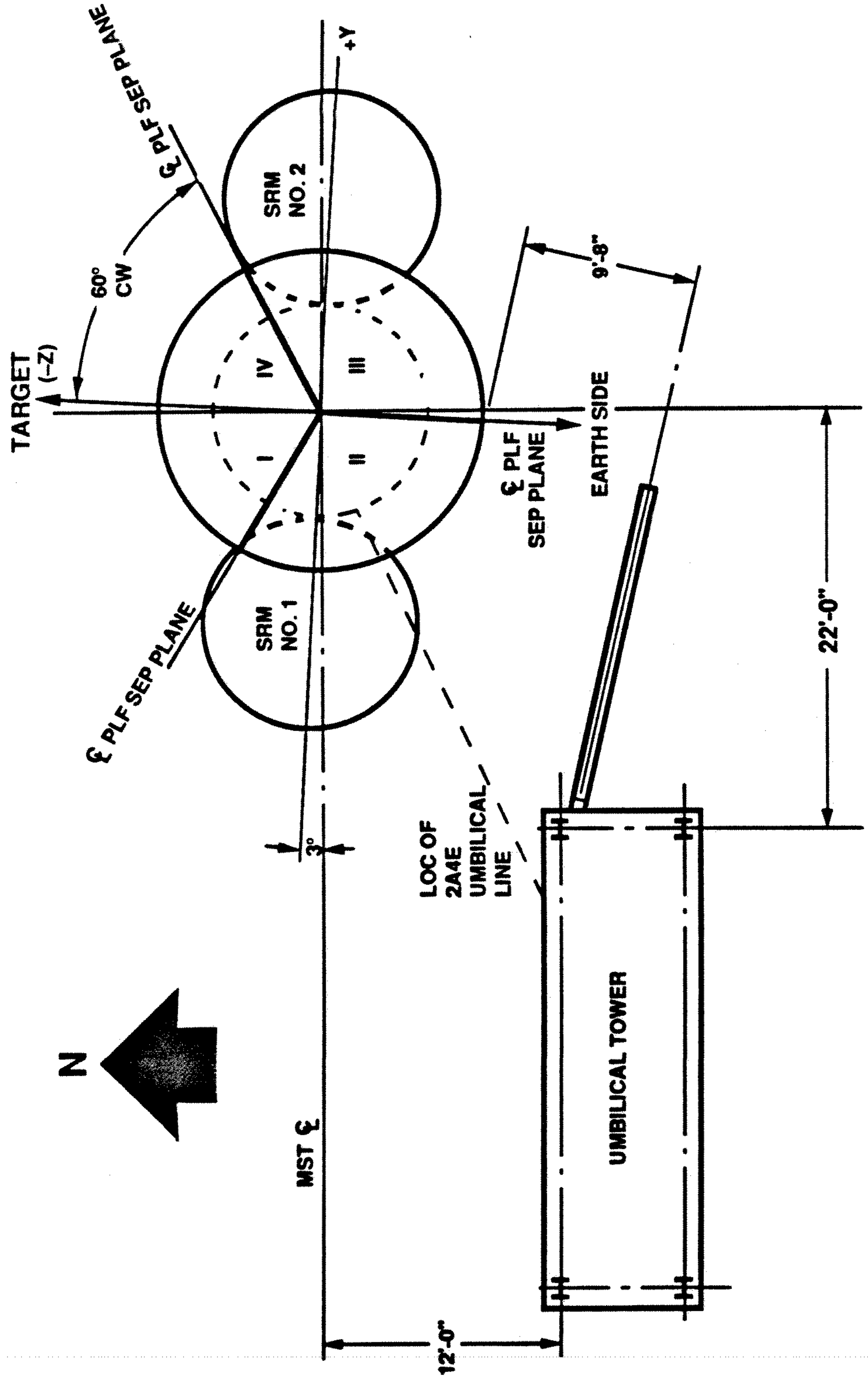


FIGURE 4.4.1.2-2 TITAN IV/NUS (403) PLAN VIEW (LOOKING TOWARD EARTH CENTER)

4.4.1.3 Azimuth Displacement

The designs of the Stage II Forward Oxidizer Skirt and the Forward Skirt Extension are such that the maximum allowable rotation around the centerline of the vehicle at VS 163.000 as referenced to VS 1389.461 does not exceed ±0.460 in. at the Outside Skin Line (OSL). This tolerance applies to a dry vehicle in the vertical position.

4.4.1.4 Mechanical Attachments

The BV shall provide for mechanical attachment of the SV as shown in Figure 4.4.1.1-2. The combined effect of all BV manufacturing tolerances shall result in less than the following misalignments at the P/L interface (VS 163.000) from the theoretical centerline. The theoretical centerline is through the launch stand SRM interface.

Lateral Translation	±1.0 in.
Angle of Tilt	±0.1° with respect to vertical
Angle of Twist	±0.5° with respect to target

4.4.1.5 Venting and Venting Loads

The structure is designed to withstand compartmental pressure loading along with vehicle flight loads. Vent Ports are provided in Compartment 2A at VS 265. The 2A Vent Area is selectable to a maximum of 260 in.^{2}. The vehicle specific 2A Vent Area is by agreement with the user and is addressed in the P/L unique ICD, reference Figure 5.2-3.

4.4.2 Payload Fairing

Reference Figure 4.4.2-1.

Venting of the SV, IUS, and PLF shall be through the vents located in the aft region cylindrical portion of PLF. PLF internal pressure decay during ascent shall not exceed 0.4 psi/sec, except for a perturbation not exceeding eight sec in duration where the rate shall not exceed 0.5 psi/sec.

4.4.3 Electrical Systems

Reference Figure 4.4.3-1.

4.4.3.1 Attitude Control

The Titan IV/NUS Flight Control System (FCS) functions to maintain the required Titan IV/NUS vehicle attitude and trajectory through all flight phases from launch to P/L separation, reference Paragraph 3.2.3.2.

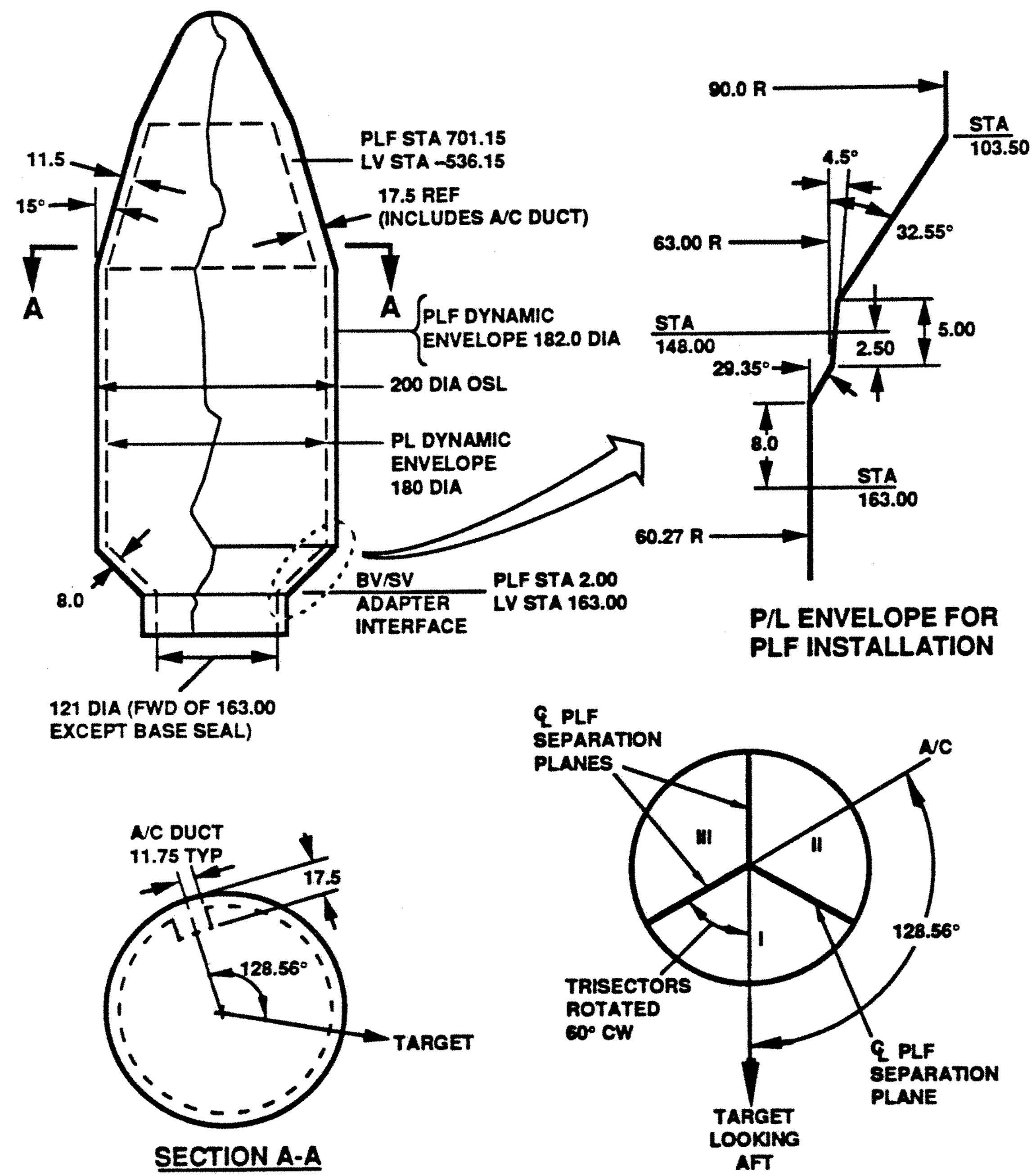


FIGURE 4.4.2-1 P/L DYNAMIC ENVELOPE FOR 403

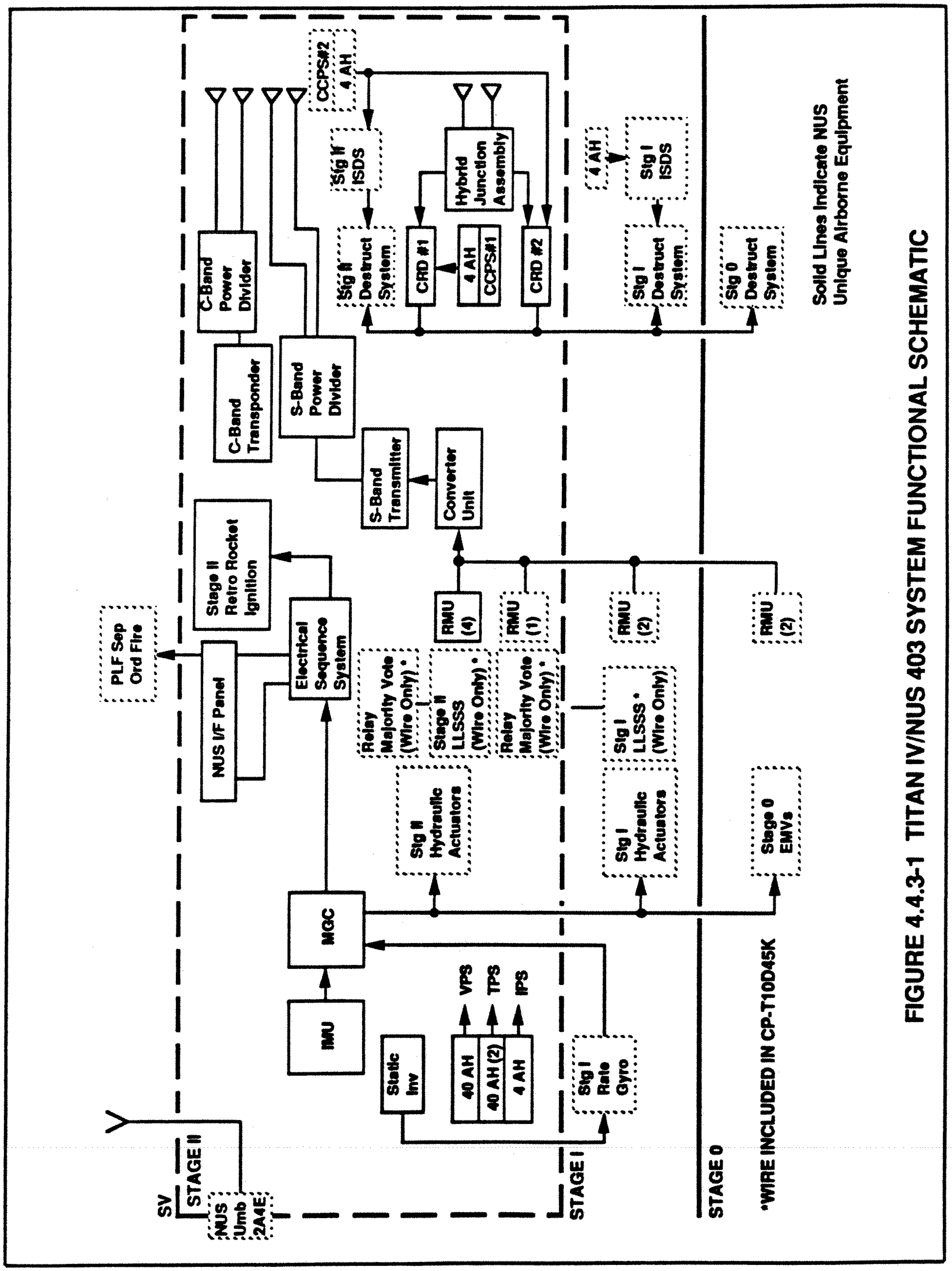


FIGURE 4.4.3-1 TITAN IV/NUS 403 SYSTEM FUNCTIONAL SCHEMATIC

4.4.3.1 Attitude Control (Continued)

Attitude command signals from the Guidance System to the FCS Actuators are attitude error signals referenced to vehicle pitch, yaw and roll axes. This is a S/W interface, programmed in the MGC S/W, which provides a flexible capability to incorporate new SC trajectory retargeting parameters, reference Figure 4.4.3.1-1.

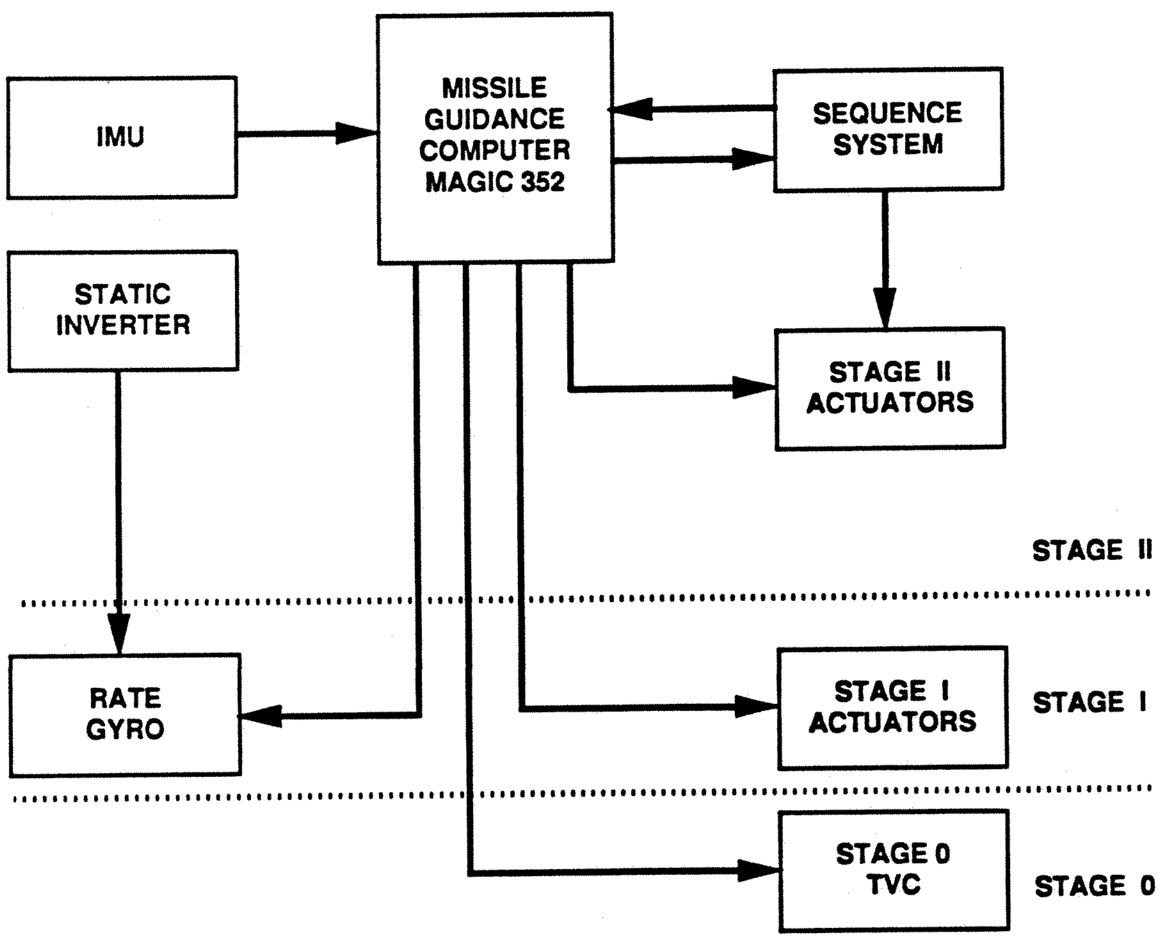


FIGURE 4.4.3.1-1 GUIDANCE AND CONTROL SYSTEM BLOCK DIAGRAM

4.4.3.2 Signal Interfaces

4.4.3.2.1 Instrumentation and Telemetry

P/L peculiar instrumentation requirements are accommodated in the Titan IV/NUS generic design by making available through a standard electrical interface panel connections for 11 SV analogs at 100 samples per sec, three SV analogs at 40 samples per sec and six SV bi-levels at 100 samples per sec.

Accommodations are provided for SV peculiar instrumentation requirements such as a Wideband Instrumentation System (WIS), diplexing of SV data through a diplexer in Titan IV, and addition of SV measurements from the SV to the Titan IV Remote Multiplexed Instrumentation System (RMIS).

The capability to install and utilize a Titan IV diplexer in the existing Titan IV Instrumentation Truss location allows reservation of the truss location for potential SV usage.

4.4.3.2.2 Commands

The electrical interfaces between the Titan IV and the SV may include Titan IV generated SV separation commands, Titan IV command destruct signals, and ISDS sense line connections.

The Titan IV/NUS sequencing system has the capability to provide staggered execution of the Stage II shutdown, Stage II retrorocket firing functions and SV separation. Also the Titan IV/NUS provides PLF separation commands.

SV separation commands per SV requirements are achieved via SV peculiar modification cabling between the Titan IV Forward Oxidizer Skirt (CP2460) standard SV Electrical Interface Panel and the SV interface. Command destruct interfaces and/or Titan IV Stage II ISDS sense line interfaces required by Range Safety for specific SVs are also SV peculiar modifications.

For SVs that have an ISDS requirement, sense lines will cross the Titan IV/NUS to the SV interface and include any SV breakup points and other requirements called for by Range Safety. For those Titan IV/NUS vehicles which fly with SVs that do not require a Titan IV Stage II ISDS, the full-up Common Core Stage II ISDS configuration will be installed with the sense line requirements satisfied by wiring at the connectors.

4.4.3.3 Electrical Subsystem

Reference Figures 4.4.3.3-1, 4.4.3.3-2, 4.4.3.3-3 and 4.4.3.3-4.

The Electrical Subsystem includes a Cable Harness which attaches to an Interconnect Harness located in compartment 2B of the Titan IV core, to the Titan IV/SV Interface Panels located on the Instrumentation Truss, to the various avionics components, and to the 2A1E and 2A2E electrical umbilicals.

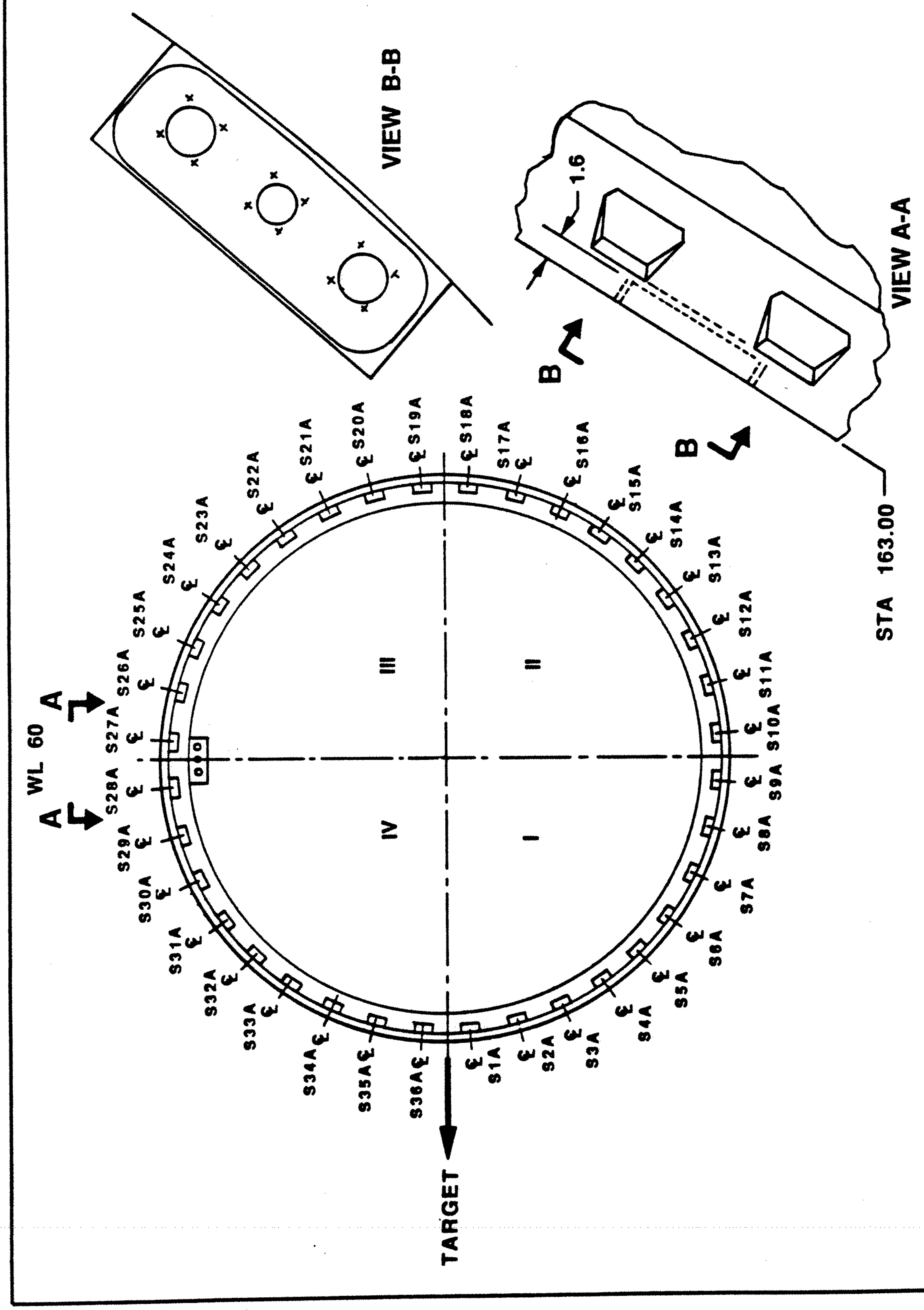
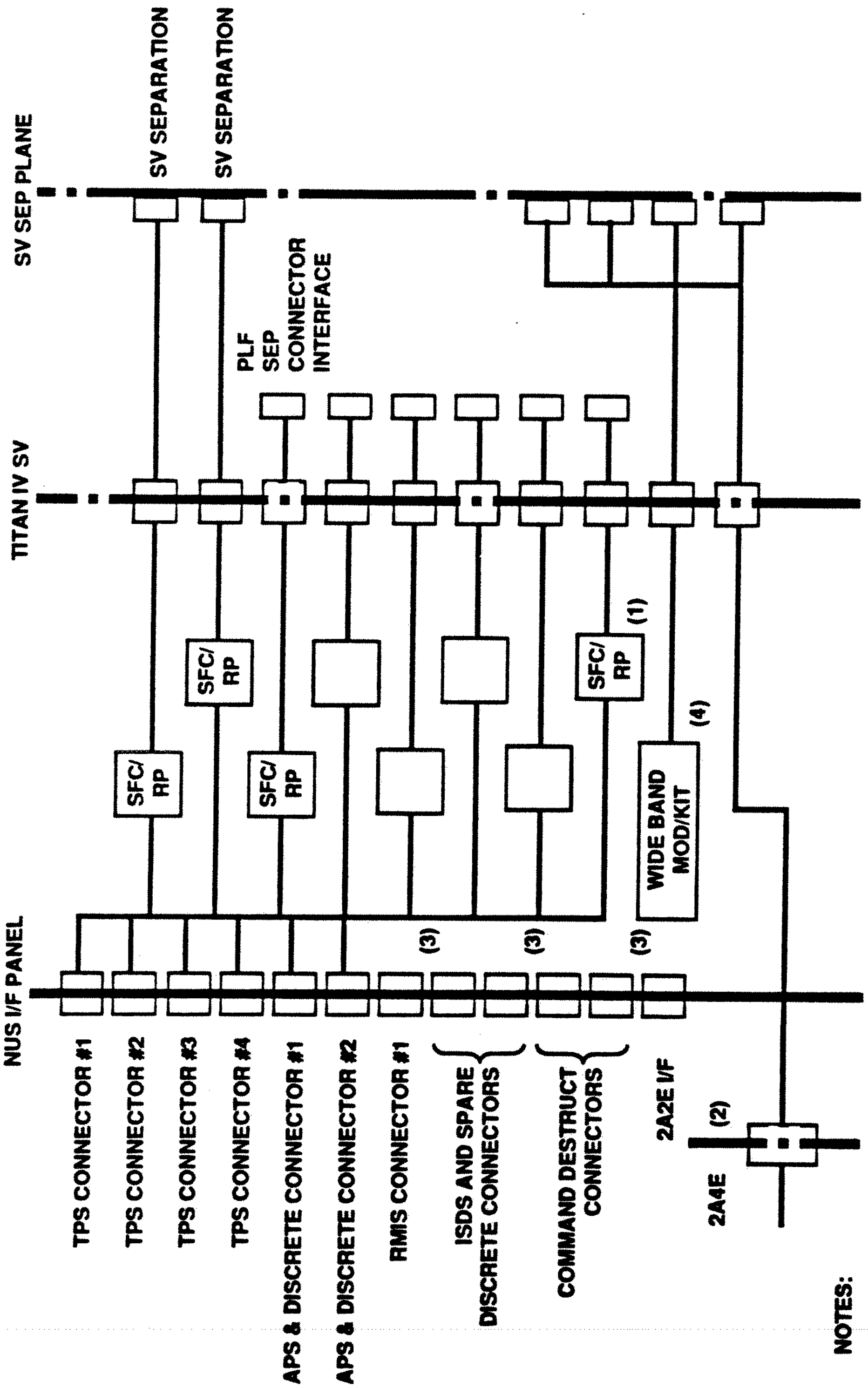


FIGURE 4.4.3.3-1 TYPICAL ELECTRICAL INTERFACE TO SV



NOTES:

(1) SFC/RP ARE SQUIB FIRING CIRCUITS AND RESISTOR PACKS FROM SV SEP, AND PLF UNLATCH AND SEPARATION
(2) 2A2E OR NEW 2A4E UMBILICAL USED AS REQUIRED
(3) CONNECTORS ARE CAPPED FOR THIS CONFIGURATION
(4) SV HAS IDENTIFIED A POTENTIAL NEED FOR WIDE-BAND SYSTEM FOR ENVIRONMENTAL DATA

FIGURE 4.4.3.3-2 TIV/NUS ELECTRICAL INTERFACE TYPICAL DESIGN

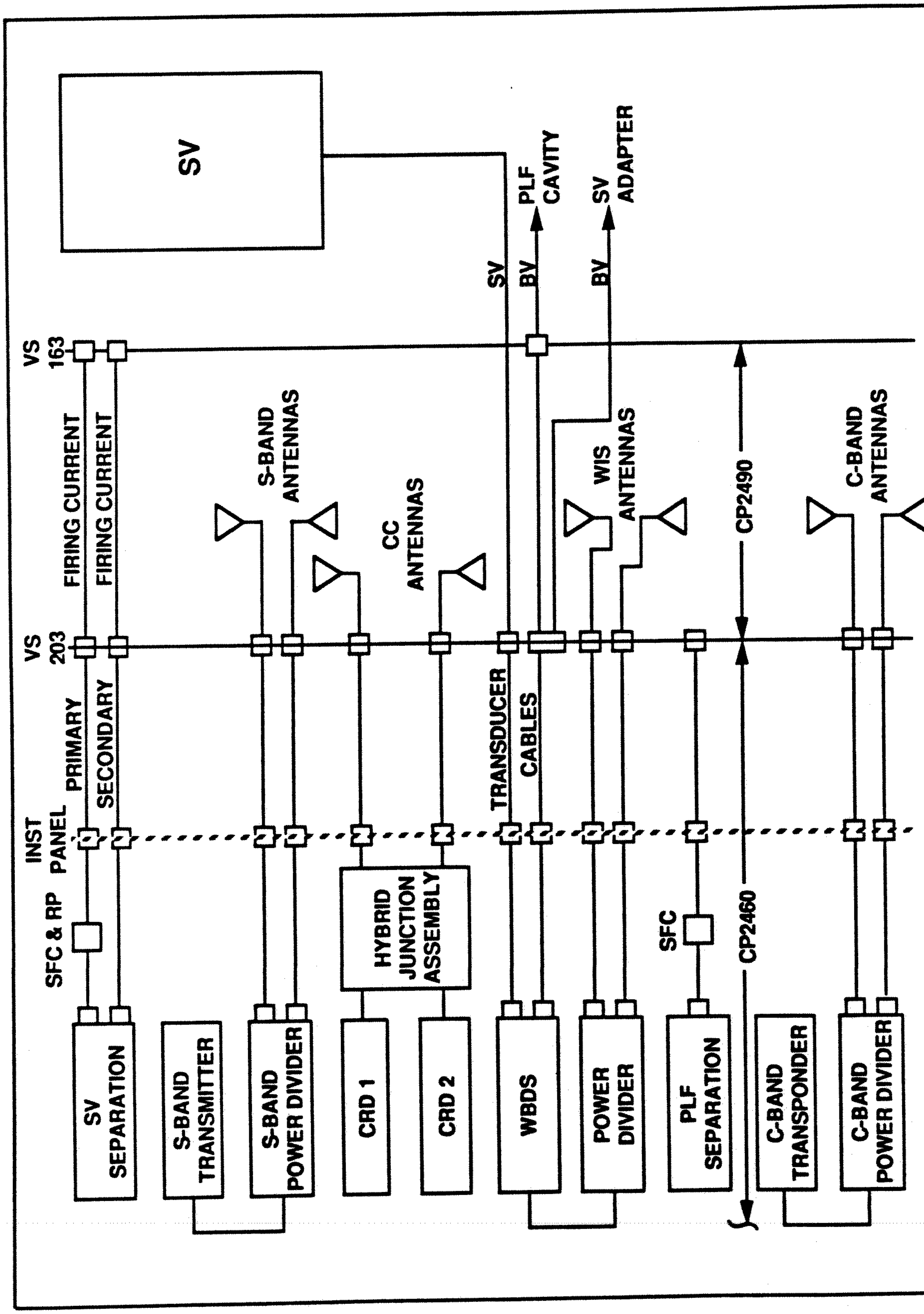


FIGURE 4.4.3.3-3 CABLING DIAGRAM TIV/NUS

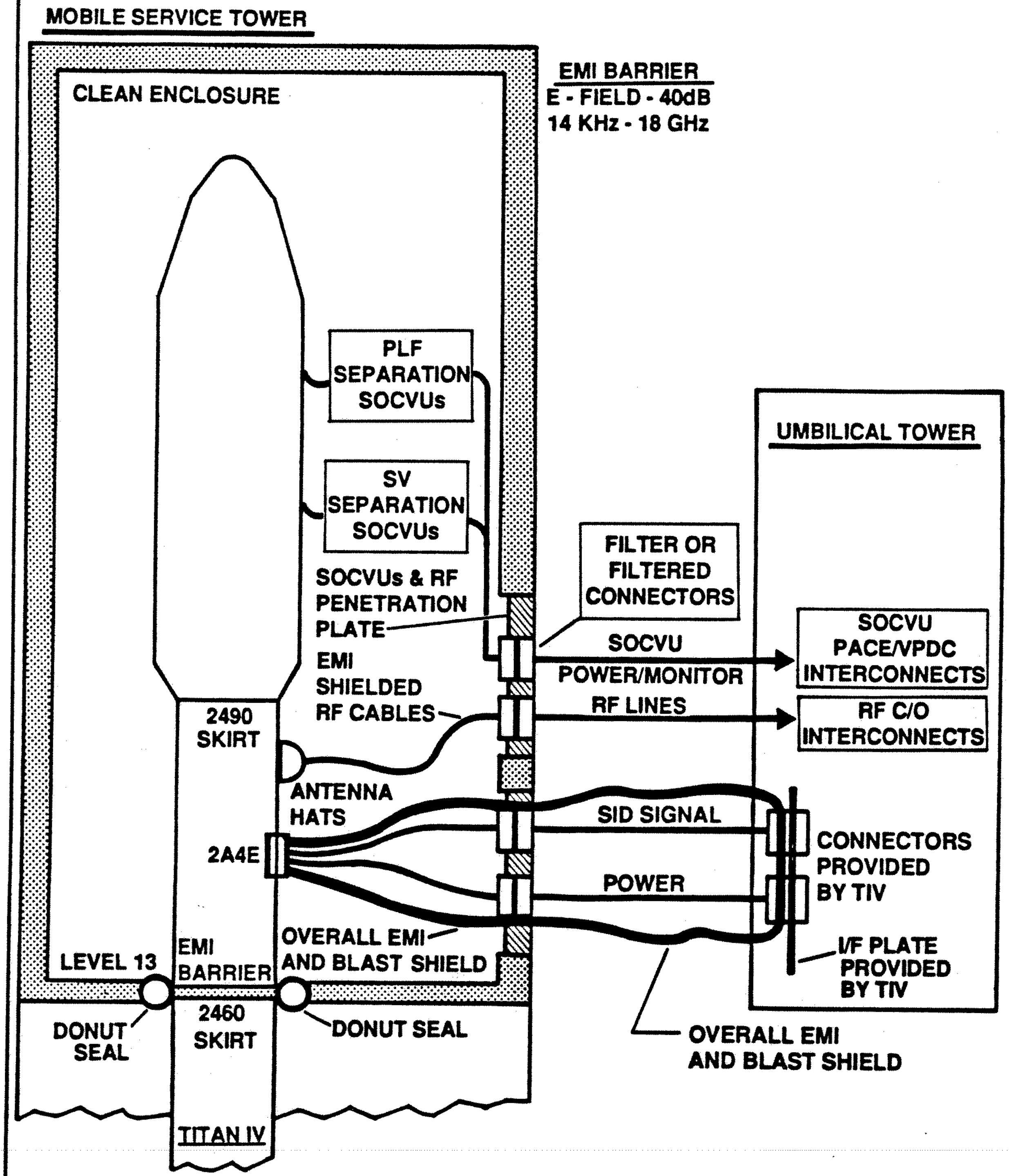


FIGURE 4.4.3.3-4 MISSION KIT UMBILICAL (WSMC)

4.4.3.3 Electrical Subsystem (Continued)

The capability exists to supply power, timing discretes, data take off, separation signals, and destruct signals to the SV via the SV interface panels.

The Titan IV/NUS standard wiring design shall be the same as the Titan IV/IUS design baseline. The specific SV/Titan IV/NUS P/L peculiar interface connector locations and orientations shall be provided in the Titan IV/NUS to SV ICDs.

Any SV requirement to route SV to Ground functions through a Titan IV/NUS Umbilical is an SV charged modification and is not part of the generic Titan IV/NUS configuration. Titan IV/NUS umbilical locations 2A2E and 2A4E have reserved positions for this requirement. Limited SV peculiar AGE wiring functions may be fitted in the Titan IV existing Umbilical 2A2E as specified in the specific SV ICD, reference Figures 3.7.3-1, 4.4.1.1-4, 4.4.1.2-2 and 4.4.3.3-2.

The Titan IV/NUS wiring shall provide a standard Interface Panel arrangement in the Stage II Truss area which brackets the requirements of several various P/Ls. Wiring, forward of the Titan IV Core 2A section standard SV Interface Panel, shall be P/L peculiar modifications.

4.4.4 Software

A software Computer Program Development Plan (CPDP), MCR-85-2504, describes the Martin Marietta standards, necessary subcontractor standards and procedures to manage, develop, control and maintain the Titan IV/NUS Boost Vehicle (BV) software and hardware.

The software CPDP defines the hardware/software integration tests for BV flight S/W and related S/W for PACE. The software validation test phase shall resolve software discrepancies discovered during interoperation tests and hardware/software acceptance testing.

4.5 No Upper Stage (NUS) SS-ELV-405 (ESMC)

Reference Figures 4.5-1 and 4.5-2.

4.5.1 Structures

4.5.1.1 Spacecraft Mechanical Interface

The SC Interface to the Titan IV/NUS configuration is through a tension joint at Titan IV VS 163.00. This joint uses 72-3/8 in. diameter bolts to attach the Forward Skirt Extension CP2490 to the Titan IV/NUS 405 Adapter, reference Figures 4.5.1.1-1, 4.5.1.1-2, 4.5.1.1-3, 4.5.1.1-4 and Table 3.2.1.2-1.

SC unique P/L adapters are not part of the Titan IV/NUS 405 System. The adapter between the SC and Titan IV is provided as GFP.

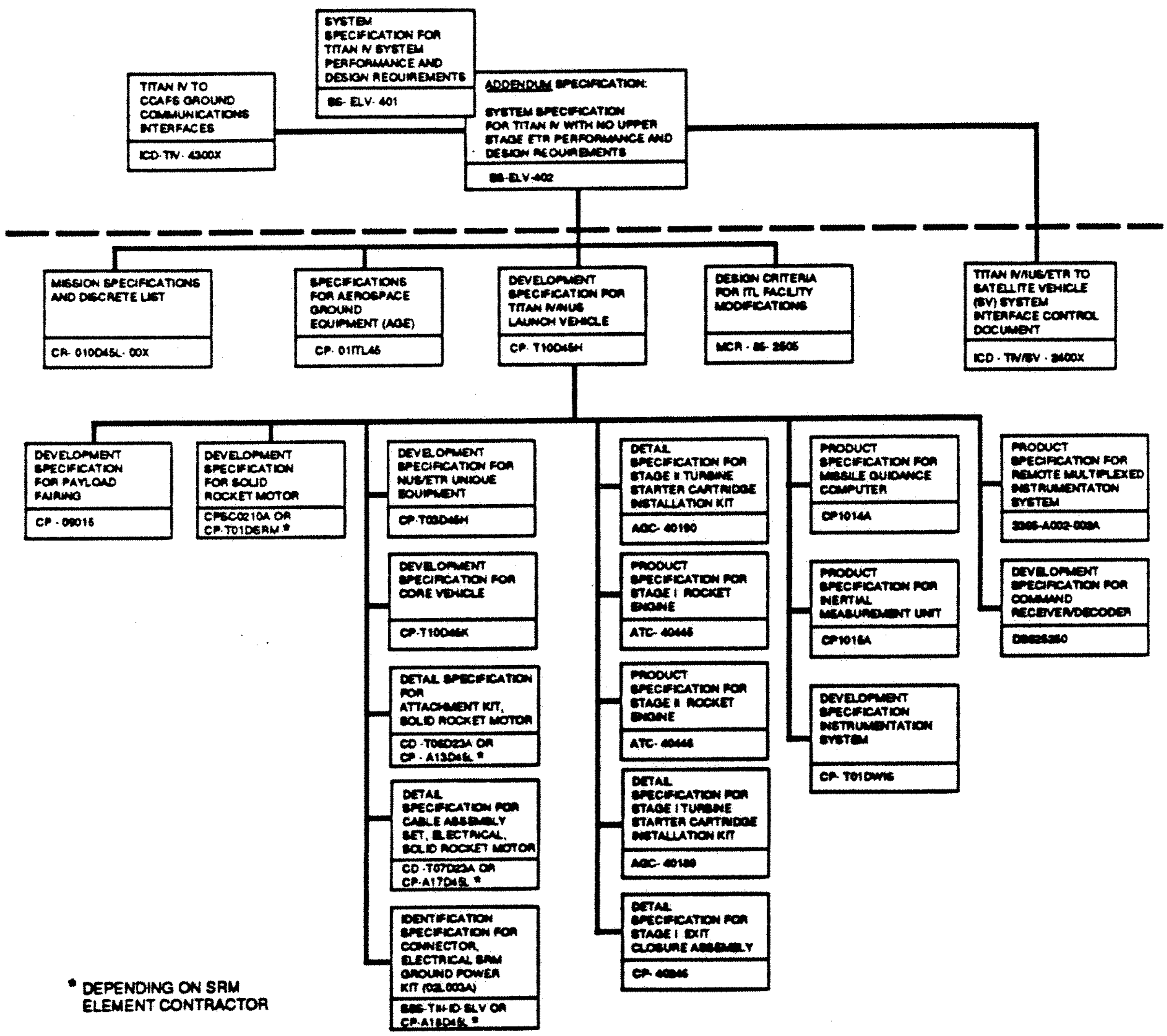


FIGURE 4.5-1 TITAN IV/NUS/ETR SYSTEM SPECIFICATION TREE

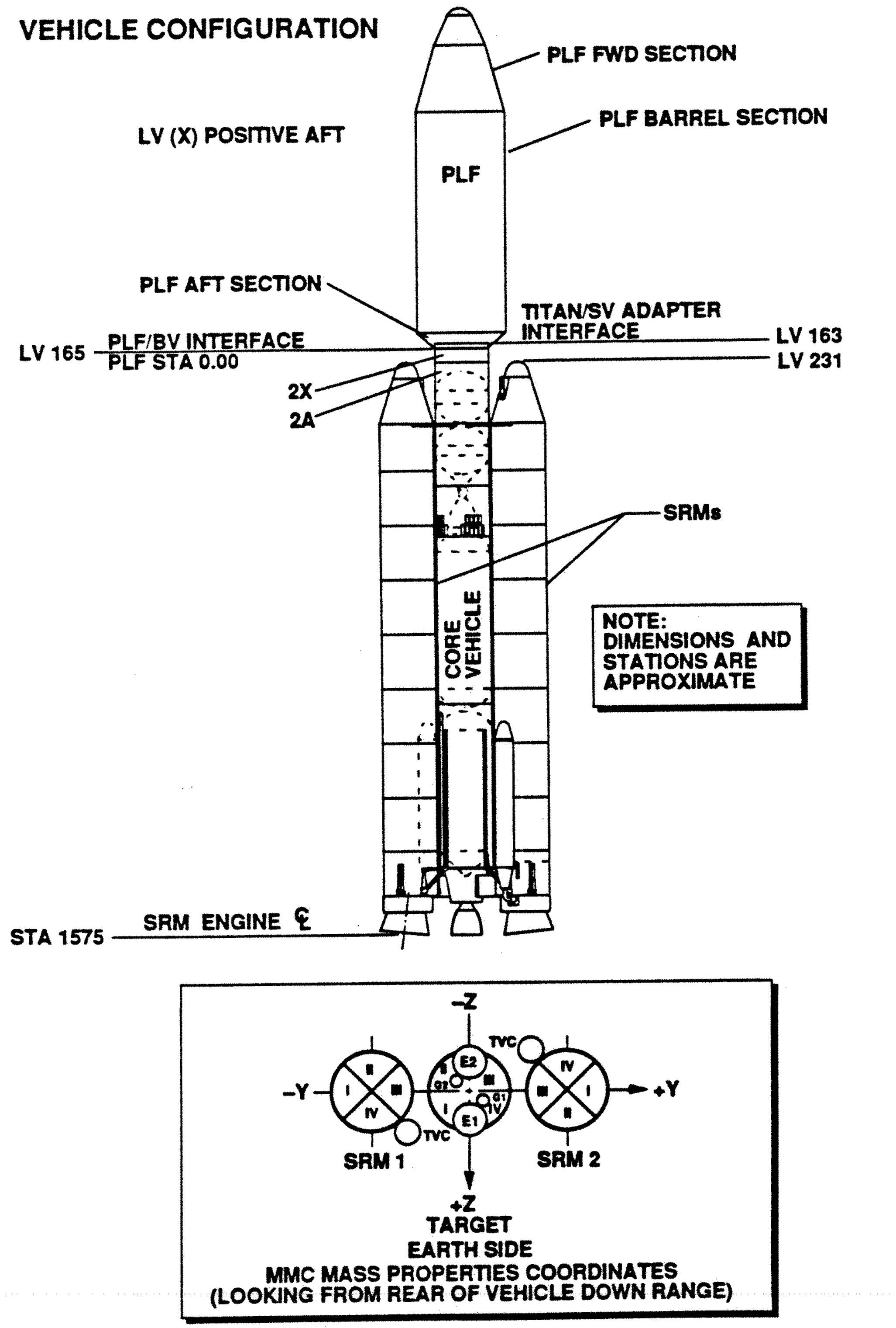


FIGURE 4.5-2 TITAN IV/NUS/ETR CONFIGURATION (405)

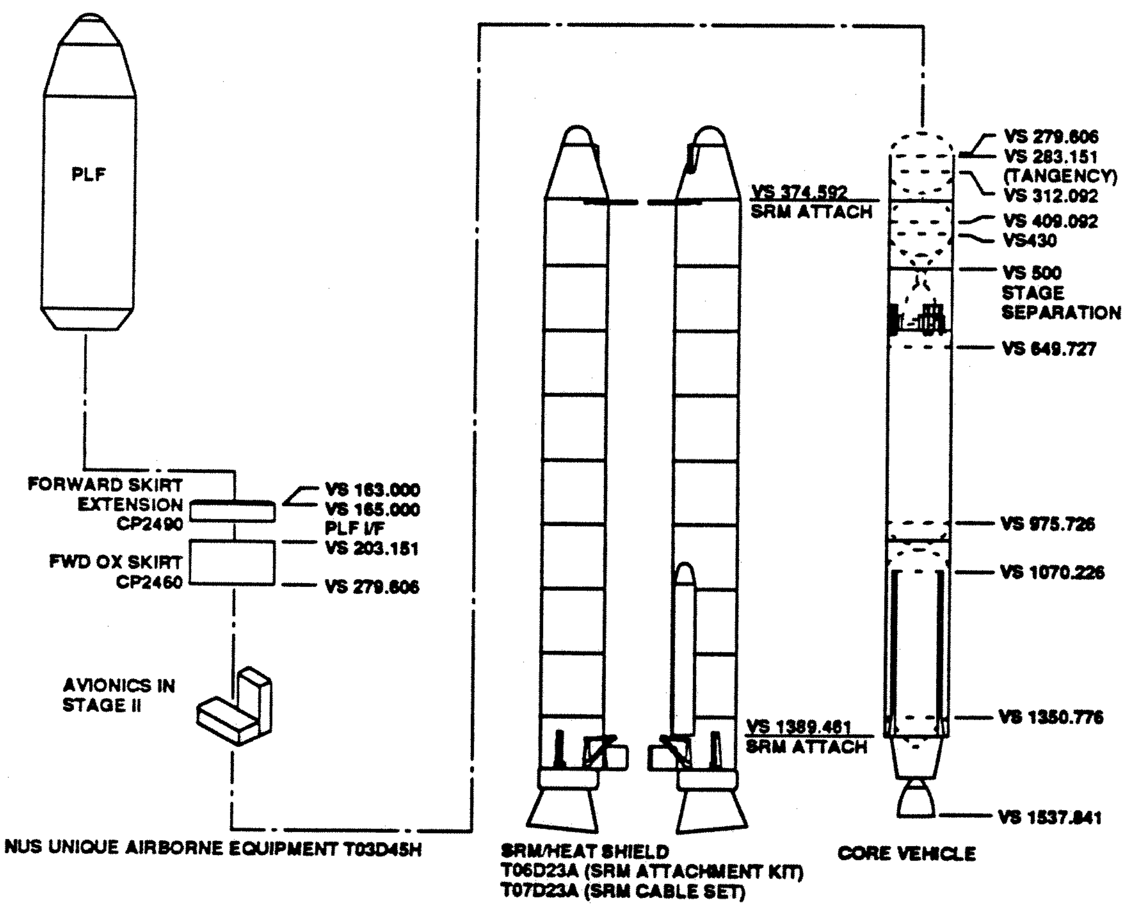


FIGURE 4.5.1.1-1 CI RELATIONSHIP WITH TITAN IV/NUS LAUNCH VEHICLE

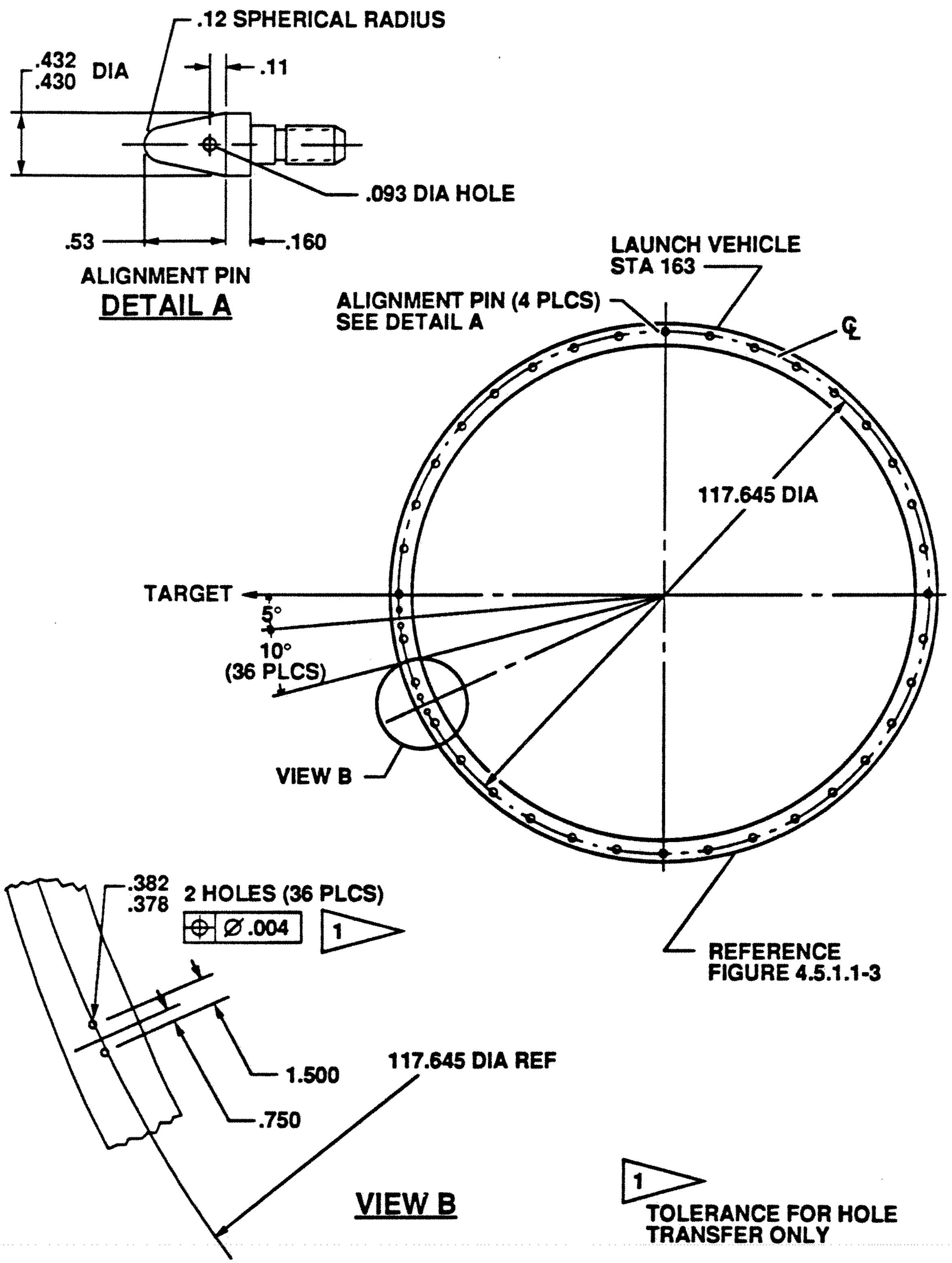


FIGURE 4.5.1.1-2 LAUNCH VEHICLE TO SV INTERFACE (405)

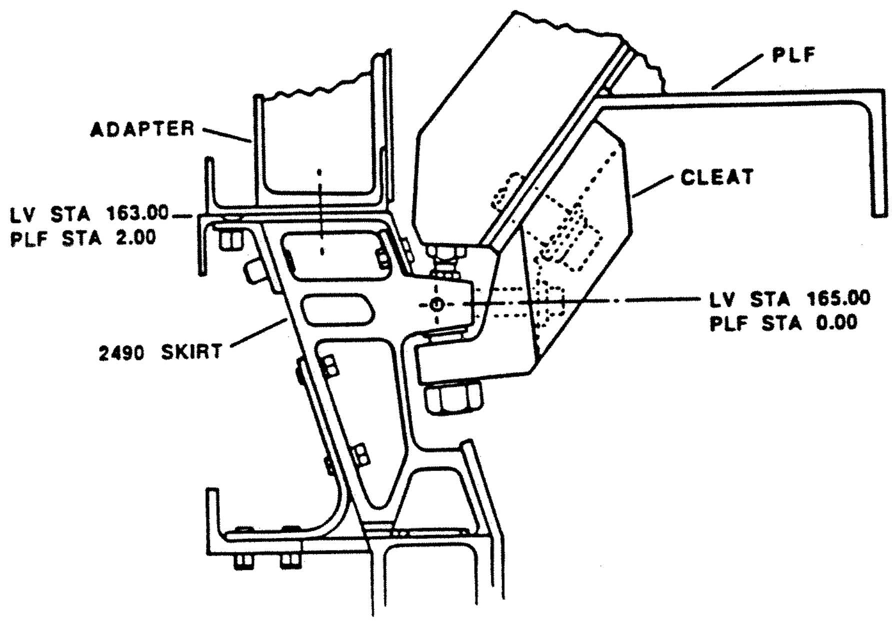


FIGURE 4.5.1.1-3 NUS 402/405 ADAPTER/PLF/2490 SKIRT INTERFACE DETAILS

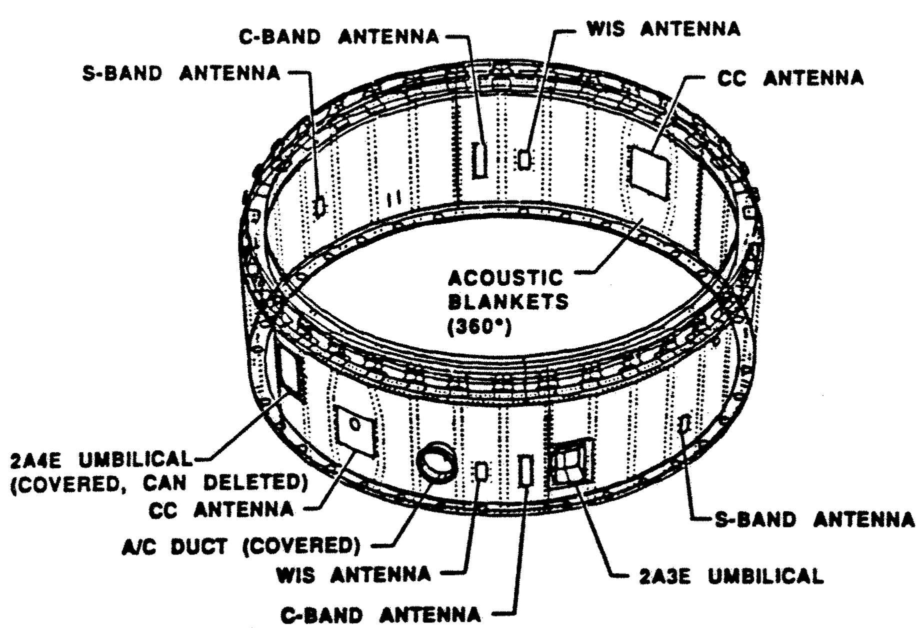


FIGURE 4.5.1.1-4 TITAN IV/NUS 405 FORWARD SKIRT EXTENSION – CP2490

4.5.1.1 Spacecraft Mechanical Interface (Continued)

The SC contractor and Martin Marietta will interchange design data which is to include structural drawings and computer models in the form of MSC/NASTRAN bulk data card images so that both parties can develop models and analyze the interface. The SC Contractor will design his structure such that all MSs are positive on both sides of the interface without Martin Marietta hardware changes.

Specific Titan IV/NUS SC Interface agreements are contained in the Mission Unique Titan IV/NUS – SC ICDs.

4.5.1.2 Alignment

Reference Figure 4.5.1.2-1.

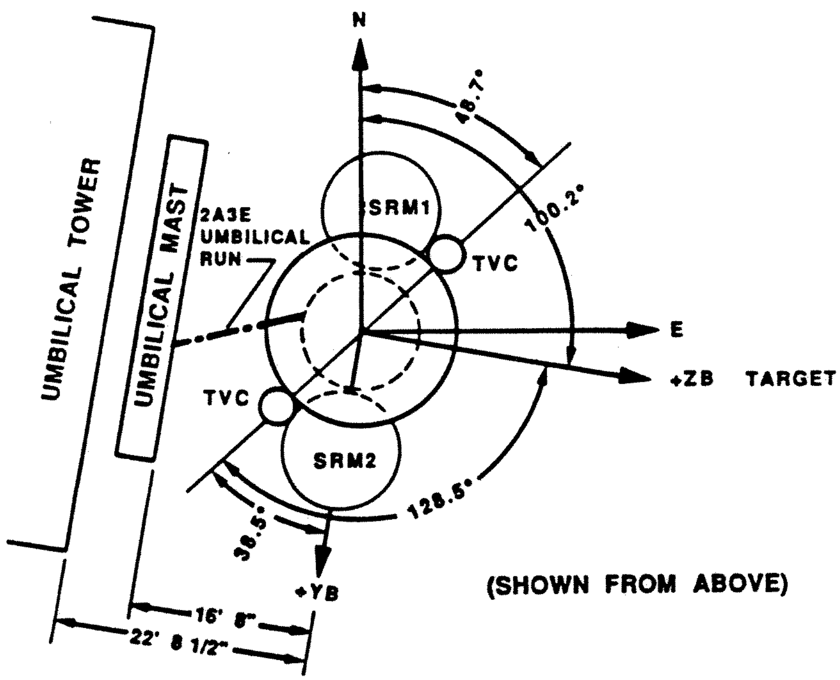


FIGURE 4.5.1.2-1 TITAN IV/NUS 405 AZIMUTH ORIENTATION ON PAD 41

4.5.1.3 Mechanical Attachments

The BV shall provide for mechanical attachment of the S/V as shown in Figure 4.5.1.1-2. The combined effect of all BV manufacturing tolerances shall result in less than the following misalignments at the S/V interface (VS 163.000):

Lateral Translation ±1.0 in.
Angle of Tilt ±0.1° with respect to vertical
Angle of Twist ±0.5° with respect to target

4.5.2 Payload Fairing

Reference Figures 4.5.2-1 PLF Dynamic Envelope

ENVELOPE DOES NOT INCLUDE THE A/C INLET ELBOW WHICH IS LOCATED AT PLF STA 410.22

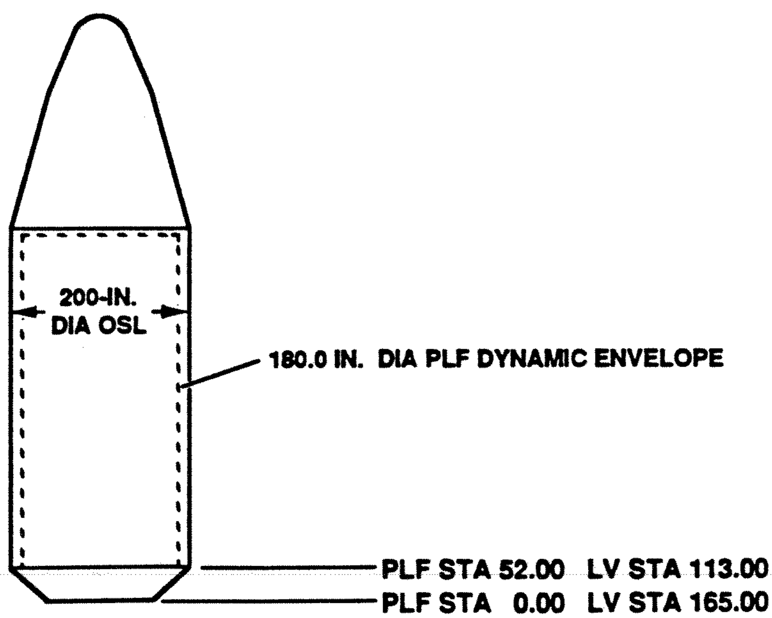


FIGURE 4.5.2-1 PLF DYNAMIC ENVELOPE (405)

4.5.2.1 Compartment Venting

Venting of the SV and PLF is through vents in the aft region of the cylindrical section of the PLF. PLF internal pressure decay during ascent will not exceed 0.5 psi/sec, except for a perturbation not exceeding three sec in duration and then the rate will not exceed 0.76 psi/sec, reference Figure 4.5.2.1-1.

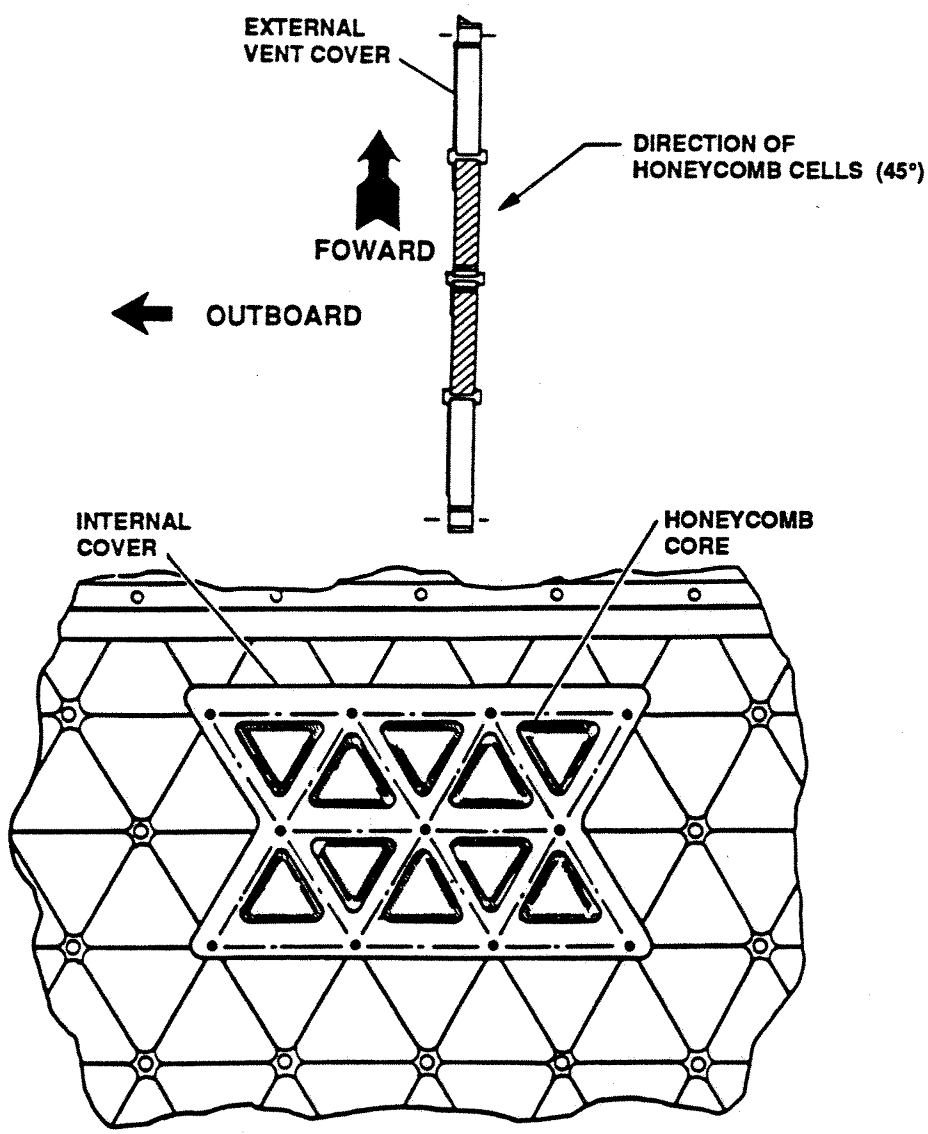


FIGURE 4.5.2.1-1 TYPICAL PLF VENT INSTALLATION

4.5.3 Electrical Systems

Reference Figure 4.5.3-1.

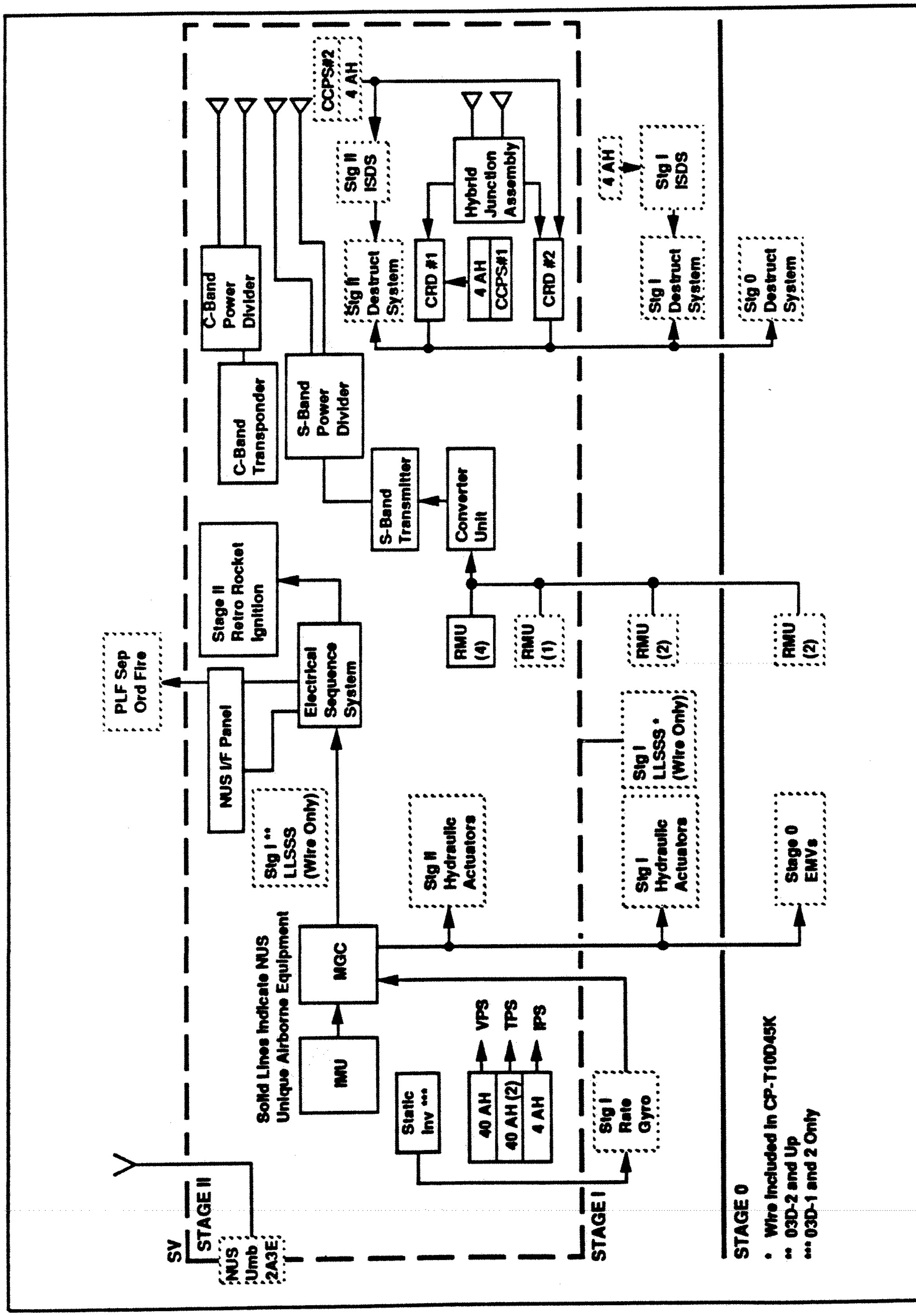


FIGURE 4.5.3-1 TITAN IV/NUS/ETR SYSTEM FUNCTIONAL SCHEMATIC (405)

4.5.3.1 Attitude Control

The Titan IV/NUS FCS is configured to function so as to maintain the required trajectory and attitude through all of the flight phases required from launch to P/L separation, reference Paragraph 3.2.3.2.

The Vehicle twist and sway induced into the IGS through wind effects or other environmental factors is monitored and corrective measures are employed to ensure compliance with twist and sway requirements.

4.5.3.2 Signal Interfaces

4.5.3.2.1 Instrumentation and Telemetry

The Titan IV and SV Instrumentation Systems other than the WIS are independent. A TLM System is provided for the Titan IV RMIS and also for WIS, reference Paragraph 3.2.3.3.

Titan IV provides the 2A3E umbilical interface to AGE which allows SC TLM and command connections. SC separation signal monitoring lines are connected to the Titan IV RMIS, reference Figure 4.5.3.2.1-1.

4.5.3.2.2 Commands

Commands between Titan IV and the SV consist of ordnance firing commands and command discretes. The Discrete Sequencing System S/W and hardware includes the capability to provide for staggered execution of the SV Separation and the Retro Rocket firing functions. Command Destruct interfaces and Stage II ISDS sense line interfaces required by ESMC Range Safety for the 405 SV are SV mission unique as is the 405 FTS detailed design, reference Figure 4.5.3.2.2-1.

The SV FTS for the Titan IV/NUS is provided as a mission unique kit, reference Figure 4.5.3.2.2-2. The SV FTS is a line-of-sight system from the Explosively Formed Projectile (EFP) to the SV. This part of the FTS can be initiated by signals from the Titan IV CRDs or from signals generated by the ISDS, reference MCR-89-2584.

4.5.3.3 Electrical Subsystem

The Titan IV/SV cabling interface for a typical mission unique kit is shown in Figure 4.5.3.3-1.

The SV to AGE umbilical 2A3E cabling runs are illustrated in Figure 4.5.3.3-2. This same 2A3E umbilical scheme is applicable to the IUS configuration.

SV wiring to the 2A2E Umbilical Interface Connector is SV peculiar.

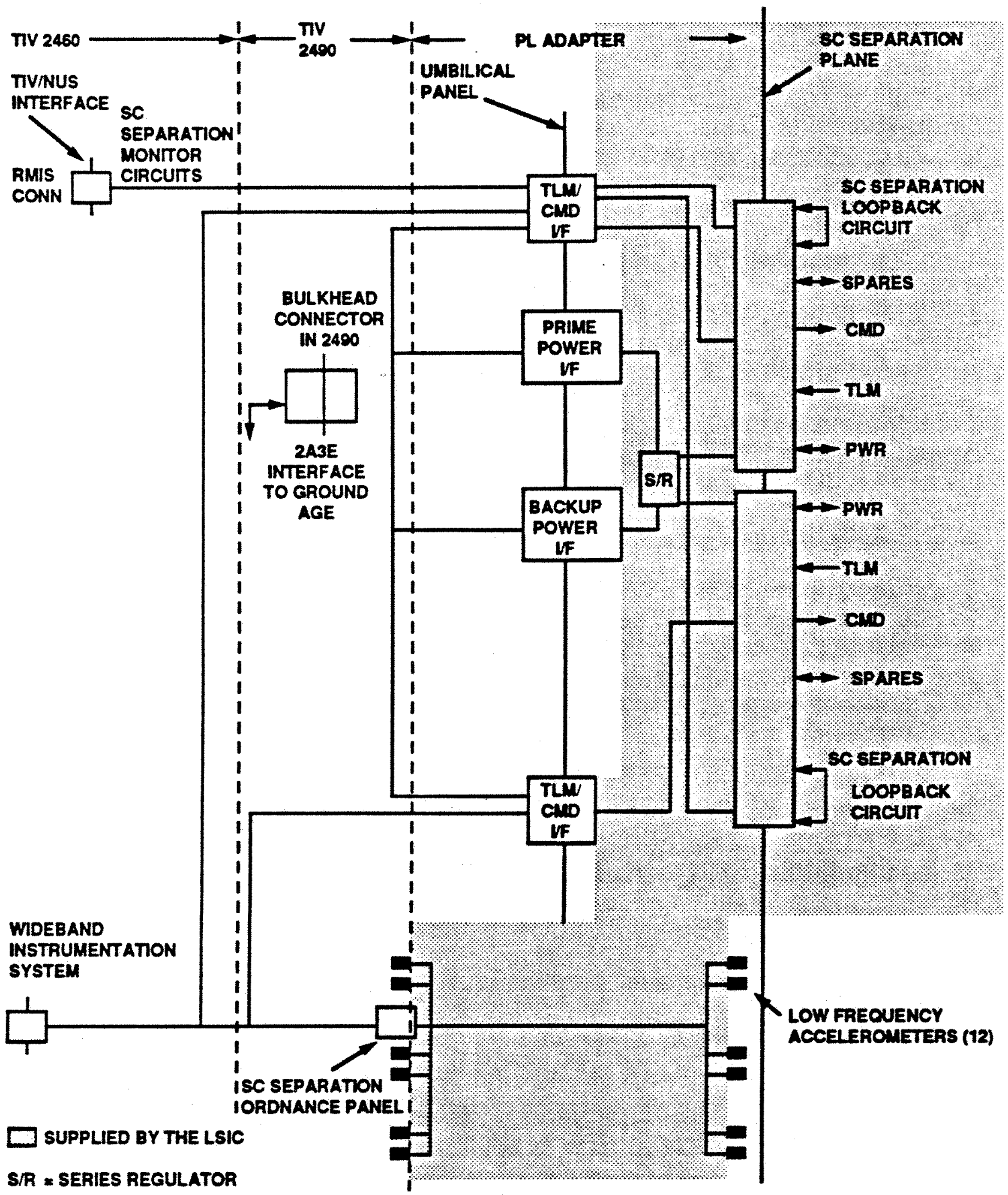
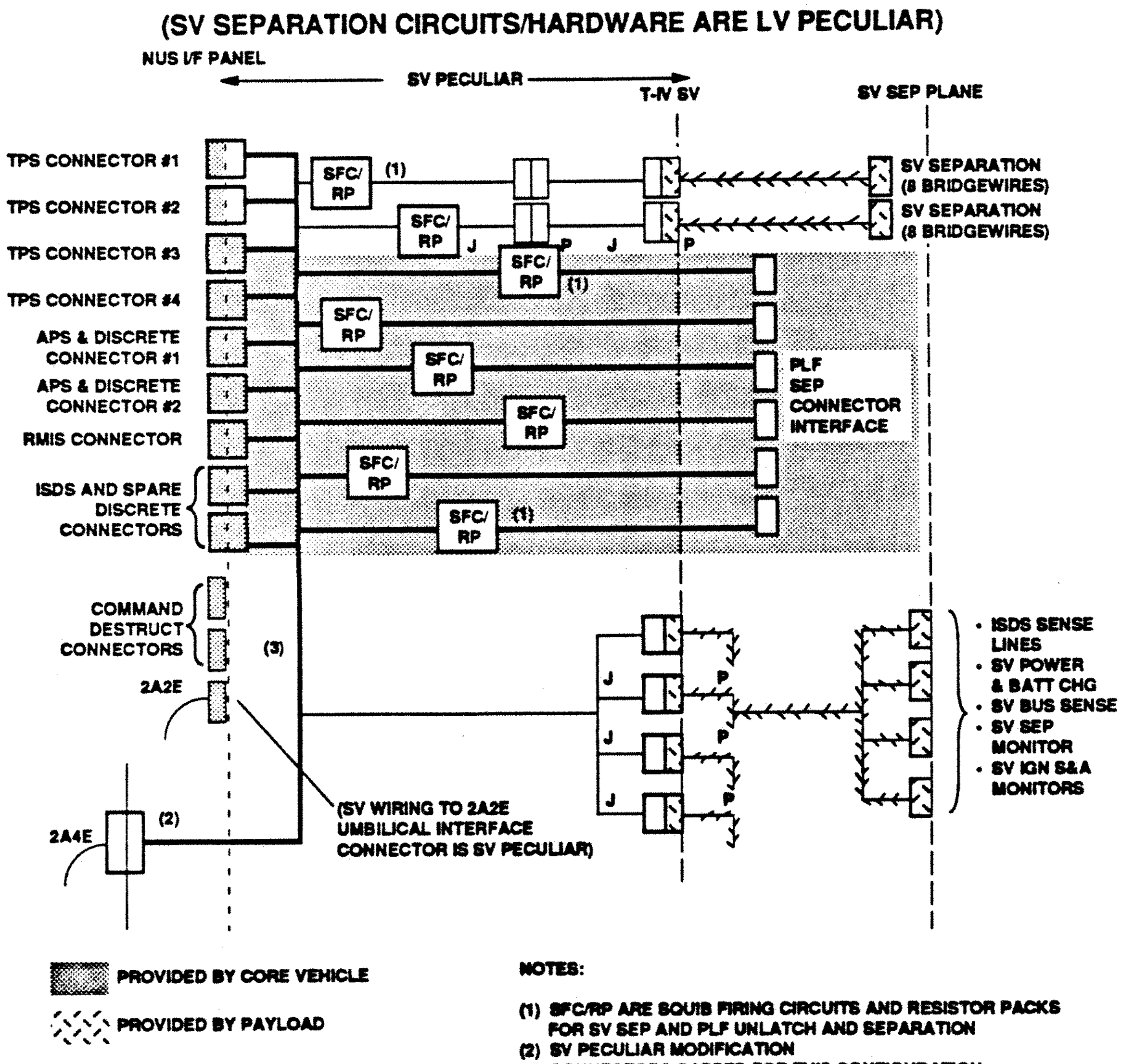


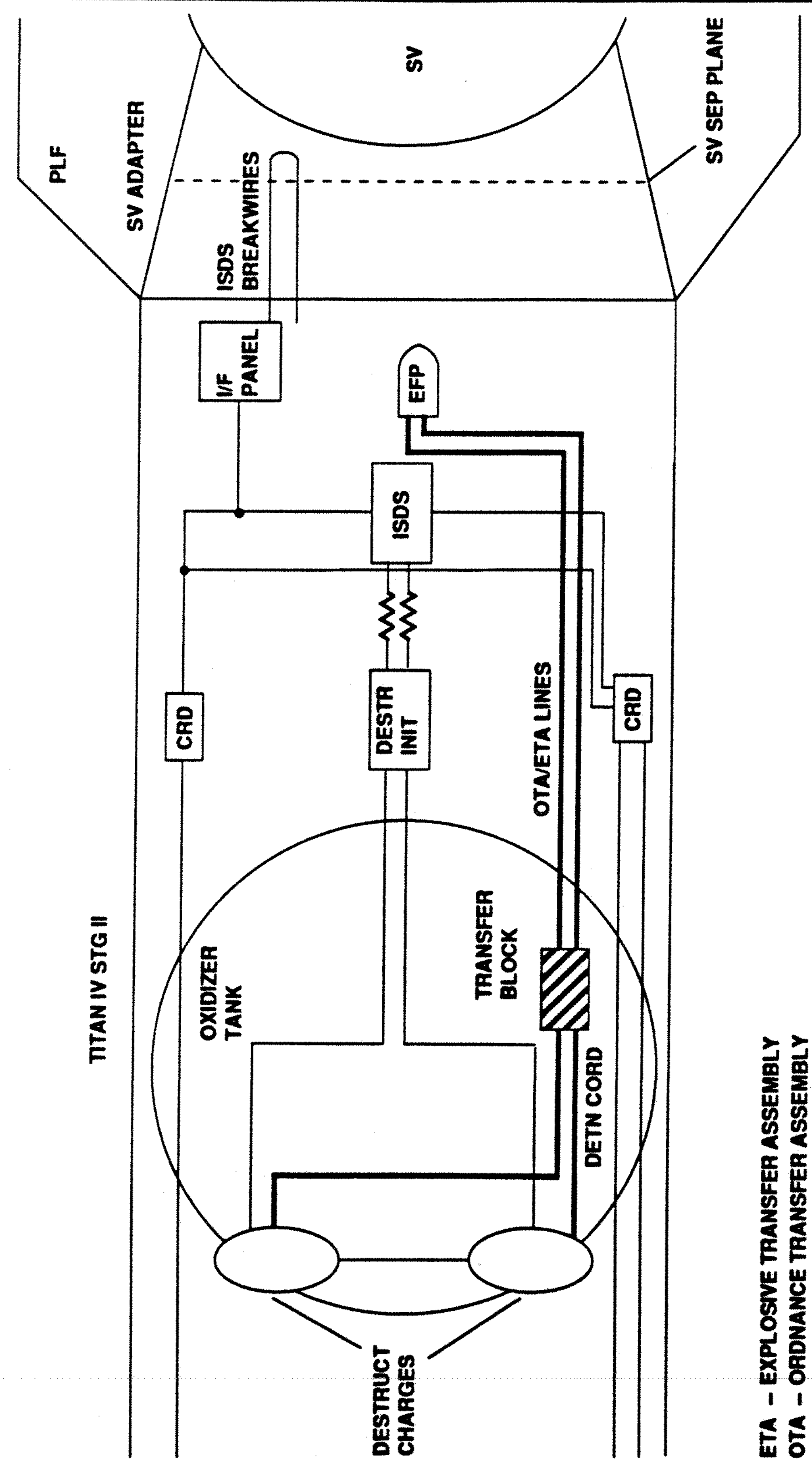
FIGURE 4.5.3.2.1-1 TITAN IV 2460 SKIRT, SPACECRAFT ADAPTER, SC SEPARATION PLANE ELECTRICAL INTERFACES



NOTES:

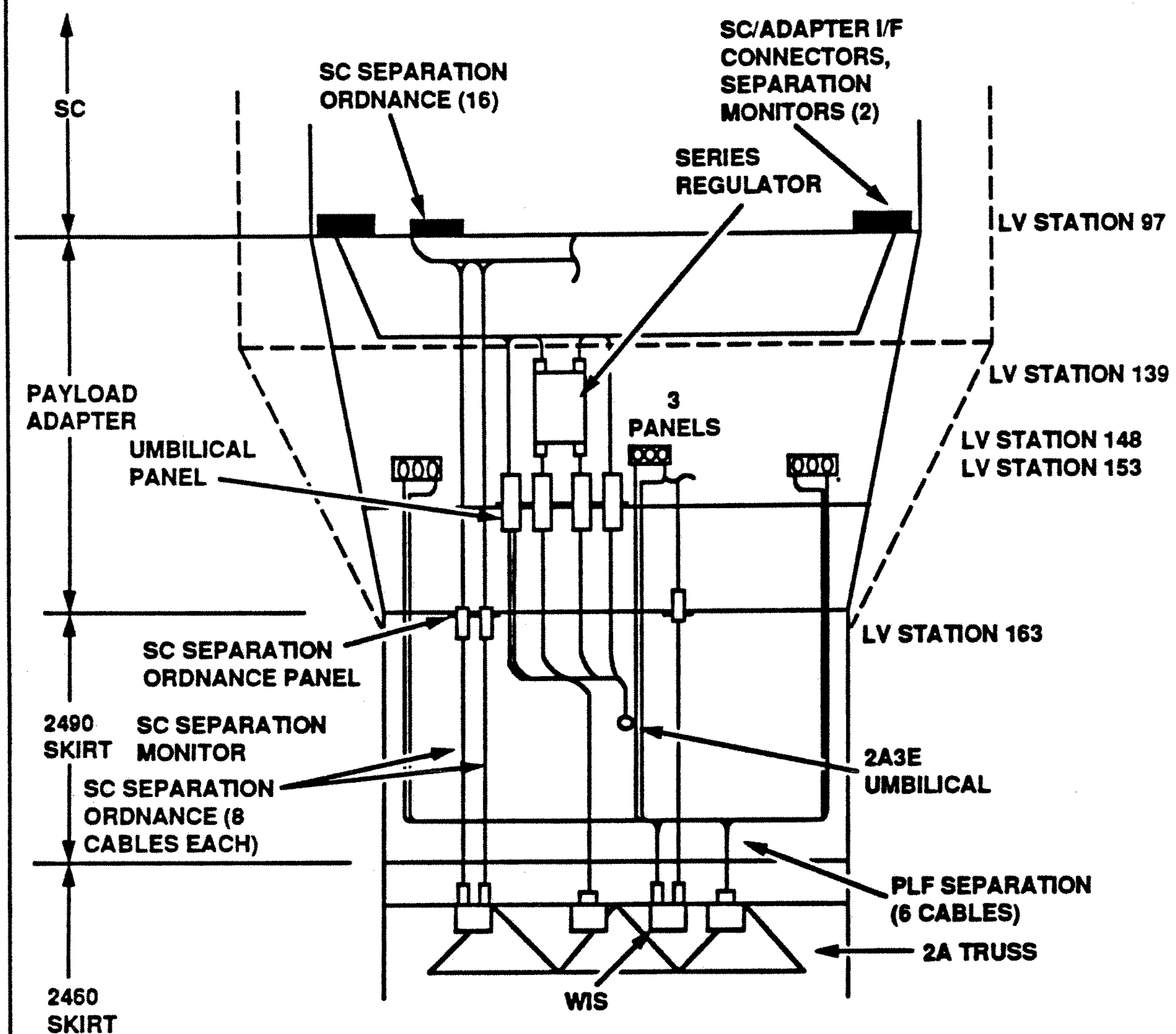
(1) SFC/RP ARE SQUIB FIRING CIRCUITS AND RESISTOR PACKS FOR SV SEP AND PLF UNLATCH AND SEPARATION
(2) SV PECULIAR MODIFICATION
(3) CONNECTORS CAPPED FOR THIS CONFIGURATION

FIGURE 4.5.3.2.2-1 TITAN IV/NUS TYPICAL ELECTRICAL INTERFACE



ETA - EXPLOSIVE TRANSFER ASSEMBLY
OTA - ORDNANCE TRANSFER ASSEMBLY
EFP - EXPLOSIVE FORMED PROJECTILE

FIGURE 4.5.3.2.2-2 NUS SS-ELV-405 SV FTS SYSTEM



NOTES:

1) SC SEPARATION ORDNANCE CABLING FROM LV STATION 97 TO THE SC SEPARATION PANEL TO BE PROVIDED BY THE LSIC. CABLING FORWARD OF AND INCLUDING THE UMBILICAL PANEL TO BE SUPPLIED BY THE LSIC. PLF SEPARATION PANELS TO BE SUPPLIED BY THE LSIC; PLF SEPARATION CABLING TO BE SUPPLIED BY TIV. THE CONNECTORS USED ON THE SC SEPARATION ORDNANCE PANEL, THE UMBILICAL PANEL AND THE PLF SEPARATION PANELS TO BE SUPPLIED BY TIV. THE SC SEPARATION ORDNANCE PANEL TO BE SUPPLIED BY TIV

FIGURE 4.5.3.3-1 TYPICAL MISSION UNIQUE KIT – ELECTRICAL INTERFACES

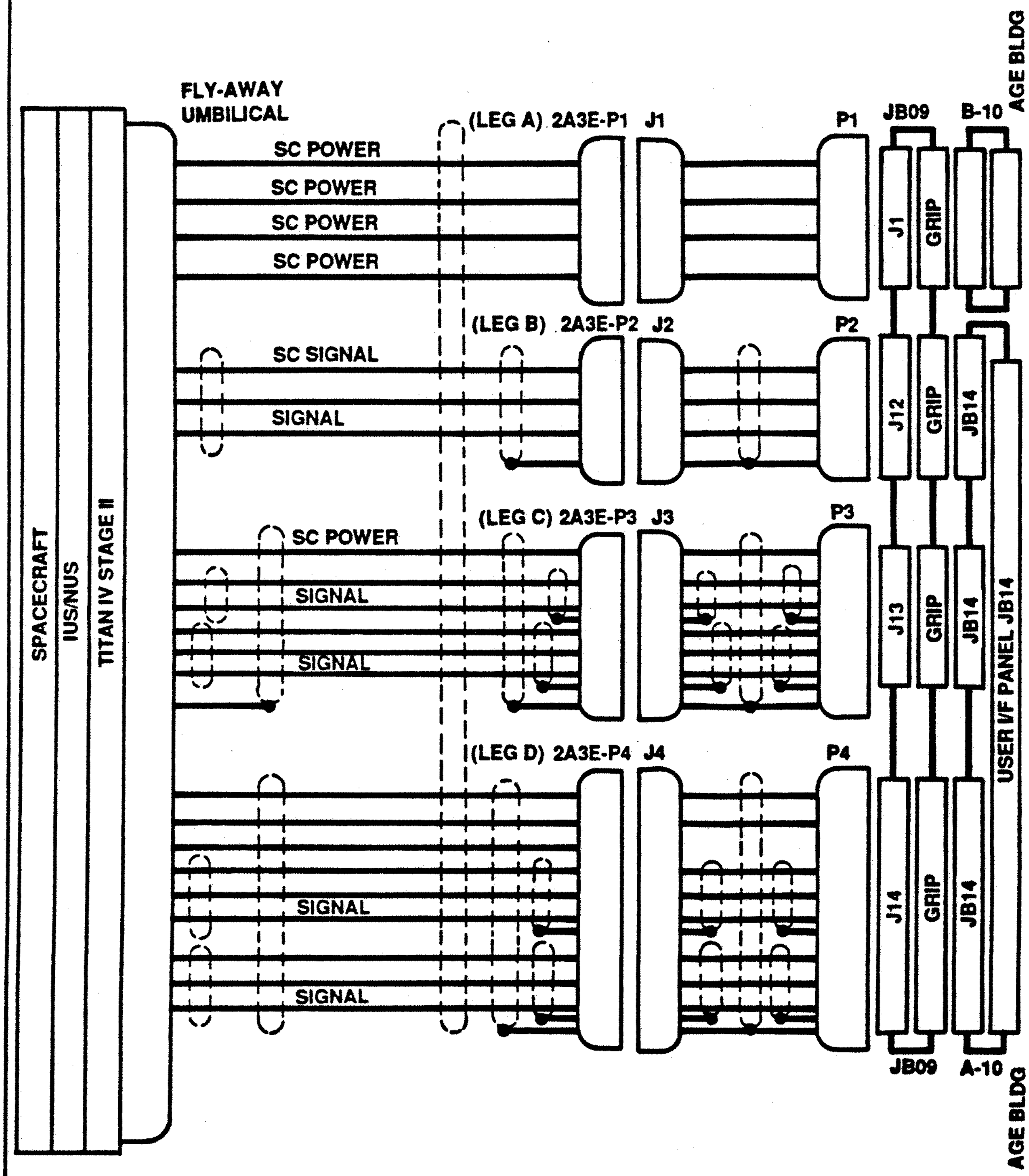


FIGURE 4.5.3.3-2 IUS/NUS CCAFS 2A3E UMBILICAL CABLES (SS-ELV-402/405)

Chapter 5

Environments

5.0 ENVIRONMENTS

5.1 Introduction

During Ground Processing, Launch and Flight, the SC is exposed to many environmental conditions/factors, which, if not properly analyzed and compensated for, could adversely affect SC performance and/or compromise its structural integrity. The general environmental factors addressed in this chapter are presented in Table 5.1-1. Reference MCR-88-2638 CCAFS/LC-41 Payload Support Capability Document and MCR-88-2639 VAFB/SLC 4E Payload Support Capability document.

TABLE 5.1-1 ENVIRONMENTAL FACTORS

	LAUNCH PHASE	
Factors	Prelaunch	Launch & Flight
Air Conditioning	•	–
Electromagnetic	•	•
Dynamic Pressure	–	•
Contamination	•	•
Acoustics	–	•
Vibration	–	•
Shock	–	•
Thermal	–	•
Acceleration	–	•
Structural Frequency	•	•
Wind	•	•
Venting	•	•

Structural frequency is included as a prelaunch factor in order to account for vehicle twist and sway. The Inertial Measurement Units of some SC require this data.

Specific environmental factors depend on SC-user requirements and mission-peculiar criteria.

5.2 Air Conditioning Capabilities – General

There are air conditioning systems at both ETR and WTR launch complexes to support the Titan IV Vehicle; both Upper Stages; the SC; the Environmental Shelter; the AGE Buildings; Vans; and Gas-Storage Facilities, reference Table 5.2-1.

TABLE 5.2-1 UNIVERSAL ENVIRONMENTAL SHELTER AIR CONDITIONING CAPABILITY

Minimum Internal Positive Pressure	0.05/in. H_2O
Flowrate	20 air changes/hour for Levels 11 – 19 6 air changes/hour for crane area
Dry Bulb Temperature	72°F ± 5°F
Relative Humidity	50% RH (Max) 35% (Min)
Filtration	Class 5,000 or better (Per FED STD 209b) At air inlet to environmental shelter

5.2 Air Conditioning Capabilities – General (Continued)

The Ground Support Air/Nitrogen Conditioning System is provided to support Upper Stage and/or Payload cooling requirements. The system is able to condition and distribute either air or nitrogen to either of two PLF umbilical hose interfaces. Performance characteristics for these air conditioners are shown in Table 5.2-2.

TABLE 5.2-2 PAYLOAD FAIRING AIR CONDITIONING CAPABILITY

ESMC Air Conditioning at Payload Fairing Inlet	
Temperature	50° to 80°F ± 1°
Filtration	Class 5,000 at Inlet
Flowrate	50 to 300 lb/min ± 11 lb
Static Pressure	60 in. H_2O (Min)
Moisture Content	N_2: –35° F Max Dew Point Air: 28 – 50% RH

Air diffusion systems designed to accommodate specific SC temperature limits, temperature gradient and air impingement velocity requirements within the PLF, are SC-unique requirements. Typical systems use ducting and a diffuser to direct inlet air into the PLF nose section. Ducting may be used to direct air to meet specific Payload or Upper Stage needs.

Figures 5.2-1 and 5.2-2 show PLF air conditioning inlet/vent locations as well as other thermal features. Figure 5.2-3 shows the baseline air conditioning/venting configuration.

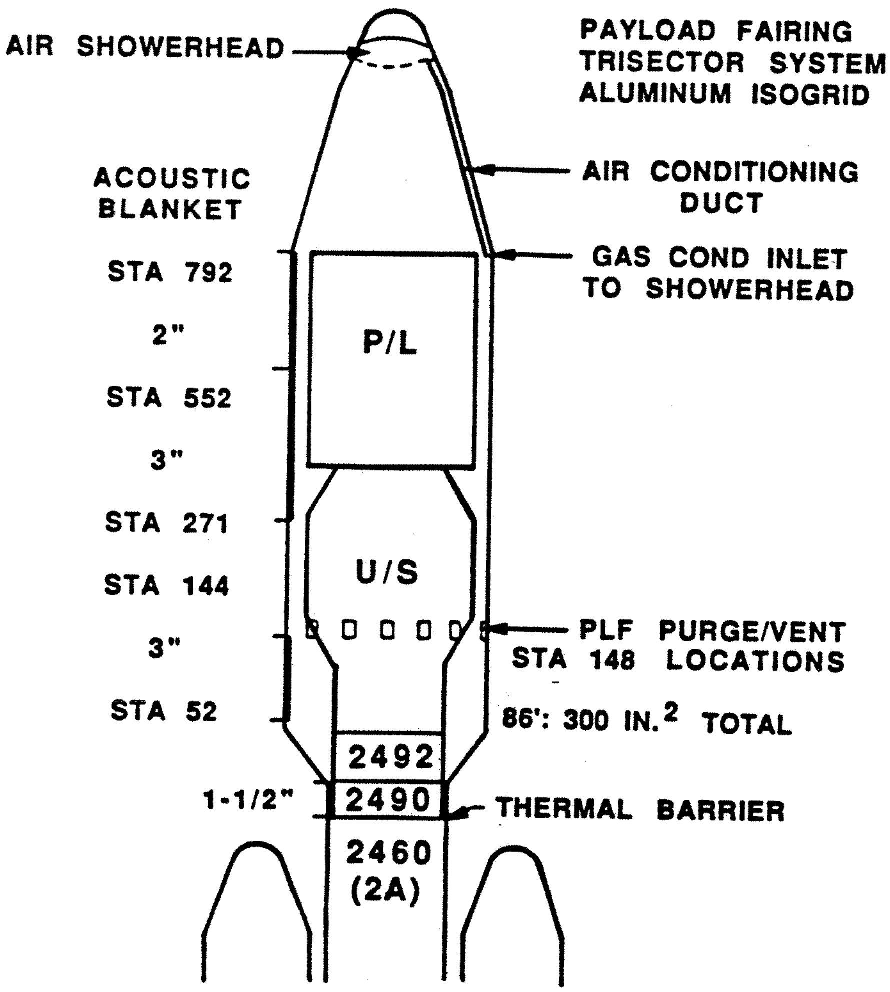


FIGURE 5.2-1 86-FOOT PLF AIR CONDITIONING AND THERMAL FEATURES

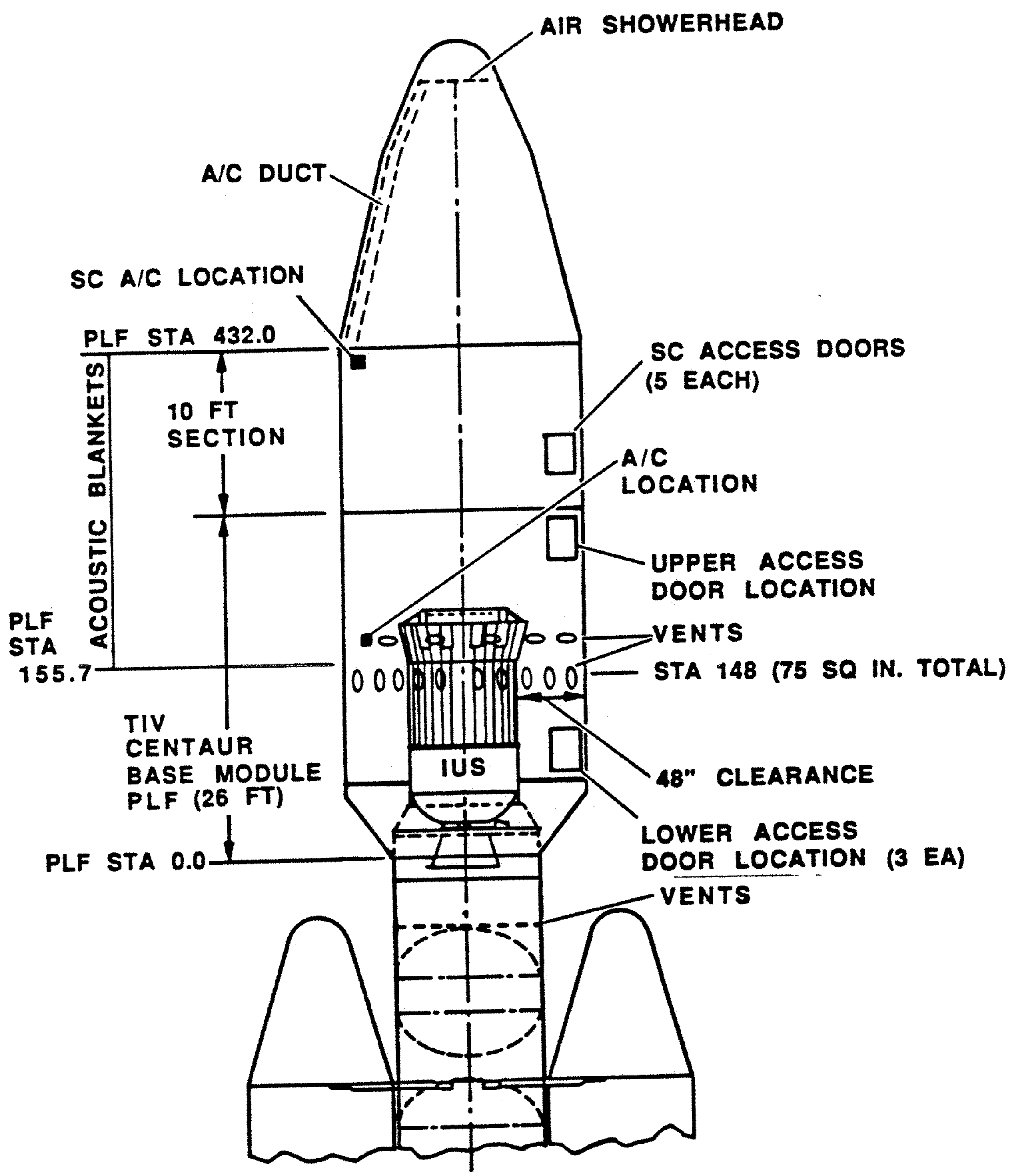
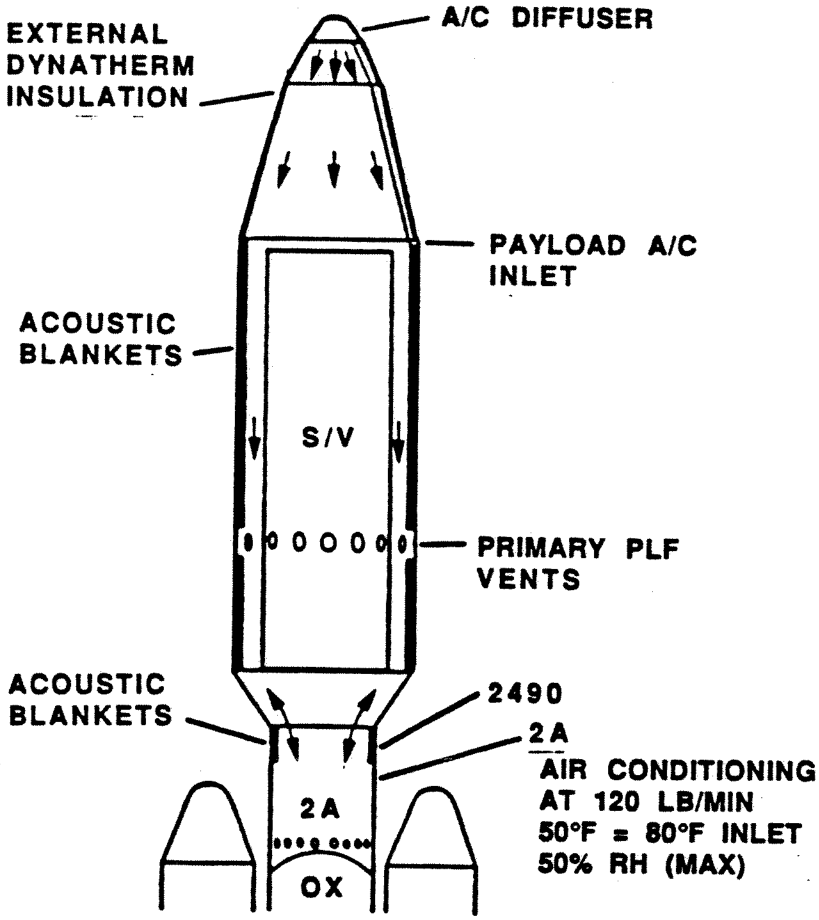


FIGURE 5.2-2 56-FOOT PLF/IUS AIR CONDITIONING AND THERMAL FEATURES



P/L AIR CONDITIONING
- 200 LB/MIN MAXIMUM
- 38 TO 100°F INLET, SELECTABLE
- 34.5° F DEWPOINT
- REDUNDANT SYSTEM
- AUXILIARY SYSTEM AVAILABLE

PLF OPTICAL PROPERTIES
- WHITE PAINTED EXTERIOR
- ISOGRID SKIN E ≤ 0.3
- ACOUSTIC BLANKET E = 0.86
- RINGS FRAMES, RAILS E = 0.15
- BOATTAIL E = 0.15

FIGURE 5.2-3 BASELINE AIR CONDITIONING CONFIGURATION

5.3 Payload Fairing and Universal Environmental Shelter Cleanliness Criteria

Both the PLF cleaning areas and the environmental shelters at the ETR and WTR are operated as class 100,000 clean rooms. Refer to the Titan IV System Contamination Control Plan, MCR-86-2550, February 1987, for the detailed cleanliness criteria and requirements for these areas.

5.4 Electromagnetic Compatibility

Reference Appendix D.

5.5 General Launch and Flight Environments

During Lift-Off and Flight, engine ignitions and staging events produce exhaust products and debris that are potential sources of SC contamination and damage. Various techniques are used to eliminate and/or minimize these hazards.

The dynamic pressure on the SC during flight depends on the trajectory profile and PLF jettison time. Jettison time normally is determined by SC aerodynamic heating limitations. For a typical mission, the pressure is less than 0.01 psf at jettison and decreases to 0.00 psf as the altitude increases.

The pressure change within the PLF during flight is a function of the trajectory, atmospheric conditions, free volume within the fairing, openings in the fairing and the adjacent launch vehicle compartment. Specific compartment pressure differentials cannot be quoted without knowledge of SC volume and leakage areas associated with the SC/Fairing combination used. The pressure change within the PLF during flight is not to exceed 0.4 psi/sec, except for a perturbation not exceeding 8 sec during which the decay rate is not to exceed 0.5 psi/sec.

The Titan Vehicle has four separation Retrorockets. The Retrorockets are located approximately 30 ft from the Upper Stage/SC separation plane for IUS and approximately 42 ft away for Centaur. The rockets are at a fixed cant angle of 45° to minimize plume impingement.

The two current solutions to the Retrorocket exhaust plume contamination problem are to employ plume deflectors on Stage II; and to utilize Retrorockets with no more than 2 percent aluminum in the propellant (when available).

The use of "clean" Retrorocket Motors will alleviate or eliminate the concern for degradation of the Centaur tank radiation shield thermal performance.

The Titan IV system specification requirements for molecular contamination for the Centaur Upper Stage are that the contamination added to the SC sensors by the Titan IV system from PLF installation, through completion of collision avoidance maneuvers, will not exceed 50 angstroms; and particulate contamination is not to reduce solar cell output by more than 1% for all Titan IV system configurations over the same period.

5.6 Acoustics, Vibration and Shock

5.6.1 Acoustics

During the liftoff event, significant acoustic energy will be created by the turbulent mixing of the exhaust stream with the atmosphere. The maximum predicted liftoff acoustic environment on the PLF external surface for a Titan IV with 7-segment UT/CSD SRMs (Type I) is presented in Table 5.6.1-1. It should be noted that the definition of "maximum predicted environment" as used herein is the statistical 95th percentile level with 50 percent confidence. With 3-segment Hercules Incorporated SRMUs (Type II), the liftoff levels are estimated to be 1 dB higher than shown at each frequency. The duration of this excitation (defined as the time during which the level is within 6 dB of its maximum value) is estimated to be approximately 5 sec.

Significant acoustic energy is also produced by separated flow and the turbulent boundary layer during the transonic and maximum dynamic pressure periods of flight. The maximum predicted flight regime acoustic environments on the PLF external surface are shown in Table 5.6.1-1. These levels are applicable for both Type I and Type II SRMs. The duration of this excitation, as defined above, is approximately 55 sec.

TABLE 5.6.1-1 MAXIMUM EXTERNAL ACOUSTIC ENVIRONMENT ON PLF EXTERNAL SURFACE

LAUNCH LEVELS		FLIGHT LEVELS		
1/3 OCTAVE BAND CENTER FREQ (Hz)	AFT BARREL SECTION (dB)	CONE/CYLINDER JUNCTION AREA (dB)	BARREL SECTION (dB)	BOATTAIL AREA (dB)
31.5	129	152	129	154
40	131	153.5	130	154
50	133	154	131	155
63	134.5	155.5	132	156
80	136	156	133	156
100	137	157	134	155
125	138	158	134.5	154
160	139	159	135.5	153
200	140	160	136	152.5
250	140	159	137	151.5
315	140.5	158	138	150.5
400	140	157	139	150
500	140	156	140	149
630	139	155.5	141	147.5
800	138	155	142	146
1000	137	154	143	144
1250	136	153.5	144	142.5
1600	134	153	143.5	141
2000	132.5	152.5	142	138
2500	130.5	152	140	136
3150	128.5	151.5	138	134
4000	126.5	150.5	137	131
5000	124.5	150	135.5	128
6300	122.5	149.5	134	126
8000	120	149	133	124
10000	118	148.5	131.5	121
OVERALL	150.5	169	152.5	165

5.6.1 Acoustics (Continued)

The maximum allowable internal acoustic environment within an empty PLF is shown in Table 5.6.1-2. Internal acoustic levels for a specific payload are dependent upon a number of parameters, such as the PLF length, the number and location of the vents and access doors, the amount and location of the installed acoustic blankets, the volume occupied and absorption properties of the spacecraft and the volumetric "fill effect" which may be experienced. Specific predictions can be made for each spacecraft for each flight regime once suficient information is provided.

TABLE 5.6.1-2 PLF MAXIMUM ALLOWABLE INTERNAL ACOUSTIC ENVIRONMENT

1/3 OCTAVE BAND CENTER FREQ (Hz)	EMPTY PLF (SPACE AVERAGE) (dB)	1/3 OCTAVE BAND CENTER FREQ (Hz)	EMPTY PLF (SPACE AVERAGE) (dB)
31.5	125	630	124.5
40	126.5	800	123
50	127	1000	121.5
63	128	1250	120
80	128.5	1600	118
100	129	2000	116.5
125	129	2500	115
160	129	3150	113
200	128.5	4000	111.5
250	128	5000	109.5
315	127.5	6300	107.5
400	126.5	8000	106
500	125.5	10000	104
		OVERALL	139.3

5.6.2 Shock

The primary sources of pyroshock during a launch occur at SRM separation, Stage I/II separation, PLF separation, Upper Stage separation (if appropriate) and P/L separation. Of these pyroshock events, the only ones significant to the spacecraft are the PLF separation, Upper Stage separation and/or P/L separation.

The satellite vehicle induced pyrotechnic shock at the SV interface with either the Centaur upper stage or the booster vehicle, as appropriate, shall not exceed the levels shown in Figures 5.6.2-1, 5.6.2-2 or 5.6.2-3. The T-IV or Centaur induced shock. at the Centaur/SV interface shall not exceed the levels shown in Figure 5.6.2-4. The T-IV booster vehicle induced shock at the booster/SV interface (VS 163) for a NUS configuration shall not exceed the levels shown in Figure 5.6.2-3.

5.6.3 Vibration

The random vibration environment which will be experienced by SV components is due to the liftoff acoustic field, the aerodynamic excitations and transmitted structure-borne vibration. These vibration responses are dependent on the dynamic properties of the SV and the component mounting configurations, and are the responsibility of the SV contractor.

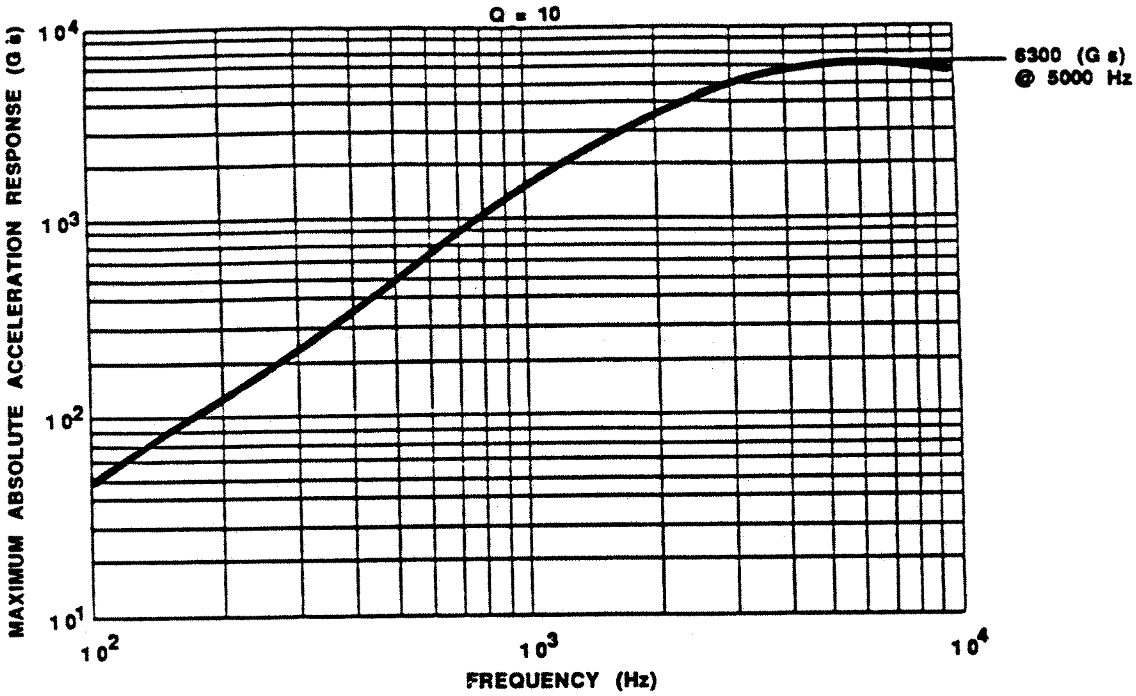


FIGURE 5.6.2-1 MAXIMUM 8-HARDPOINT INTERFACE SHOCK FROM SPACECRAFT

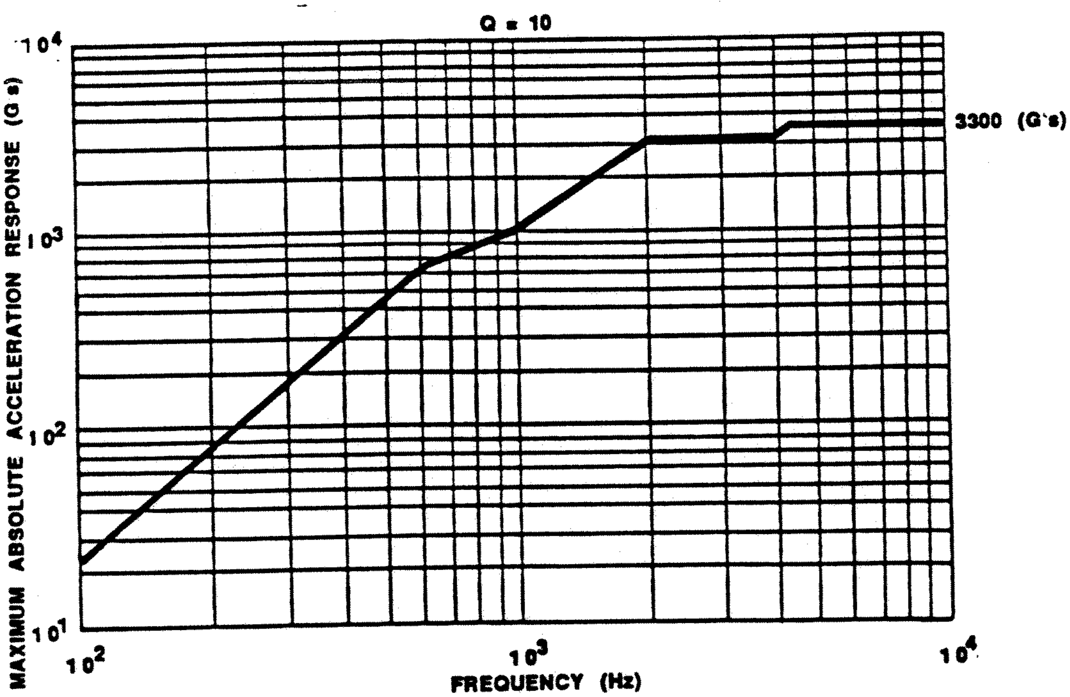


FIGURE 5.6.2-2 MAXIMUM 22-HARDPOINT INTERFACE SHOCK FROM SPACECRAFT

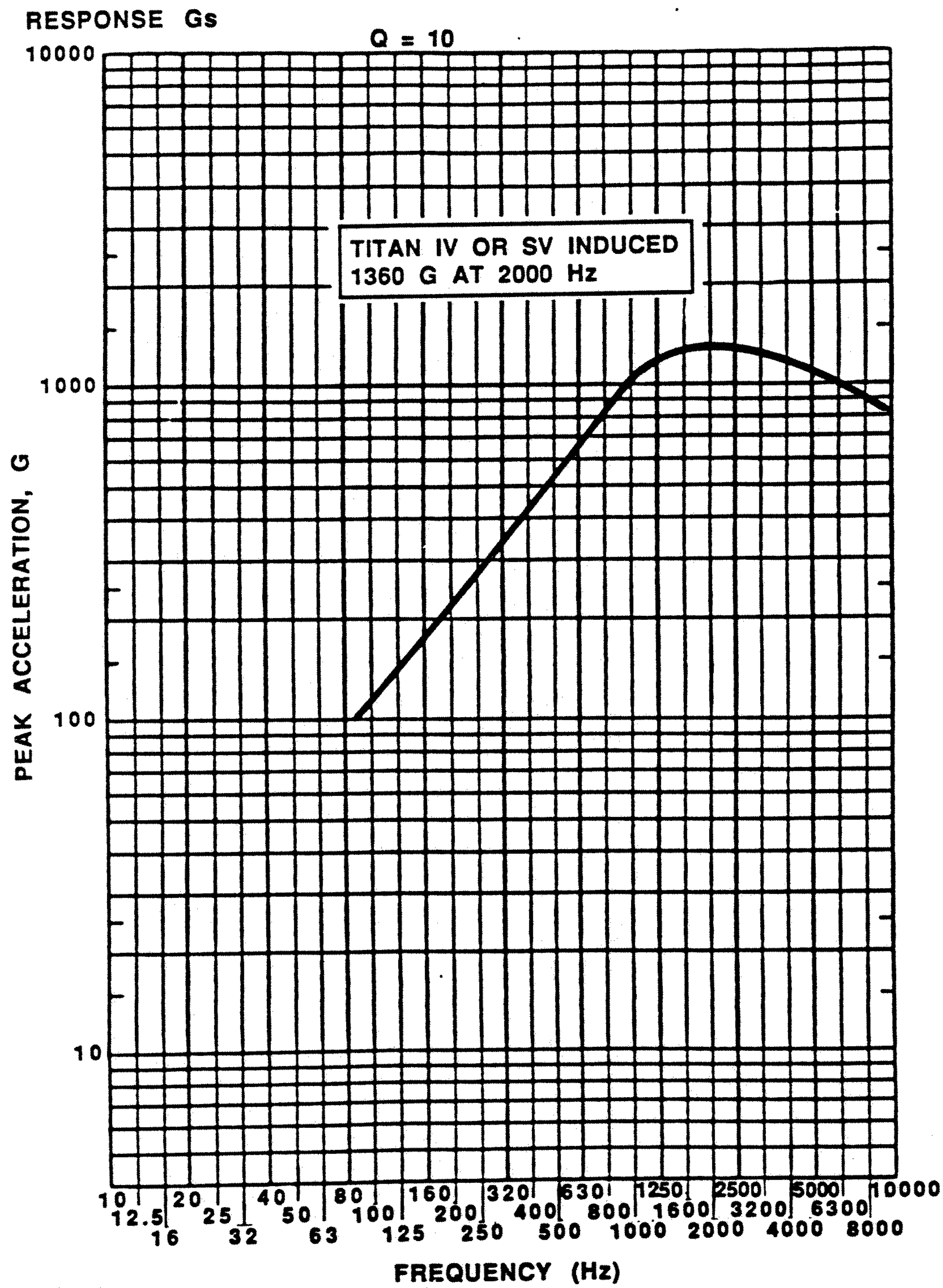


FIGURE 5.6.2-3 MAXIMUM SHOCK RESPONSE AT LAUNCH VEHICLE/ SV INTERFACE (VS 163)

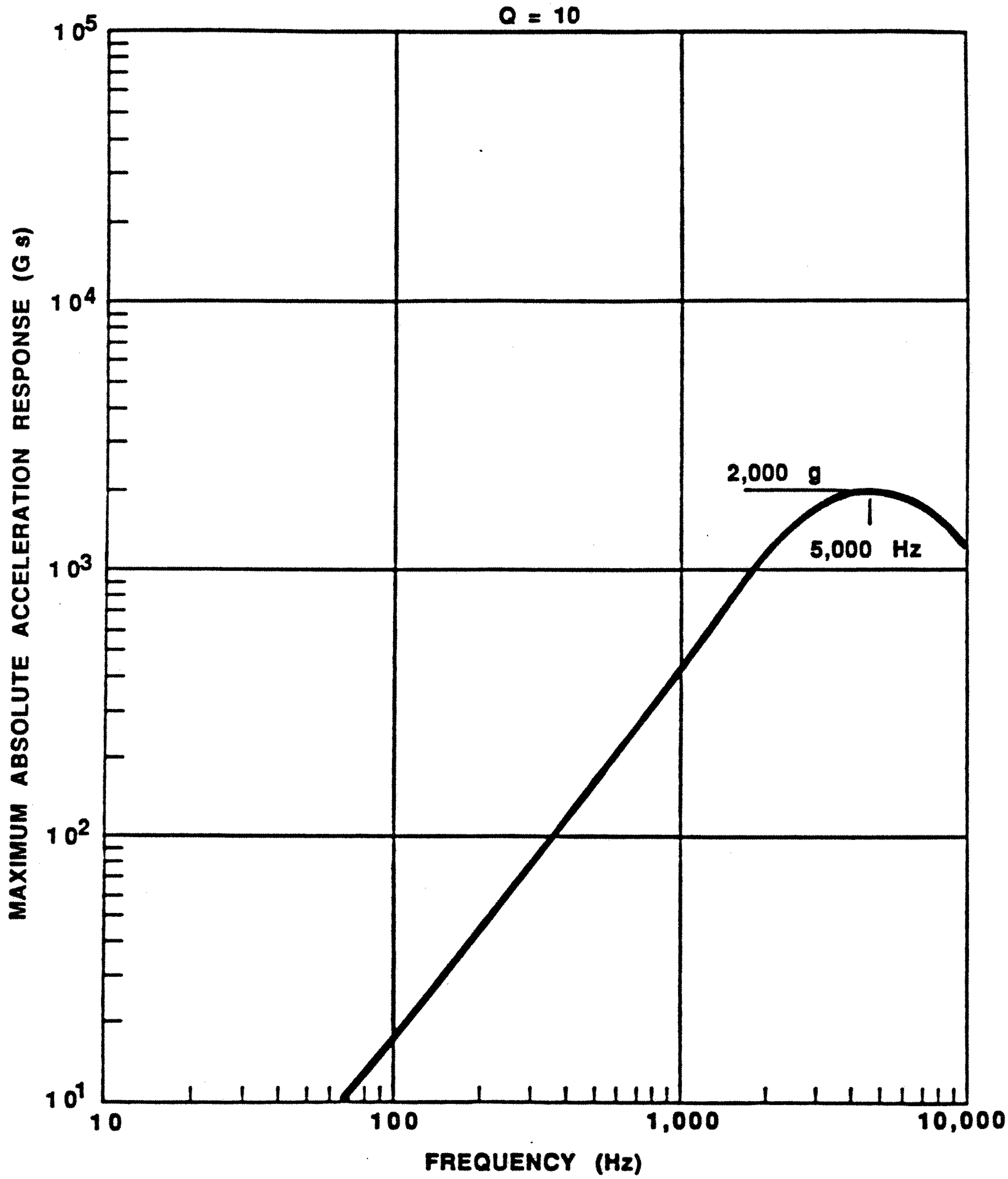


FIGURE 5.6.2-4 MAXIMUM 8 AND 22 HARDPOINT INTERFACE SHOCK FROM TIV OR CENTAUR

5.6.4 Liftoff Instrumentation System

A LOIS will be employed for some first flights and will be effective for the few seconds before the vehicle height, above the pad during launch, causes separation of the LOIS umbilical.

5.7 Sustained Acceleration

The Titan IV and vehicle mounted equipment are capable of withstanding the maximum longitudinal acceleration levels experienced during flight.

The SS-ELV-405 configuration represents the worst case for maximum longitudinal acceleration G-loading. This is due to the lower overall vehicle weight involved.

The maximum acceleration values are:

	SRM	SRMU
Stage 0	3.25 G	3.45 G
Stage I	5.00 G	5.00 G
Stage II	4.50 G	4.59 G

This is not a design environment for payloads, but is for the Titan IV Booster components. For Payload factor, see paragraph 5.9.

5.8 Thermal

5.8.1 Payload Fairing/Spacecraft

The thermal environment imposed on a Spacecraft during Launch and Flight is comprised of three different phases. During the initial portion of the ascent trajectory, the Spacecraft is enclosed in a PLF which prevents direct exposure to the ascent aerodynamic heating environment. Heat transfer to the Spacecraft during this period is primarily by radiation from PLF walls.

The PLF wall temperature history is a function of the ascent trajectory, PLF wall heat capacity and wall insulation (if any). Design ascent wall temperatures for the Titan IV/86-ft PLF are shown in Figure 5.8.1-1. This Figure can also be used for the Titan IV/IUS PLF (56-ft) wall temperatures. Radiation impact to the payload is moderated by installation of acoustic blankets in the cylindrical region of the PLF as shown in Table 5.2-1 and Table 5.2-2. The acoustic blanket temperature profile is also shown in Figure 5.8.1-1. The emissivity of the inner surface of the acoustic blanket is 0.86 ± 0.05. The emissivity of the unblanketed inner isogrid surfaces of the PLF is equal to or less than 0.30.

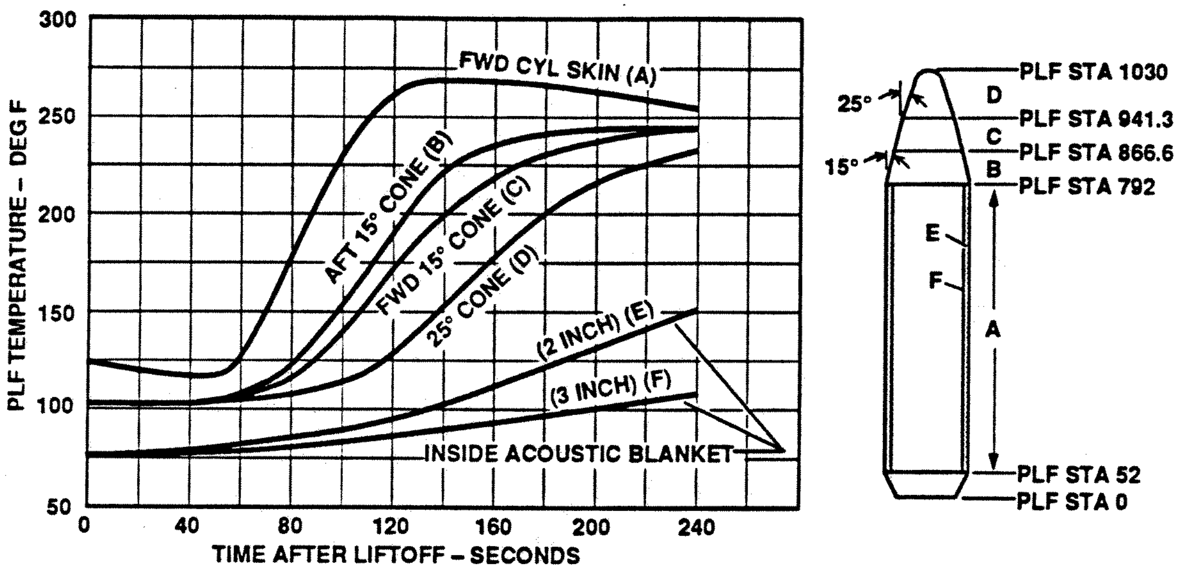


FIGURE 5.8.1-1 PLF TEMPERATURE ASCENT PROFILES ETR TITAN IV SYNTHETIC GENERIC TRAJECTORY – 86' PLF

5.8.1 Payload Fairing/Spacecraft (Continued)

The second phase of the ascent thermal environment imposed on the Spacecraft occurs at the time of PLF separation. The PLF is jettisoned as soon as possible for performance considerations, but is delayed until FMH and dynamic pressure effects are acceptable to the Spacecraft. After PLF jettison, the Spacecraft is exposed to the effects of Free Molecular Heating (FMH). The dispersed FMH is payload dependent and usually on the order of one-half (1/2) the solar constant (200 to 250 Btu/ft^2-hr). Following PLF separation, the free molecular heating decreases and then increases. A second peak occurs at the time of park orbit insertion. With the Upper Stage, a third peak may be experienced near perigee of the transfer orbit. The relative magnitude of the latter two peaks depends on the perigee altitude and transfer burn timing. If these peaks exceed allowable limits, they can be lowered by increasing the park orbit perigee.

The final phase of flight has ascent heating resulting from direct and reflected solar radiation, and direct earth infrared radiation. These radiation levels can be controlled during the park and transfer orbits by means of thermal control maneuvers by an Upper Stage or the Spacecraft. Typical maneuvers performed in previous missions include toasting, metronome, rotisserie and continuous roll, reference Figure 5.8.1-2.

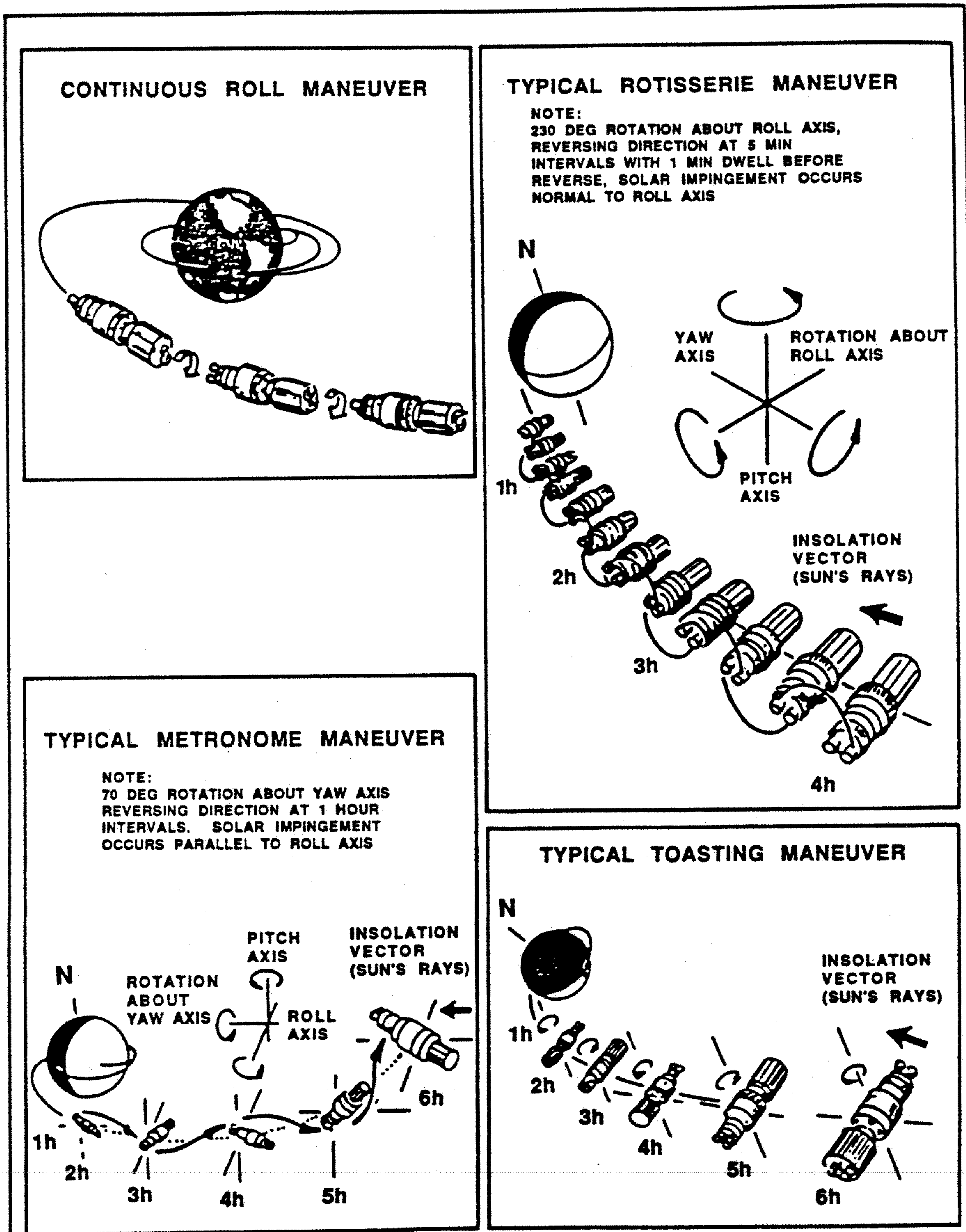


FIGURE 5.8.1-2 TYPICAL THERMAL MANEUVERS

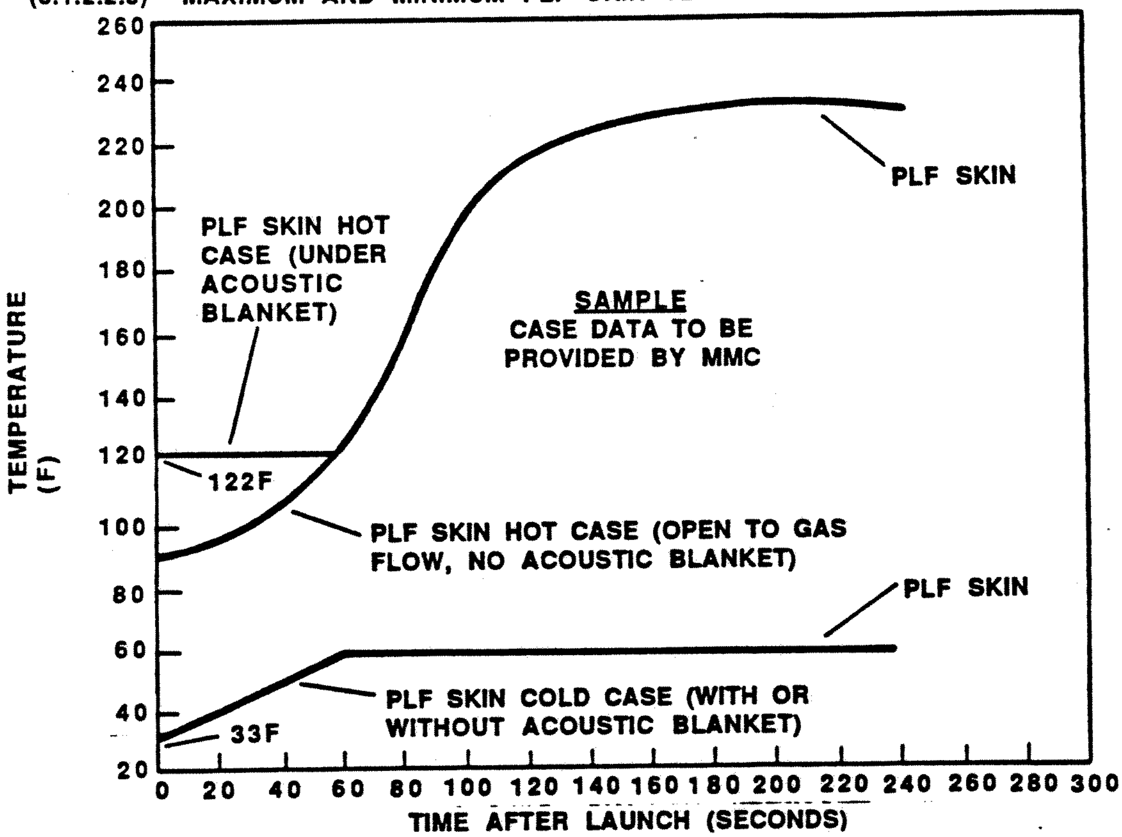
5.8.2 Centaur/Spacecraft

The average temperature of the 8 SV/Centaur interface mechanical attachments for the 10-ft diameter interface will be in the range of (–105° F to +100° F) when the attach points are considered adiabatic. At the time of SV/Centaur separation, the maximum temperature difference between any two attach points will not exceed 100° F.

The temperatures at the SV Adapter/Centaur interface at the 22 attach points (Titan IV STA –105.519) during flight will be in the range of (–175° F to 100°F). The temperature difference between any two of the 22 attach points does not exceed (100°F). During ground processing from PLF closure to launch, the 22 attach points temperature extremes are (–75° to 125° F).

The generic Centaur Thermal Requirements up to PLF jettison are identified in Figures 5.8.2-1 and 5.8.2-2.

- (3.1.2.2.3) MAXIMUM AND MINIMUM PLF SKIN TEMPERATURES:



- (3.1.2.2.3) EFFECTIVE ACOUSTIC BLANKET AND PLF SKIN EMITANCES:
  - BLANKET = 0.86
  - PLF SKIN = 0.30

FIGURE 5.8.2-1 GENERIC CENTAUR THERMAL REQUIREMENTS

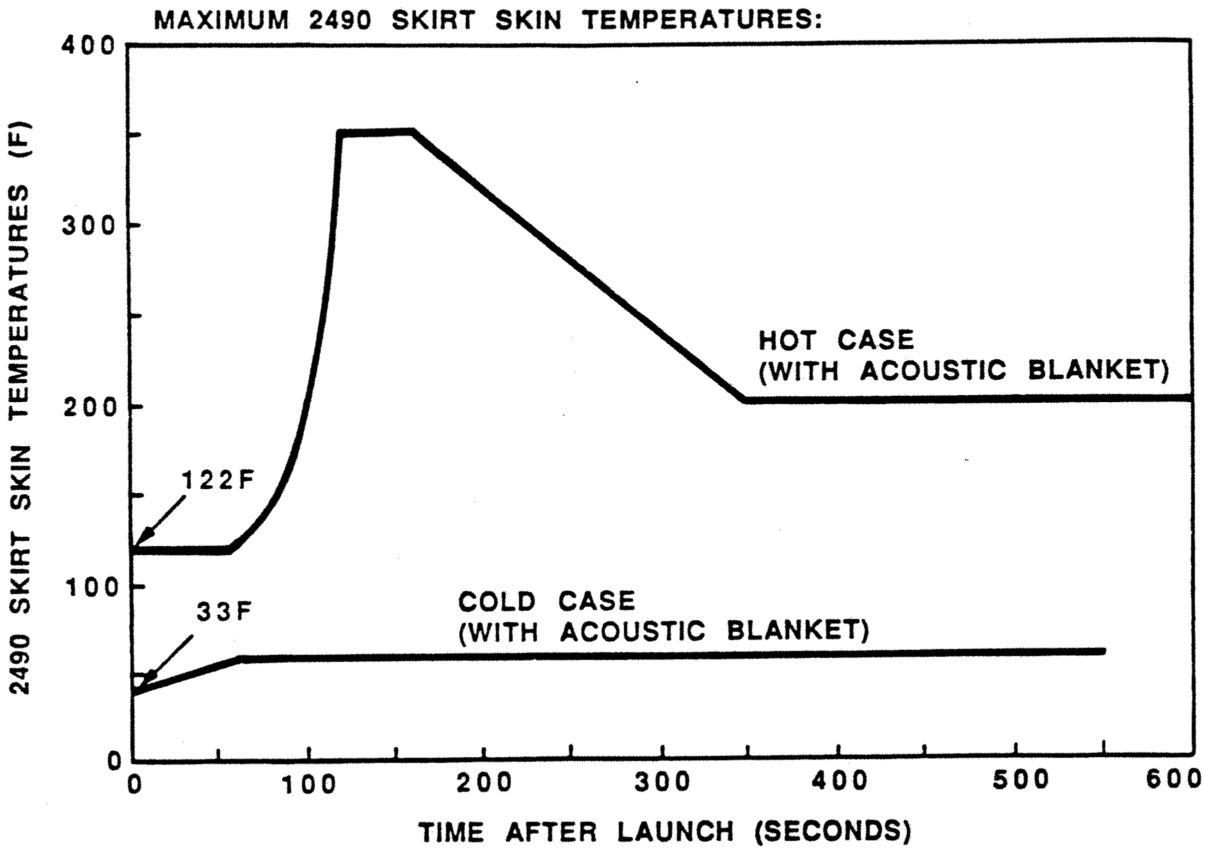


FIGURE 5.8.2-2 GENERIC CENTAUR THERMAL REQUIREMENTS

5.8.2 Centaur/Spacecraft (Continued)

While in the park orbit, the Centaur shall have the capability to be oriented with the roll axis normal to the ecliptic plane within ±10 deg and to maintain a commanded roll rate selected between 0.5 to 1.0 deg/sec in either direction. The Centaur shall also have the capability to be oriented with the forward-directed roll axis coincident with the velocity vector ±10 deg.

While in the transfer orbit, the angle between the Centaur roll axis and the solar vector shall be selectable by the user program within the range between 80 and 100 deg. Allowable operation tolerance shall be ±6 deg. The Centaur shall be capable of providing a commanded roll rate selectable between 0.5 and 1.0 deg/sec in either direction.

5.8.2.1 Rocket Engine Module (REM) Analysis

The Rocket Research Corporation (RRC) will provide the REMs for the Titan/Centaur. RRC will provide the rocket engine assembly thermal model to GDSS, who in turn will perform the analysis to predict REM thermal performance during Titan/Centaur environmental conditions.

5.8.3 IUS/Spacecraft

The IUS and Spacecraft are thermally isolated with respect to radiation heat transfer by a multilayer insulation blanket with an aluminized Beta cloth outer layer. On the Spacecraft side, the effective emittance is less than 0.02. The IUS cavity temperatures can range between 35 and 109°F. The thermal interface between IUS and the Spacecraft is subject to analyses performed by the IUS and the Spacecraft contractors.

5.8.4 Aerothermodynamic Trajectory Histories

Reference Figure 5.8.4-1 for TIV/401, TIV/402 and TIV/405 Aerothermodynamic Histories with SRMs.

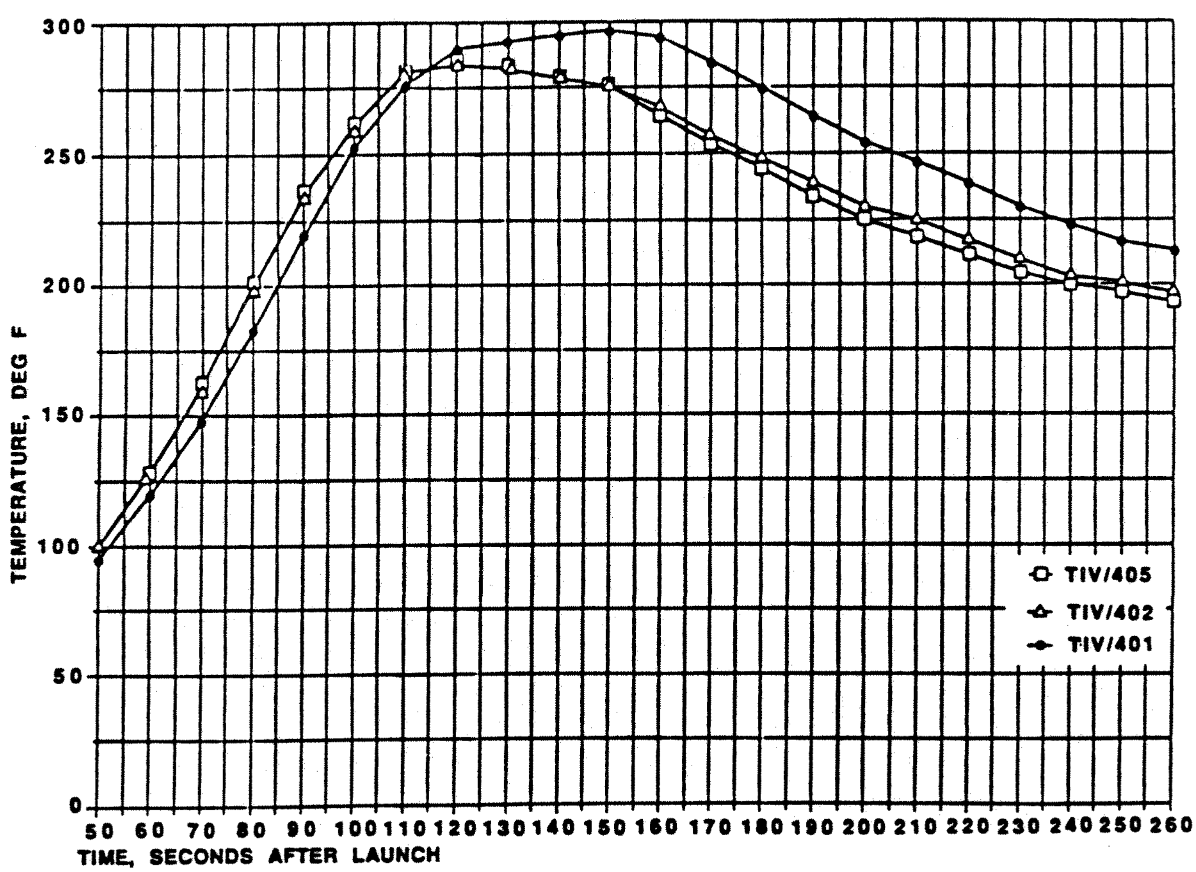


FIGURE 5.8.4-1 AEROTHERMODYNAMIC TRAJECTORY HISTORIES

5.9 Design Load Factors

Load factors are provided in Table 5.9-1 for the current booster configuration and initial sizing and/or evaluation of Spacecraft primary structure. Spacecraft response is a function of Spacecraft weight, stiffness and lateral/axial coupling as well as booster configuration. The load factors are intended to provide a conservative design envelope which includes a range of variation in these parameters. Transient loads analyses performed during the development of a Spacecraft will provide the detailed member loads required for complete design and evaluation of the structure including interface loads and envelope clearance.

TABLE 5.9-1 SV LOAD FACTORS FOR TITAN IV

Event	Axis	Steady State Value Gs	Dynamic Value Gs
Lift-off	Axial	1.7	±1.5
	Lateral	0	±3.5
	Torsion	0	±.05/in.
Max Air Loads	Axial	2.0	±1.0
	Lateral	0	±3.0
	Torsion	0	±.05/in.
Stage I Burnout	Axial	0 to 4.0	±2.5
	Lateral	0	±2.0
Stage II Burnout	Axial	0 to 3.0	± 2.5
	Lateral	0	±1.5

Total axial load factors can be obtained from the table by the addition of the steady-state and dynamic components. Lateral loads at the extremities of the Spacecraft may exceed the c.g. load factors. The following equation may be used to estimate the distribution of lateral loads as a function of longitudinal distance (x) from the Spacecraft/Launch Vehicle Interface, where g_{cg} is the lateral c.g. load factor from the load factor table and x_{cg} is the distance from the interface to the Spacecraft c.g.

$$g_x = (3/5)\, g_{cg} * (x/x_{cg}) + (2/5)\, g_{cg}$$

5.9 Design Factors (Continued)

The following frequency criteria are recommended to minimize Vehicle/SC interaction:

1. Maintain lateral first mode frequency above 2.5 Hz.

   – Low frequency lateral modes can affect Launch Vehicle Control System performance.

2. Significant SV lateral excitation can occur in the 4-10 Hz range due to the liftoff and maximum airload events.

3. Significant SV axial excitation can occur in the 17-24 Hz range due to Stage I Shutdown thrust oscillations.

5.9.1 Verification Loads Cycle

To complete the Verification Loads Cycle (VLC) on a timely basis it is imperative that the SV adapter model and the SV model from the user be available at least six months prior to the scheduled VLC loads delivery date.

The SV load model and the flight control modes are also needed to complete the autopilot verification.

5.10 Solid Rocket Motor Ignition Induced Overpressure Suppression System

a. The SRM start transient induced overpressure is a component of the Titan IV liftoff forcing function as is the SRM/SRMU start-up nozzle flow separation. An Overpressure Suppression System (OSS) is being developed to reduce the peak impulse of the SRM start transient a minimum of 50% (60% reduction required for SLC-4E) on LC-40, SLC-4E and is in place and operational on LC-41.

   Other liftoff forcing function components are wind induced oscillation and launch stand misalignment. These components of course are not affected by the OSS.

b. The purpose of the OSS is to reduce the SRM/SRMU ignition transient overpressure resulting from interaction between SRM/SRMU exhaust and launch pad exhaust duct.

c. The reduction is achieved by injecting water droplets into the launch plume.

5.11 Winds

5.11.1 Ground Wind Constraints

The Titan IV Vehicle configurations have sufficient structural capabilities to withstand the loads imposed by the 99.9% (ESMC) surface wind profiles for prelaunch and liftoff conditions. These wind profiles are defined in MMC Document No. MCR-76-553 "Atmosphere and Wind Criteria for the Titan 34D Vehicle," 1976. The Vehicle/Umbilical Tower collision constraint requires wind placarding over a limited wind azimuth range at ESMC.

5.11.2 Winds Aloft

The Titan IV vehicle is very sensitive to aerodynamic loading during Stage 0 Flight, and, therefore, the wind environment during launch must be taken into account. Aerodynamic loading is kept to a minimum by measuring the wind environment approximately two hours prior to launch and then tailoring a set of pitch and yaw steering commands and initial rollout/pitchover maneuvers to the observed wind. These commands are verified via 6-D simulation and then loaded into the on-board flight computer approximately 1/2 hr prior to launch. During launch countdown after T–2, additional wind data will be gathered to verify that actual persistence effects are not exceeding design persistence allowances.

Reference: Aerospace TOR-0075 (5702-02)-1 "Titan IIIE/Centaur Wind Persistence Study," 23 April 1975.

5.11.2.1 Titan IV Winds Aloft Placarding Process

The Titan IV DOL winds aloft placarding process involves generating vehicle steering parameters to minimize aerodynamic loading on the Titan IV vehicles during Stage 0 flight. This process uses measured wind data from balloons released up to T–2:30, for steering and measured wind data from balloons released T–2:30, T–2:05, T–1:20 and T–1:00 for placarding of airloads and the TVC system. The placard values which support a launch Go or No-Go assessment are defined by analyses performed prior to and during targeting. These analyses utilize 99% probability design wind profiles, a library of actual winds, definitions of 3-sigma wind persistence and 3-sigma hardware dispersions to determine Go or No-Go launch conditions. The DOL simulation activities and placard analyses are performed in the Titan IV Winds Room located at the Denver Division South Park West (SPW) facility.

A placard represents the level of criticalness for a particular simulation parameter for a nominal vehicle in the presence of winds. If the launch day wind causes a particular simulation parameter for a nominal vehicle to reach the critical placard value, then exactly enough reserve exists in the launch vehicle system to ensure that a 3-sigma vehicle, in the presence of a 3-sigma environment, will still satisfy mission requirements. The simulation parameters used for the Titan IV program are associated with structural loads and the TVC system. The magnitudes of these parameters, computed in trajectory simulations made during the countdown, are compared with pre-determined constraints in order to recommend a launch Go or No-Go.

5.11.2.1 Titan IV Winds Aloft Placarding Process (Continued)

In order that a more definitive quote (compared to a simple Go or No-Go) may be given, the status is reported in terms of Capability Ratios (CR) for each of the constraints. A CR is the value of the parameter (as computed in the trajectory simulation) divided by the predetermined maximum allowable value. If the CRs should be, at any time during the trajectory, greater than 1.0, a launch scrub may be recommended. In this situation, as time permits, new steering parameters will be generated and replacarding will be performed using these updated parameters based on the data from later balloon releases. The trajectory program is structured so that a time history of each of the CRs is in the computer, and the highest value of each is presented in a summary display in the winds monitoring room. This procedure comprises the placarding analysis.

On Day-of-Launch, the steering parameters contained in the Titan IV booster vehicle guidance computer (MGC) are updated based on the results of an optimization simulation containing the measured wind data. This simulation determines the steering parameters that will result in minimum aerodynamic loading on the Titan IV vehicle for a given measured wind.

The steering parameter update procedure is performed following receipt and processing of wind data from balloons released at T–24:00, T–6:30, T–4:30 and T–2:30. Independent verification and validation of the software parameters used for the steering and separate trajectory validation are also performed with the wind data from these balloons. The placard simulations are performed for each set of wind data based on the corresponding set of steering parameters. The final placard simulations, generated with wind data from balloons released closest to launch, are essential to maintain vehicle reliability and launch availability.

For a nominal, on-time launch, the final Go or No-Go recommendation is based on the combined results of the placard simulations generated using T–2:30 steering parameters for the T–2:30, T–2:05, T–1:20 and T–1:00 balloons. These results, which determine the Go or No-Go conditions, are passed from SPW to MMAG management at the launch site. MMAG management then provides a Go or No-Go recommendation to the USAF Launch Test Director.

Chapter 6

Launch Preparation Operations at CCAFS

6.0 LAUNCH PREPARATION OPERATIONS AT CCAFS

6.1 Introduction

This chapter addresses the Receipt-to-Launch support at Cape Canaveral Air Force Station (CCAFS), Florida, reference Figures 6.1-1, 6.1-2 and 6.1-3. Reference Payload Support Capability document CCAFS/LC-41 MCR-88-2638.

6.2 Integrate-Transfer-Launch (ITL)

The ITL concept utilized at CCAFS is designed to provide for four launches per year for that series of the first 23 Titan IV launch vehicles built and scheduled for launch at CCAFS. The ITL facilities principally consist of the Vertical Integration Building (VIB), Solid Rocket Motor Processing Facilities, Solid Rocket Motor Assembly Buildings (SMAB and SMARF) (one for SRMs and one for SRMUs), Titan Transporter Railroad System, and Launch Complex 41 (LC-41).

LC-41 mainly consists of the Launch Pad, Mobile Service Tower (MST), Umbilical Tower (UT), AGE Building, Air Conditioning Shelter, Gas Storage Area, propellant holding areas and miscellaneous service facilities, reference Figures 6.2-3, 6.2-2 and 6.2-1.

In this ITL concept, the Core Vehicle Stages are received, assembled vertically on a mobile transporter and checked out in the VIB. The Core Vehicle is then transported to the SMAB or the SMARF via the railroad facilities. After receiving and inspection, the SRMs are partially assembled and checked out in the SMAB. The SRMUs will be partially assembled and checked out in the SMARF. The Core Vehicle and SRMs/SRMUs will be joined on the transporter and the combination moved to LC-41, positioned, configured to the UT and then enclosed by the MST.

In the MST, the remaining SRM/SRMU components are installed, the upper stage (if required) is installed and the configuration checked out. The Spacecraft is then installed and mated to the Launch Vehicle assembly and checked out. When access requirements permit, the PLF is installed.

The ITL concept requires equipment to support a vehicle in launch readiness condition at the launch complex and to support a vehicle undergoing preliminary checkout in the VIB. This requirement is met by mounting checkout and support electronics in two vans which are connected to the transporter prior to vehicle erection and remain connected to that transporter and vehicle through Checkout and Launch. Control Center electronics and the Instrumentation Ground Station are located in the VIB and can be used for Launch or for VIB testing.

General hazards and safety considerations will be covered in in the Martin Marietta System Safety Engineering Manual M64-125 and the ACRBC, Appendix C, MMC Safety Organization and General Hazards/Safety Considerations.

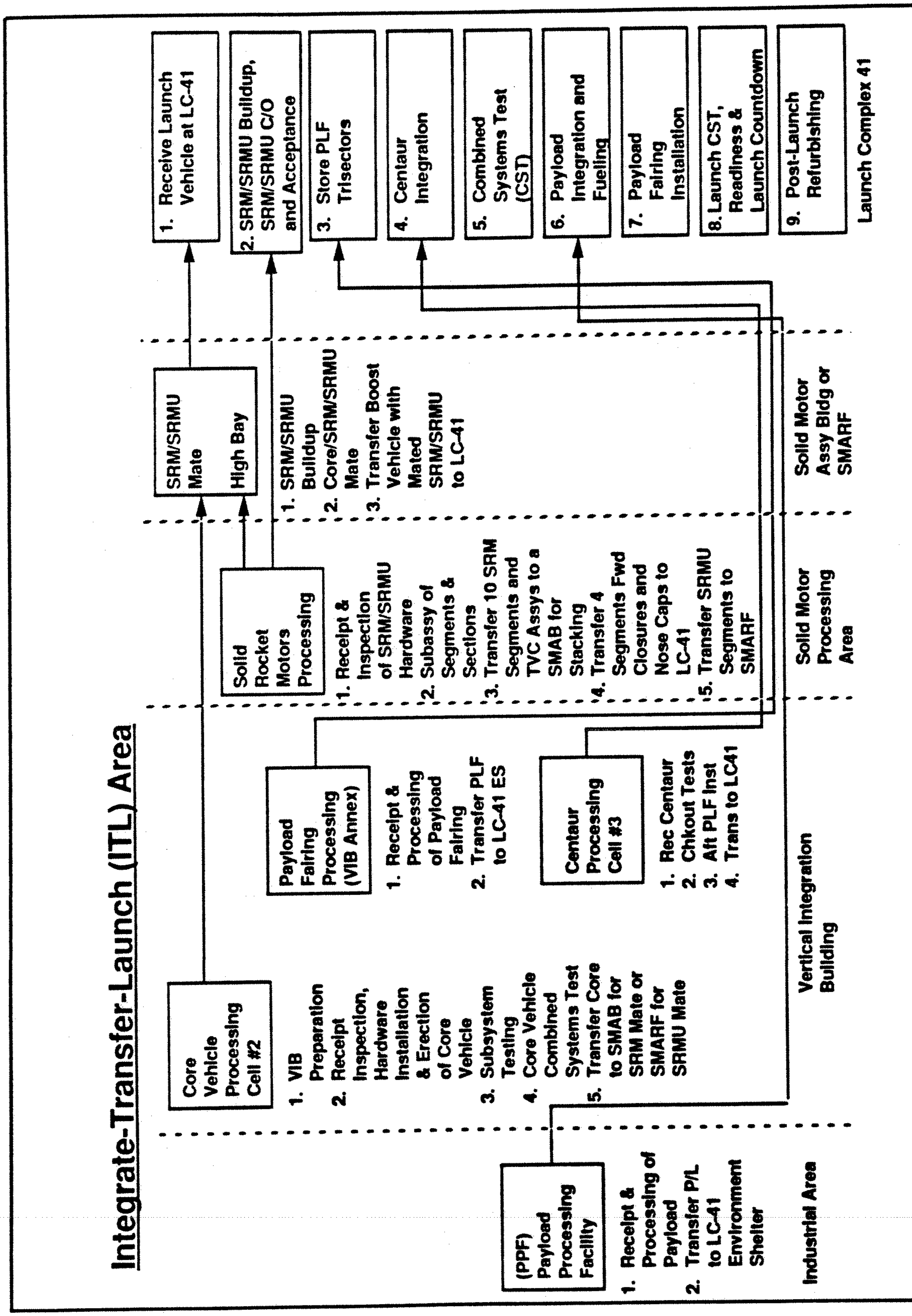


FIGURE 6.1-1 RECEIPT-THROUGH-LAUNCH ACTIVITIES – CENTAUR

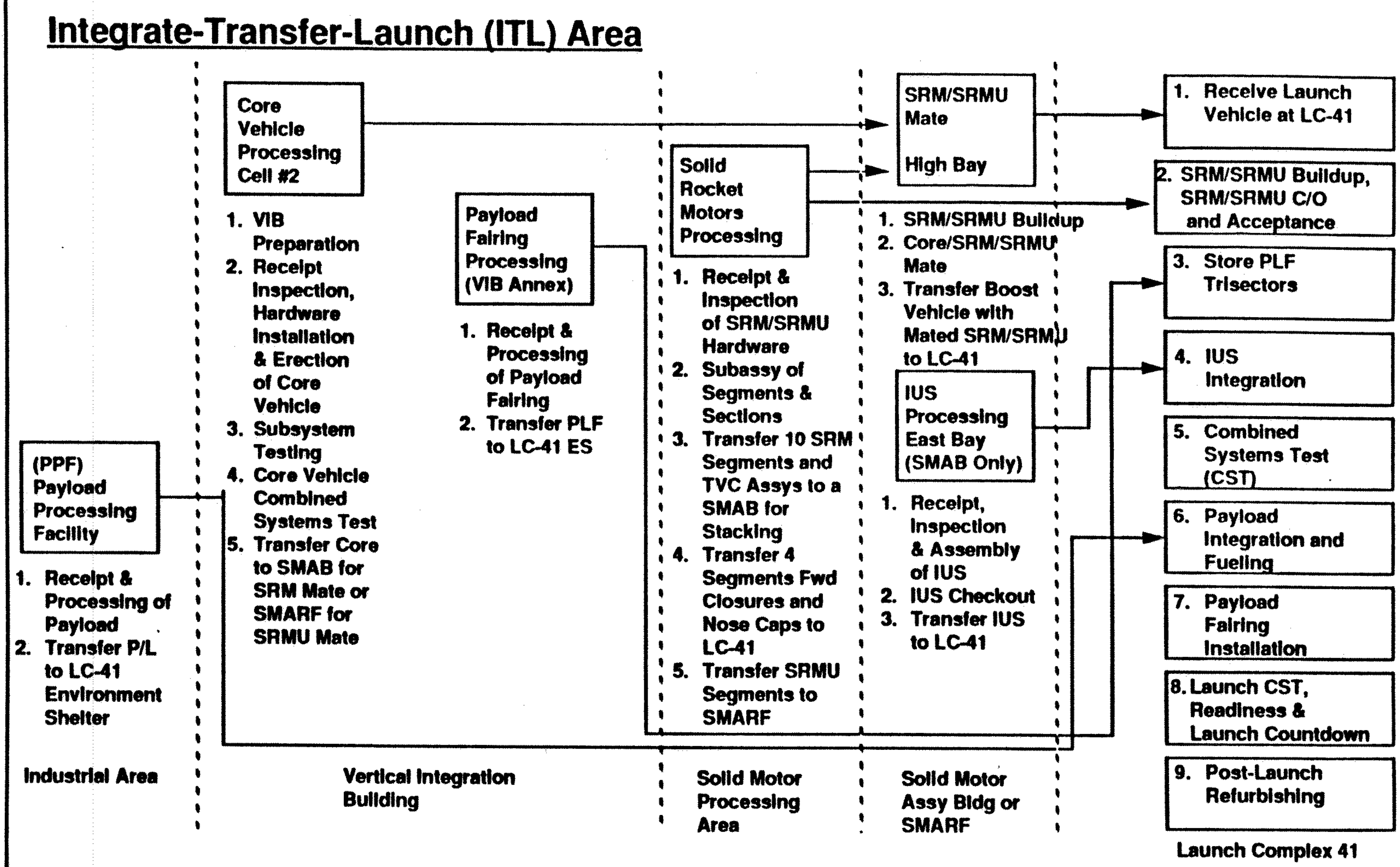


FIGURE 6.1-2 RECEIPT-THROUGH-LAUNCH ACTIVITIES – IUS

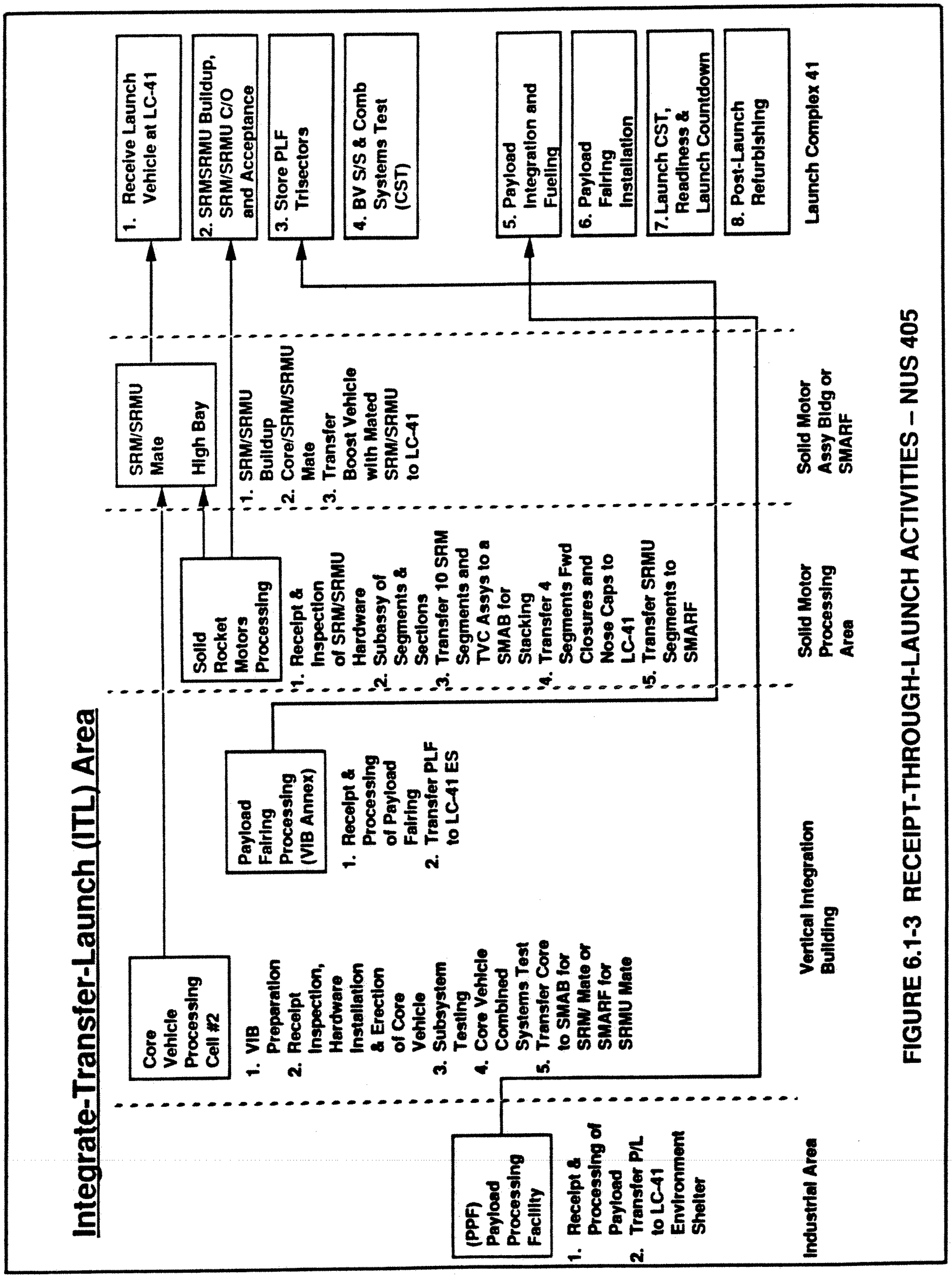


FIGURE 6.1-3 RECEIPT-THROUGH-LAUNCH ACTIVITIES – NUS 405

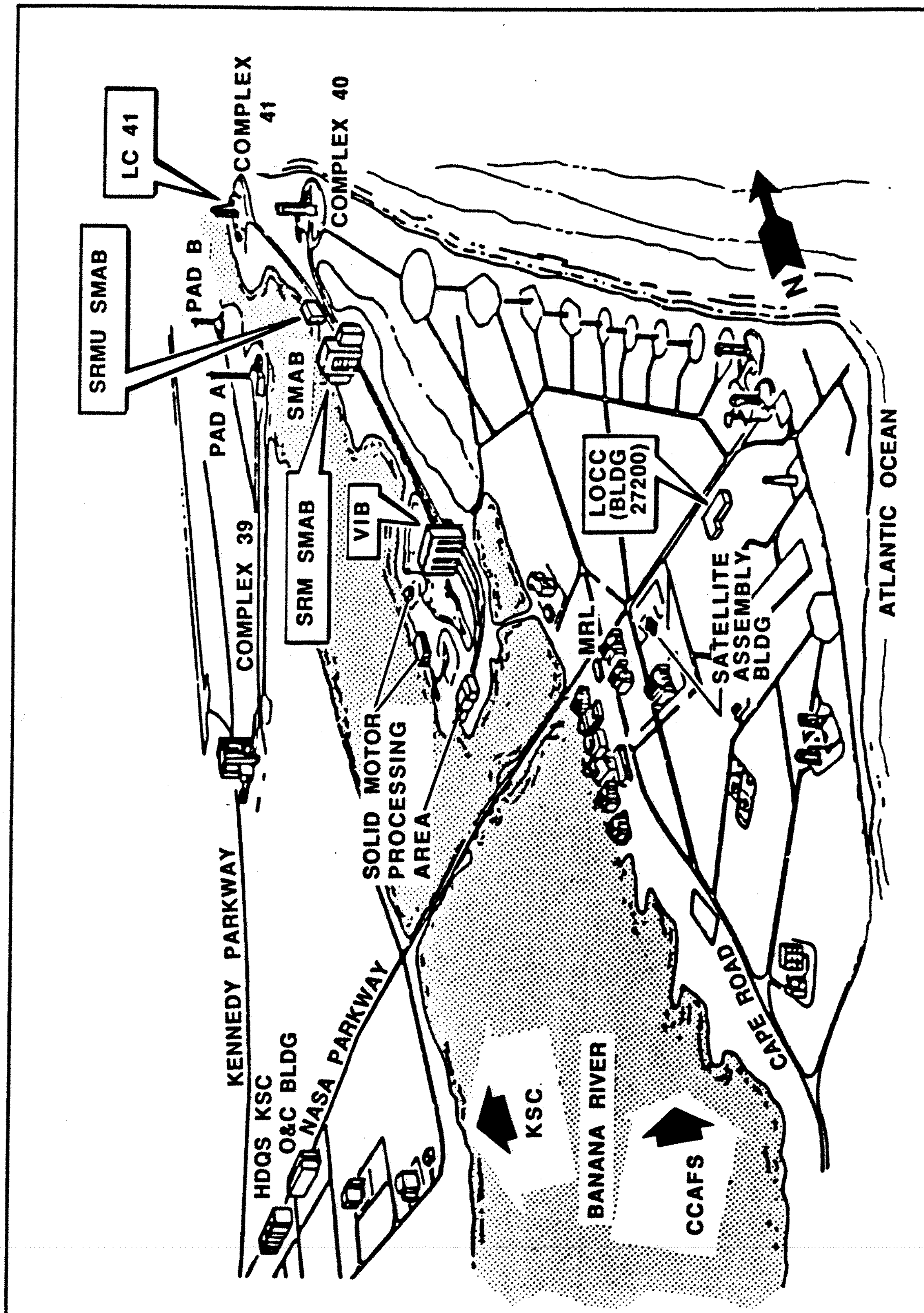


FIGURE 6.2-1 CAPE CANAVERAL AIR FORCE STATION

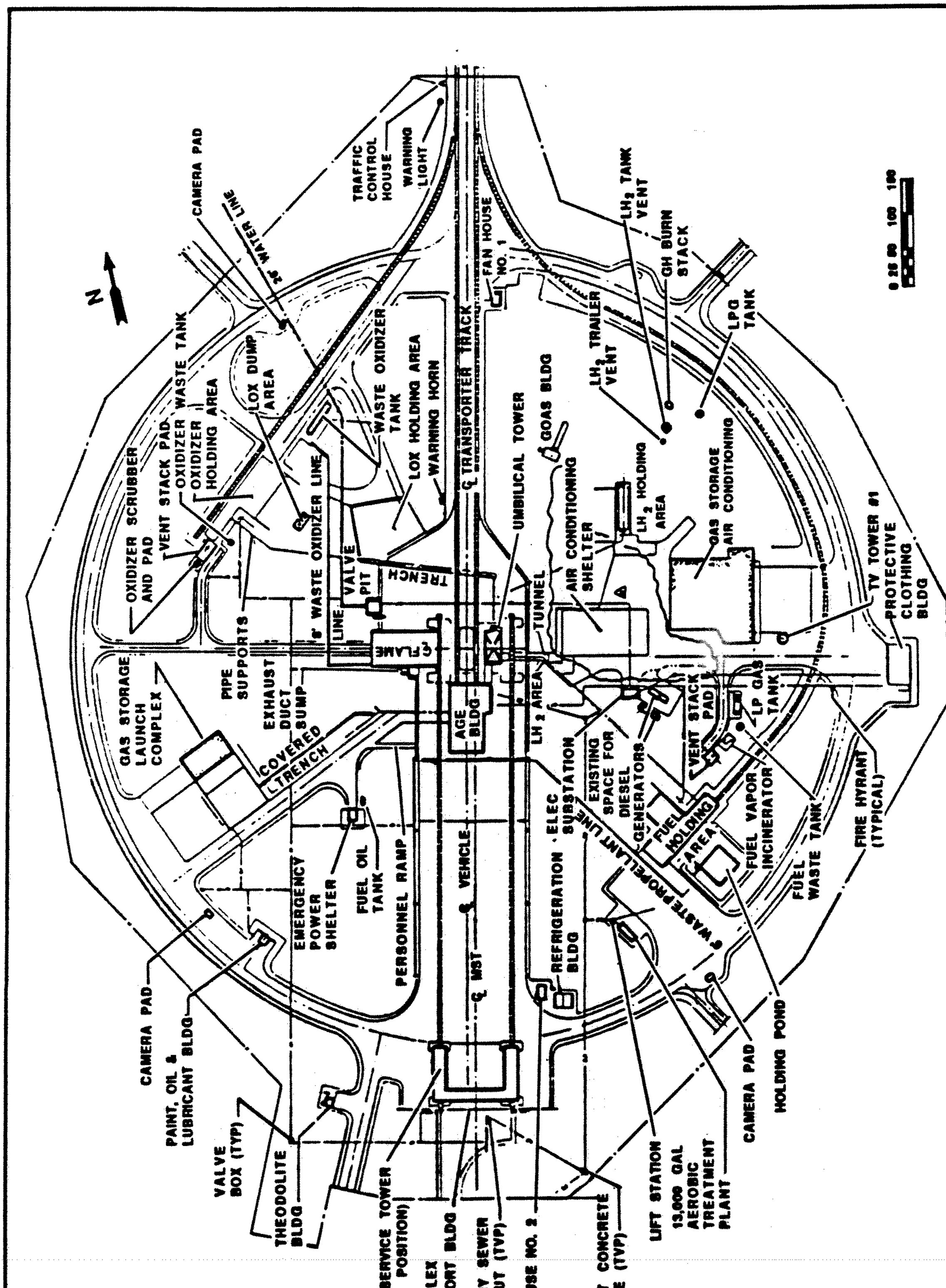


FIGURE 6.2-2 LAUNCH COMPLEX 41

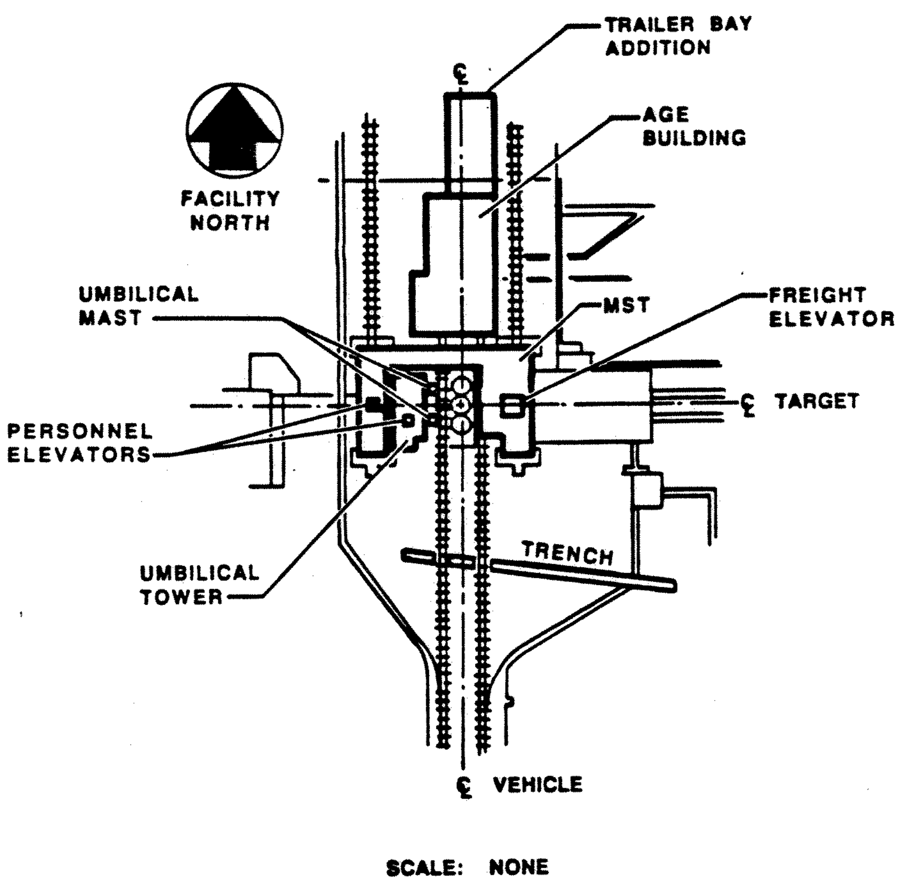


FIGURE 6.2-3 KEY PLAN LAUNCH COMPLEX 41

6.2.1 Test Sequence

After the required Upper Stage is installed at the MST, the Acceptance Combined Systems Test (A/CST) is performed. The A/CST is the final integrated test of the Titan IV/Upper Stage and along with the previous subsystem tests demonstrates the acceptability of the LVS for customer program requirements.

The SV is next installed, the required interface testing performed, and the PLF installed. The Launch CST is now performed and provides a final launch readiness verification of all Titan IV/Upper Stage/SV Systems.

After a successful Launch CST, a Readiness Countdown is performed during which ordnance is installed and connected, propellants loaded, stray voltage tests performed, command and control checks performed, and Launch Countdown started.

The Launch Countdown includes SV tests, MST move to park position, IMU alignment and calibration, vehicle verification, command control checks, Terminal Countdown and vehicle launch initialization.

6.2.2 Mobile Service Tower (MST)

The MST provides facilities for mating the IUS or the Centaur and/or the SC to the Titan IV, and for the servicing and checkout of the complete integrated flight vehicle system. Work platforms at nineteen levels provide access to the vehicle, upper stages, SC and PLF. A 15-ton hook on the bridge crane is used to handle PLF and SC. The bridge crane also has a 50-ton hook which is used for installation of the IUS and Centaur. The entire structure is mounted on four self-propelled, electric motor-driven trucks that ride on rails. Just prior to launch, the MST is moved from the service position to its park position, north of the launch mount, reference Figure 6.2.2-1.

The Universal Environmental Shelter (UES) is an integral part of the MST and provides a controlled environment for the SC and Upper Stage and supports temporary PLF storage. Umbilical slots on the west side of the shelter are provided for electrical, mechanical and air conditioning umbilicals for the Upper Stage and SC. These umbilicals can be left connected while the MST is moved back for launch. One large door on the shelter's south side is open to receive the PLF, SC and Upper Stages.

Nine facility work platforms in the environmental shelter provide access to the Upper Stage, SC and PLF. The platforms are approximately 10-ft apart in elevation, and each has an area of approximately 30-ft by 40-ft. The platforms are hinged and folded up to permit installation of the Vehicle, Spacecraft and PLF. Platform adapters and auxiliary working platforms are used in conjunction with the facility platforms to provide 360-degrees access at each level, reference Figure 6.2.2-2, 6.2.2-3 and 6.2.2-4.

There are three elevators at the LC-41 Launch Complex, a freight elevator and a personnel elevator in the MST, and a personnel elevator in the UT, reference Fig 6.2-3.

The capacity of the MST freight elevator is 12,000 lb and the loading is given in the design criteria. The elevator has an 8-ft by 8-ft door opening and is available for transporting Upper Stages/SC, AGE and Test Equipment.

Refer to Integrate-Transfer-Launch Facility Modifications/Facility Design Criteria – Titan IV/CCAFS document MCR-85-2505, Revision W of 9 June 1989, for UES Requirements and Specifications.

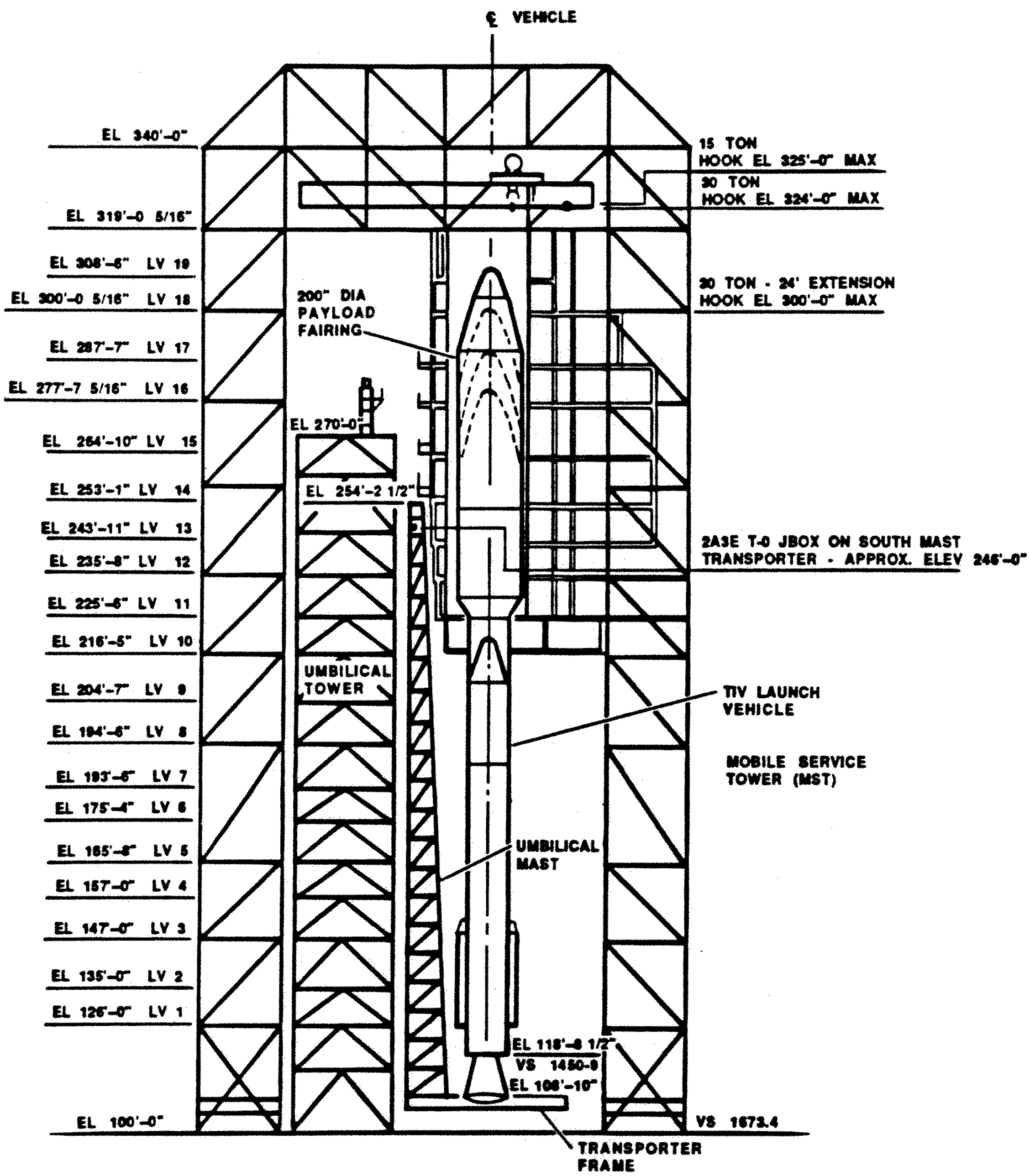


FIGURE 6.2.2-1 SOUTH COMPOSITE ELEVATION MST

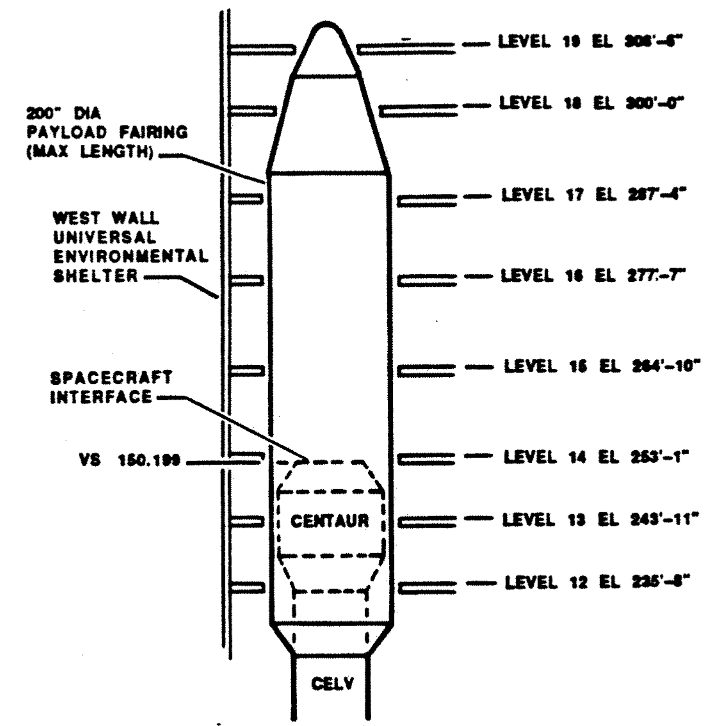


FIGURE 6.2.2-2 UES PLATFORM ELEVATION FOR TITAN IV/CENTAUR

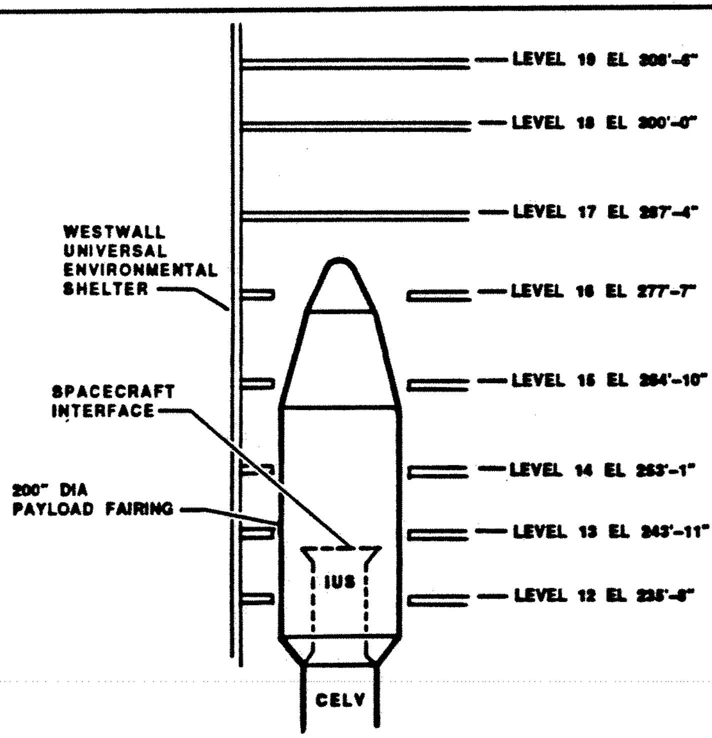


FIGURE 6.2.2-3 UES PLATFORM ELEVATION FOR TITAN IV/IUS

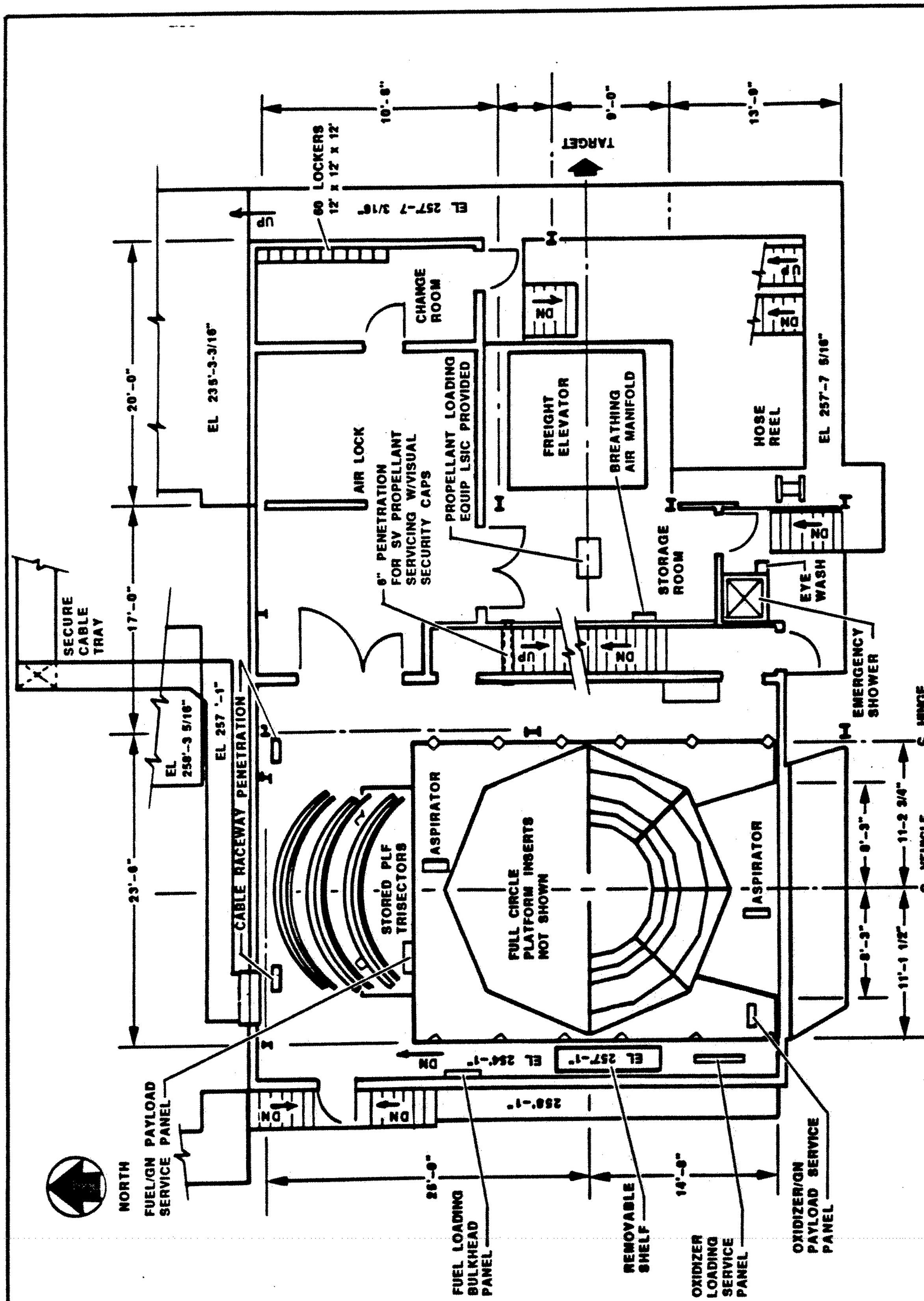


FIGURE 6.2.2-4 PARTIAL PLAN MST LEVEL-14 EL 253'-1" UNLESS NOTED

6.2.3 Umbilical Tower (UT)

The UT provides connections for propellants, pressurization gases and conditioned air to both the Launch Vehicle and to the PLF. Installations on the tower have been provided to accommodate both manual and launch-disconnected umbilicals.

The UT has sliding platforms that provide access to the Launch Vehicle and the umbilical installations. The upper platforms provide access to umbilical installations for the Upper Stages, SC and PLF, reference Figures 6.2.3-1 and 6.2.3-2.

6.2.4 Satellite Assembly Building (SAB), Building 49904

The SAB is operated by the Spacecraft Division of the 6555 ASTG. The SAB consists of administrative areas, high bay, low bay, two control centers and the Eastern Vehicle Checkout Facility (EVCF). The SAB can be used to build up and checkout P/Ls prior to mating with the Titan IV booster at LC-41.

6.2.5 Vertical Integration Building (VIB)

The VIB is used to receive, inspect, assemble, erect and check out the Titan Core Vehicle. For Titan IV, the VIB has two assembly and checkout cells; one for the core vehicle, Cell 2; and one for the Centaur, Cell 3. Additionally, the VIB contains a low bay area, the backup launch control center, the Titan IV Core checkout computer system and an IUS checkout station. The VIB houses the PLF Processing Facility where receiving, inspection and the PLF launch preparation operations take place. A low bay area near each cell provides space for the Checkout and Instrumentation AGE Vans.

6.2.5.1 Inertial Upper Stage (IUS) Checkout Station (COS)

The IUS COS is the instrumentation/telemetry "ground station" used to process the Inertial Upper Stage. The COS is located on the 4th Floor, NW corner of the VIB (Room 415), and consists of a Radio Frequency Interference (RFI) shielded room called the "Red Room" and a support room called the "Black Room". The "Red Room" is further divided into three separate control stations and an Advanced Decom System (ADS-100). Each control station is capable of independent operation that allows for checkout of multiple IUS(s) at several locations. During a launch, the three stations and ADS room exclusively support the operation. The "Black Room" contains two multiple-rack equipment bays used to process/record data entering/exiting the COS.

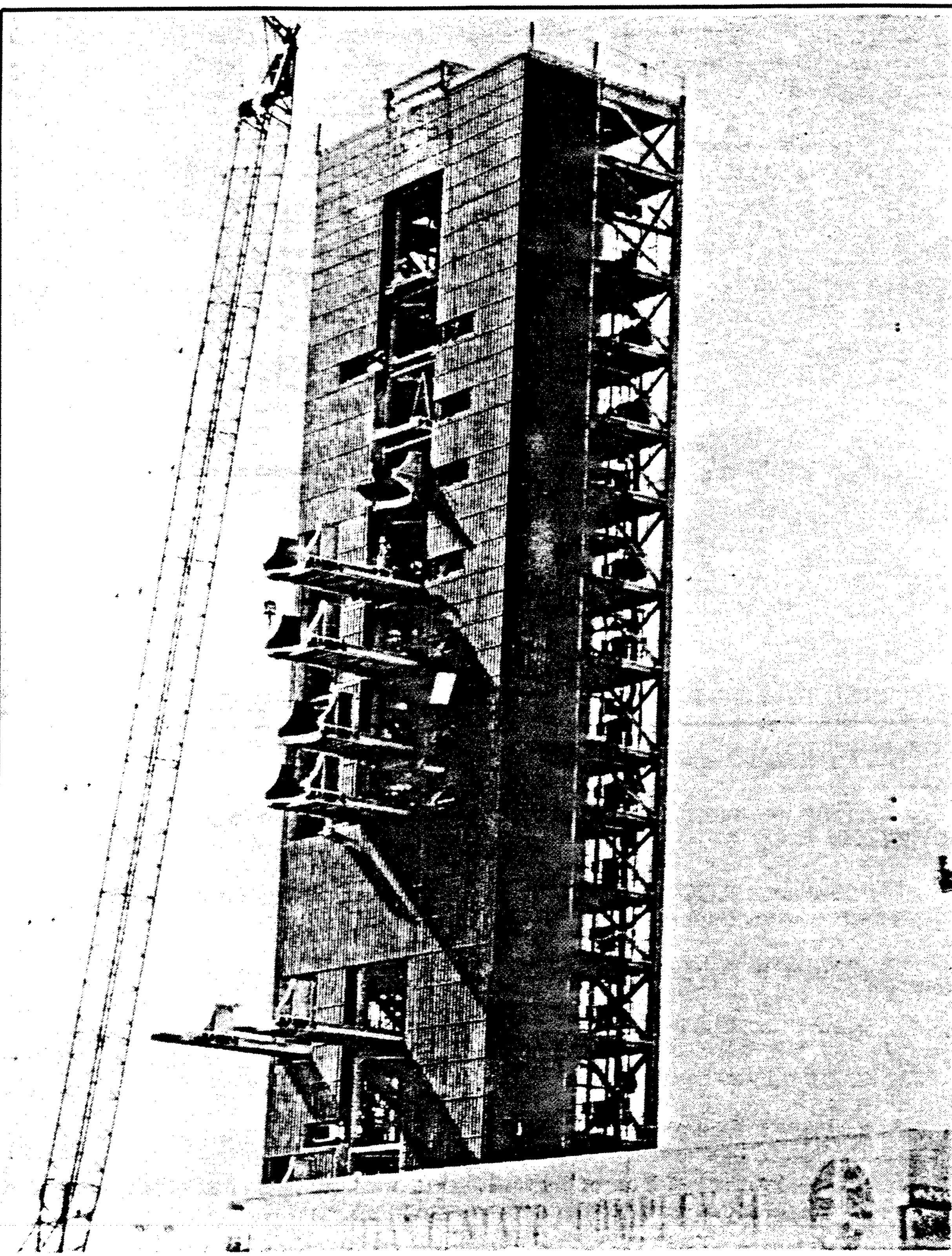


FIGURE 6.2.3-1 LC-41 UT/RETRACTABLE PLATFORMS

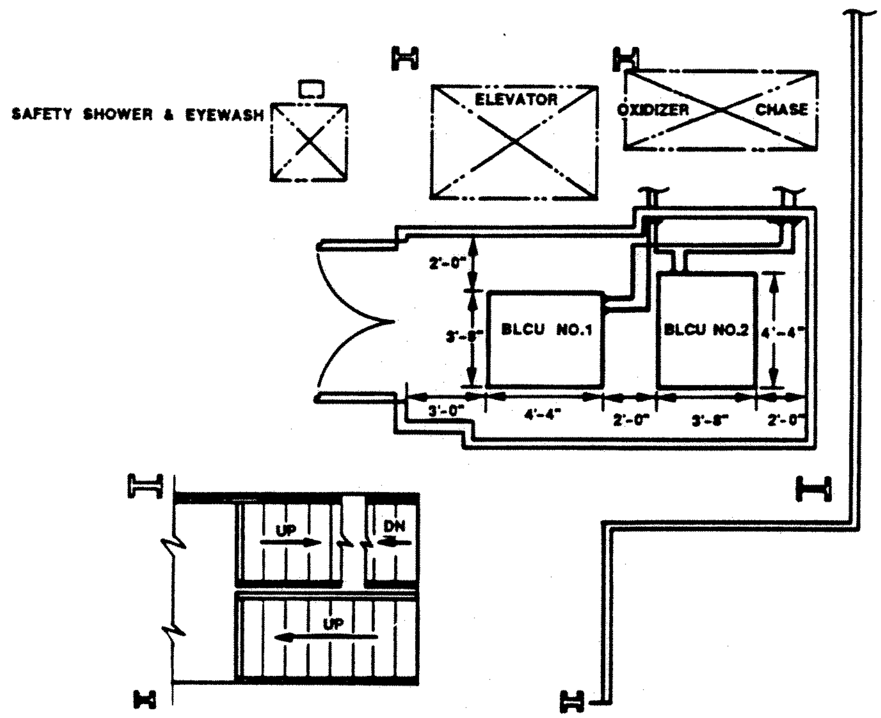


FIGURE 6.2.3-2 PARTIAL PLAN UT LEVEL 12 - EL 235'-8"

6.2.6 Launch Operations Control Center (LOCC)

The Titan IV Launch Operations Control Center (LOCC) is located in Building 27200.

The purpose of LOCC is to provide a consolidated location for checkout and launch operations for:

a. Titan IV Subsystem Testing in VIB cell 2 and at LC-41

b. Centaur Subsystem Testing in VIB cell 3 and at LC-41

c. Combined System/Subsystem Tests at LC-41

d. Launch Operations at LC-41

Operationally, the first floor of the facility is used to support Titan IV and Centaur Upper Stage processing, checkout and launch operations.

6.2.6 Launch Operations Control Center (LOCC) (Continued)

The second floor is used for the Launch Management Control Center (LMCC), Command Management Control Center (CMCC) and a Payload Information Room. The LMCC area in the LOCC does not replace the LMCC in the Test Group Support Facility (TGSF). The LOCC LMCC enhances the 6555 ASTG capabilities to provide a multiple operation support environment dependent on P/L program(s) requirements.

The second floor is also divided into functional areas, but unlike the first floor, will be used to support all payload and launch operations managed by the 6555 ASTG at CCAFS, reference Figure 6.2.6-1.

6.2.7 Solid Motor Assembly Buildings (SMABs)

The Standard SRM SMAB is a multi-use facility consisting of an IUS assembly and checkout facility (east bay) (which uses the checkout equipment in the VIB), an SRM (5- segment only) assembly area, and a Titan IV/SRM mate and checkout area.

The center high bay has a 305-ton bridge crane with a 50-ton crane on the same carriage. The 50-ton crane is used to stack five of the seven SRM segments from rail cars into the SRM assembly cells on the east side of the high bay. Assembled SRMs are then mated to the Titan Core using the 305-ton crane. The remaining two solid segments and the forward closure are installed at the Launch Pad using the MST 50-ton crane.

The SMARF will facilitate processing of the SRMUs and Titan IV/SRMU mate and checkout activities.

6.2.8 AGE Building

The Aerospace Ground Equipment (AGE) building is a two-story, reinforced concrete structure located between the MST rails adjacent to the launch pad. The upper story, which is level with the launch pad surface, houses the two Titan AGE Vans and has provisions for two additional vans for either Upper Stage or Spacecraft AGE.

The AGE building has an upper level (Level A) and a basement level (Level B).

Facilities for Payload User Electrical AGE equipment is provided in the A-10 Payload User's Room.

The facilities in the B-10 Payload User's Room are similar to those in the A-10 room.

Room A-10 is a class 300,000 clean room with a maintained relative humidity within 25-60% and a temperature within 65-70°, reference Figure 6.2.8-1.

The Propellant Transfer and Pressurization Control Set is located in the AGE building and is used to control and monitor propellant loading, unloading and tank pressurization.

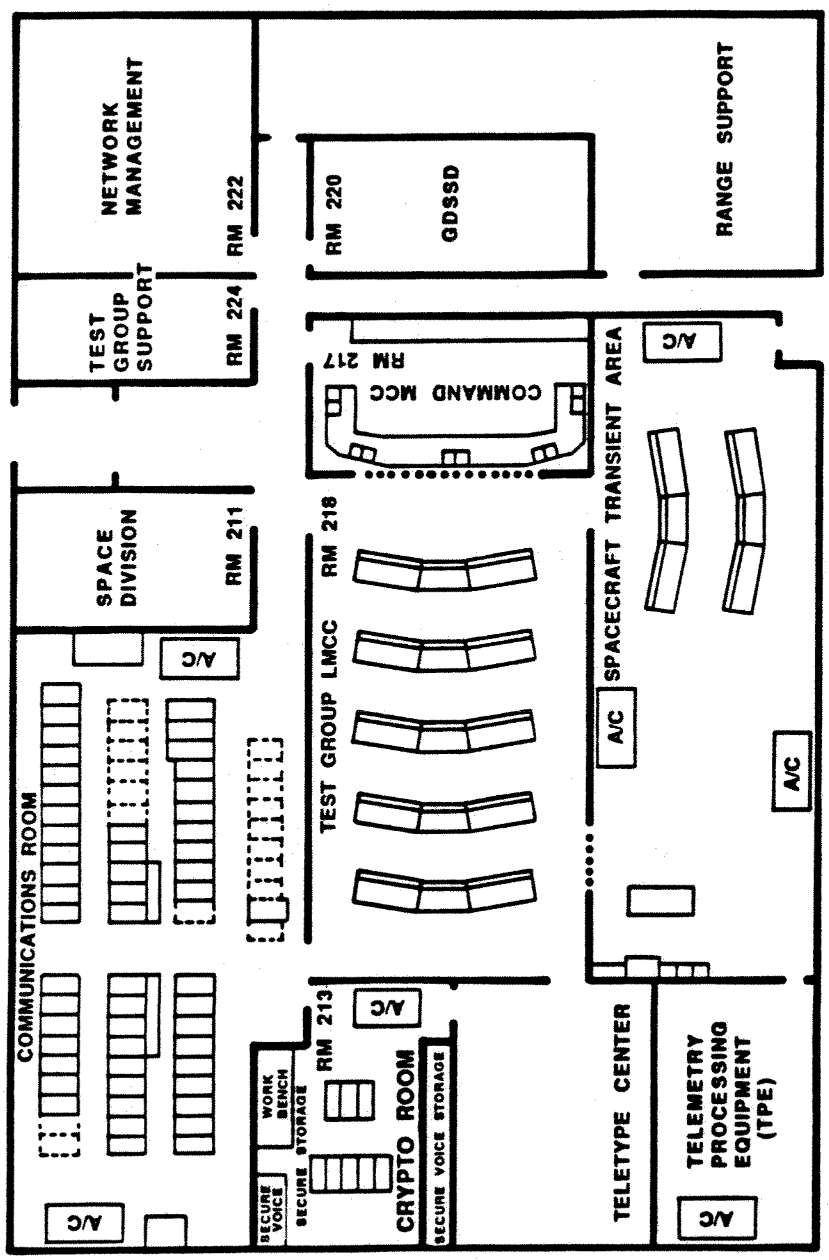


FIGURE 6.2.6-1 LOCC, LMCC AND CMCC AREA – SECOND FLOOR

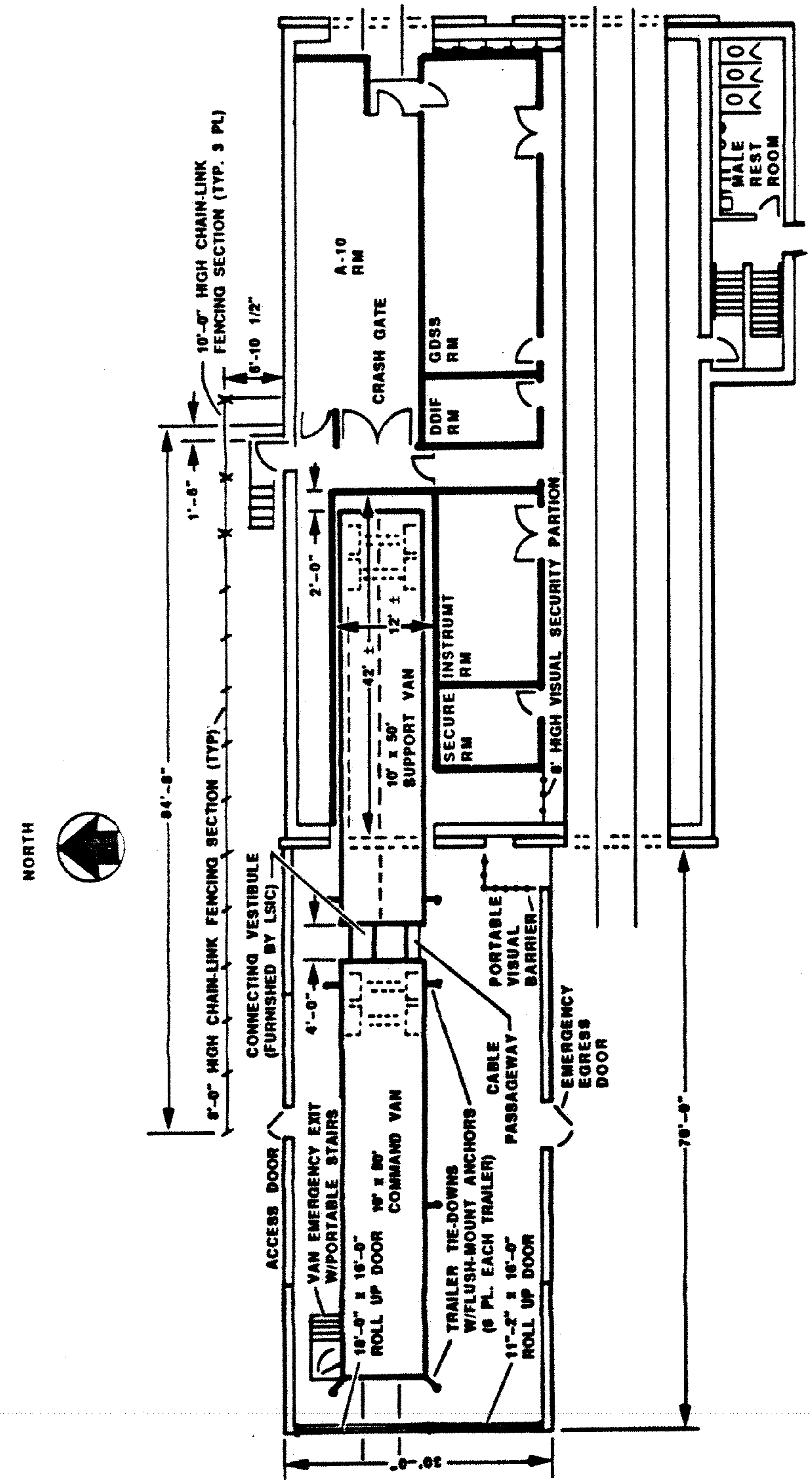


FIGURE 6.2.8-1 AGE BUILDING FIRST LEVEL

6.2.9 Gas Storage and Propellant Holding Areas

The Gas Storage Area contains storage vessels for high-pressure nitrogen and helium. Separate Fuel and Oxidizer Holding Areas contain facilities to store, transfer and unload propellants for the Launch Vehicle including the SRM Thrust Vector Control (TVC) tanks.

6.2.10 Titan Transporter

The purpose of the Titan Transporter Railroad System is to transport the vertically assembled Titan Core from the VIB to the SMAB/SMARF, where SRM/SRMU assemblies are mated on the Core. From the SMAB/SMARF the partially assembled Booster Vehicle is transported to the LC-41 launch pad. The Titan Core and SRM/SRMU assemblies are vertically positioned on the transporter which is set on the rail car wheeled undercarriage. The transporter supports the LV and Spacecraft from buildup until Launch.

The undercarriage assembly is equipped for raising the transporter off its support bearings, support and transport it, and position and lower it onto its destination support bearings. An undercarriage assembly jacking provision allows lateral alignment of the transporter as necessary. The undercarriage is removed after the transporter is secured to the launch pad support piers.

The AGE Vans, which are assigned to the Titan Vehicle, are mounted on rail car wheels and remain with the transporter . When the Titan IV vehicle is placed at the proper LC-41 launch pad position, the AGE Vans will also be properly positioned.

6.2.11 Facilities Not in ITL Area

Additional support facilities are located in the ESMC industrial area. Availability and type of facilities are coordinated with the 6555th Aerospace Test Group. Some Department of Defense (DOD) Spacecraft use the Satellite Assembly Building (SAB) and other range services and facilities for assembly and/or checkout. Some noncomplex Spacecraft are transported directly to LC-41 for integration with the Upper Stage and Titan IV. Reference Figure 6.2.11-1 which illustrates Range Instrumentation Facilities available in the Atlantic area.

6.3 Aerospace Ground Equipment (AGE)

6.3.1 ITL Electrical-Electronic AGE

ITL Electrical-Electronic AGE falls into three categories: 1) Titan IV/Centaur unique, 2) Titan IV/IUS unique and 3) common. This equipment supports prelaunch checkout, launch control and monitoring. Fixed Electrical-Electronic AGE is located in the VIB, SMAB and AGE buildings. Van-mounted Electrical-Electronic AGE is located in the VIB during the buildup and initial checkout of the vehicle and is later moved with the vehicle to the AGE building at the Launch Complex.

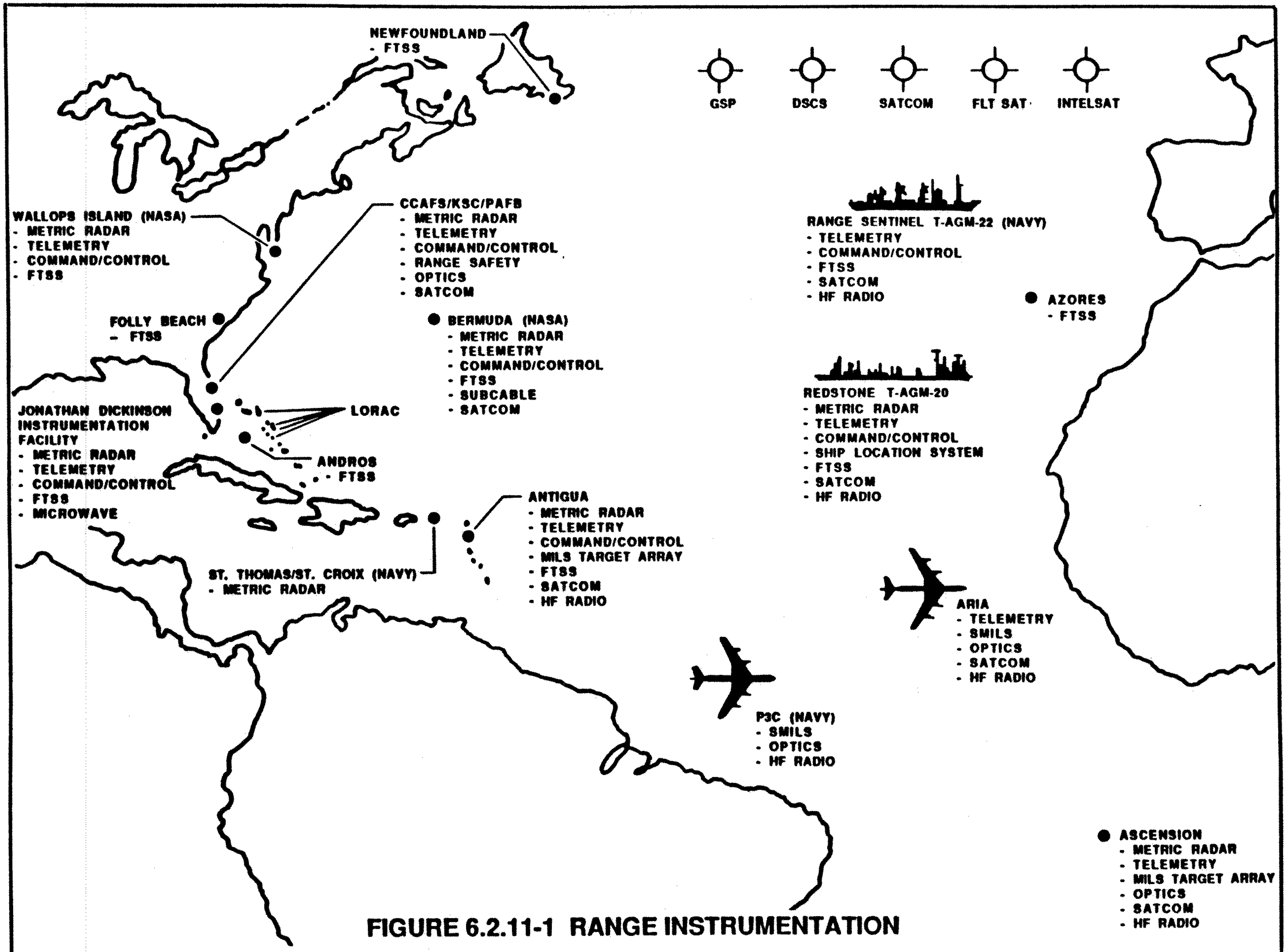


FIGURE 6.2.11-1 RANGE INSTRUMENTATION

6.3.1.1 IUS Unique AGE

The IUS Electrical-Electronic AGE is used for stand-alone testing of the IUS at the SMAB prior to mating with the Titan IV at LC-41. Also, stand-alone tests of the IUS, in addition to integration tests of the IUS, Titan IV and SC are performed at LC-41.

The IUS COS, located in the VIB, is the major item of IUS Electrical – Electronic AGE that supports IUS checkout and launch activities at ESMC. It is the "control center" for tests involving the IUS and is connected by land lines to the remote equipment located in the AGE building at LC-41. The land lines carry IUS telemetry data, IUS commands and remote ground support equipment control and status.

6.3.1.2 Centaur Unique AGE

The Centaur Electrical – Electronic AGE is used to perform stand-alone tests in Cell 3 of the VIB. Stand-alone tests are also performed at the Launch Complex in addition to integrated tests with the Titan IV and the SC. Land lines transfer command, monitor and telemetry data between the VIB and the AGE building at LC-41.

6.3.1.3 Common AGE

Common ESMC Electrical-Electronic AGE consists of prelaunch checkout units, launch control and monitoring sets, instrumentation, and data and power support. Common AGE is located in the vans, in the VIB, SMAB/SMARF and in the AGE building. Equipment which interfaces directly with the launch vehicle is located in the vans.

6.3.1.3.1 Prelaunch Checkout

Prelaunch Checkout Equipment consists of the Programmable Aerospace Control Equipment (PACE), a Flight Safety System Checkout and Control Monitor Group, a Pulse Tracking Control Monitor Group and a Combined Systems Test Simulator Set (CSTSS).

6.3.1.3.2 Launch Control and Checkout AGE

The launch control and checkout AGE incorporates those pieces of equipment necessary to initiate a launch sequence, control the sequence, transmit command signals, monitor functions and isolate malfunctions.

Launch Control and Monitoring Electrical – Electronic AGE includes the PACE, a Launch Control Console and a Flight Safety System Control Monitor Group.

6.3.1.3.2 Launch Control and Checkout AGE (Continued)

PACE functions as a readiness monitor and the master countdown sequencer for the Titan IV ground equipment, the Launch Vehicle, the Upper Stage and the SC. It automatically time-sequences launch events from initiation of the Terminal Countdown through Liftoff. It monitors and displays countdown status, hold status and criteria violations in real-time. PACE computers and peripherals are located in the VIB backup Launch Control Center. Additional peripherals are located in the LOCC.

The launch countdown can be controlled from the backup Launch Control Console in the VIB or from the Launch Control Console in the LOCC. The Launch Control Console controls ground or external power application and displays overall status, initiates the automatic countdown and serves as the communications center for the launch facility. The console also monitors ground power, hold indications, SRM/SRMU parameters and the status of the Water System at the Launch Complex.

The monitor group for the T&FS System monitors the arming and safing of the Titan IV ISDS, the Command Destruct System and the SRM Ignition System during the Terminal Countdown.

6.3.1.3.3 AGE Vans

AGE vans provide protection and space for electronic AGE used during checkout and launch operations. The Titan vans are capable of being towed on hard surfaced roads and on the standard-gage railroad tracks. The vans will be used at the VIB; and then remain connected to the transporter, both electrically and mechanically, in transit to the launch complex. These vans will occupy the west track in the AGE building at the complex. The van air conditioning system maintains temperature and humidity control in the vans at all times. The normal van complement consists of an Instrumentation Van and a Launch Control Van.

6.3.1.3.4 Inertial Guidance System AGE

The IGS AGE consists of those units necessary to erect, load and monitor the IGS. This equipment consists of the Guidance Control Monitor Group (GCMG) and the Telemetric Data Monitor Group (TDMG). The GCMG is located in the Instrumentation Van at ESMC.

6.3.1.3.5 Tracking and Flight Safety AGE

The T&FS AGE consists of three units, the Flight Safety Checkout Control Monitor Group, the T&FS Monitor Group and the Pulse Tracking Control Monitor Group. All of these units are housed in the Launch Control Vans at ESMC.

6.3.1.3.6 Telemetry/Instrumentation AGE

The Telemetry/Instrumentation AGE includes numerous units that transmit, decode, modulate, reproduce and record telemetered information from the airborne systems. These functions are performed during countdown and flight of the vehicle. Some of the telemetry equipment is located in the Instrumentation Van at ESMC.

6.3.1.3.7 Test Tools and Simulators

In addition to the major pieces of electronic AGE already described, a variety of simulators and special test equipment is required during a launch cycle. Payload simulators and SRM simulators are used during system tests prior to connection of these components. Ordnance simulators such as Standard Ordnance Checkout and Verification Units (SOCVUs) are used for all major system tests to verify firing circuits.

6.3.2 Mechanical/Structural AGE

Titan IV launch vehicle system mechanical AGE provides transportation, assembly and support facilities, pressurization gases and air conditioning, propellant services, handling equipment such as bridge cranes and elevators and personnel and equipment protection.

Chapter 7

Launch Operations at CCAFS (LC-41)

7.0 LAUNCH OPERATIONS AT CCAFS LC-41

7.1 Introduction

This chapter presents the launch operations management, launch operations documentation, and test operations and support at Cape Canaveral Air Force Station (CCAFS). Reference Titan IV Program System Test Plan Eastern Test Range Vehicles MCR-84-2506A and Launch Operations Requirements Document MCR-88-2519.

Some changes in organization roles and responsibilities occur at "Phase Point". The definition of Phase Point is different for CCAFS and Vandenberg Air Force Base (VAFB). Phase Point at CCAFS starts with the physical mating of the Satellite Vehicle Adapter (truss) to the 2490 Forward Skirt Extension for SS-ELV-405 or to the Upper Stage for SS-ELV-401 and SS-ELV-402 configurations. (Note: The 8-point truss for Centaur is installed at the LC-41 and the 22-hard point truss is installed in Cell 3 in the VIB).

During the Countdown Sequence, various test and operational functions are accomplished. The Countdown Sequence is normally a four-phase event and is accomplished during a three-day time period (R-2, R-1 and R-Day). Phase One consists of Launch Vehicle and Space Vehicle tests. Phase Two consists of ordnance connection with the exception of the destruct initiators, the Space Vehicle Destruct System, and the SRM igniter Safe & Arm devices and staging motor interfaces.

Phase Three consists of Launch Vehicle and Space Vehicle preparations. Phase Four consists of Launch Vehicle and Space Vehicle readiness checks, destruct initiator connection, space vehicle destruct system connection; and in the TIV, the SRM Igniter S&A device connection and staging motor interface connection, Mobile Service Tower (MST) removal, final verification, and terminal count.

7.2 Launch Operations Management

7.2.1 General

Test requirements for launch operations testing shall be identified by the ACRBC, Launch Test Directive, Interface Test Specification, CI Specifications, engineering drawings and other applicable Government documentation.

7.2.2 Test/Launch Operations and Schedule

The 6555th ASTG will exercise technical test and schedule control over all system testing and will ensure that the contractors maintain integrity of the system hardware, software and checkout equipment.

7.2.3 Flight Vehicle Integrated Test Operations

7.2.3.1 Joint/Integrated Test Conduct

At Phase Point, joint/integrated testing at the Launch Site will be controlled by the Test Group Launch Controller.

7.2.3.2 Integrated Test Scheduling

Meetings shall be held and chaired at a place and time as assigned by the Test Group for interchange of status, scheduling, test, and facility interface matters. Attendees shall consist of a representative from each of the affected contractors and a representative from all other agencies actively engaged in Launch Operations.

7.2.3.3 Launch Site Reviews

7.2.3.3.1 Incremental Reviews

Incremental Reviews are designed to verify systems readiness at a given time and to insure readiness to proceed to the next major increment.

7.2.3.3.1.1 Readiness Reviews

Readiness Reviews are scheduled in advance of the milestone event to allow sufficient time for open or action items to be normally completed without schedule impact. The Satellite Vehicle Mate Readiness Review is held prior to transportation of the Satellite Vehicle to VAFB.

7.2.3.3.1.2 Launch Readiness Review

The Test Director presents the Space Division Commander and Mission Director with a final Launch Readiness Review (LRR) at R-1 day. This review culminates the prelaunch review cycle and results in a joint readiness statement by the Space Division Commander, Mission Director, and Program Director.

7.2.3.4 Integrated Test Management

The key positions in the Integrated Test Management Structure at CCAFS are shown in Figure 7.2.3.4-1.

7.2.3.4.1 Mission Director – USAF

The Mission Director is designated by the USAF Space Division Commander, in coordination with the Spacecraft SPO Director and is responsible for total mission success.

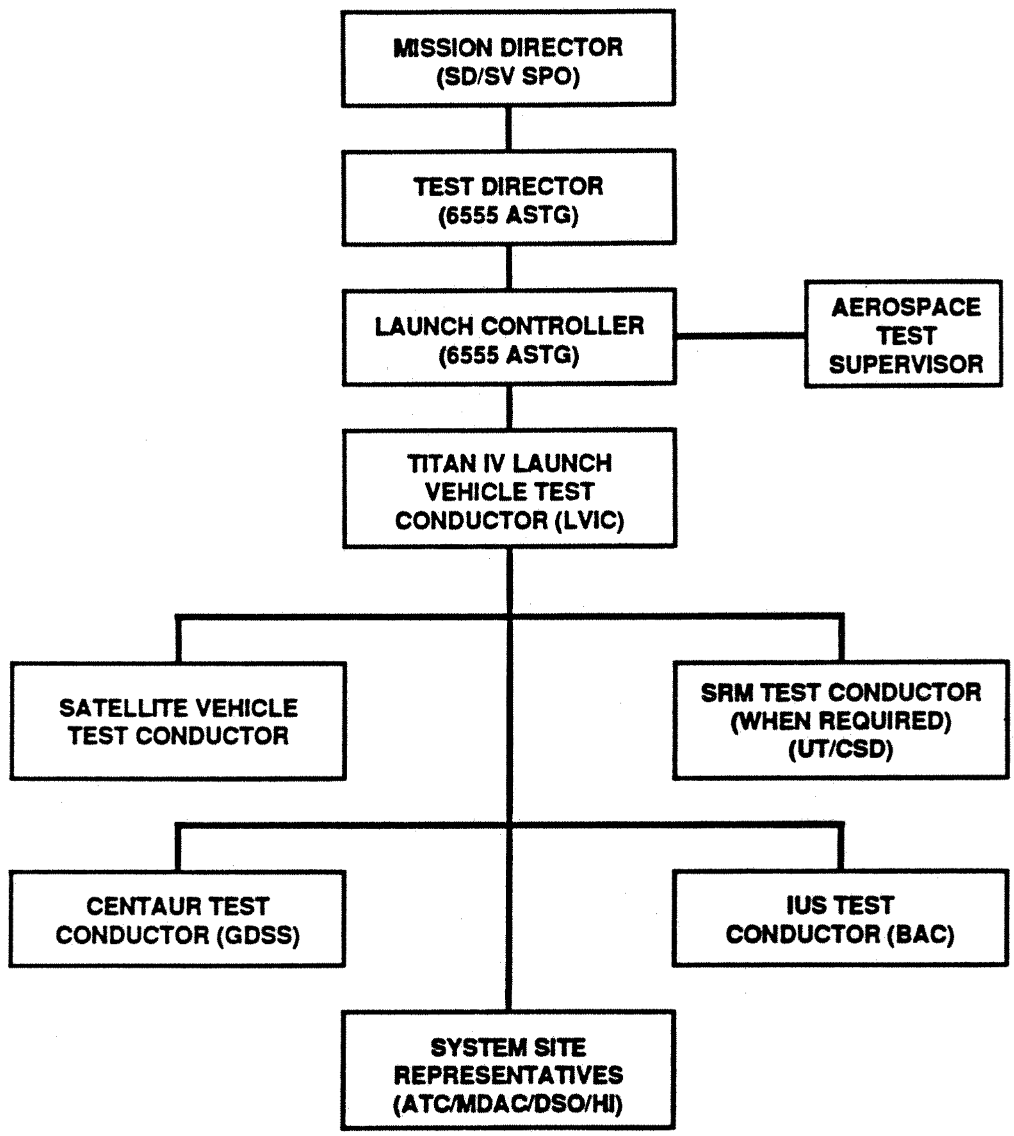


FIGURE 7.2.3.4-1 INTEGRATED TEST MANAGEMENT – CCAFS

7.2.3.4.2 Test Director – USAF

The Test Group Test Director is responsible to the Space Division for the accomplishment of the overall Launch Operations objectives. The Test Director receives inputs from the Launch Controller, the Range Control/Safety Officer, etc., and evaluates the data to determine if the test objectives outlined in the System Test Objectives (STOs)can be met. The Test Group Commander, or his designee, serves as the Test Director and is responsible to the Mission Director for the accomplishment of the overall Launch Operations objectives.

7.2.3.4.3 Launch Controller – USAF

The Test Group Launch Controller supervises Launch Operations and exercises overall technical test control. The Launch Controller is responsible to the Test Director for the technical and operational readiness of the hardware and resources, and for managing the integrated test in progress as necessitated by other Range commitments, external factors that might affect the outcome of the test/operation, or other activities that are being performed concurrently. The Launch Controller is responsible for safety, security, and integrated schedules.

7.2.3.4.3.1 Test Controllers

Test Group Test Controllers are responsible for technical test control of specific systems test and related activities. Test Controllers will assist the Launch Controller during integrated systems testing and launch countdown and provide Go/No-Go recommendations.

7.2.3.4.4 Aerospace Corporation Test Supervisor

This person is the senior Aerospace representative assigned to integrated test/launch operations. The Test Supervisor is assisted by a staff of system specialists who will be stationed in critical test areas for real time evaluation of test operations. A readiness recommendation is required from the Aerospace Test Supervisor prior to the Launch Controller's approval of any test.

7.2.3.4.5 Satellite Vehicle Test Conductor (SVTC)

The Satellite Vehicle Test Conductor (SVTC) is the P/L contractor person responsible for the conduct of all S/V tests and for the conductance of S/V launch countdown. The SVTC is responsible to and keeps the S/V Countdown Controller and the LV Test Conductor informed at all times of the current status of the S/V and potential problems. The SVTC is designated to have technical and operational responsibility for:

a. Readiness of the satellite AGE to support testing.
b. Conducting S/V tests and countdown.
c. Integration/coordination of S/V directives and operating procedures with other agencies.

7.2.3.4.6 Titan IV Test Conductor

The LVIC Test Conductor (TC) is directly responsible for the conductance of the overall LV Receipt-through-Launch testing and countdown activities. The TC is responsive to test support requirements of Subcontractor Test Conductor/Representatives and keeps the Launch Controller and the Boost Vehicle Countdown Controller (BVCC) informed of the current status of the Titan IV LV and potential problems. At Phase Point, the TC must have approval of the Launch Controller for deviations from the integrated test procedures. He is designated to have operational responsibility for:

a. Readiness of the launch site to support tests.
b. Conduct of Titan IV LV tests.
c. Test preparations, schedule maintenance and test of the Titan IV LV, associated Aerospace Ground Equipment (AGE), ground station instrumentation and launch site facilities.
d. Conduct of Titan IV LV Launch Countdown.

7.2.3.4.7 Boeing Aerospace Test Conductor

The Boeing Aeorspace Test Conductor is responsible for the conductance of all IUS tests and the conduct of the IUS launch countdown.

7.2.3.5 Anomalies

In case of equipment anomalies encountered during test procedure run, a Procedure History Sheet (PHS), Martin Automatic Reporting System (MARS) and/or other applicable subcontractor/associate contractor and equivalent documentation is used to initiate troubleshooting activities during integrated tests until the anomaly is resolved.

7.2.3.6 Post-Test Requirements

Following completion of Integrated Test and Launch Countdown, the Program Engineer schedules a meeting with associate/participating contractors for post-test critique. System integrity, as defined in procedures, is maintained following a test. Break of integrity may be reason for retest and requires Air Force Quality Assurance notification. Final acceptance of Post S/V Mate integrated test results and approval to proceed is to be made by the Test Group at the post testing meeting.

7.2.3.7 Titan IV Launch Vehicle Recycle, Backout and Safing

The LVIC/LSIC and all Titan IV LV subcontractors have recycle, backout, and safing procedures to follow in the event of critical malfunctions or test interruptions of the Titan IV Launch Vehicle and associated systems during Countdown Operations.

7.2.3.8 S/V Recycle and Safing

The SV Contractors/Representatives prepare recycle, backout and safing procedures for the S/V and associated systems to use in the event of critical malfunctions or test interruptions during Countdown Operations.

7.2.4 Working Groups

7.2.4.1 Launch Test Working Group (LTWG)

The LTWG implements all Space Launch Vehicle and associated spacecraft programs at the launch base and manages those functions which involve the coordinated activities of the various agencies associated with the Prelaunch, Launch, and Post Launch activities at the Launch Base. Accomplishment of the System Test Objectives, as specified for the Titan IV Launch Vehicle, and by associated spacecraft System Program Offices (SPO) is a primary goal of the LTWG. The LTWG is responsible for those functions related to the Titan IV Launch Vehicle and associated spacecraft programs at the Launch Base from the initial planning phases (including hardware acceptance) through launch of vehicles, acquisition of flight data, and the submission of applicable reports. The period of activity of the LTWG extends throughout the interval of the Titan IV and associated spacecraft programs.

7.2.4.2 Facility Working Group (FWG)

The FWG is responsible for reviewing and making recommendations on all Facility Change Requests (FCR) and Facility Engineering Change Proposals (FECP) submitted, and for maintaining configuration of all active Titan IV facilities. LVIC is responsible for controlling facility configuration and for scheduling implementation of changes. FWG membership consists of representatives from participating government agencies and contractors and includes an LSIC representative.

7.2.4.3 Safety Working Group (SWG)

The SWG is responsible for establishing/implementing safety policy and requirements for the Titan IV program. The Pad Safety Standard Operating Procedures (SOPs) and Launch Site Range Safety Regulations serve as the media for defining safety policy and requirements for the Launch Sites. The SWG is chaired by the responsible Test Group and membership is open to all program participants.

7.3 Launch Operations Documentation

7.3.1 Titan IV Master Scheduling

The Titan IV LVIC Level 5 Master Schedule is the document which sets forth the master plan which reflects major milestones for Launch Vehicle operations. This document is prepared by LVIC and approved by 6555 ASTG.

7.3.2 Titan IV Integrated Program Schedules

The Titan IV LVIC Level 7 Program Schedule document is the master plan and site scheduling document which reflects launch vehicle delivery, test operations, and launch operations at the launch base. This document is prepared by LVIC and concurred with by 6555 ASTG.

7.3.3 Daily Integrated Test Schedules

The daily LVIC Level 3 Launch Vehicle test schedules for Titan IV Launch Vehicle will identify the testing and operations planned over a specified period. It will also identify all support requirements for schedule tasks that are not already dedicated. The daily integrated launch vehicle test schedule is prepared by LVIC and concurred with by 6555 ASTG.

7.3.4 Acceptance, Checkout, Retest, Backout Criteria (ACRBC)

The ACRBC, prepared by LVIC, establishes minimum integrated Titan IV Launch Vehicle checkout and testing requirements and criteria at the Launch Site.

7.3.5 Launch Test Directive (LTD)

The LTD shall establish testing and constraint requirements on Titan IV Launch Vehicles. The LTD is based on information contained in the System Test Objectives, ACRBC and other contractor prepared approved inputs. Major milestones, test procedures and constraints will be identified in the LTD or in a referenced document thereof. The LTD will be prepared by the Aerospace Corporation and approved by 6555 ASTG.

7.3.6 System Test Objectives (STO)

The STO report is published by the SPO to define the mission objectives to be followed during Prelaunch, Launch, and Flight of Titan IV Space Launch Vehicles.

7.3.7 Test Procedures

7.3.7.1 Receipt-to-Launch Test Sequence

The test sequence is a Receipt-to-Launch logic network which identifies vehicle test activities that are scheduled at CCAFS. It is a LVIC procedure which delineates all Launch Vehicle activity by test procedure number.

7.3.7.2 Titan IV Launch Vehicle Test Procedures

Included are: Core Vehicle, Core Guidance, Liquid Rocket Engine, Payload Fairing, Launch Vehicle, and Solid Rocket Motor Test Procedures.

7.3.7.2.1 Centaur Systems Procedures

Centaur System Operations are conducted by GDSS in accordance with the GDSS Integrated Test Plan Section 8.0 – Ground Operations Plan (Launch Site) and Section 9.0 – Test Parameters and Redlines (Launch Site). These procedures are subject to Martin Marietta review and approval, and 6555 ASTG coordination.

7.3.7.3 IUS Test Procedures

IUS operations are to be conducted by Boeing Aerospace in accordance with Detail Operating Procedures (DOP) and SOPs. SOPs are written for recurring tasks necessary to perform the DOPs. SOPs cannot be run alone but must be embedded within the DOPs.

7.3.7.4 Titan IV Flight Vehicle Test Procedures

7.3.7.4.1 Integrated Test Procedures

Integrated tests include such tests as Combined Systems Test, Umbilical Drop Test, EMI, Launch Countdown, Flight Events Demonstration, Composite Electrical Readiness Test, etc., where a vehicle countdown and/or countup is conducted.

7.3.7.5 Countdown Manual

The Countdown Manual provides a step-by-step integrated check list of all system operations from the start of the count through launch. In addition, it provides contingency procedures for recycle and backout in the event of countdown termination. The Countdown Manual is prepared by the LVIC from inputs from all participating agencies.

7.3.8 Range Documentation

7.3.8.1 Program Requirements Document (PRD)

The early planning information contained in the Program Introduction and Statement of Capability is expanded in the PRD to a complete and detailed statement of range user requirements.

7.3.8.2 Program Support Plan (PSP)

The PSP documents are submitted by ESMC to SD and the 6555 ASTG, responding to how the requirements of the PRDs will be implemented.

7.3.8.3 Operations Requirements (OR) Document

The OR document outlines Range Support Requirements for a major test or flight plan and is derived from the PRD, PSP and inputs from all contractors. The LVIC, associate contractors (as applicable), and S/V Contractor provide OR inputs to the 6555 ASTG.

7.3.8.4 Operations Directive (OD)

The OD is the final document in the Universal Documentation System (UDS) and is the official Range Directive which mobilizes and assigns the resources necessary to support the Range User's requirements shown in the OR.

7.4 Test Operations and Support

7.4.1 Titan IV/IUS Test Team

The Titan IV/IUS Test Team, under the direction of the Air Force Test Director (AFTD), manages the day-to-day integrated processing activities to include Build-up, Checkout, and Launch of the Titan IV/IUS Booster as well as scheduling, and on-line test operations at CCAFS. It consists of the U.S. Government (6555 ASTG, ESMC) and its Technical Advisor, the Aerospace Corporation (ASC), and the various aerospace contractors, i.e., Martin Marietta Astronautics Group (MMAG), Aerojet Tech Systems Corporation (ATC), Boeing Aerospace (BA), Chemical Systems Division (CSD), etc., and other support elements required to conduct daily vehicle processing operations.

7.4.2 Payload Test Team

A senior member of the 6555 ASTG Spacecraft Division is the Payload Test Director (PLTD). Under his guidance, the team manages the day-to-day activities of processing the P/L at CCAFS. All work on the P/L is performed by a Payload Support Contractor (PSC). The PSC is managed/supported by a designated 6555 ASTG Project Engineer (PE) and a team of Air Force Test Controllers (AFTCs) that oversee all P/L stand-alone and integrated test operations. Reference DOD Generic Communications Plan Applicable to Titan IV/NUS/IUS Centaur (CCAFS), reference Figure 7.4-1.

7.4.3 Post Test and Launch Evaluation

Data is collected following test or launch for evaluation by the responsible system contractor under the cognizance of SD.

7.4.4 Operations Support

7.4.4.1 Titan Integrated Corrective Action System

The Integrated Corrective Action System is in accordance with the requirements of the System Effectiveness Plan.

7.4.4.2 Contractor Maintenance Area

A Contractor Maintenance Area (CMA) is operated by LVIC. The CMA services are made available to program contractors in the event of emergency.

7.4.4.3 Crew Certification

At the Launch Readiness Review, each prime contractor certifies that its launch crew personnel are qualified to perform Launch Operations.

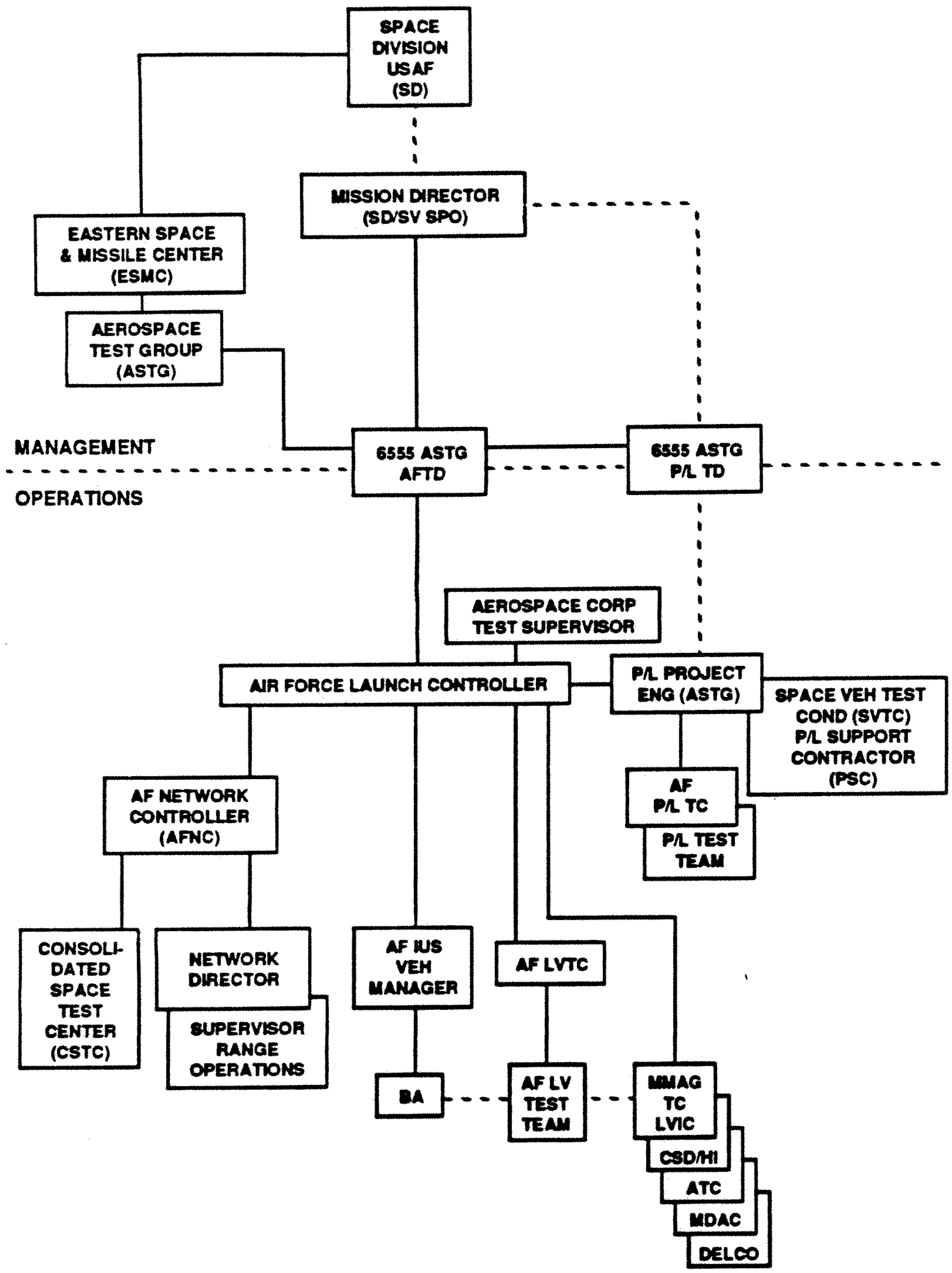


FIGURE 7.4-1 TITAN IV/IUS/PAYLOAD TEST TEAM

Chapter 8

Launch Preparation Operations at VAFB

8.0 LAUNCH PREPARATION OPERATIONS VAFB

8.1 Introduction

This chapter describes the receipt and checkout of the LV and PLs as related to the Western Space and Missile Center (WSMC) facilities and ground systems at SLC-4E, reference Figure 8.1-1 and Figure 8.1-2. Also reference the Payload Support Capability Document VAFB/SLC-4E MCR-88-2639.

8.2 Titan IV/Payloads Receipt-to-Launch Sequence

The Titan IV Space Launch Vehicle components are received at VAFB via air and surface transportation. Stage I & II of the core vehicle, arrive at the Vehicle Assembly Building/Horizontal Test Facility (VAB-HTF) Building 8401, where the LREs are received and installed and hydraulic components are installed and tested. The SRMs live components are received at the Receiving Inspection & Storage (RIS) Building 945, and inert components are received at the Motor Inert Component Storage Area (MIS). SRMUs are received and processed at Building V-31.

After initial receipt, inspection & testing, the LV components are transported to SLC-4E where they are assembled and checked out on the launch mount.

Titan IV assembly, checkout and launch processing initializes the integrate-on-pad concept at SLC-4E. Assembly begins with the SRMs/SRMUs which mechanically interface with the launch mount followed by Titan Core Stage I & II which mechanically interface with the SRMs/SRMUs. Checkout of the LV is accomplished systematically beginning with components testing, followed by subsystem and system tests and culminating in a Combined System Test (CST) to verify the LV integrity prior to mating of the P/L.

P/Ls arrive at VAFB and optionally may be off loaded to a PPF at SLC-6 for unique operations; or, proceed directly to SLC-4E where they are mated to the LV. After P/L mate, P/L subsystem tests are performed and the P/L-LV electrical interface connectors are mated. When final direct access requirements to the P/L are complete the PLF aft and forward segments are installed around the P/L. P/L testing will be integrated into the LV-P/L-PLF test sequence of the Integrated Systems Test (IST) and the Countdown Procedure. Functional Flow sequences, timelines and support services for the P/L are developed in detail in P/L-unique Launch Base Support Plans.

Time lines for accomplishing Titan IV vehicle systems receipt-to-launch functions are based upon facilities and resource capabilities to support two launches per year for the first 23 Titan IV LVs built and scheduled to be launched at WTR.

General hazards and safety considerations will be covered in the Martin Marietta System Safety Engineering Manual M64-125 and the ACRBC, Appendix C, MMC Safety organization and General Hazards/Safety considerations. Also reference Titan IV System Contamination Control Plan – WTR, MCR-88-2524.

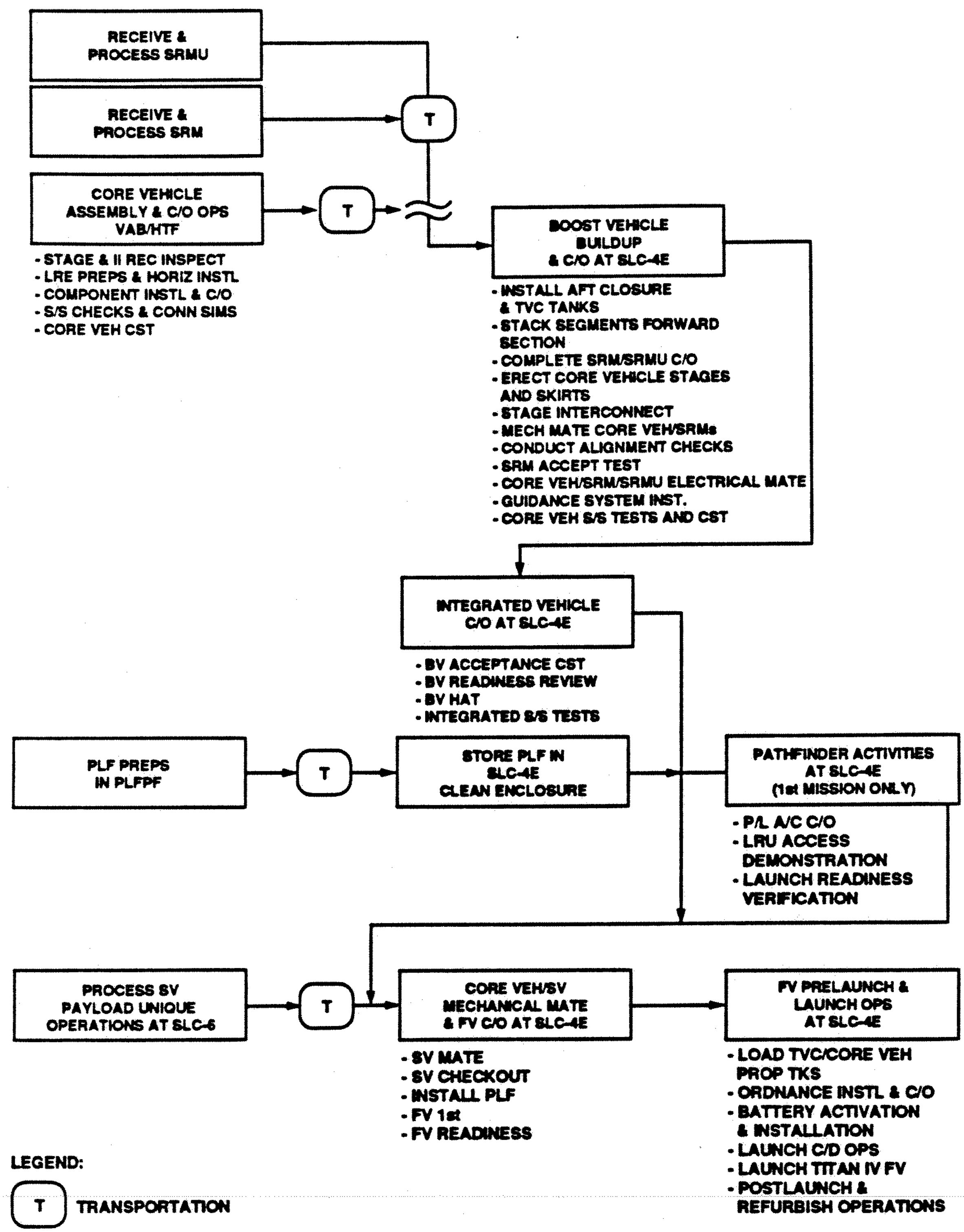


FIGURE 8.1-1 TITAN IV/NUS VAFB RECEIPT-TO-LAUNCH FLOW

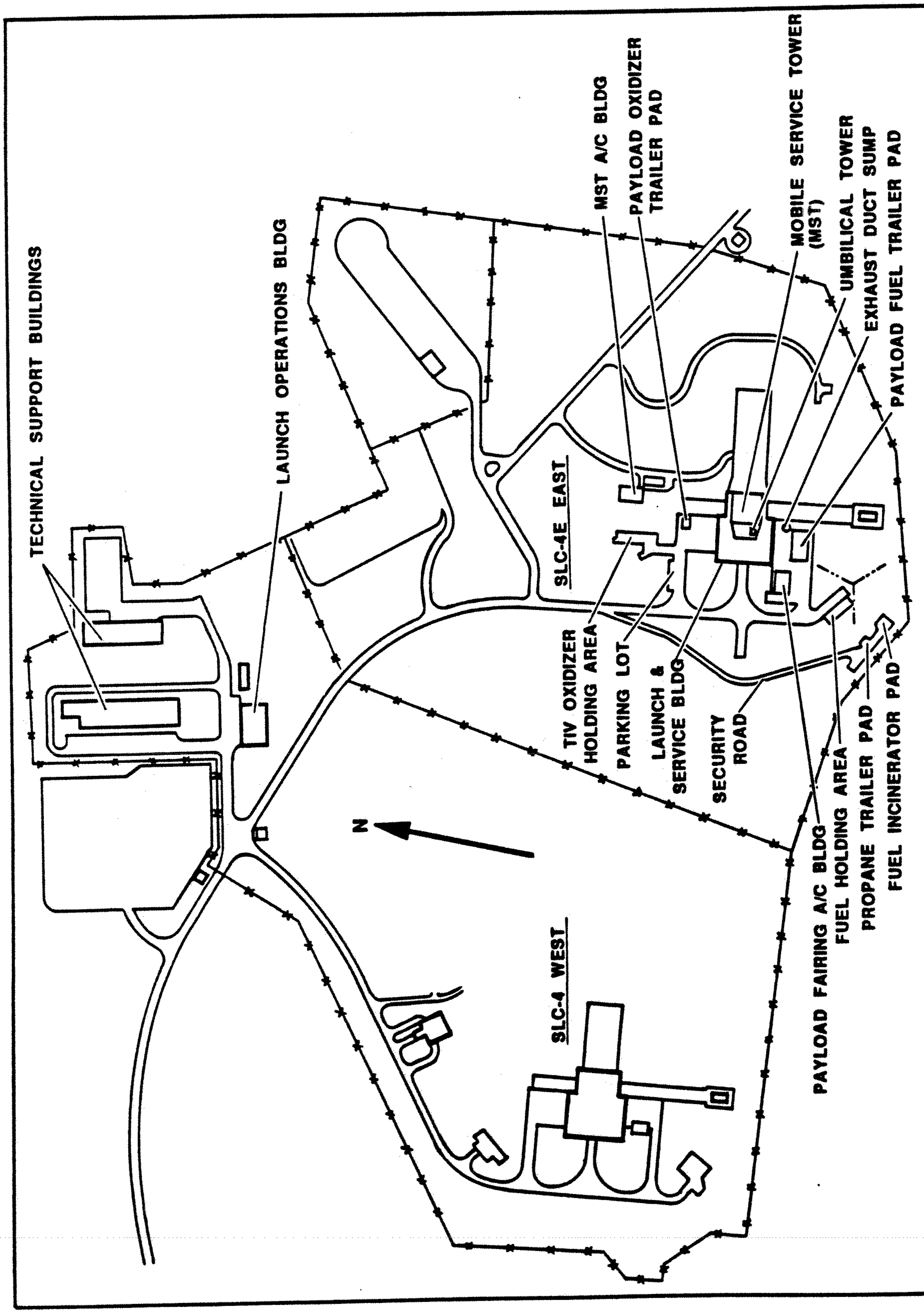


FIGURE 8.1-2 SLC-4E SITE PLAN

8.3 WSMC-Provided Payload Service/Support Facilities

Facilities of interest to the P/L User include the following.

8.3.1 Mobile Service Tower (MST)

The SLC-4E MST provides facilities for erection and buildup of the LV and PLF mating, as well as P/L servicing and checkout. Work platforms at strategic levels allow personnel and equipment access to various vehicle levels. The platforms fold or retract to clear the LV and P/L when the MST is retracted.

The clean enclosure area within the upper portion of the MST, provides 100,000 ppm clean environment and secure area for P/L activities. The north elevator in the MST provides personnel and freight access to the P/L platforms. Stairs and ladders also provide access to various MST levels.

Accessibility for installing the P/L and PLF is provided by means of a large door on the west side of the clean enclosure. Hoisting the PLF and/or the P/L is accomplished by the overhead MST cranes. The entire structure is mounted on trucks that ride on fixed rails. Just before launch, the MST is moved from its service position at the launch pad to its parked position, reference Figures 8.3.1-1 and 8.3.1-2.

8.3.1.1 WSMC-Provided Service/Support Systems

Service and support systems are provided to all Titan IV users at SLC-4E. Distribution of commodities locations are shown within the primary P/L service area in Figure 8.3.1-1. Specifications for these systems are detailed in the "Titan IV WTR Payload Generic Interface Control Document". Users desiring to review/analyze the validation of these systems as installed may request the applicable Ground System Test Procedures (as run) by subject system and listed in the "Titan IV Implementation Plan", Section VIII, reference P/L Support Capability Doc VAFB SLC-4E MCR 88-2639.

8.3.2 Umbilical Tower (UT)

The UT supports routing of the P/L and LV servicing capabilities from ground systems to the flight hardware for electrical components, propellants, pressurization gases, and conditioned air. The basic structure is an open bridgework with the north and east sides covered to provide plume flame and blast protection during vehicle liftoff, reference Figure 8.3.2-1.

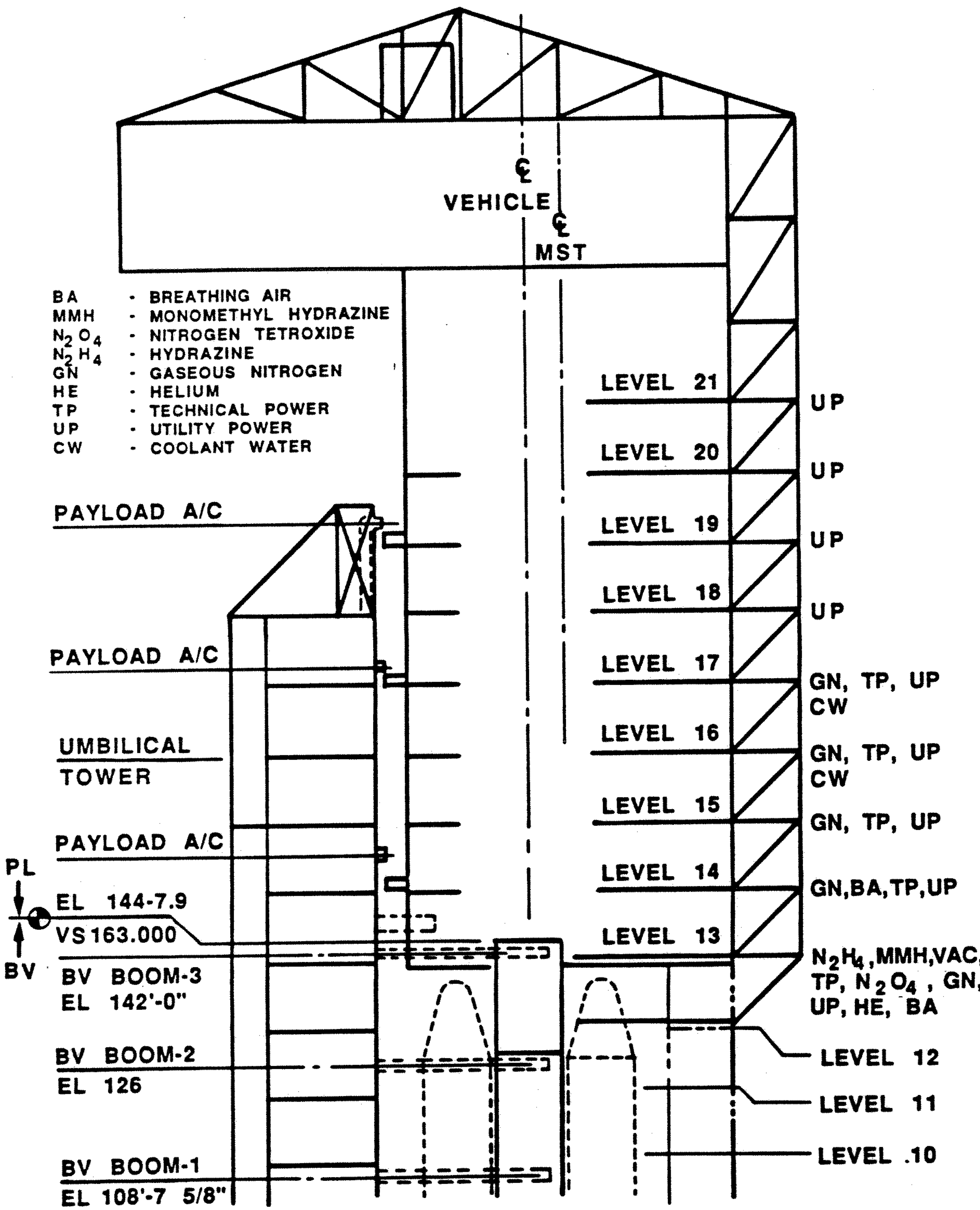


FIGURE 8.3.1-1 UMBILICAL/COMMODITY DISTRIBUTION BY LEVEL

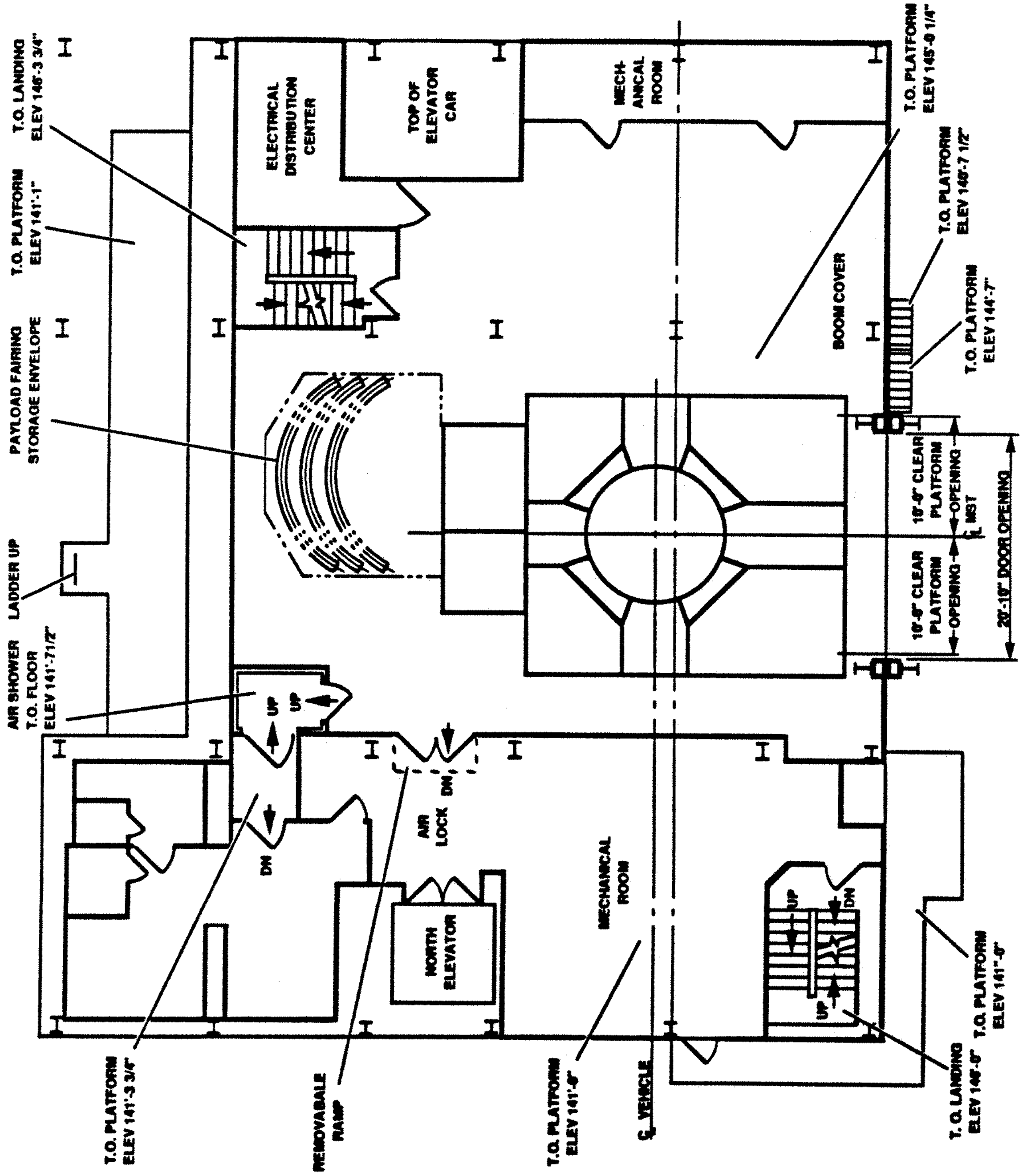


FIGURE 8.3.1-2 MST LEVEL 13 – TYPICAL PAYLOAD CHECKOUT AREA

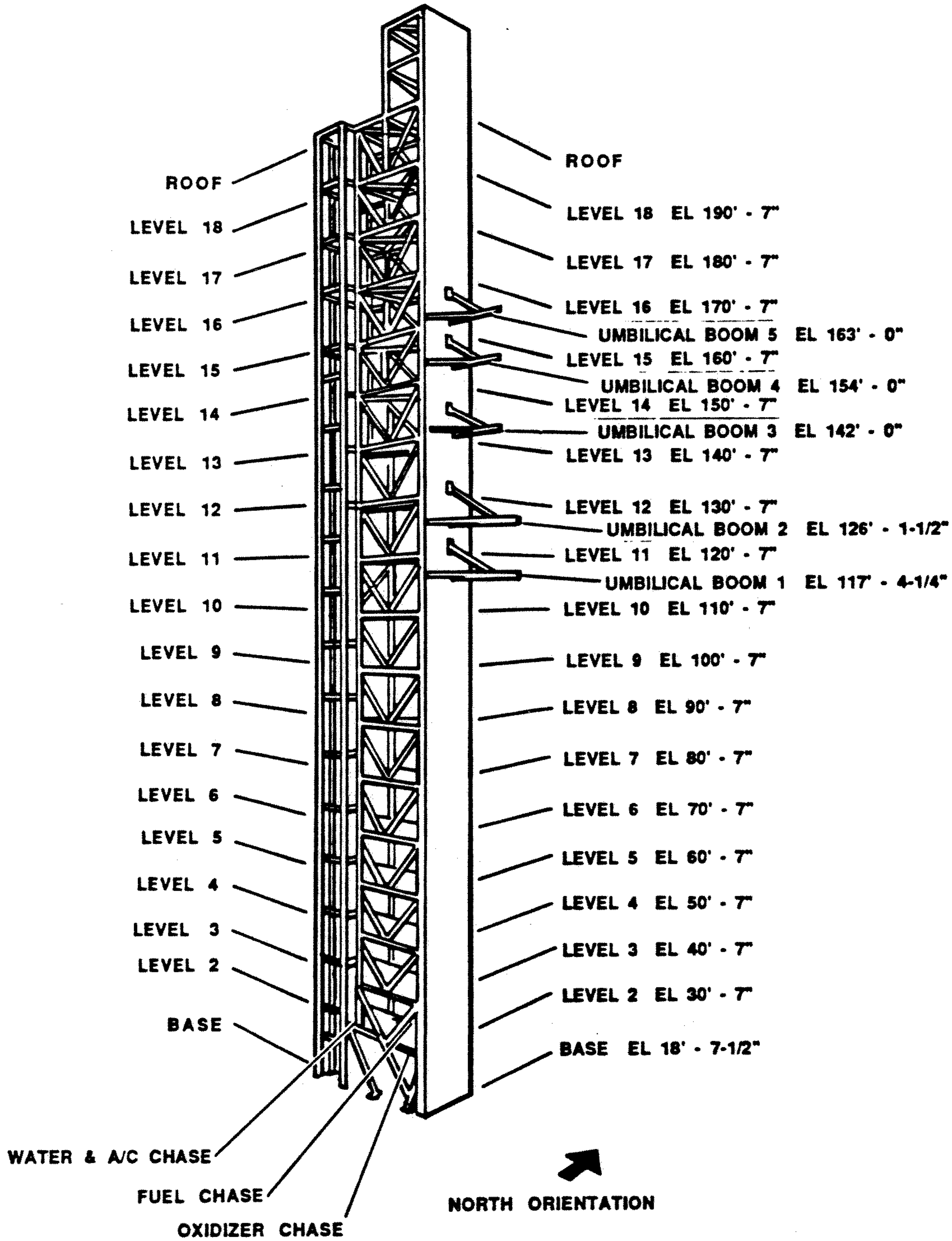


FIGURE 8.3.2-1 UMBILICAL TOWER

8.3.3 Launch Operations Building (LOB)

The LOB is a two-level reinforced concrete structure located north of the launch pad. The systems located in the basement level include: (1) the power supply and distribution system for DC power requirements for the LOB, (2) the ground instrumentation equipment, and (3) the PACE computer and peripheral equipment. The Launch Control Center (LCC) is located on the ground floor and includes the launch control console, facility control console, flight safety control monitor groups, and complex safety officer console, reference Figure 8.3.3-1.

8.3.4 Launch and Service Building (LSB)

The LSB is a single-level structure adjacent to the launch pad that contains the electronic and mechanical equipment necessary to bring the vehicle to readiness for launch countdown, reference Figure 8.3.4-1. Equipment that directly interfaces with the LV is located in the LSB. Equipment consists of PACE VCA/LCA Racks, a Flight Safety Checkout, Control Monitor Group, a GCMG, and a CSTSS. The LSB has a P/L support area (20 x 20 ft) for common use and has a propellant transfer and pressurization control set, nitrogen pressure controller, and air conditioning system. The LSB includes a room to store P/L hydrazine propellant and related propellant equipment.

8.3.5 Vehicle Assembly & C/O Building – Horizontal Test Facility (VAB-HTF)

At this facility the Titan IV Core Stages I & II are received & inspected, and engines checked out and installed. Stage II is partially linked to the forward skirt extension to support testing by use of its antennas. The SRM and PLF interfaces are simulated with test tooling and AGE. A "Hanger Queen" MGC and IMU are mounted on a test fixture and connected. Once this configuration is connected to static ground, test operations are performed.

8.3.6 Payload Air Conditioning System

The PLF and MST clean enclosure are supplied with conditioned air. The P/L air conditioning system provides class 5,000 clean air directly to P/L interfaces. The MST air conditioning building maintains class 100,000 clean environment in the MST clean enclosure.

8.3.7 Payload Propellant Load Systems

The P/L propellant load systems provide service to the P/L from 3 locations

a. The P/L oxidizer (N_2O_4) trailer pad, north of the LSB near the oxidizer holding area.

b. The fuel (MMH) trailer pad, south of the LSB near the fuel waste tank.

c. The propellant (N_2H_4) transfer unit, located in Room 104 of the LSB.

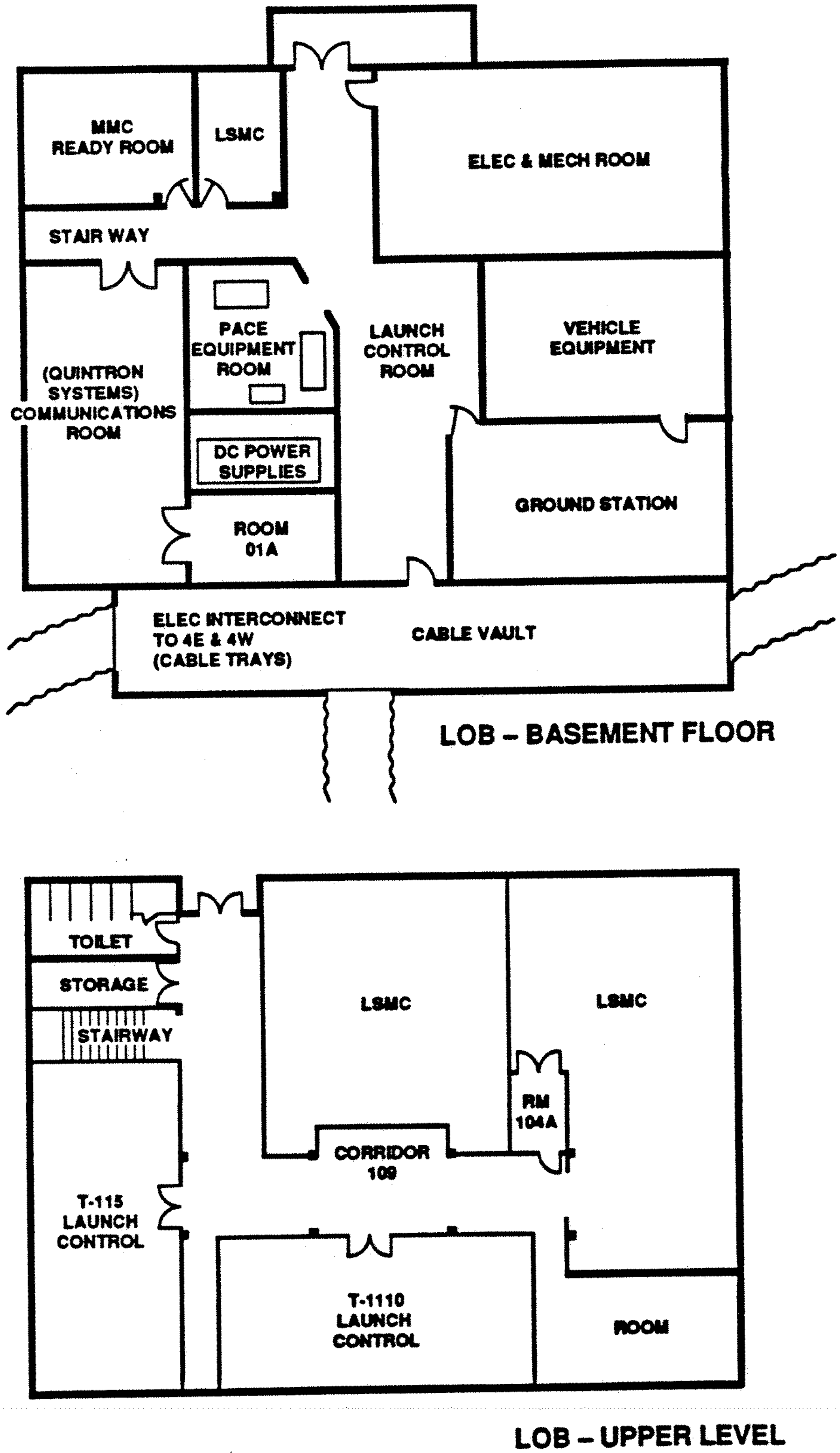


FIGURE 8.3.3-1 LAUNCH OPERATIONS BUILDING (LOB)

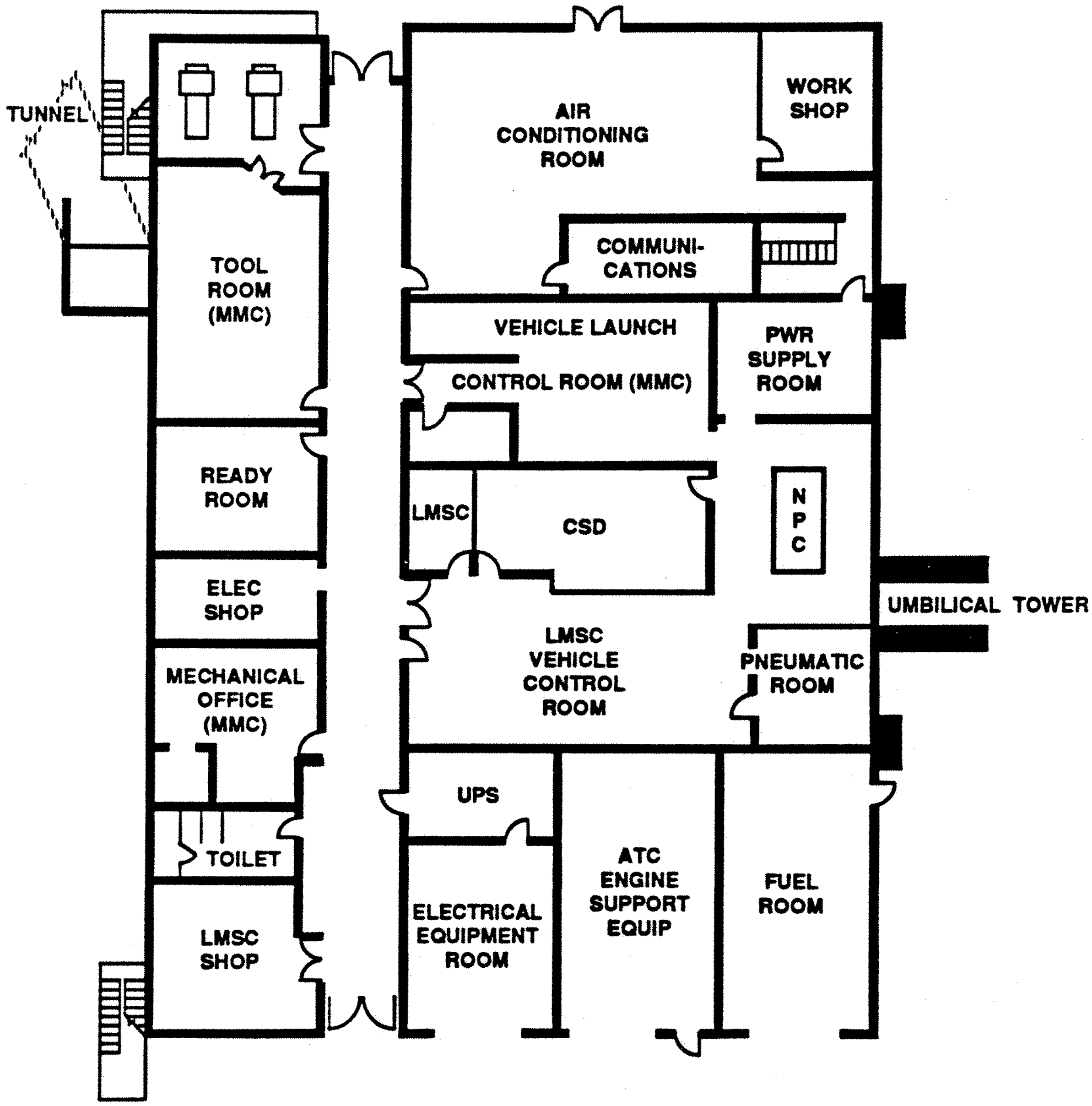


FIGURE 8.3.4-1 LAUNCH AND SERVICE BUILDING (LSB)

8.3.8 Payload Fairing Processing Facility (PLFPF)

The PLF will be received and processed in the PLFPF. In the PLFPF, all pre-launch processing for the PLF will be accomplished. This includes mechanical preparations, thermal coating application, electrical functional checks, final cleaning and weighing. The forward and aft PLF section will be transported to the launch complex and stored in the MST clean enclosure until required for installation on LV.

8.3.9 Offsite Interface Systems/Facilities

Booster and P/L telemetry are routed by land line from the vehicle to the SLC-4E Launch Operations Building for subsystems checkout (see Figure 8.3.9-1). P/L data can also be relayed to other facilities via microwave lines for prelaunch testing and launch from telemetry transmitters on the vehicle. Telemetry data is received and recorded at the Vandenberg Telemetry Receiver Site (VTRS) and the Vandenberg Tracking Station (VTS – a subsystem of the Air Force Satellite Control Facility). Booster telemetry is decoded and displayed by the Building 7000 Telemetry Integrated Processing System (TIPS). P/L data can also be decoded and displayed at TIPS or can be relayed to offsite user facilities. The ARIA aircraft receives and records booster and P/L data after the vehicle is out of range of the Vandenberg stations.

8.4 Aerospace Ground Equipment (AGE) at VAFB

8.4.1 VAFB/CCAFS AGE Differences

The electrical AGE in use at VAFB was adapted from the equipment in use at CCAFS and is similar in design and function. Significant differences are listed below:

a. The end items that are mounted in vans at the Cape are permanently installed in the LSB at VAFB.

b. The DTS is not used because discrete signals can be transmitted 1200 ft to the LOB without signal conditioning.

c. The diesel generator system located near SLC-4E is utilized to provide countdown electrical power to the complex and is considered as prime launch power. Backup power is supplied from a commercial line by Pacific Gas and Electrical Company.

8.4.1.1 Electrical-Electronic AGE

Electrical-Electronic AGE consists of PACE, a Flight Safety Checkout and Control Monitor Group, a Tracking & Flight Safety Monitor Group, a GCMG, a Combined Systems Test Simulator Set (CSTSS), and Power Distribution and Support Equipment. Equipment that directly interfaces with the LV is located in the LSB.

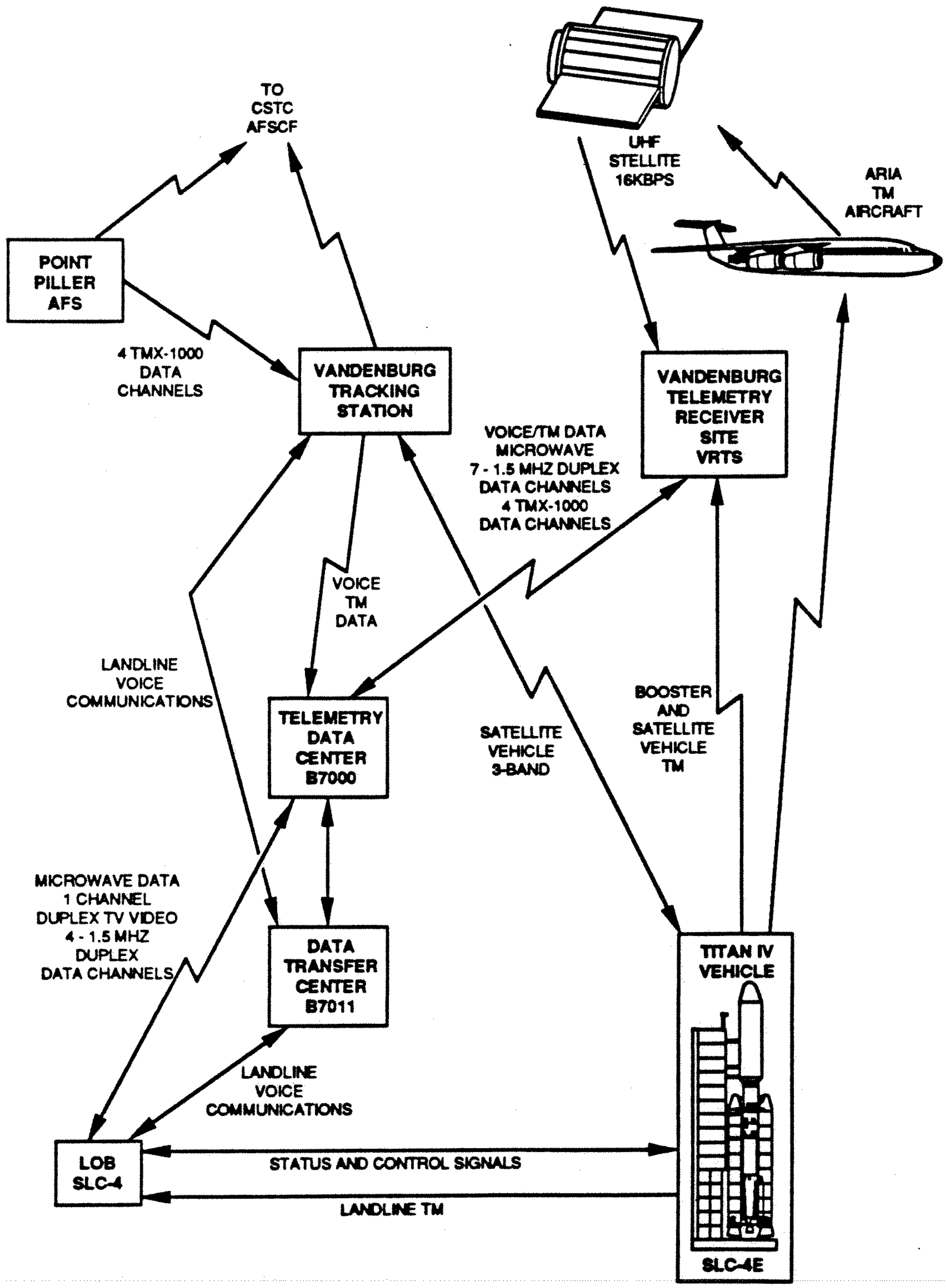


FIGURE 8.3.9-1 OFFSITE COMMUNICATIONS INTERFACES

8.4.1.2 Programmable Aerospace Control Equipment (PACE)

PACE is a computer driven command, control, checkout, and data monitoring system that performs automated end-to-end testing of vehicle subsystems. PACE functions as a readiness monitor and master countdown sequencer for the Titan IV GSE, LV and P/L. It automatically times the sequence of launch events from initiation of the terminal countdown through liftoff. It monitors and controls countdown status, hold status, and criteria violations in real time. PACE is made up of three minicomputers with three CRT/keyboards, two magnetic tape units, two line printers, two disk drives, Launch Control Assembly (LCA) and Vehicle Checkout Assembly (VCA). The computer and peripherals are located in the control center, and interface with the vehicle in the LSB is via the LCA and VCA. Tests are computer sequenced using vehicle test language application software to apply stimuli and monitor the outputs for the correct response. Go/No-Go status is displayed for the test in process with indications of channel response and any malfunctioning component. During the test, analog and digital data are recorded on magnetic tape and can be printed for post-test evaluation. Criteria violations are printed in real time during the test and will result in a hold or initiation of corrective sequences.

8.4.1.3 Launch Control and Monitoring

The launch countdown is controlled from the LCC in the LOB. This console controls vehicle power application, displays overall vehicle AGE Go/No-Go status, initiates terminal countdown, and provides manual abort capability. The monitor group for the T&FS system controls and monitors arm/safe commands for the Titan IV ISDS, arm/safe commands for the CRDs and engine shutdown circuitry. The console also monitors ground power, hold indications, SRM parameters, and IGS Go/No-Go status.

Support equipment includes ground instrumentation in the Martin Marietta Ground Station and power distribution and control. Equipment in the LOB Ground Station receives and decommutates PCM data from the booster TM system. This data is transmitted open-loop by the vehicle antenna or land line PCM output of the booster TM system for real-time recording and/or display. P/L data can be integrated into the LV PCM system and monitored in the Ground Station.

8.4.2 Mechanical/Structural AGE

Titan IV LV system mechanical AGE provides transportation, assembly and support facilities, pressurization gases and A/C, propellant services, handling equipment such as braidge cranes and elevators and personnel and equipment protection.

Chapter 9

Launch Operations at VAFB (SLC-4E)

9.0 LAUNCH OPERATIONS AT VAFB (SLC-4E)

9.1 Introduction

This chapter presents the Launch Operations Management, Launch Operations Documentation and Test Operations and Support at Vandenberg Air Force Base (VAFB). Reference Titan IV Program System Test Plan Western Test Range Vehicles MCR-87-2507, and Launch Operations Requirements Document MCR-88-2519.

Some changes in organizational roles and responsibilities occur at "Phase Point". The definition for Phase Point at VAFB is the completion of the LV CST or the start of the Satellite Vehicle mating process, whichever occurs first.

9.2 Launch Operations Management

9.2.1 General

Test requirements for launch operations testing shall be identified by the ACRBC, Launch Test Directive, Interface Test Specification, CI Specifications, engineering drawings and other applicable Government documentation.

9.2.2 Test/Launch Operations and Schedule

The Test Group will exercise technical test and schedule control over all system testing and will ensure that the contractors maintain integrity of the system hardware, software, and checkout equipment.

9.2.3 Flight Vehicle Integrated Test Operations

9.2.3.1 Joint/Integrated Test Conduct

Joint/integrated testing at the launch site will be controlled by the Test Group Launch Controller. Anomalies occurring during the conduct of a test will be reported to the appropriate LV/SV Test Conductor who will immediately inform the Launch Controller. The LV/SV TC will initiate necessary actions to coordinate the isolation and correction of Titan IV Launch Vehicle anomalies.

9.2.3.2 Integrated Test Scheduling

Meetings shall be held and chaired at a place and time as assigned by the Test Group for interchange of status, scheduling, test, and facility interface matters. Attendees shall consist of a representative from each of the affected contractors and a representative from all other agencies actively engaged in Launch Operations.

9.2.3.3 Launch Site Reviews

9.2.3.3.1 Incremental Reviews

Incremental Reviews are designed to verify Systems Readiness at a given time and to insure readiness to proceed to the next major increment.

9.2.3.3.1.1 Readiness Reviews

Readiness reviews are scheduled in advance of the milestone event to allow sufficient time for open or action items to be normally completed without schedule impact. The Satellite Vehicle Mate Readiness Review is held prior to transportation of the Satellite Vehicle to VAFB.

9.2.3.3.1.2 Launch Readiness Review

The Test Director presents the Space Division Commander and Mission Director with a final Launch Readiness Review (LRR) at R-1 day. This review culminates the prelaunch review cycle and results in a joint readiness statement by the Commander, Mission Director and Program Director.

9.2.3.4 Integrated Test Management

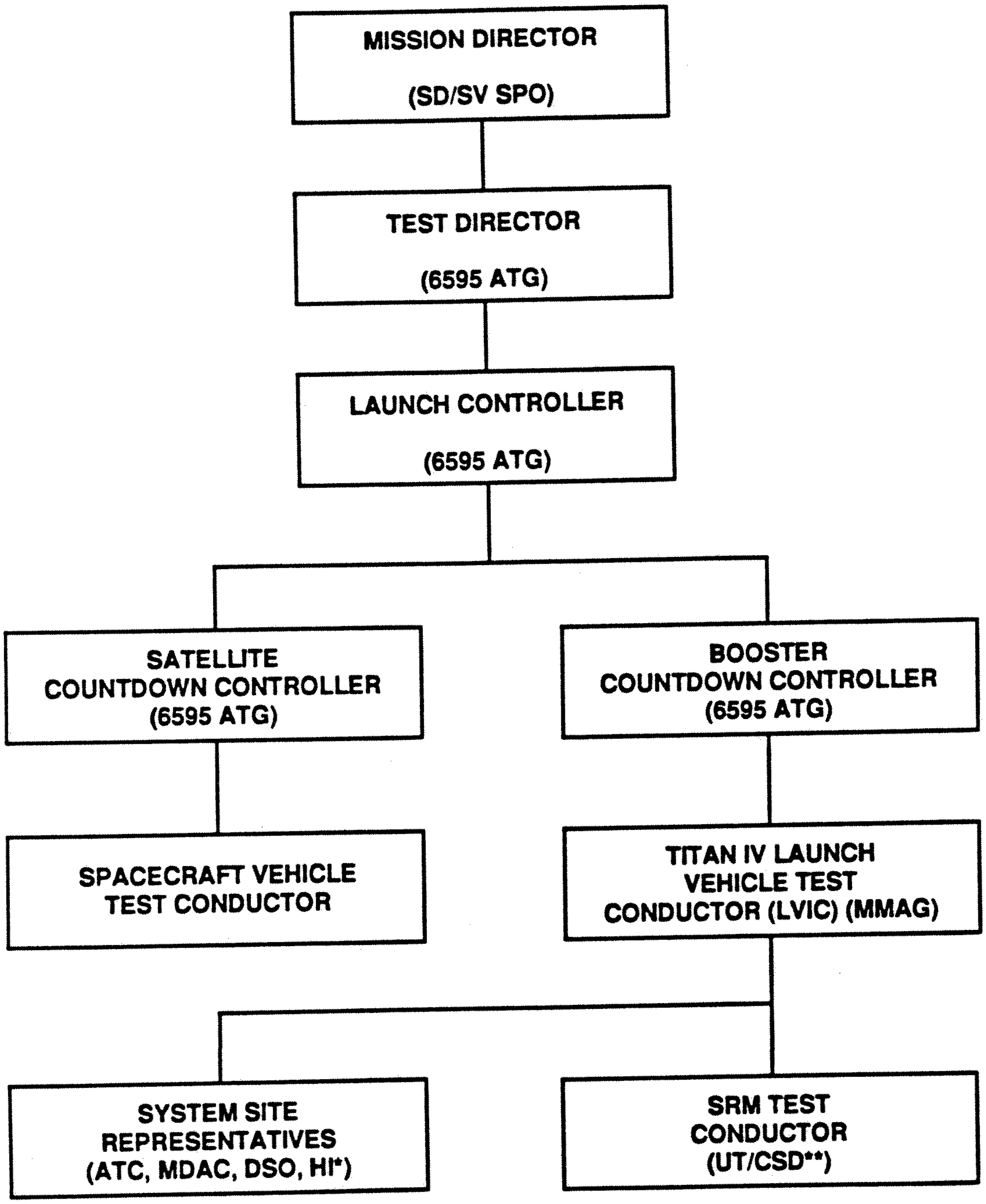
The key positions in the Integrated Test Management structure at VAFB are shown in Figure 9.2.3.4-1.

9.2.3.4.1 Mission Director – USAF

The Mission Director is designated by the USAF Space Division Commander, in coordination with the spacecraft SPO Director and is responsible for total Mission Success.

9.2.3.4.2 Test Director USAF

The Test Group Test Director is responsible to the Commander for the accomplishment of the overall launch operations objectives. The Test Director receives inputs from the Launch Controller, the Range Control/Safety Officer, etc. and evaluates the data to determine if the test objectives outlined in the System Test Objectives can be met. The Test Group Commander, or his designee, serves as the Test Director and is responsible to the Mission Director for the accomplishment of the overall launch operations objectives.



* FOR HI SRMU FLIGHTS ONLY

** FOR CSD SRM FLIGHTS ONLY

FIGURE 9.2.3.4-1 INTEGRATED TEST MANAGEMENT – VAFB

9.2.3.4.3 Launch Controller – USAF

The Test Group Launch Controller supervises launch operations and exercises overall technical test control. The Launch Controller is responsible to the Test Director (TD) for the technical and operational readiness of the hardware and resources and for managing the integrated test in progress as necessitated by other Range commitments, external factors that might affect the outcome of the test/operation, or other activities that are being performed concurrently. The Launch Controller is responsible for safety, security, and integrated schedules.

9.2.3.4.3.1 Test Controllers

Test Group Test Controllers are responsible for technical test control of specific systems test and related activities. Test Controllers will assist the Launch Controller during integrated systems testing and launch countdown and provide GO/NO-GO recommendations.

9.2.3.4.4 Booster Countdown Controller (BCC) – USAF

The Test Group BCC supervises the conduct of LV Launch Operations. He is responsible for the operational readiness of the LV. The BCC attends all Technical Interchange Meetings (TIMs) involving LV airborne equipment. He keeps the Launch Controllers apprised of problems and test results which affect the readiness of the LV. The BCC serves as the Air Force point of contact for all LVIC technical inputs.

9.2.3.4.5 Satellite Countdown Controller (SCC) – USAF

The Test Group SCC supervises the conduct of Satellite Launch Operations. He is responsible to the Launch Controller for the operational readiness of the S/V. The SCC attends all meetings involving satellite readiness. He keeps the Launch Controller apprised of all problems and test results which affect S/V readiness. The SCC serves as the Air Force point of contact for all S/V technical inputs.

9.2.3.4.6 Aerospace Corporation Technical Staff

An Aerospace Technical Representative is stationed at all critical areas, reviews anomalies/test results, and makes recommendations to the TD and Launch Controller.

9.2.3.4.7 Spacecraft Vehicle Test Conductor (SVTC)

The Spacecraft Vehicle Test Conductor (SVTC) is the P/L contractor person responsible for conduct of all SV tests and for the conduct of the SV Launch Countdown. The SVTC is responsible to and keeps the SV Countdown Controller and LV Test Conductor informed at all times of the current status of the SV and potential problems. The SVTC is designated to have technical and operational responsibility for:

a. Readiness of the Satellite AGE to support testing.
b. Conducting SV tests and countdown.
c. Integrated coordination of SV Directives and Operating Procedures with other agencies.

9.2.3.4.8 Titan IV Test Conductor (TC)

The LVIC TC is directly responsible for the conduct of the overall LV Receipt-to-Launch testing and countdown activities. The TC is responsive to test support requirements of Subcontractor TC/Representatives and keeps the Launch Controller and the BCC informed of the current status of the Titan IV Vehicle and potential problems. At Phase Point, the TC must have approval of the Launch Controller for deviations from the Integrated Test Procedures. He is designated to have operational responsibility for:

a. Readiness of the launch site to support tests.
b. Conduct of Titan IV LV tests.
c. Test preparations, schedule maintenance and test of the Titan IV LV, associated AGE, ground station instrumentation and launch site facilities.
d. Conduct of Titan IV LV Launch Countdown and direction of Launch Site Post-Launch Refurbishment.

9.2.3.5 Anomalies

In case of equipment anomalies encountered during test procedure run, a Procedure History Sheet (PHS), MARS and/or other applicable subcontractor/contractor documentation is used to initiate troubleshooting activities during integrated tests until the anomaly is resolved.

9.2.3.6 Post Test-Requirements

Following completion of integrated tests and launch countdown, Program Engineering schedules a meeting with participating contractors for Post Test Critique. System integrity, as defined in procedures, is maintained following a test. Break of integrity may be reason for retest and will require AF Quality Assurance notification. Final acceptance of integrated test results and approval to proceed is made by the Test Group at the Post Test Meeting.

9.2.3.7 Titan IV Launch Vehicle Recycle, Backout and Safing

The LVIC/LSIC and all Titan IV Launch Vehicle subcontractors have recycle, backout, and safing procedures to follow in the event of critical malfunctions or test interruptions of the Titan IV Launch Vehicle and associated systems during Countdown Operations.

9.2.3.8 SV Recycle and Safing

The SV Contractors prepare recycle, backout and safing procedures for the S/V and associated systems to use in the event of critical malfunctions or test interruptions during Countdown Operations.

9.2.4 Working Groups

9.2.4.1 Launch Test Working Group (LTWG)

The LTWG implements all Space Launch Vehicle and associated spacecraft programs at the Launch Base and manages those functions which involve the coordinated activities of the various agencies associated with the Prelaunch, Launch and Post Launch activities at the Launch Base. Accomplishment of the System Test Objectives, as specified for the Titan IV Launch Vehicle, and by associated spacecraft SPO is a primary goal of the LTWG. The LTWG is responsible for those functions related to the Titan IV Launch Vehicle and associated spacecraft programs at the Launch Base from the initial planning phases (including hardware acceptance) through launch of vehicles, acquisition of flight data, and the submission of applicable reports. The period of activity of the LTWG extends throughout the interval of the Titan IV and associated spacecraft programs.

9.2.4.2 Facility Working Group (FWG)

The FWG is responsible for reviewing and making recommendations on all Facility Change Requests (FCR) and Facility Engineering Change Proposals (FECP) submitted, and for maintaining configuration of all active Titan IV facilities. LVIC is responsible for controlling facility configuration and for scheduling implementation of changes. The FWG is chaired by the Test Group for all items relating to the Titan IV Program. LVIC serves as the recorder and as a member to work all items relating to the Titan IV program. FWG membership includes an LSIC Representative.

9.2.4.3 Safety Working Group (SWG)

The SWG is responsible for establishing/implementing safety policy and requirements for the Titan IV Program. The Pad Safety SOPs and Launch Site Range Safety Regulations serve as the media for defining safety policy and requirements for the Launch Sites. The SWG is chaired by the responsible Test Group and membership is open to all program participants.

9.3 Launch Operations Documentation

9.3.1 Titan IV Integrated Program Schedules

The Titan IV LVIC Level 7 Program Schedule Document is the master plan and site scheduling document which reflects major milestones for LV delivery, test operations and launch operations at the Launch Base. This document is prepared by LVIC and is used as input to the 6595 ATG Integrated Program Milestone Schedule. This 6595 ATG schedule is the official master plan and site scheduling document which reflects major milestones for all flight vehicle processing activities and Launch Operations. It integrates the schedules of all contractor and support agencies. It provides authority for contractor launch base work planning to meet milestones and implements scheduling direction from Space Division.

9.3.2 Daily Integrated Test Schedules

The Daily LVIC Level 8 Launch Vehicle Test Schedules for Titan IV Launch Vehicle identifies the testing and operations planned over a specified period. It also identifies all support requirements for schedule tasks. The daily integrated Launch Vehicle Test Schedule is assembled, published and distributed by LVIC and used as input to 6595 ATG Daily Integrated Launch Base Schedule. This 6595 ATG schedule is the official daily work schedule that integrates all Titan IV Launch Base activities.

9.3.3 Acceptance, Checkout, Retest, Backout Criteria (ACRBC)

The ACRBC, prepared by LVIC, establishes minimum integrated Titan IV LV, checkout and testing requirements and criteria at the Launch Site.

9.3.4 Launch Test Directive (LTD)

The LTD input, provided by the LVIC, identifies launch constraints on Titan IV Launch Vehicles. The LTD is based on information contained in the ACRBC and other approved documents.

9.3.5 Launch Constraints Document (LCD)

The LCD is prepared by the LVIC. The LVIC provides LCD inputs in the form of a Launch Test Directive for the Titan IV Launch Vehicle. The SV Contractor/Representative provides LCD inputs for the Satellite Vehicle. The LCD is approved by the PPO, SD and the 6595 ATG prior to publication.

9.3.6 System Test Objectives (STO)

The STO report is published by the SPO to define the mission objectives to be followed during Prelaunch, Launch and Flight of Titan IV Space Launch Vehicles. It contains general information for mission planning and support.

9.3.7 Test Procedures

9.3.7.1 Receipt-to-Launch Test Sequence

The test sequence is a Receipt-to-Launch logic network which identifies vehicle test activities that are scheduled at VAFB. It is an LVIC procedure which delineates all Launch Vehicle activity by test procedure number.

9.3.7.2 Titan IV Launch Vehicle Test Procedures

These procedures include Core Vehicle, Core Guidance, Liquid Rocket Engine, Payload Fairing, Launch Vehicle, and Solid Rocket Motor test procedures.

9.3.7.3 Titan IV Flight Vehicle Test Procedures

9.3.7.3.1 Integrated Test Procedures

Integrated Test Procedures are procedures used to operate and test the complete Flight Vehicle in a launch configuration. Integrated test procedures, i.e., IST and Countdown Manuals, are prepared by the LVIC from 6595 ATG and participating contractors' input, for recurring and nonrecurring test activity, including Launch Countdown. Integrated test procedures for the Titan IV Flight Vehicle Program are reviewed and approved by the LVIC, ATG and applicable participating organizations.

9.3.7.3 2 Joint Test Procedures

Joint Test Procedures are procedures used to test systems within the Flight Vehicle in which more than one contractor has responsibility for performance, i.e., PLF Mate, SV Mate, etc. Joint procedures are prepared by the contractor holding the prime responsibility and approved by the supporting contractor(s) and the 6595 ATG.

9.3.7.4 Countdown Manual

The Countdown Manual provides a step-by-step integrated check list of all system operations from the start of the count through launch. In addition, it provides contingency procedures for recycle and backout in the event of Countdown Termination. The Countdown Manual is prepared by the LVIC for inputs from all participating agencies. The Countdown Manual for the Titan IV Flight Vehicle program is reviewed and approved by the LVIC, Test Group, and applicable participating organizations.

9.3.8 Range Documentation

9.3.8.1 Program Requirements Document (PRD)

The early planning information contained in the Program Introduction and Statement of Capability is expanded in the PRD to a complete and detailed statement of Range User requirements. The PRD consists of a series of standardized UDS form pages, each pertaining to some specific requirement, such as trajectory data, telemetry data or supplies/services needed. After preparation, the PRD is submitted by LVIC to SD for approval and processing.

9.3.8.2 Program Support Plan (PSP)

The PSP documents are submitted by WSMC to SD and 6595 ATG, responding to how the requirements of the PDRs will be implemented.

9.3.8.3 Operations Requirements (OR) Document

The OR document outlines Range Support Requirements for a major test or flight plan and is derived from the PRD, PSP and inputs from all contractors. The LVIC, contractors as applicable, and S/V Contractor provide OR inputs to the 6595 ATG.

9.3.8.4 Operations Directive (OD)

The OD is the final document in the UDS and is the Official Range Directive which mobilizes and assigns the resources necessary to support the Range User's requirements shown in the OR.

9.4 Test Operations and Support

9.4.1 Vehicle Operations

Vehicle Operations consists of Receipt, Vehicle Erection, Subsystem Installation as required, Checkout and Launch of Titan IV and attendant operations, maintenance, post-launch securing and post-launch refurbishment of AGE.

9.4.2 Post-Test and Launch Evaluation

Data is collected following test or launch for evaluation by the responsible system contractor under the cognizance of SD.

9.4.3 Operations Support

9.4.3.1 Titan Integrated Corrective Action System

The Integrated Corrective Action System is in accordance with the requirements of the System Effectiveness Plan.

9.4.3.2 Contractor Maintenance Area

A CMA is operated by LVIC. These services are made available to program contractors in the event of emergency.

9.4.3.3 Crew Certification

At the LRR, each prime contractor certifies that its Launch Crew Personnel are qualified to perform Launch Operations.

Appendix A

Spacecraft
Questionaire

The items listed on the spacecraft questionnaire are representative of most spacecraft requirements; however, additional requirements, when applicable, should be included when the questionnaire is submitted. The questionnaire is to be completed by the Launch Systems Integration Contractor (LSIC) and must by validated by the Spacecraft Program Office (SPO); it must be revised if mission or spacecraft requirements change. All interface requirements identified in the completed questionnaire will be documented in an interface control drawing. The format and paragraph numbers shown below should be followed for traceability.

1.0 MISSION REQUIREMENTS

1.1 Launch Schedule

1.2 Spacecraft Orbit Parameters

1.2.1 Apogee and Limits

1.2.2 Perigee and Limits

1.2.3 Inclination and Limits

1.2.4 Eccentricity and Limits

1.2.5 Argument of Perigee

1.2.6 Longitude of Ascending Node

1.2.7 Right Ascension of Ascending Node (RAAN)

1.3 Launch Window Parameters

1.3.1 Orbit Lifetime

1.3.2 Eclipse Time Constraints

1.3.3 Orbit Inclination Limits versus Time

1.3.4 Ascending Node

1.3.5 Window Duration (over a year)

1.4 Spacecraft Separation Parameters

1.4.1 Velocity Limits

1.4.2 Rate-of-Angular-Change Limits (pitch, yaw, roll)

1.4.3 Timing (multiple-release time delays)

1.4.4 Latitude and Limits

1.4.5 Longitude and Limits

1.4.6 Orientation and Limits (pitch, yaw, roll axes)

1.4.7 Separation Directions (e.g., in plane, perpendicular to plane, and northerly

1.4.8 Acceleration Constraints during Deployment

1.4.9 Attitude Constraints during Deployment

1.4.10 Pyro Shock Environment

1.5 Preseparation Functions

1.5.1 Equipment Deployment Identity (e.g., spatial envelope, c.g. changes)

1.5.2 Equipment Deployment Timing (e.g., time before separation, time between events)

1.5.3 Acceleration Constraints (pitch, yaw, roll, longitudinal) during and after deployment

1.5.4 Equipment Deployment Constraints

1.5.5 Attitude Constraints before Deployment

1.6 Post-separation Requirements

1.6.1 Operations (e.g., complete shutdown, special orientation)

1.6.2 Post-separation Maneuver Requirements (e.g., time delays, separation distances before initiation, orientation, new orbit identification, minimum velocity additions)

1.6.3 Acceleration Constraints after Deployment

1.6.4 Attitude Constraints after Deployment

1.7 Special Trajectory Requirements

1.7.1 Airload Limits

1.7.2 Free Molecular Heating Constraint

1.73 Thermal Maneuvers

1.7.4 Spacecraft Release Over Telemetry or Tracking Ground Station

1.7.5 Maximum or Minimum Position Drift Rate (synchronous orbit only)

1.7.6 Telemetry Maneuvers

1.7.7 Payload Release in RV View of Tracking Satellite

1.7.8 Solar Insulation Vector

2.0 SPACECRAFT CHARACTERISTICS

2.1 Physical Envelope

2.1.1 Shape (e.g., dimensional assembly drawing)

2.1.2 Volume

2.1.3 Coordinate System (spacecraft relative to launch vehicle)

2.1.4 Kick Stage (i.e., manufacturer's designation, thrust, specific impulse, burn action time, and propellant off-load limit

2.2 Mass Properties

2.2.1 Weight (e.g., total, separable, and retained masses)

2.2.2 Center of Gravity (e.g., cg locations about the three reference axes of total, separable, and retained masses)

2.2.3 Moment of Inertia (e.g., inertia about the three reference axes of total, separable, and retained masses)

2.2.4 Product of Inertia (e.g., product of inertia with respect to the three-axis c.g. of the total, separable, and retained masses

2.2.5 Propellant Slosh Characteristics

2.3 Structural Stiffness

2.3.1 Natural Frequencies below 50 Hz

2.3.2 Mode Shapes below 50 Hz

2.3.3 Interface Stiffness Matrices

2.3.4 Mass Matrices (dynamic model)

2.3.5 Degree-of-Freedom Table

2.3.6 Dynamic Model Coordinate System

2.3.7 Elastic Mode Matrix to 50 Hz

2.3.8 Constraint Mode Matrix

2.3.9 Modal Damping Matrix

2.3.10 Discrete Acceleration Transformation Matrices

2.3.11 Deflection and Clearance Loss Transformation Matrices

2.3.12 Load Transformation Matrices

2.4 Handling Constraints

2.4.1 Ground Transport (e.g., orientation with respect to earth)

2.4.2 Handling Limits (e.g., acceleration constraints)

2.5 Critical Orientation - During Checkout, Prelaunch and On-Orbit

2.5.1 Antennas (e.g., location, direction, beamwidth)

2.5.2 Sensors (e.g., location, look angle)

2.5.3 Solar Arrays (e.g., location, size)

2.5.4 Other Critical Elements

2.6 Ordnance Items

2.6.1 Identity and Function (quantity and type in accordance with AFETRM 127-1, manufacturers part number, location, usage, etc.)

2.6.2 Installation (i.e., when, where, and by whom supplied, installed, and connected)

2.6.3 Shorting Plugs (i.e., when, where, and by whom supplied and installed)

2.6.4 Electrical Characteristics (e.g., no-fire current, all-fire current, bridgewire resistance pin-to-case resistance)

2.6.5 RF Susceptibility (i.e., pin-to-case, pin-to-pin, and bridgewire-to-bridgewire RF susceptibility and input impedance at the following frequencies: dc, 243, 2300, 5000, 5700, and 900 MHz)

2.6.6 Electrostatic Sensitivity Data

2.6.7 AKM Flight Termination Data (identify type, manufacturer's part number, location, initiation method, etc.)

2.7 Special Safety Items

2.7.1 Item Identity (e.g., cryogenics, corrosive fluids, high pressure gases, nuclear items)

2.7.2 Item Characteristics (e.g., quantity, pressure, radiation levels)

2.8 Thermal Characteristics

2.8.1 Thermal Math Model (i.e., an analytical tool containing the thermal nodes and conductors; the emissivity, absorptivity, conductivity, and resistivity characteristics; and the thermal node view factors of the spacecraft)

Complexity of model will be dependent on accuracy needed. A model of boundary nodes may suffice; however, no model may be required for launch vehicle purposes

2.8.2 Heat Sources (i.e., identity, location, orientation; and amount, times, and duration of heat generated)

2.8.3 Node Limits (i.e., maximum and minimum allowable temperatures per node)

2.8.4 Sun Angle ( ) and Limits

2.9 Contamination Control

2.9.1 Ground Conditions (e.g., constraints before and after PLF installation)

2.9.2 Inflight Conditions (e.g., during ascent trajectory prior to and after jettisoning PLF)

2.9.3 Critical Surfaces (e.g., type, size, location)

2.9.4 Surface Sensitivity (e.g., susceptibility to propellants, gases and exhaust products)

2.10 Separation System

2.10.1 Separation Mechanism (e.g., design of fasteners, clamps, pyro devices, springs, lanyards, and loose hardware)

2.10.2 Separation Loads (e.g., reactions, shock, spinup torques)

2.10.3 Separation Rates (e.g., longitudinal, lateral and radial)

2.10.4 Separation Method (e.g., discrete or uplink required)

2.11 Grounding Philosophy

2.11.1 Structure (e.g., use of structural grounds, current levels)

2.11.2 Electrical Equipment (e.g., grounding technique for black boxes, power supplies)

2.11.3 Single-Point Ground (e.g., location, related equipment)

2.12 RF Radiation

2.12.1 Characteristics (e.g., power levels, frequency, duration)

2.12.2 Locations (e.g., location of spacecraft when radiating, location of receivers)

2.12.3 Checkout Requirements (e.g., open-loop, closed-loop, prelaunch, and ascent trajectory)

3.0 AVE REQUIREMENTS

3.1 Mechanical Interfaces

3.1.1 Mounting Points (e.g., location, number, size, type of mounting hardware)

3.1.2 Mounting Alignment (e.g., rotational and planar position requirements relative to local horizon)

3.1.3 Mounting Surface Characteristics (e.g., coplanarity, flatness)

3.1.4 Service Items (e.g., location, size, and type of commodity lines, special attachments)

3.1.5 Connector Details (e.g., manual, flyaway, umbilical, or inflight operation; manufacturer's part number)

3.1.6 Commodity Requirements (e.g., type, pressure, flowrate, temperature, quantity, timing)

3.2 Electrical Interfaces

3.2.1 Connector Items (e.g., location, function)

3.2.2 Connector Details (e.g., flyaway, umbilical, or inflight operation; manufacturer's part number; potting type; type of potting compound; wire size; pin assignment)

3.2.3 Bonding Requirements (e.g., interface materials, finishes)

3.2.4 Shielding Requirements (e.g., each conductor, entire connector, shield grounding, maximum shield ground length, shield ground through connector)

3.3 Electrical Power - Requirements

3.3.1 28-vdc Control Power (i.e., current, duration, function, time, tolerances)

3.3.2 28-vdc Ordnance Power (i.e., current, duration , function, time, tolerances)

3.3.3 28-vdc Instrumentation Power (i.e., current, duration, function time, tolerances)

3.3.4 10-vdc Instrumentation Power (i.e., current, duration, function, time, tolerances)

3.3.5 Other Power Requirements

3.3.6 Overcurrent Protection Requirements,

3.3.7 Number, Types, and Sequences of Discretes

3.3.8 Redundancy Requirements

3.4 Telemetry Requirements

3.4.1 Measurement Identity (e.g., temperature, pressure, switch closure)

3.4.2 Signal Characteristics (i.e., discrete bilevel, ananlog maximum response, sampling rate, condition represented by on-off or high-low state)

3.4.3 Signal Conditioning Requirements (e.g., input impedance, impedance circuit load limits, overcurrent protection, signal-to-noise ratio)

3.4.4 Circuit Identity (e.g., connector number, pin number, wire size)

3.4.5 Time of Use (e.g., during checkout, prelaunch, inflight)

3.4.6 Tempest

3.4.7 Encryption

3.5 Ordnance Safing/Arming

3.5.1 Item Identity (i.e., function, location)

3.5.2 Accessibility (e.g., direct access, remote circuitry, manual or automated activation)

3.5.3 Sequence (i.e., installation and removal times)

3.6 Material Requirements

3.6.1 Compatibility (e.g., finishes, electrolytic aspects, chemical)

3.6.2 Outgassing (i.e., constraints on nature and quantity of constituents)

3.6.3 Contamination (e.g., retrometer) Constraints

3.7 Preflight Environment

3.7.1 Requirements (e.g., cleanliness, temperature, relative humidity, air conditioning and air impingement limits in each area from spacecraft delivery through terminal countdown)

3.7.2 Lost Environment (i.e., effects on payload if environment is lost, and recovery plan)

3.7.3 Monitoring and Verification Requirements

3.8 Payload Fairing (PLF) Requirements

3.8.1 Heating (e.g., temperature and temperature change rate limits)

3.8.2 Venting Characteristics (i.e., quantity, timing, and nature of gases vented from payload)

3.8.3 Pressure (i.e., pressure limits inside PLF)

3.8.4 RF Transmission (e.g., size, location and orientation of RF-transparent windows in PLF)

3.8.5 PLF Separation (e.g., desired separation altitude, shock, and acoustics limits)

3.8.6 Other PLF Constraints

3.8.7 Acoustic Blankets (may require further analysis)

3.8.8 Access Door Panels

3.8.9 Bolt-Ons

4.0 AGE/FACILITY REQUIREMENTS

4.1 Spacecraft/Launch Vehicle Integration

4.1.1 How Accomplished (i.e., sequence from spacecraft delivery through attachment to launch vehicle)

4.1.2 Handling Equipment (e.g., description, who supplies and operates, when used, where stored)

4.1.3 Spacecraft Covers (e.g., description, who supplies and uses, when used, where stored)

4.1.4 Spatial Envelopes (e.g., dimensions and clearance requirements for containers, handling equipment, and payload covers; work area requirements)

4.1.5 Fairing Installation (i.e., desired mode of payload fairing Installation; e.g., segment-by-segment, or as a unit)

4.1.6 Equipment Disposition (e.g., disposition of containers, handling equipment, and spacecraft covers after the environmental shelter is closed and/or after the PLF is installed)

4.1.7 Support Services (i.e., services required and expected request means; e.g., range program requirements document, interface specification, etc.)

4.2 Spacecraft Checkout AGE

4.2.1 Use and Storage (i.e., identity, locations of use and storage)

4.2.2 Installation Criteria (e.g., weight; volume; tiedown requirements; who supplies, installs, operates, and removes; when supplied, installed, used and removed; post-use disposition)

4.2.3 Compatibility (e.g., compatibility with range safety requirements and launch vehicle propellants)

4.2.4 Space Requirements (i.e., unusual work/operating space requirements)

4.2.5 Payload Checkout Procedures

4.3 Preflight Environment Protection

4.3.1 Equipment (i.e., identity, characteristics, and location of required environmental protection equipment)

4.3.2 Operation (i.e., who supplies, installs, operates and removes; when supplied, installed, used and removed; post-use disposition)

4.4 Spacecraft Access

4.4.1 Mechanical Attachment (i.e., requirements for direct access from delivery through attachment of the spacecraft to the launch vehicle)

4.4.2 Before Fairing Installation (i.e., requirements for direct access after closing environmental shelter closure and before installing PLF)

4.4.3 After Fairing Installation (i.e., requirements for direct access after PLF is installed)

4.4.4 Terminal Countdown (i.e., requirements for direct access during the terminal countdown)

4.5 Umbilicals

4.5.1 Identity (i.e., function and location)

4.5.2 Source (e.g., spacecraft or launch vehicle AGE)

4.5.3 Support and Control (e.g., structural support requirements, retraction mechanisms)

4.5.4 Installation (e.g., when and by whom supplied and installed)

4.6 Hard Lines

4.6.1 Mobile Service Tower (i.e., identity and requirements for service or safety plumbing, cabling and conduit)

4.6.2 Umbilical Tower (i.e., identify and requirements for service or safety plumbing, cabling and conduit)

4.6.3 Installation (e.g., when and by whom supplied and installed)

4.7 Commodities

4.7.1 Identity (e.g., quantity of gases, propellants, chilled water)

4.7.2 Source (e.g., supplied by launch vehicle common services or spacecraft)

4.7.3 Timing (i.e., time that commodities are required and expected transfer-time duration)

4.8 Electrical Power

4.8.1 AGE Requirements (e.g., voltage and frequency limits, consumption characteristics of power for spacecraft-related facility items at the Satellite Assembly Building, Satellite Operations Center, and Launch Complex)

4.8.2 Facility Requirements (e.g., voltage and frequency limits, consumption characteristics of power for spacecraft-related facility items at the Satellite Assembly Building, Satellite Operations Center, and Launch Complex)

4.8.3 Continuous Power; UPS, Standby Diesel Generator; Purity

4.9 RF Transmission

4.9.1 Antennas (i.e., function, location, physical characteristics, beam-width, beam direction, frequencies and relationship to other antennas and receivers/ transmitters)

4.9.2 Transmission (e.g., frequency power)

4.9.3 Operation (e.g., nature of use, operating time requirements)

4.10 Monitors and Control

4.10.1 Signals (i.e., identity of monitor signals from spacecraft and/or AGE during readiness and terminal countdown, and time that signals are to be monitored)

4.10.2 Transmission (e.g., via payload telemetry, launch vehicle telemetry, and landline)

4.10.3 Evaluation (e.g., location of data evaluation center, evaluation responsibility, measurement limits, go/no-go constraints)

4.10.4 Launch Control Center (i.e., description and use of launch control center by spacecraft)

4.11 Security

4.11.1 Proposed Security Plan (e.g., visual barrier requirements, special clearances, and security requirements imposed on Titan II SLV

4.11.2 Identification of Special Physical Access and Data Access Security Requirements during all Operations

5.0 SYSTEM TESTS

5.1 Test Plan (i.e., discussion of each test that can affect or be affected by the launch vehicle, such as - structural static test, ground vibration survey, electromagnetic compatibility, RF interference, environmental demonstration, interface compatibility, installation verification and combined system test. Discussion should include details of test hardware, simulators and pages needed, testing responsibility, etc.)

5.2 Test Schedule (i.e., time-based schedule showing test duration and milestones)

6.0 LAUNCH OPERATIONS

6.1 Flow Chart (i.e., detailed sequence and time span of all spacecraft-related launch site activities, including AGE installation, facility installations and activities, spacecraft testing, and spacecraft servicing)

6.2 Restrictions (e.g., launch site activity limitations, constraints on launch vehicle operations, security requirements, personnel access limitations, safety precautions)

6.3 Support (e.g., personnel, communications, data reduction, requirements from other contractors)

6.4 Terminal Countdown (i.e., detailed sequence and time spans of all spacecraft-related events, including recycling requirements, ordnance installation, equipment activation, personnel activities, etc.)

6.5 Launch and Flight (e.g., requirements for real-time data readout, postflight data analysis, data distribution, postflight facilities and AGE disposition)

Appendix B

Independent Verification and Validation

Centaur Engineering Analysis Support – Independent Verification and Validation

Analex

1. Flight Control Stability
2. Centaur Flight Critical DCU Software
   - DCU Airborne Software
   - Software IV&V Planning
   - Specification Analysis
   - Software Math Models and Tools
   - Software Quality Assurance
   - Flight Software Interfaces
   - IMU Calibration and Alignment Model

Structural Dynamics Research Corporation (SDRC)

1. Centaur Loads and Dynamics IV&V
   - Model Verification
   - Loads Validation

MDAC – Core Loads and Dynamics Independent Verification and Validation

1. Model Verification
2. External Loads Verification
3. Internal Loads Verification
4. Substructural Model Development and Validation
5. Model Data Validation
6. External Loads Validation
7. Integrated Booster Models
8. Validate Internal Loads and Load Combinations

DELCO – Core Software Independent Verification and Validation

1. Requirements Identification and Traceability
2. Specification Analysis
3. Flight Program Analysis and Compliance Verification
4. Validation of Airborne Software to Insure Compliance with Mission Requirements
5. TAG Cycle Verification and Validation
6. Participate in Day of Launch Support

Martin Marietta, Orlando

1. Autopilot Stability
2. Guidance Error Analysis – Core
3. Limit Cycle Analysis

Martin Marietta, Denver

1. Injection Error Analyses – Centaur
2. CRD

Appendix C

Weight Charged to Spacecraft

The following list of items are representative of those things that are typically charged against the SC (S/V, P/L) weight allocation. In general anything added that is not needed for the basic BV flight, is chargeable to the S/V

1. Modification to CP 2460, 2490, 2491, 2492, and 2500 Skirts

2. Modification to additions to Titan IV electrical cabling/harnesses

3. Pyrotechnic Controllers
   - Harness, Cables
   - Support Brackets, Panels and Disconnects

4. Additional Batteries and Associated Hardware

5. Air Conditioning
   - Umbilical Access in PLF
   - Ducting, Showerhead
   - Diffusers
   - Baffles, Deflectors

6. Liquid Cooling Lines to Equipment in Spacecraft

7. Acoustic Attenuator Baffles

8. PLF Unique Acoustic Blankets

9. Instrumentation
   - Measurement Devises
   - Cabling
   - Attachment Hardware

10. PLF Tailored Access Doors

11. PLF Vent Modifications

12. Electrical Umbilical PLF Interfaces

13. Telemetry Kits

14. Spacecraft Separation Mechanisms and Associated Hardware/Cabling

15. RF Shielding/Barrier

16. SC Adapter and Attachment Hardware

17. PLF Separation Pull-away Hardware Attached to the SC Adapter and Related TIV Interface Connectors

18. Plume Deflector Shields/Mechanisms

19. RIG Interface Hardware

20. Power Transfer Hardware

21. Thermal Insulation

22. PLF Bolt-ons

23 Additional Attitude Control System Hardware for Certain Mission Unique Maneuvers

24. SC Snubber

25. Special EMC/EMI Required Hardware

26. SC Adapter Changes due to Human Factors Engineering, Maintainability, Alignment and Thermal Radiation Path Requirements

27. S/V Separate Mechanism

28. 2A3E Umbilical SV Interface H/W

29 Flight Termination System

30. WIS

31. Personnel Access Doors

32. Vent Covers

33. Acoustic Blankets on Doors

Appendix D

EMC/EMI Interface Guidelines

APPENDIX D

EMC/EMI INTERFACE GUIDELINES FOR TITAN IV PAYLOADS

1.0 INTRODUCTION

This document provides guidelines and data for the development of the EMC/EMI interface requirements and the compatibility analysis between the payload and each of three Titan IV launch vehicle configurations.

These three Titan IV configurations are represented in Figure 1 and are the:

A. Titan IV w/Centaur Upper Stage (Centaur
B. Titan IV w/No Upper Stage (NUS)
C. Titan IV w/Inertial Upper Stage (IUS)

These three Titan IV launch vehicle configurations are referred to as "Centaur", "NUS", and "IUS", respectively, throughout this document.

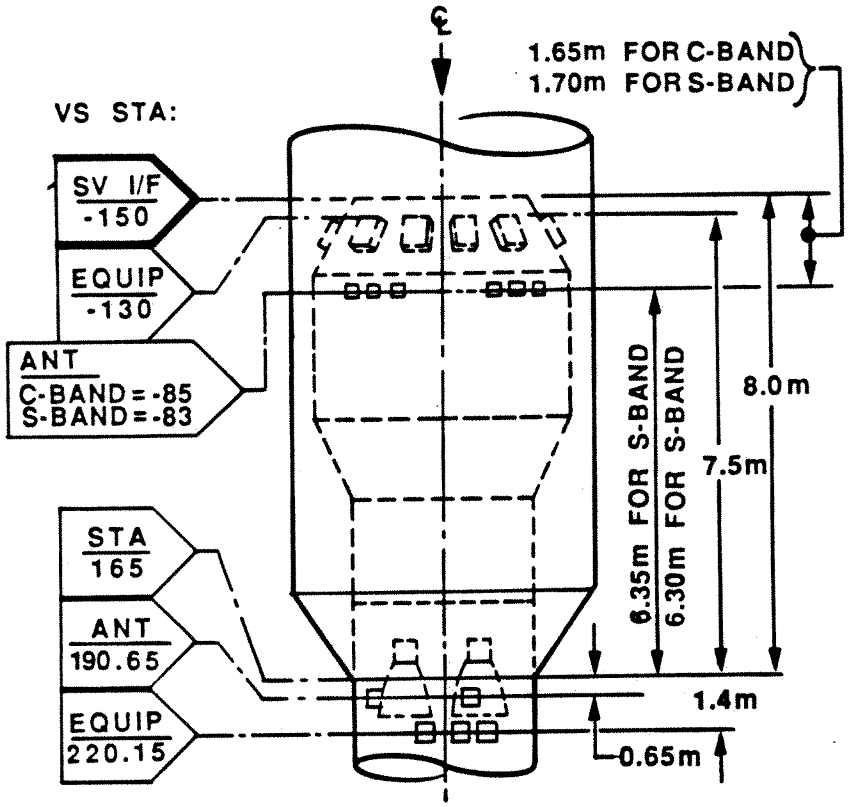


FIGURE 1A CENTAUR CONFIGURATION

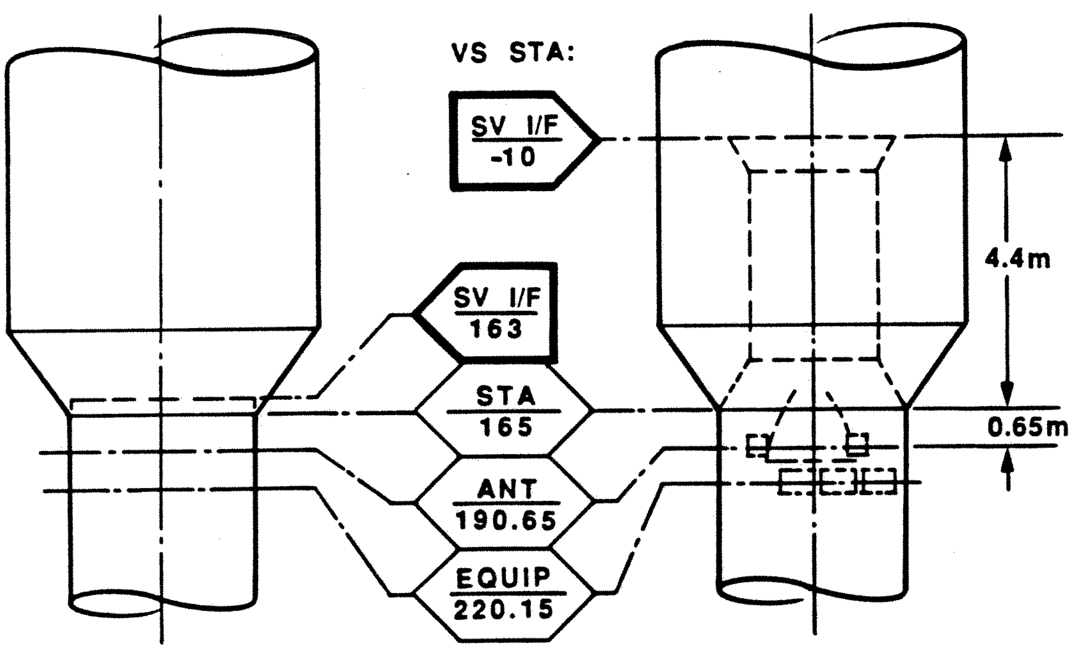


FIGURE 1B NUS CONFIGURATION

FIGURE 1C IUS CONFIGURATION

FIGURE 1 SV-LAUNCH VEHICLE INTERFACES

2.0 SOURCE DOCUMENTS

The upper stage contractors EMC control plans are the EMC requirements source for each of the three Titan IV configurations:

CONFIGURATION	CONTROL PLAN
Centaur	GDSS-TC-86-007 EMC Control Plan for the TIV/ Centaur Program 27 Jan 88
NUS	MCR-85-2506
IUS	D290-10068-1

2.1 Applicable Documents

The following documents are applicable to the EMC/EMI interface to the extent specified herein:

NUMBER & VERSION	TITLE
MIL-STD-461B 1 April 1980	Electromagnetic Interface Characteristics, Requirements for Equipment
MIL-STD-1541 15 October 1973	Electromagnetic Compatibility Requirements for Space Systems
MIL-B-5087B 31 August 1970	Bonding, Electrical, and Lightning Protection, for Aerospace Systems

3.0 TITAN IV CONFIGURATIONS/PAYLOAD GENERIC EMC/EMI INTERFACE

The EMC/EMI interface documentation for the Titan IV configurations can be generated using the unclassified Titan IV EMC baseline data from Figures 2 through 19.

This baseline data is specified at the P/L and Titan IV(SV I/F) interface plane depicted in Figure 1 for each of the three launch vehicle configurations.

3.1 Unintentional Emissions and Susceptibility

Conducted and radiated unintentional emissions and susceptibility levels are discussed in the following paragraphs by reference to Figures 2 through 19. These figures, except for Figure 17, are NOT adjusted for the applicable 6, 10 and 20 dB EMI Safety Margins (EMISM).

3.1.1 Titan IV Conducted Emissions

The maximum conducted emission values presented at the P/L interface plane ("SV I/F" in Figure 1) by the Titan IV vehicle are given in Figures 2 through 7 for the three LV configurations.

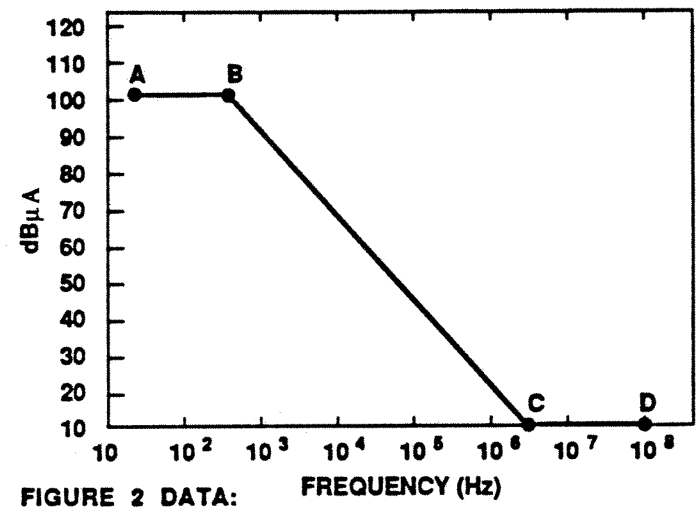


FIGURE 2 DATA:

A: 100 dBμA @ 30 Hz
B: 100 dBμA @ 3 kHz
C: 10 dBμA @ 2 MHz
D: 10 dBμA @ 100 MHz

FIGURE 2 CENTAUR NARROWBAND (N/B) CONDUCTED EMISSIONS AT THE S/V I/F

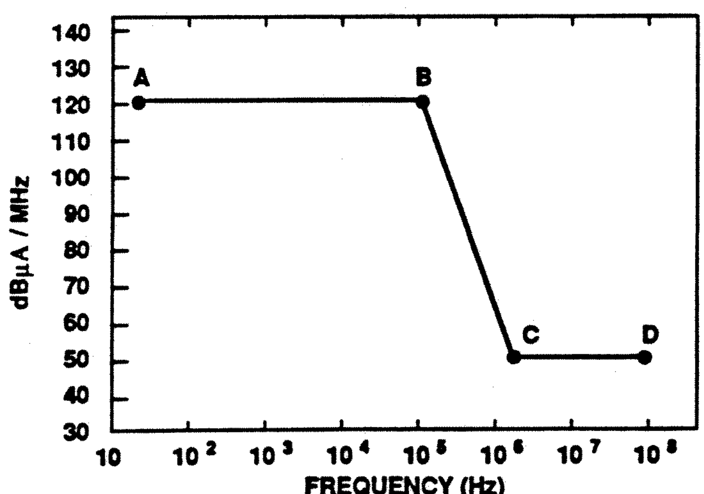


FIGURE 3 DATA:

A: 120 dBμA @ 15 kHz
B: 120 dBμA @ 100 kHz
C: 50 dBμA @ 2 MHz
D: 50 dBμA @ 100 MHz

FIGURE 3 CENTAUR BROADBAND (B/B) CONDUCTED EMISSIONS AT THE S/V I/F

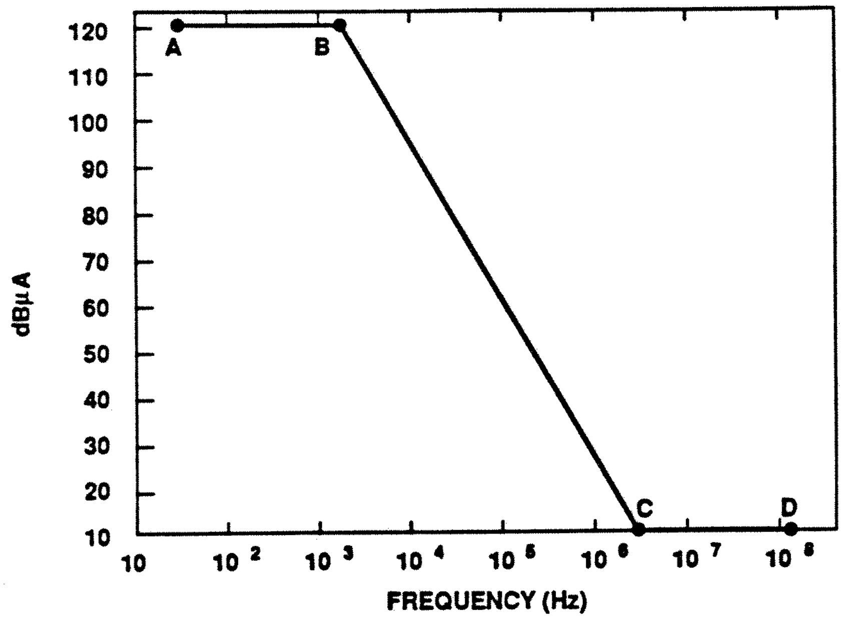


FIGURE 4 DATA:

A: 120 dBµA @ 30 Hz
B: 120 dBµA @ 1 kHz
C: 10 dBµA @ 2 MHz
D: 10 dBµA @ 100 MHz

FIGURE 4 NUS N/B CONDUCTED EMISSIONS AT THE S/V I/F

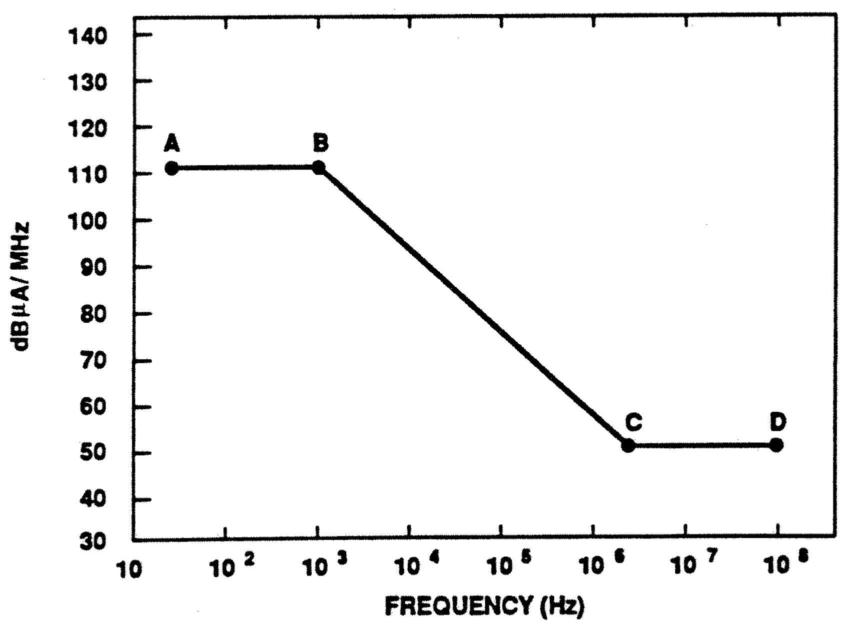


FIGURE 5 DATA:

A: 110 dBµA/MHz @ 30 Hz
B: 110 dBµA/MHz @ 1 kHz
C: 50 dBµA/MHz @ 2 MHz
D: 50 dBµA/MHz @ 100 MHz

FIGURE 5 NUS B/B CONDUCTED EMISSIONS AT THE S/V I/F

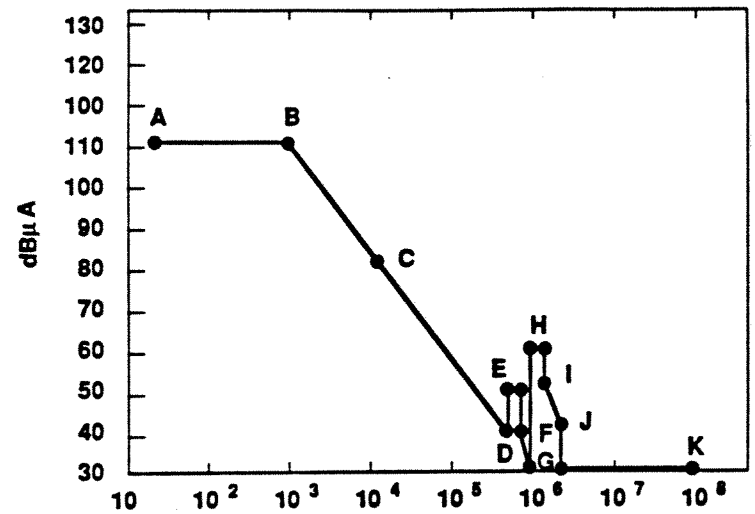


FIGURE 6 DATA:

A: 130 dBµA @ 30 Hz
B: 130 dBµA @ 2 kHz
C: 80 dBµA @ 20 kHz
D: 39 dBµA @ 700 kHz
E: 50 dBµA @ 700 kHz TO 800 kHz
F: 39 dBµA @ 800 kHz
G: 30 dBµA @ 1 MHz
H: 61 dBµA @ 1 MHz TO 1.5 MHz
I: 54 dBµA @ 1.5 MHz
J: 41 dBµA @ 3 MHz
K: 30 dBµA @ 3 MHz TO 100 MHz

FIGURE 6 IUS N/B CONDUCTED EMISSIONS AT THE S/V I/F

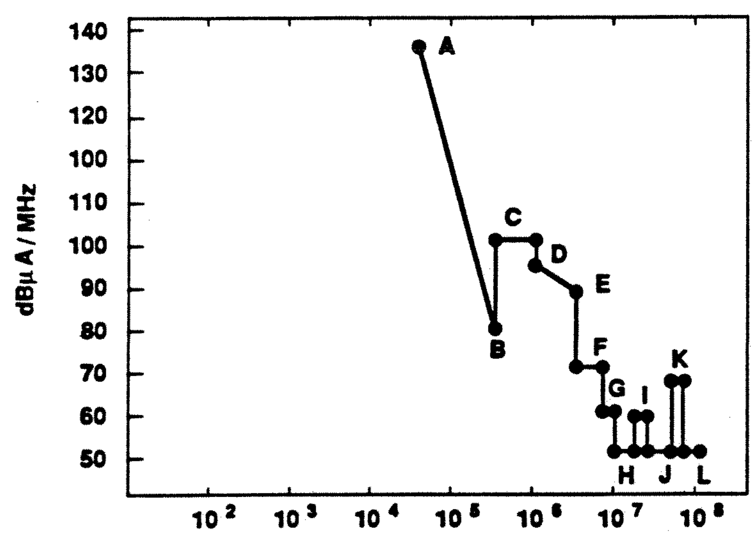


FIGURE 7 DATA:

A: 134 dBµA/MHz @ 20 kHz
B: 79 dBµA/MHz @ 350 kHz
C: 102 dBµA/MHz @ 350 kHz TO 1 MHz
D: 94 dBµA/MHz @ 1 MHz
E: 87 dBµA/MHz @ 4 MHz
F: 71 dBµA/MHz @ 4 MHz TO 8 MHz
G: 65 dBµA/MHz @ 8 MHz TO 10 MHz
H: 50 dBµA/MHz @ 10 MHz TO 15 MHz
I: 57 dBµA/MHz @ 15 MHz TO 17 MHz
J: 50 dBµA/MHz @ 17 MHz TO 35 MHz
K: 66 dBµA/MHz @ 35 MHz TO 40 MHz
L: 50 dBµA/MHz @ 40 MHz TO 100 MHz

FIGURE 7 IUS B/B CONDUCTED EMISSIONS AT THE S/V I/F

3.1.2 Titan IV Conducted Susceptibility

The allowable conducted emission values presented to the titan IV at the SV I/F (interface) are given in Figures 8, 9 and 10 for the three LV configurations.

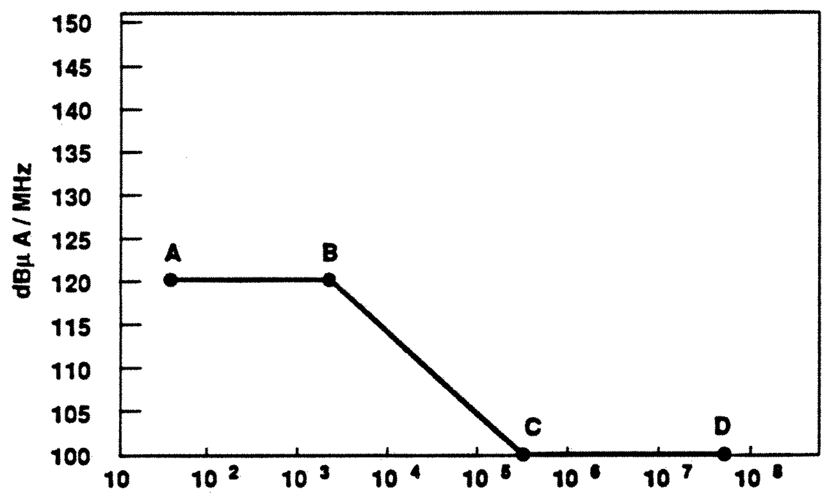


FIGURE 8 DATA:

A: 120 dBμA @ 60 Hz
B: 120 dBμA @ 15 kHz
C: 100 dBμA @ 150 kHz
D: 100 dBμA @ 400 kHz

FIGURE 8 CENTAUR CONDUCTED SUSCEPTIBILITY AT THE S/V I/F

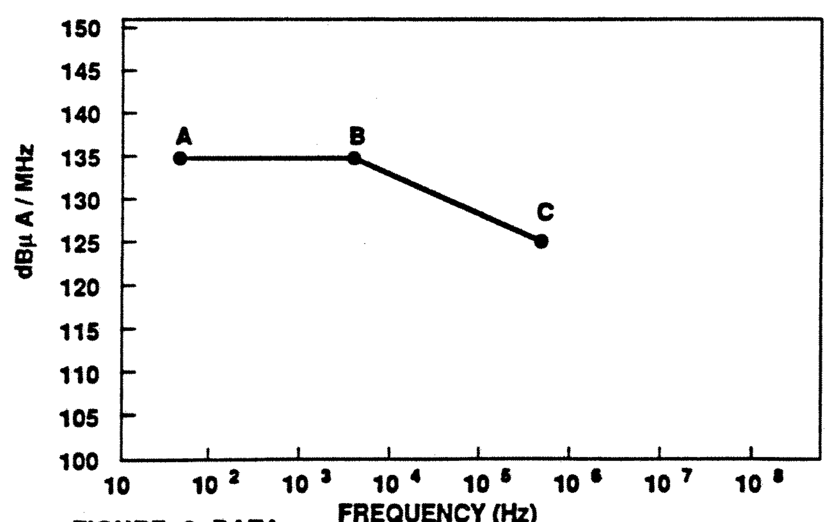


FIGURE 9 DATA:

A: 135 dBμA @ 30 Hz
B: 135 dBμA @ 1.5 kHz
C: 126 dBμA @ 150 kHz

FIGURE 9 NUS CONDUCTED SUSCEPTIBILITY AT THE S/V I/F

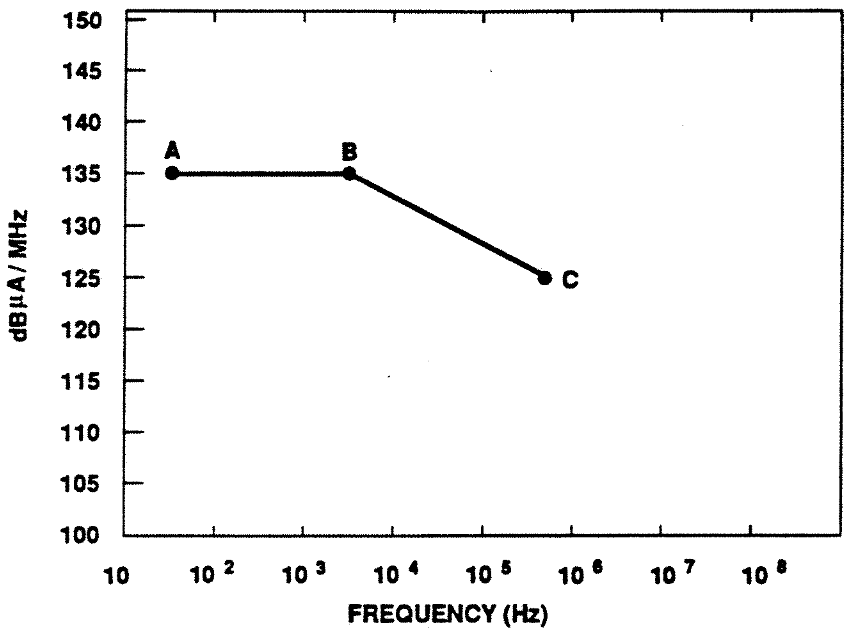


FIGURE 10 DATA:

A: 135 dBµA @ 30 Hz

B: 135 dBµA @ 1.5 kHz

C: 126 dBµA @ 150 kHz

FIGURE 10 IUS CONDUCTED SUSCEPTIBILITY AT THE S/V I/F

3.1.3 Titan IV Unintentional Radiated Emissions

The maximum unintentional radiated emission values presented to the SV I/F by the Titan IV vehicle are given in Figures 11 through 16 for the three LV configurations.

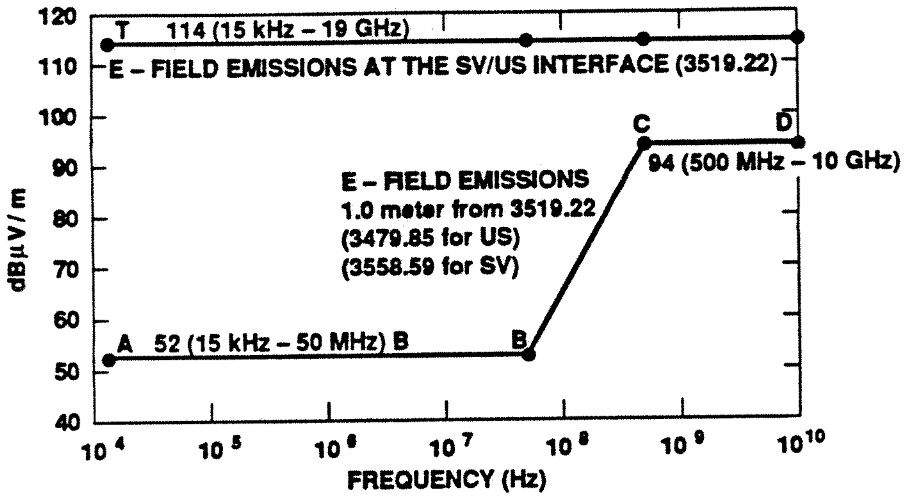


FIGURE 11 DATA:

A: 52 dBµV/m @ 15 kHz
B: 52 dBµV/m @ 50 MHz
C: 94 dBµV/m @ 500 MHz
D: 94 dBµV/m @ 10 GHz
T: 114 dBµV/m FROM 15 kHz THRU 10 GHz

FIGURE 11 CENTAUR N/B RADIATED EMISSIONS AT THE S/V I/F

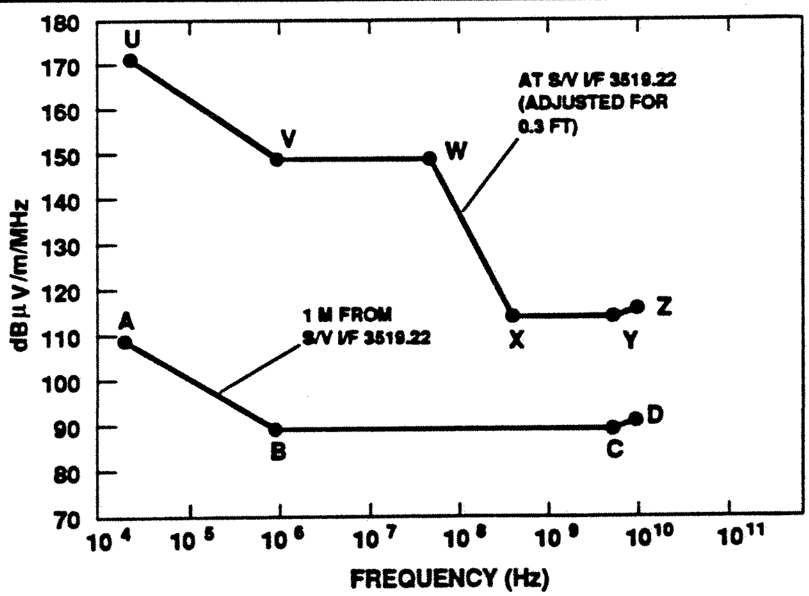


FIGURE 12 DATA:

A: 110 dBµV/m/MHz @ 15 kHz
B: 90 dBµV/m/MHz @ 1 MHz
C: 90 dBµV/m/MHz @ 8.5 GHz
D: 92 dBµV/m/MHz @ 10 GHz
U: 172 dBµV/m/MHz @ 15 kHz
V: 152 dBµV/m/MHz @ 1 MHz
W: 152 dBµV/m/MHz @ 50 MHz
X: 111 dBµV/m/MHz @ 500 MHz
Y: 111 dBµV/m/MHz @ 8.5 GHz
Z: 113 dBµV/m/MHz @ 10 GHz

FIGURE 12 CENTAUR B/B RADIATED EMISSIONS AT THE S/V I/F

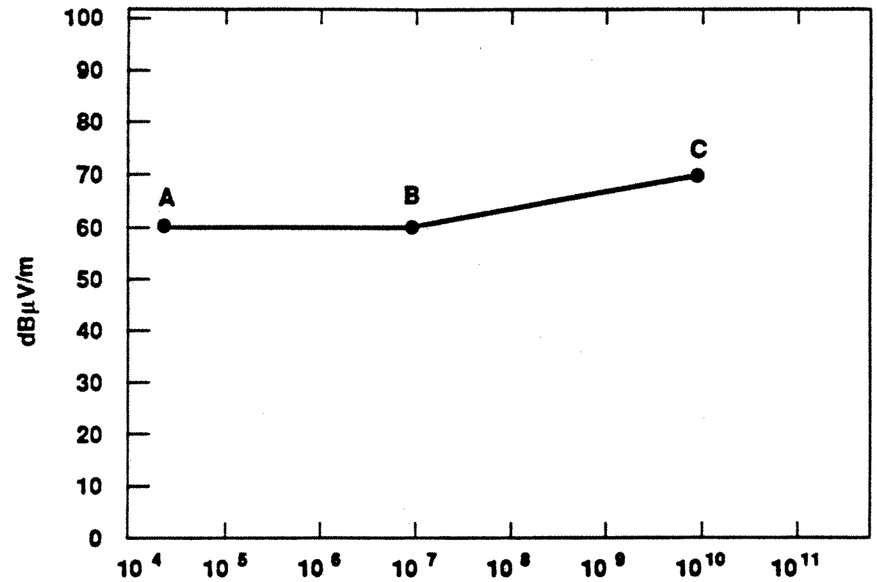


FIGURE 13 DATA:

A: 60 dBµV/m @ 15 kHz
B: 60 dBµV/m @ 10 MHz
C: 70 dBµV/m @ 10 GHz

FIGURE 13 NUS N/B RADIATED EMISSIONS AT THE S/V I/F

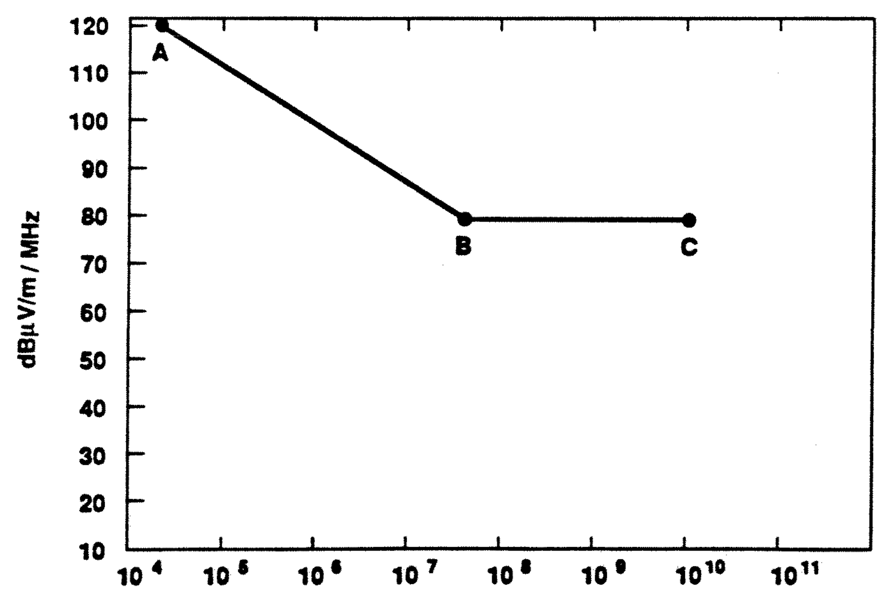


FIGURE 14 DATA:

A: 120 dBµV/m/MHz @ 15 kHz
B: 80 dBµV/m/MHz @ 20 MHz
C: 80 dBµV/m/MHz @ 10 GHz

FIGURE 14 NUS B/B RADIATED EMISSIONS AT THE S/V I/F

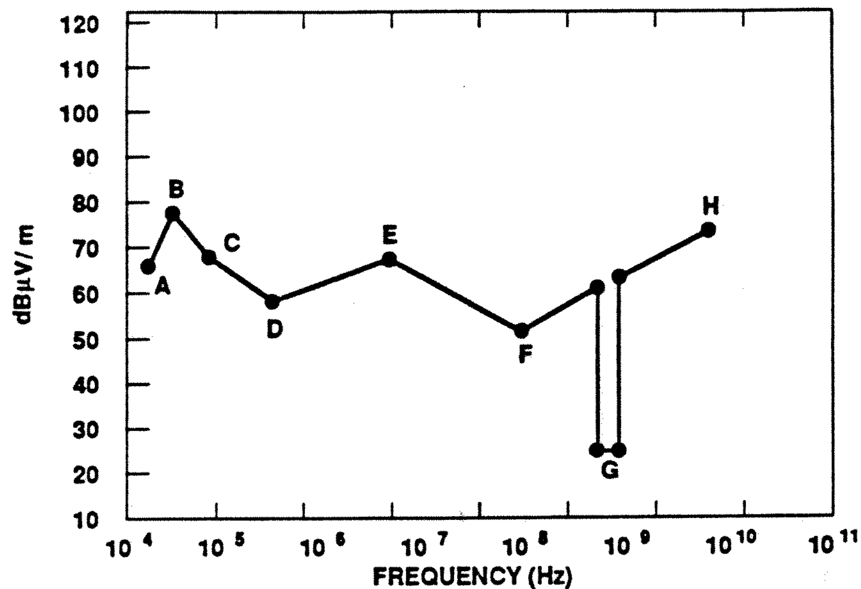


FIGURE 15 DATA:

A: 67 dBµV/m @ 14 kHz
B: 76 dBµV/m @ 30 kHz
C: 68 dBµV/m @ 80 kHz
D: 57 dBµV/m @ 500 kHz
E: 67 dBµV/m @ 8 MHz
F: 52 dBµV/m @ 40 MHz
G: 25 dBµV/m @ 1.77 TO 2.3 GHz
H: 75 dBµV/m @ 40 GHz

FIGURE 15 IUS N/B RADIATED EMISSIONS AT THE S/V I/F

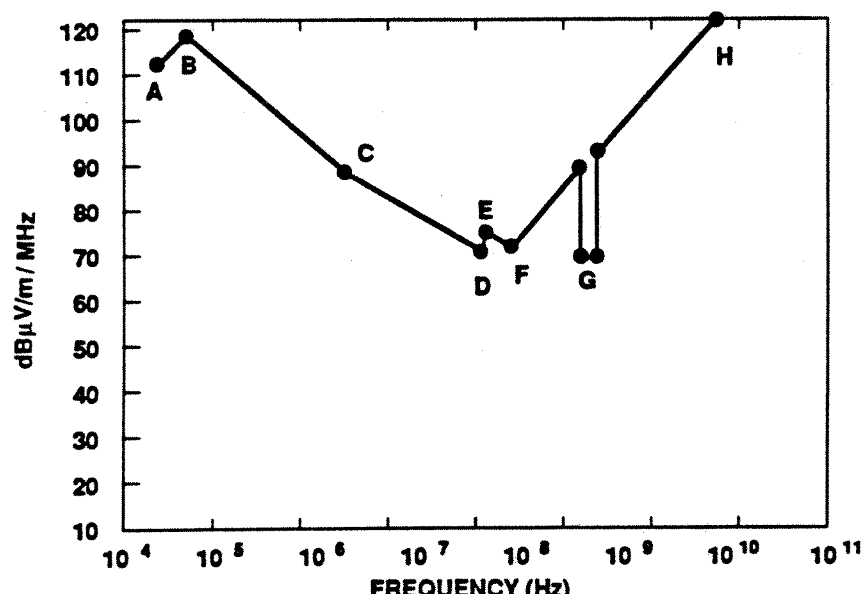


FIGURE 16 DATA:

A: 110 dBµV/m/MHz @ 14 kHz
B: 117 dBµV/m/MHz @ 30 kHz
C: 87 dBµV/m/MHz @ 80 kHz
D: 71 dBµV/m/MHz @ 500 kHz
E: 75 dBµV/m/MHz @ 100 MHz
F: 72 dBµV/m/MHz @ 200 MHz
G: 70 dBµV/m/MHz @ 1.77 TO 2.3 GHz
H: 120 dBµV/m/MHz @ 40 GHz

FIGURE 16 IUS B/B RADIATED EMISSIONS AT THE S/V I/F

3.1.4 Titan IV Radiated Susceptibility

The allowable radiated emission values presented to the Titan IV at the SV I/F are given in Figures 17, 18 and 19 for the three LV configurations.

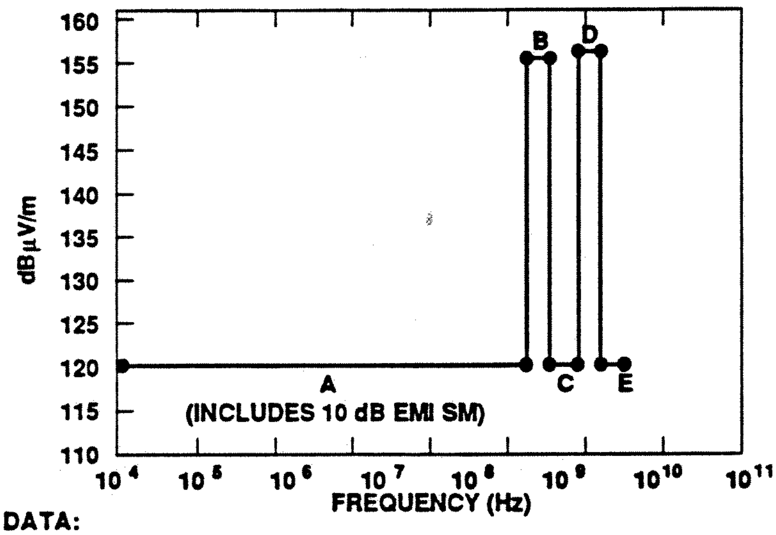


FIGURE 17 DATA:

A: 120 dBμV/m @ 14 kHz TO 2 GHz
B: 156.6 dBμV/m @ 2.0 GHz TO 2.5 GHz
C: 120 dBμV/m @ 2.5 GHz TO 5.4 GHz
D: 157 dBμV/m @ 5.4 GHz TO 5.9 GHz
E: 120 dBμV/m @ 5.9 GHz TO 10 GHz

FIGURE 17 CENTAUR RADIATED SUSCEPTIBILITY AT THE S/V I/F

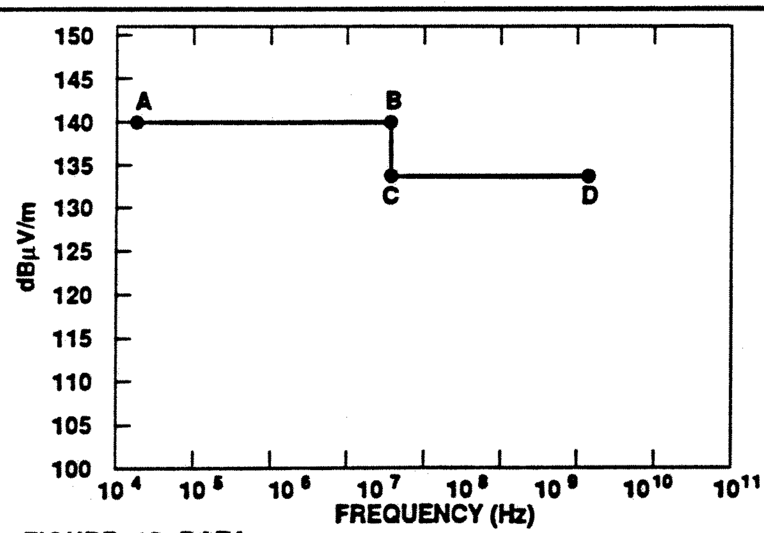


FIGURE 18 DATA:

A: 140 dBμV/m @ 15 kHz
B: 140 dBμV/m @ 35 MHz
C: 134 dBμV/m @ 35 MHz
D: 134 dBμV/m @ 10 GHz

FIGURE 18 NUS RADIATED SUSCEPTIBILITY AT THE S/V I/F

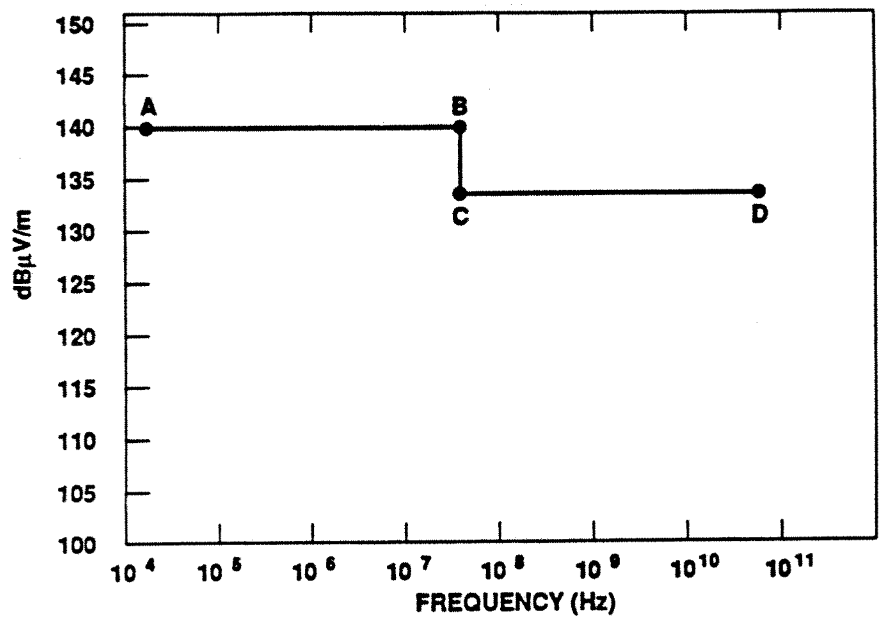


FIGURE 19 DATA:

A: 140 dBµV/m @ 15 kHz
B: 140 dBµV/m @ 35 MHz
C: 134 dBµV/m @ 35 MHz
D: 134 dBµV/m @ 40 GHz

FIGURE 19 IUS RADIATED SUSCEPTIBILITY AT THE S/V I/F

3.1.5 Payload Intentional Emissions

Payloads equipped with transmitters that potentially can exceed the 3.1.4 values require additional analysis/tests. These analysis/tests shall verify that the 3.1.4 limits are not exceeded over the following frequency bandwidths:

CONFIGURATION	FREQUENCY (fc)	BANDWIDTH (fl) - (fu)
Centaur	416.5 MHz	45 kHz
	5727.5 MHz	60 kHz
NUS	416.5 MHz	45 kHz
	5765.0 MHz	60 kHz
IUS	416.5 MHz	45 kHz
	5765.0 MHz	60 kHz

3.1.6 Titan IV Intentional Emissions

The intentional emissions for each Titan IV vehicle configuration, at the SV interface, subsequent to PLF ejection, are given in the indicated tables:

a. CENTAUR S-band:	Table 3.1.6-I
b. CENTAUR C-band peak power:	Table 3.1.6-II
c. CENTAUR C-band average power:	Table 3.1.6-III
d. NUS S-band:	Table 3.1.6-IV
e. NUS C-band peak power:	Table 3.1.6-V
f. NUS C-band average power:	Table 3.1.6-VI
g. IUS S-band:	Table 3.1.6-VII
h. IUS C-band peak power:	Table 3.1.6-VIII
i. IUS C-band average power:	Table 3.1.6-IX
j. CENTAUR S-band WIS:	Table 3.1.6-X
k. NUS S-band WIS:	Table 3.1.6-XI
l. IUS S-band WIS:	Table 3.1.6-XII
m. TIV Core S-band at the CENTAUR SV I/F:	Table 3.1.6-XIII

TABLE 3.1.6-I

CENTAUR S-BAND INTENTIONAL EMISSIONS AT THE I/F PLANE

TX power:	12.6 dBW
Power transmitted from the Antenna:	8.4 dBW
Path to I/F:	1.7 m
At the I/F:	
E-Field:	8.5 V/m 138.6 dBμV/m
Power density:	0.19 W/m^2

TABLE 3.1.6-II

CENTAUR C-BAND PEAK POWER INTENTIONAL EMISSIONS AT THE I/F PLANE

TX power:	25.4 dBW
Power transmitted from the Antenna:	12.4 dBW
Path to I/F:	1.65 m
At the I/F:	
E-Field:	13.8 V/m 142.8 dBμV/m
Power density:	0.5 W/m^2

TABLE 3.1.6-III

CENTAUR C-BAND AVERAGE POWER INTENTIONAL EMISSIONS AT THE I/F PLANE

TX power:	.69 W
Power transmitted from the Antenna:	–14.6 dbW
Path to I/F:	1.65 m
At the I/F:	
E-Field:	0.62 V/m 115.8 dBμV/m
Power density:	0.001 W/m^2

TABLE 3.1.6-IV

NUS S-BAND INTENTIONAL EMISSIONS AT THE I/F PLANE

TX power:	13.0 dBW
Power transmitted from the Antenna:	6.9 dBW
Path to I/F:	0.7 m
At the I/F:	
E-Field:	17.28 V/m 144.75 dBμV/m
Power density:	0.79 W/m^2

TABLE 3.1.6-V

NUS C-BAND PEAK POWER INTENTIONAL EMISSIONS AT THE I/F PLANE

TX power:	30 dBW
Power transmitted from the Antenna:	9.2 dBW
Path to I/F:	0.7 m
At the I/F:	
E-Field:	22.5 V/m 147.0 dBμV/m
Power density:	1.3 W/m^2

TABLE 3.1.6-VI

IUS C-BAND AVERAGE POWER INTENTIONAL EMISSIONS AT THE I/F PLANE

TX power:	3.0 dBW
Power transmitted from the Antenna:	–17.8 dBW
Path to I/F:	0.7 m
At the I/F:	
E-Field:	1.0 V/m 120.1 dBμV/m
Power density:	0.003 W/m^2

TABLE 3.1.6-VII

IUS S-BAND INTENTIONAL EMISSIONS AT THE I/F PLANE

TX power:	13.0 dBW
Power transmitted from the Antenna:	6.9 dBW
Path to I/F:	5.1 m
At the I/F:	
E-Field:	2.38 V/m 127.5 dBμV/m
Power density:	0.015 W/m^2

TABLE 3.1.6-VIII

IUS C-BAND PEAK POWER INTENTIONAL EMISSIONS AT THE I/F PLANE

TX power:	30.0 dBW
Power transmitted from the Antenna:	9.2 dBW
Path to I/F:	5.1 m
At the I/F:	
E-Field:	3.1 V/m 129.8 dBµV/m
Power density:	0.025 W/m^2

TABLE 3.1.6-IX

IUS C-BAND AVERAGE POWER INTENTIONAL EMISSIONS AT THE I/F PLANE

TX power:	3.0 dBW
Power transmitted from the Antenna:	–17.8 dB W
Path to I/F:	5.1 m
At the I/F:	
E-Field:	0.14 V/m 102.8 dBµV/m
Power density:	0.00005 W/m^2

TABLE 3.1.6-X

CENTAUR S-BAND WIS INTENTIONAL EMISSIONS AT THE I/F PLANE

TX power:	13.4 dBW
Power transmitted from the Antenna:	7.3 dBW
Path to I/F:	8.7 m
At the I/F:	
E-Field:	1.47 V/m 123.4 dBμV/m
Power density:	0.0057 W/m^2

TABLE 3.1.6-XI

NUS S-BAND WIS INTENTIONAL EMISSIONS AT THE I/F PLANE

TX power:	30.4 dBW
Power transmitted from the Antenna:	7.3 dBW
Path to I/F:	0.7 m
At the I/F:	
E-Field:	18.1 V/m 145.2 dBμV/m
Power density:	0.87 W/m^2

TABLE 3.1.6-XII

IUS S-BAND WIS INTENTIONAL EMISSIONS AT THE I/F PLANE

TX power:	13.4 dBW
Power transmitted from the Antenna:	7.32 dBW
Path to I/F:	5.1 m
At the I/F:	
E-Field:	2.5 V/m 128.0 dBμV/m
Power density:	0.016 W/m^2

TABLE 3.1.6-XIII

TIV CORE S-BAND INTENTIONAL EMISSIONS AT THE CENTAUR SV I/F PLANE

TX power:	13.0 dBW
Power transmitted from the Antenna:	6.9 dBW
Path to I/F:	8.7 m
At the I/F:	
E-Field:	1.4 V/m 122.9 dBμV/m
Power density:	0.0052 W/m^2

3.2 EMI Safety Margin (EMISM)

MIL-STD-1541 requires that EMISMs be established for critical signal, power, and ordnance circuits. EMISM is the ratio between the susceptibility threshold and the interference level. For new designs, the required minimum EMISMs are 6 dB for critical signal and power circuits and 20 dB for ordnance circuits. Previously qualified ordnance designs which flew on earlier Titan vehicles had to meet only a 6 dB EMISM.

Centaur configuration new designs require a 10 dB EMISM for analyses and a 6 dB for tests. Requests have been received from various payload representatives to use 12 dB as an analytical EMISM. Where practical, analyses will provide data for both 10 and 12 dB EMISMs.

In Figures 2 through 19, the emission and the susceptibility levels have not been adjusted for the EMISM. Therefore, the sum of the interference level and the EMISM must be less than or equal to the susceptibility levels in Figures 8, 9, 10, 17, 18, and 19. (Exception – Figure 17 is adjusted for 10 dB EMISM susceptibility.)

3.3 Bonding

On the Titan IV vehicle, all electrical/electronic equipment shall be installed so that there will be a continuous, low-impedance path from the equipment enclosure to the basic vehicle structure. This low-impedance path shall have a maximum DC bonding resistance of 10 milliohms across the mated surface as demonstrated by contractor test.

3.4 Grounding

Titan IV power and signal returns are tied to a Single-Point-Ground (SPG) system except as noted below:

3.4.1 Centaur SPG Grounding Exceptions

a. The engine igniters on the RF10A-3-3A are internally grounded.

b. The battery supply to the range safety receiver is tied to chassis ground of the FT, but is isolated from Centaur's system power.

c. The SV battery power return through the PTU is referenced to ground through a one megohm resistor.

d. The System Electronics Unit/Inertial Reference Unit (SEU/IRU) power returns are grounded at the SEU chassis and isolated from Centaur battery.power returns through a DC-to-DC converter.

3.4.2 Titan IV Core (NUS) SPG Grounding Exceptions

The transient power bus provides the Titan IV ordnance power. This bus is referenced to the SPG through a 10K ohm resistor.

3.4.3 IUS Single Point Grounding Exceptions

IUS SPG exceptions: The Titan IV and IUS shall each employ SPG for prime electrical power and circuits operating below 100 kHz.

a. Isolation of one megohm minimum shall be maintained between IUS prime electrical power and IUS structure when the IUS system ground reference is not connected to the IUS structure SPG

b. For signal interfaces, DC isolation greater than 10 megohms shall be provided within the IUS. Returns shall be grounded to structure in the IUS and Titan IV.

c. Each ISDS and Command Destruct Function Circuit shall be isolated from all other IUS circuitry and IUS structure as follows:
   1) Destruct System Armed 100K to 500K ohms
   2) Destruct System Safed 100K to 500K ohms
   3) Destruct System Fire 100K to 500K ohms

d. Each separation fire circuit shall be isolated from all other IUS structure by a resistance of at least 2 megohms.

e. Excluding the IUS electroexplosive device separation ordnance, all power wiring crossing the Titan IV/IUS electrical interface shall be isolated from the respective vehicle's structure and power systems by a minimum of 10 megohms.

3.5 Isolation

The power between the Titan IV LV and the P/L shall be isolated by providing a return line for each signal crossing the SV interface. Power isolation between the Titan IV LV and the P/L, with the SPG disconnected, shall be less than one megohm for the Centaur configuration and less than ten megohms for the NUS and the IUS configurations.

3.6 Electrostatic Charge Prevention

Conductive surfaces are to be grounded in accordance with MIL-B-5087B Class S. New designs will conform to the 10^9 ohm-cm surface resistivity requirement. No design changes will be made solely to meet the 10^9 ohm-cm surface resistivity requirement for existing or modified designs.

3.7 Shielding

3.7.1 Centaur Shielding

For audio circuits (F < 100 kHz), the power or signal wire shall be twisted with its return. The return shall be referenced to the SPG. Shields which are used to reduce radiated interference from an audio frequency circuit shall be grounded at the source end only. The shield for a sensitive circuit shall be grounded at the load or receiver end only.

Shields on RF carrying wires or on circuits which are sensitive to RF shall be grounded at both ends and at intermediate points along the length of the shield as necessary. All shield grounds are attached to the basic structure or chassis. Whenever only two ground points are used, they shall be located at each end of the RF shield. The coax or wires shall be routed as close as possible to the ground plane to minimize the closed ground-loop area.

Individual twisted wire pairs of digital data signals shall be shielded if they connect to circuits with impedances greater than 100 ohms or if they are sensitive to high frequencies. High-data-rate circuits shall be treated as RF circuits.

3.7.2 NUS Shielding

Two types of signals are known to cross the NUS S/V I/F. Ordnance circuit signals shall be carried on twisted shielded pairs with the shield grounded at both ends of the cable. Discrete event signals, e.g., PLF separation and P/L separation, may be neither shielded nor twisted.

3.7.3 IUS Shielding

The IUS circuitry has been divided into six sensitivity categories. Circuits have been assigned to these categories by their frequency, voltage and current characteristics.

3.7.3.1 IUS Sensitivity Categories

Six categories of IUS wiring have been established:

CATEGORY	DESCRIPTION
I	Power and control signals
II	Digital signals
III	Low-frequency signals (F < 100 kHz)
IV	Electro-explosive device (EED) circuits
V	High-frequency signals (F < 1 MHz)
VI	Secure communications

3.7.3.2 IUS Circuit Category Assignments

IUS circuit category assignments, shielding, and grounding requirements are listed below in Table 3.7.3.1–I.

TABLE 3.7.3.1-I IUS CIRCUIT CATEGORY ASSIGNMENTS

FREQUENCY OR RISE FALL TIME	VOLTAGE CATEGORY			WIRE/ SHIELD	SHIELD GROUNDING
DC	<10V & <5 Amp		III	TWS	SPG
DC	Other		I	TW	N/A
F <100 kHz, &/or tr/tf > 10 µs	a.	≤5V	III	TWS	SPG
	b.	>5V & <25V	II	TWS	SPG
	c.	>25V	I	TW	N/A
F >100 kHz & F < 1 MHz, or tr/tf ≤ 10 µs	a.	≤1V	III	TWS	MPG Note 2
	b.	>1V & ≤10V	II	TWS	MPG Note 2
	c.	>10V	V	Note 3	MPG
>1 MHz	All		V	Note 3	MPG

NOTES:

1. Legend:
   SPG = Single Point Ground
   MPG = Multiple Point Ground
   TW = Twisted
   TWS = Twisted & Single Shielded

2. "Twinax" should be used in installations where cabling capacitance is critical.

3. High frequency circuits, with the exception of those using waveguide, shall use shielded coaxial cable, shielded balanced cable, or balanced cable with characteristic impedances of 100 ohms or less.

4. All EED circuits are classified as Category IV and shall use TWS with MPG.

5. Unencryped classified data (Clear Test) is defined as Category VI. All the above wire, shielding, and grounding requirements apply. Additional Category VI requirements include:

   A. Overall shields are required on wire bundles.

   B. Unclassified data circuits will not be included in Category VI wire bundles.

3.8 RF Environments

Estimates for the East Coast and the West Coast launch facilities worst case RF environments are given in Tables 3.8-I and 3.8-II, respectively. These data were derived from a study done for the Shuttle program and are provided for information only. Sources are grouped by frequency. E-Field values are recalculated estimates for the indicated facilities. Blank spaces indicate that the data is not available or not applicable. Brackets "[ ]" indicate that the data was based on assumed duty cycle parameters. Source names are not necessarily their official ones.

TABLE 3.8-I EAST COAST RF ENVIRONMENTS FOR LC-41

Source	Frequency (MHz)	Measured E-Field Peak (Wm)	Theoretical E-Field Peak (Wm)
Range Safety Command (Omni)	416.5		0.03
Range Safety Command	416.5	0.0507	0.03
USNS REDSTONE RS Command	416.5		0.02
KSC TACAN	1146		0.01
FPS-66 SAGE Radar	1265–1345	3.47	9.4
GPS Ground Antenna Uplink	1783	2.6	
MILA Uplinks	2034.2–2108.3		6.6
TPN-19 Surveillance Radar	2700–2900		4.2
GPN-20 Air Surveillance Radar	2750	2.75	5.0
1.6 Lighthouse RS Radar	3056	0.861	3.0
SEEK SKYHOOK (TARS) @ 10K FT	3230 -		1.4
USNS REDSTONE FPS-26	5400		98.9
USNS REDSTONE FPS-16	5400		57.1
WSR-74C Weather Radar	5625–5900	7.41	10.6
0.14 Tracking Radar	5690–5900	24.5	103.0
1.16 Tracking Radar	5690	30.5	76.5
1.17B Tracking Radar	5690	27.5	64.9
19.14 Tracking Radar	5690	90.2	211.0
19.17 Tracking Radar	5690		80.1
TPN-19 Approach Radar	9050		10.7
Condo 8	9410		2.0
MSBLS	15400–15700		0.9

*NOTE These Sources are deleted – no longer operational

TABLE 3.8-II WEST COAST RF ENVIRONMENT FOR SLC-4E

Equipment	Low – High Frequency (MHz)	Peak Power (MW)	Avg Power (KW)	Ant. Gain (dB)	Distance (m)	Theoretical E-Field Peak (V/m)	Measured E-Field Peak (V/m)	Avg Intensity (V/m)	Modulation
ARSR-1E	1280.0 – 1350.0	4.000	2.400	34.0	5170.13	106.19	5.50	2.60	Pulse
ARSR-1E *	1280.0 – 1350.0	0.450	0.270	34.0	5170.13	35.62	1.74	0.87	Pulse
FPS-77 (WX)	5450.0 – 5650.0	0.300	0.195	35.8	17052.29	10.85	17.18	0.28	Pulse
GPM-12 (RAPCON)	2800.0 – 2800.0	0.425	0.320	34.0	10922.31	16.39	4.80	0.44	Pulse
FPS-16-1	5400.0 – 5900.0	1.200	1.200	43.5	7087.83	126.66	11.48	4.01	Pulse
FPS-16-2	5400.0 – 5900.0	1.200	1.200	43.5	7082.84	126.75	-	4.01	Pulse
TPQ-18	5400.0 – 5900.0	2.500	5.000	51.0	4607.13	666.96	462.38	29.83	Pulse
HAIR	5400.0 – 5900.0	3.000	8.000	53.0	14105.09	300.43	275.40	15.51	Pulse
GRW-5 (TITAN)	8530.0 – 8990.0	0.200	0.020	44.0	20085.72	19.33	34.67	0.19	Pulse
GRQ-2 (ATLAS)	9220.0 – 9220.0	0.060	0.140	43.0	4498.22	42.13	0.08	2.04	Pulse
SGLS-14	1750.0 – 1850.0	0.010	10.000	35.0	20026.29	1.54	-	1.54	FM
SGLS-46	1750.0 – 1850.0	0.010	10.000	45.0	23539.16	4.14	-	4.14	FM
SGLS-60	1750.0 – 1850.0	0.010	10.000	42.7	23370.28	3.20	-	3.20	FM
CT-1	406.5 – 416.5	0.010	10.000	23.0	18068.73	0.43	-	0.43	FM
CT-2	406.5 – 423.0	0.010	10.000	23.0	5189.10	1.49	-	1.49	FM
CT-3	406.5 – 416.5	0.010	10.000	23.0	4574.87	1.69	-	1.69	FM
S-Band (10m)	2040.0 – 2107.0	0.010	10.000	45.0	12344.00	7.89	-	7.89	CW, PM, PSK
USB (9m)	2040.0 – 2107.0	0.020	20.000	42.8	12344.00	8.66	-	8.66	CW, PM, PSK
COLLINS 205/310V	3.0 – 30.0	0.045	22.000	12.0	11887.00	0.39	-	0.27	AM
Black Mtn. H/F	2700.0 – 2800.0	3.600	2.100	38.5	77249.00	11.32	-	0.27	Pulse
Black Mtn. ARSR-1E	1280.0 – 1320.0	5.000	3.000	34.0	77249.00	7.95	-	0.19	Pulse

*ARSR-1E may be operated at lower power

Notes:
1) Measured values are outside at level 15 of the MST (accuracy +/– 3 dB).
2) Radars are normally operated at lower power levels than rated peak power.
3) Measured values are with emitters at full power. (Report available upon request.)

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3.9 Lightning Protection

MIL-STD-1541 requires that bonding be provided for the vehicle and all external electrically conducting objects to prevent damage to the vehicle and ground equipment from the effects of lightning strokes and arc-discharges.

The following measures have been taken to lessen the hazards of a lightning strike.

- Transzorbs on Guidance System uplink and downlink circuits
- Aluminum wrap on all electrical umbilicals and external test cables
- Air terminal on Umbilical Tower - used when MST retracted in Park position*
- Operational Constraints
  - Electrical umbilicals and guidance system disconnected between VIB and PAD*
  - Lightning damage search procedure performed if there is credible evidence of a "Near Strike"
  - Launch constraints – meteorological constraints established at site and along flight path to minimize the hazards to launch vehicles (after launch) from vehicle triggered lightning and natural lightning.

*ESMC only

3.10 Launch Pad RF Protection

C-band RF intensities are held below 1 Volt/meter in the Universal Environmental Shelter (UES) at LC-41 by the Eastern Test Range (ETR) emitter controls. The controls are in the form of software masks at each site that removes radar transmitter power when the antenna crosses specified azimuth/elevation limits. These masks are removed on launch day and are controlled to the limits defined by the booster/spacecraft Program Requirements Document (PRD).

3.11 Titan IV Payload Fairing RF Protection

To preclude damage to spacecraft circuits from transponder RF energy reflected from the payload fairing during payload fairing separation, Range Safety has instituted a procedure to inhibit the transponder interrogation command for the period of time involved, reference Paragraph 4.2.3.2.4.