SPECIFICATION NUMBER: IS-DMSP-888B
CODE IDENTIFICATION NUMBER: 18713
27 February 1998

DEFENSE METEOROLOGICAL SATELLITE PROGRAM (DMSP)

Interface Specification for DMSP Block 5D-3 Spacecraft (S16-S20) to Operational Linescan System to Mission Sensors

C. I. 20025564G1 through G5
Contract F04701-89-C-0029
CDRL Item 007A2

Prepared for:

Space and Missile Systems Center
2420 Vela Way, Suite 1467-A8
Los Angeles AFB
El Segundo, CA 90245-4659

Prepared by:

Lockheed Martin Missiles & Space
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P.O. Box 800
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LOCKHEED MARTIN

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INTERFACE SPECIFICATION FOR

DMSP BLOCK 5D-3 SPACECRAFT (S16 TO S20)

TO OLS TO MISSION SENSORS

C. I. 20025564G1 through G5

CONTRACT F04701-89-C-0029

CDRL ITEM 009A2

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27 Feb 1998
Date

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J. L. Levatich
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27 Feb 1998
Date

AF-CDRL-8253
6 May 1998

SMC/CIIS
Attn: M. Fierro (DMO)
2420 Vela Way Suite 1467-A8
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Subject: Contract F04701-89-C-0029,
CDRL Item 007A2 (DI-E-3106/T)
Specification Maintenance Document
(FOR INFORMATION)

Reference: (a) ECP 103, Updates and Corrections to IS-DMSP-888 (Interface Specification, DMSP Spacecraft to OLS to Mission Sensors), dated 20 October 1997, transmitted via LMMS Letter AF-CDRL-8133 dated 3 November 1997

(b) USAF Contractual Document Action 0029-97-005, dated 27 February 1998, re: Approval of ECP 103, Updates and Corrections to IS-DMSP-888

Enclosure: (1) Approved Specification Change Notice 001 to IS-DMSP-888 Rev B, Interface Specification, DMSP Spacecraft to OLS to Mission Sensors, dated 27 February 1998

Dear Capt. Keller,

Reference (a) was approved with comments via the Reference (b) letter. Lockheed Martin Missiles & Space (LMMS) has incorporated the USAF's DMSP Configuration Control Board (CCB) comments and, in accordance with the subject CDRL item requirements, is submitting five (5) copies and one (1) reproducible of Enclosure (1) for your information. In addition, LMMS is distributing Enclosure (1) to all current holders of the specification.

If there are any questions or comments concerning this document, please direct contractual issues to me at (609) 490-2176 or technical issues to Mr. J. Levatich at (609) 490-3827.

Sincerely,

S. E. Hopkinson
DMSP Contracts Manager

xc: w/encl.

DCMC/DCMDE-RVNE (MS 5A)

INTERFACE CHANGE NOTICE
(SEE MIL-STD-490 FOR INSTRUCTIONS)

DATE PREPARED 27 February 1998

1. ORIGINATOR NAME AND ADDRESS	2.	3. CODE IDENT.	4. SPEC NO.
Lockheed Martin Missiles & Space P.O. Box 800 Princeton, NJ 08543-0800	☐ PROPOSED ☒ APPROVED	18713 5. CODE IDENT. 18713	IS-DMSP-888 6. SCN NO. 001

7. SYSTEM DESIGNATION	8. RELATED ECP NO.	9. CONTRACT NO.	10. CONTRACTUAL ACTIVITY
DMSP	103	F04701-89-C-0029	

11. CONFIGURATION ITEM NOMENCLATURE	12. EFFECTIVITY
Block 5D-3 Spacecrafts	5D-3 Spacecrafts S16 through S20

THIS NOTICE INFORMS RECIPIENTS THAT THE SPECIFICATION IDENTIFIED BY THE NUMBER (AND REVISION LETTER) SHOWN IN BLOCK 4 HAS BEEN CHANGED. THE PAGES CHANGED BY THIS ICN BEING THOSE FURNISHED HEREWITH AND CARRYING THE SAME DATE AS THIS ICN. THE PAGES OF THE PAGE NUMBERS AND DATES LISTED BELOW IN THE SUMMARY OF CHANGED PAGES, COMBINED WITH NON-LISTED PAGES OF THE ORIGINAL ISSUE OF THE REVISION SHOWN IN BLOCK 4, CONSTITUTE THE CURRENT VERSION OF THIS SPECIFICATION.

13. ICN NO.	14. PAGES CHANGED (INDICATE DELETIONS)	* S	* A	15. DATE
001	ENTIRE DOCUMENT TRANSMITTED HEREIN (IS-DMSP-888B).	X		27 Feb 98

16. TECHNICAL CONCURRENCE	DATE
Approved by USAF DMSP CCB	27 Feb 98

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* "S" indicates supersedes earlier page. "A" indicates added page.

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FIGURES

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1.0 SCOPE

1.1 General. This Interface Specification (IS) establishes the requirements for the interfaces between:

a. the Defense Meteorological Satellite Program (DMSP) Block 5D-3 Spacecraft (S16-S20) and the Operational Linescan System (OLS)

b. the DMSP Block 5D-3 Spacecraft (S16-S20) and DMSP mission sensors

c. the OLS and DMSP mission sensors

1.2 Interface documentation. This IS establishes requirements for the interfaces described in Section 1.1, and is supplemented by a set of Interface Control Documents (ICDs). The interface documentation relationship is described in Lockheed Martin document ICD-88801.

1.2.1 Interface Specification. The IS may be changed in response to changes in system requirements in either the spacecraft or the sensors. Changes to the IS shall be proposed by ICN and issued by ICN, and shall be controlled by the DMSP Configuration Control Board (CCB).

1.2.2 Interface Control Documents. Each mission sensor will have an Interface Control Document (ICD) which will contain sensor specific requirements for interface with the spacecraft and with the OLS. A separate ICD will contain Spacecraft/OLS requirements. The ICDs for each sensor are listed in Appendix A. The ICDs will be changed as necessary to incorporate changing or new interface definitions as they evolve from agreements between the affected contractors and/or government agencies. Proposed ICD changes shall be submitted to the DMSP Space Segment Integrator. A Space Segment Interface Control Working Group (ICWG) Technical Interchange Meeting (TIM) will be convened to review proposed changes and to summarize all anticipated impacts. Consensus TIM approval of any change, without cost impact, shall result in change approval. Consensus TIM disapproval of any change shall result in change disapproval. All nonconsensus ICD changes and all changes with cost impacts shall be submitted to, and resolved by, the Air Force DMSP System Program Office (SPO). Once approved, the Space Segment Integrator shall produce and distribute formal change notices to all ICD holders.

1.3 Items description

1.3.1 General. The sensors consist of the Operational Linescan System (OLS) and a set of mission sensors (SSPs). Each of the sensors are delivered as Government Furnished Equipment (GFE) to the spacecraft contractor. A summary of the functional interface elements is shown in Figure 1.

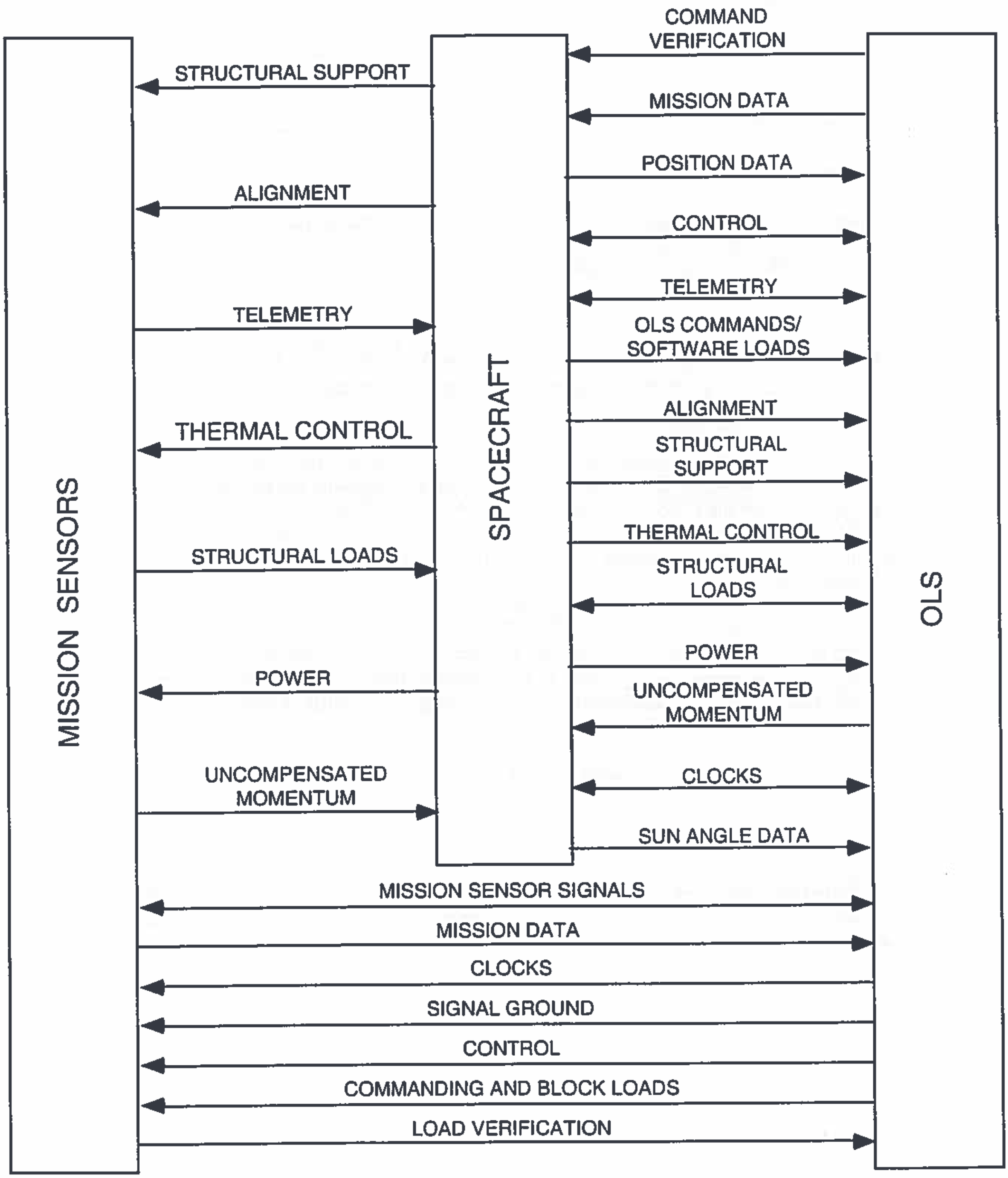


Figure 1. Spacecraft/Sensors Functional Interfaces

1.3.2 Operational Linescan System. The OLS consists of a number of individually packaged units. These units perform all functions necessary for gathering and processing OLS data and for formatting, storing, encrypting, and controlling both OLS and mission sensor data. For reference, throughout this document, the four OLS primary tape recorders are collectively abbreviated as PRxs, and the four encryption boxes (or KG boxes) are collectively abbreviated as BBxs. BB1, BB2, and BB3 will be GFE. BB4 will be procured by the OLS contractor.

1.3.3 Mission sensors. The mission sensors are independent sensors providing data about either in situ or remote geophysical phenomenon or device technology. Their identity, scientific description, characteristics and the like are described in the individual ICDs listed in Appendix A.

1.3.4 Spacecraft. The spacecraft serves as a stable platform for structural support and alignment of sensor units and provides the necessary communications, control, power, and thermal control.

2.0 APPLICABLE DOCUMENTS

2.1 Government documents. The following documents form a part of this interface specification to the extent specified herein. In the event of conflict between the documents referenced herein and the contents of this specification, the contents of this specification shall be considered a superseding requirement.

SPECIFICATIONS:

IS-YD-821B 15 Jan 77 SCN 001 5 Nov 80 SCN 002 1 Apr 87 SCN 004 8 Aug 83 SCN 005 1 Nov 83 SCN 006 15 Apr 87 SCN 007 1 Jun 89 SCN 010 9 Mar 90 SCN 011 7 Mar 91 SCN 012 15 Oct 91	DMSP Data Specifications

STANDARDS:

ASTM-E595-77 25 Feb 77	Test Method for Total Mass Loss and Collected Volatile Condensable Materials from Outgassing in a Vacuum Environment

OTHER PUBLICATIONS: Interface Control Documents

ICD-88801 2 April 1992	Interface Control Document, Space Segment Documentation Relationship
ICD-88802 13 January 1993 ICN 001 25 May 1994	Interface Control Document, DMSP Block 5D-3 (S16-S20)/OLS
ICD-88803 19 March 1993	Interface Control Document, DMSP Block 5D-3 (S16-S20)/Mission Sensor SSM

ICD-88804 8 October 1993 ICN 001 19 May 1994	Interface Control Document, DMSP Block 5D-3 (S16-S20)/Mission Sensor SSIES3
ICD-88805 29 January 1993	Interface Control Document, DMSP Block 5D-3 (S16-S20)/Mission Sensor SSJ5
ICD-88806 20 August 1992	Interface Control Document, DMSP Block 5D-3 (S16-S20)/Mission Sensor SSMIS
ICD-88807 26 June 1996	Interface Control Document, DMSP Block 5D-3 (S16-S20)/ Mission Sensor SSF
ICD-88809 14 March 1995	Interface Control Document, DMSP Block 5D-3 (S16-S20)/Mission Sensor SSULI
ICD-88810 2 November 1993 ICN 001 22 May 1994 ICN 002 1 April 1996	Interface Control Document, DMSP Block 5D-3 (S16-S20)/Mission Sensor SSUSI

(Copies of military standards may be obtained from the following:

Naval Publications and Forms Center/Code 3015
5801 Tabor Ave.
Philadelphia, PA 19120

Copies of specifications, interface control documents and other government documents may be obtained from the following:

Space and Missiles Systems Center (SMC)
2420 Vela Way, Suite 1476-A8
Los Angeles AFB
El Segundo, CA 90245-4659)

2.2 Non-government documents. The following documents form a part of this specification to the extent specified herein. In the event of conflict between the documents referenced herein and the contents of this specification, the contents of this specification shall be considered a superseding requirement.

None.

3.0 INTERFACE REQUIREMENTS

3.1 Physical. The physical interfaces, and requirements for the interfaces, between the spacecraft, mission sensors, and OLS are as described herein. In general, the spacecraft contractor shall provide for the mounting of sensor units such that the field of view requirements and alignment requirements of each sensor are satisfied. The spacecraft shall provide, as a minimum, the volume and mounting areas required for each sensor unit and shall provide additional clearance, as necessary, for installation, connector access, thermal blanketing, dynamic excursions, and any other required clearances.

3.1.1 Mass properties. The specific mass characteristics for each sensor are defined in the individual spacecraft/sensor ICD listed in Appendix A.

3.1.1.1 Payload weight allocations. The Satellite payload weight allocations are as follows:

a. 315 lbs allocated to the OLS.
b. 439 lbs allocated to:
   (1) Mission sensor complement and command and telemetry encryption and decryption units
   (2) Spacecraft growth required as a result of either the addition of new mission sensors or contracted spacecraft design enhancements
   (3) The Glare Obstructor (GLOB)
   (4) All balance weights required by the Satellite.

3.1.1.2 Harness weight accounting. Harness weights shall be accounted for as follows:

a. All harnesses that connect components of a mission sensor system shall be provided by the mission sensor contractor and shall be included in that sensor's weight budget. Any harnesses to be included in the spacecraft weight budget shall be documented as such in the applicable sensor ICD listed in Appendix A.
b. All harnesses that connect components of the OLS shall be provided by the OLS contractor and shall be included in the OLS weight budget.
c. Harness between the OLS or mission sensors and the spacecraft shall be provided by the spacecraft contractor and shall be included in the spacecraft weight budget.
d. Harness between the OLS and each mission sensor shall be provided by the spacecraft contractor and shall be included in the spacecraft weight budget.
e. The OLS test harness shall be provided by the spacecraft contractor and shall be included in the spacecraft weight budget.

3.1.2 Mechanical interface. The spacecraft shall provide the volumetric envelope and mounting area, and shall accommodate the Field of View requirements specified in the mission sensor and OLS ICDs listed in Appendix A.

3.1.2.1 Structural and thermal loads. The spacecraft structure subsystem shall provide mounting of OLS and mission sensors in a manner that meets the structural and thermal requirements of the mission.

These mounting requirements are detailed in the individual ICDs listed in Appendix A, and form the basis for the design of the particular sensor interfaces.

3.1.2.2 Fields of view (FOV). The spacecraft contractor shall provide arrangement of components and sensors such that the fields of view required by the OLS remains unobstructed, the fields of view required by the mission sensors are unobstructed to the greatest practical extent, and glare/glint is controlled. Detailed field of view requirements are documented in the appropriate ICD listed in Appendix A.

3.1.2.3 Uncompensated momentum. As a design goal, each sensor shall minimize uncompensated momentum. The total uncompensated momentum from all sensor sources shall be less than 60.3 inch-pound-seconds about any axis. Details of individual sensor uncompensated momentum are described in the appropriate ICD listed in Appendix A.

3.1.3 Alignment. The alignment of any operating sensor shall be maintained with other sensors operating, non-operating, failed, or absent, in part or entirely.

3.1.3.1 Alignment axes

3.1.3.1.1 OLS primary coordinate axes. The coordinate axes X_p, Y_p, Z_p, of the Sensor Scanning Subsystem (SSS) of the Operational Linescan System (OLS) are the primary alignment reference axes for the Satellite.

The orientation of the primary axes are defined by a set of secondary reference axes, X_r, Y_r, Z_r, and a set of rotation angles d_{xy}, d_{xz}, d_{yz} from the secondary axes. The relationship of these axes and angles is shown in Figure 2.

3.1.3.1.2 OLS secondary axes. The secondary reference axes, X_r, Y_r, Z_r comprise the normals to the X, Y, and Z surfaces, respectively, of the secondary reference optical cube of the SSS.

The optical cube, the values of d_{xy}, d_{xz}, d_{yz}, and the order of rotation shall be furnished for each OLS by the OLS Contractor.

3.1.3.1.3 Mission sensor coordinate axes. The X, Y, Z coordinate axes of each mission sensor are defined by a set of secondary reference axes X_r, Y_r, Z_r and a set of rotation angles d_{xy}, d_{xz}, d_{yz} from the secondary axes.

The relationships of these axes and angles is exactly analogous to that shown in Figure 2 where the X, Y, Z axes of a mission sensor are analogous to the X_p, Y_p, Z_p axes of Figure 2. The sense and nominal direction of the X, Y, Z axes of each mission sensor are shown in the individual ICDs listed in Appendix A.

3.1.3.1.4 Mission sensor secondary axes. The secondary reference axes X_r, Y_r, Z_r, for each mission sensor shall be defined by the normals to the X, Y, Z surfaces, respectively, of the secondary reference optical cube (or secondary reference surfaces) of each mission sensor. A secondary reference surface shall be defined for each mission sensor, in the applicable mission sensor ICD listed in Appendix A, along with the values of d_{xy}, d_{xz}, and d_{yz} and the order of rotation. For mission sensors whose secondary axes are defined by normals to an optical cube, an optical cube shall be supplied by the mission sensor contractor with each sensor unit.

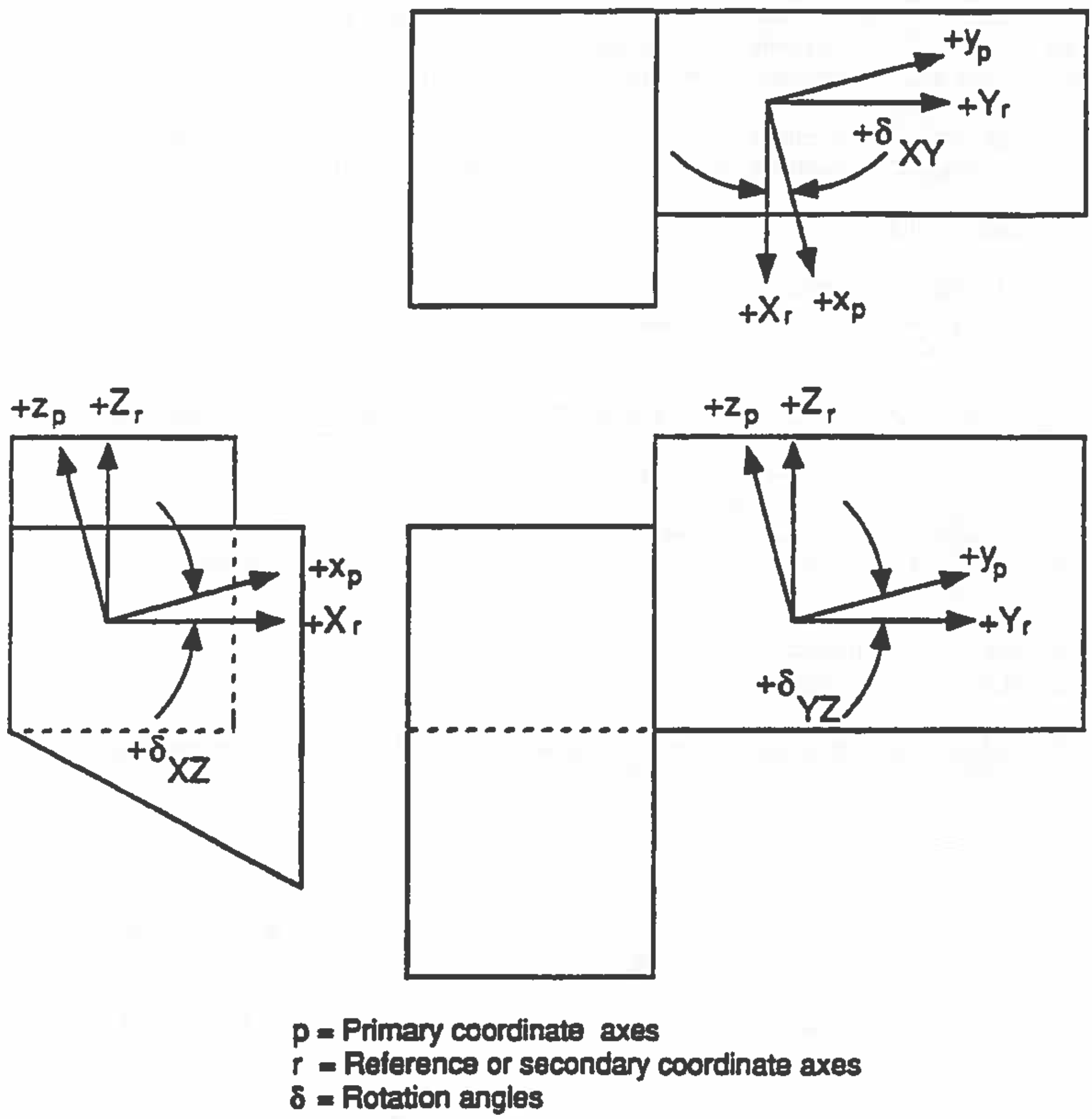


Figure 2. Alignment Coordinates

3.1.3.2 Alignment requirements

3.1.3.2.1. Primary axes. The spacecraft shall maintain the attitude of the satellite such that the primary axes X_p, Y_p, Z_p, of the SSS remain aligned with the X, Y, Z reference axes of the geodetic local vertical within 0.01 degrees. This attitude shall be maintained during all orbital conditions for the life of the Satellite except during start -up and shut-down of the mission sensor SSMIS. The tolerance is three-sigma per axis, and includes all errors due to mounting, structural alignment, attitude determination and control, thermal distortions, and all other error sources affecting SSS orientation.

3.1.3.2.2. Mission sensor axes. The spacecraft shall maintain, under all orbital conditions, the X, Y, Z axes of each mission sensor aligned with the X_p, Y_p, Z_p reference axes of the SSS to the tolerance requirements specified in the applicable sensor ICD listed in Appendix A.

3.1.3.3 Alignment knowledge. The spacecraft contractor shall provide sensor and component mounting arrangement such that measurement of actual alignment of the sensors is possible when the sensors are integrated with the spacecraft. Details of the tolerance for component mounting and the alignment knowledge for each sensor are described in the ICDs listed in Appendix A.

3.1.4 Materials

3.1.4.1 Outgassing. In accordance with ASTM-E595-77, materials used in the fabrication of the spacecraft or the sensors shall not have outgassing characteristics greater than the following:

0.1% volatile condensable material (VCM) loss in vacuum
1% total mass loss (TML) in vacuum

unless they are suitably encased to prevent contamination of adjacent equipment.

3.1.4.2 Dissimilar metals. Use of dissimilar metals in direct contact shall be avoided in all interfaces. Metals used shall be of the corrosion resistant type or shall be suitably treated and controlled to resist corrosion, except where untreated surfaces are needed to meet electrical grounding requirements. Interface details are contained in the individual sensor ICDs listed in Appendix A.

3.1.4.3 Magnetic materials. The total magnetic moment of any individual sensor shall not exceed 0.1 ampere- turn- meter^2 along any axis. The mission sensor SSMIS is an exception to this requirement, having been granted relief to 0.4 ampere-turn-meter^2. Detailed sensor magnetic characteristics are defined in the applicable sensor ICD listed in Appendix A.

The total magnetic moment from all sensors shall not exceed 1.0 ampere- turn- meter^2 along any axis. The maximum magnetic moment difference between any modes of operation for the OLS and each mission sensor shall be 0.01 ampere-turn-meter^2.

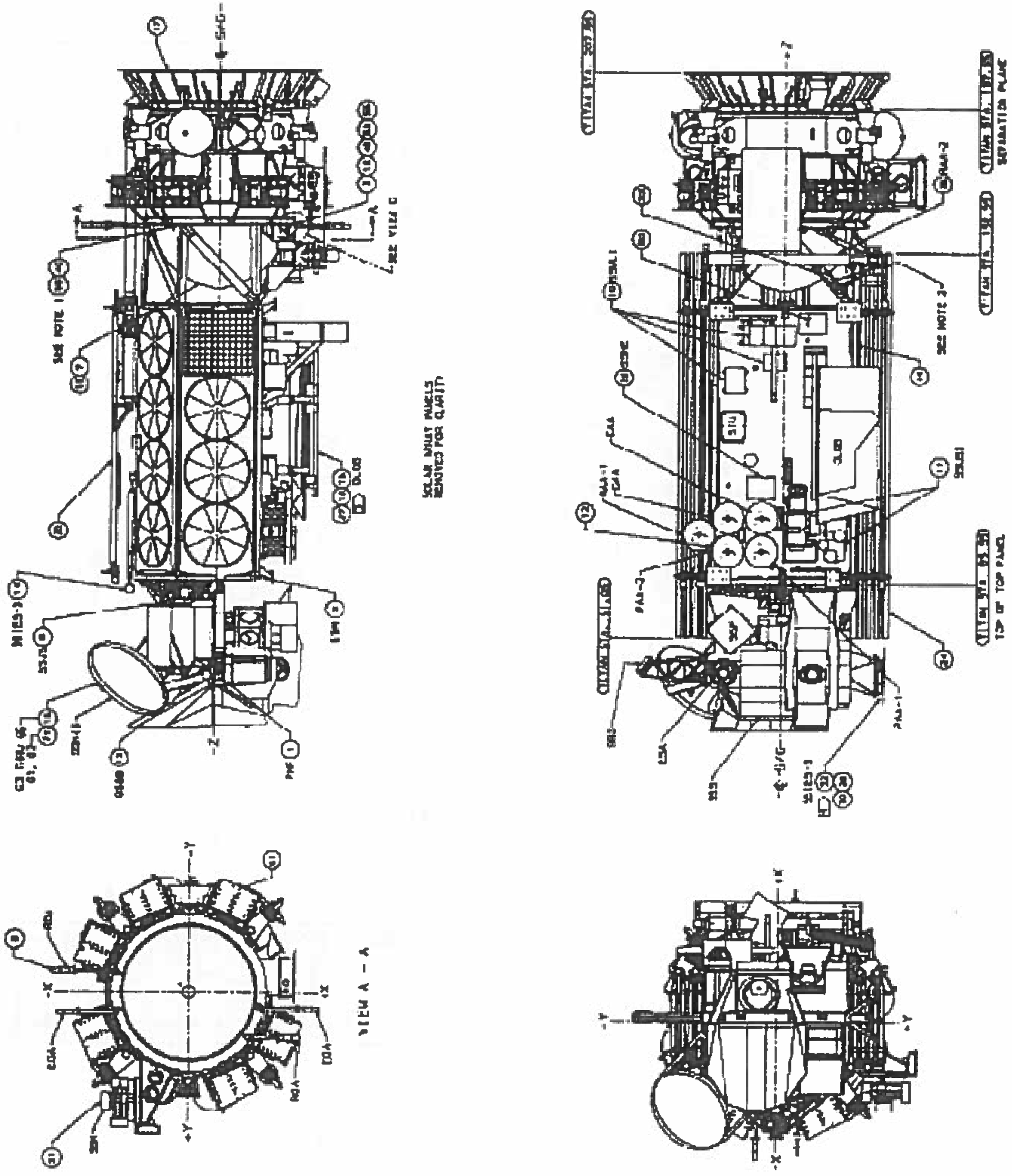
3.1.5 Deployment characteristics. The spacecraft deployment characteristics, to the extent that they may affect the sensors, are shown in Figures 3 through 11.Sensor and sensor cover deployment characteristics are described in the individual sensor ICDs listed in Appendix A.

3.1.6 Structural and thermal models. Requirements for sensor structural and/or thermal models, when necessary, shall be as defined in each sensor ICD listed in Appendix A.

3.1.7 Actuator location. The location of actuators shall permit access for installation and arming, if required, while the sensor is mounted on the spacecraft.

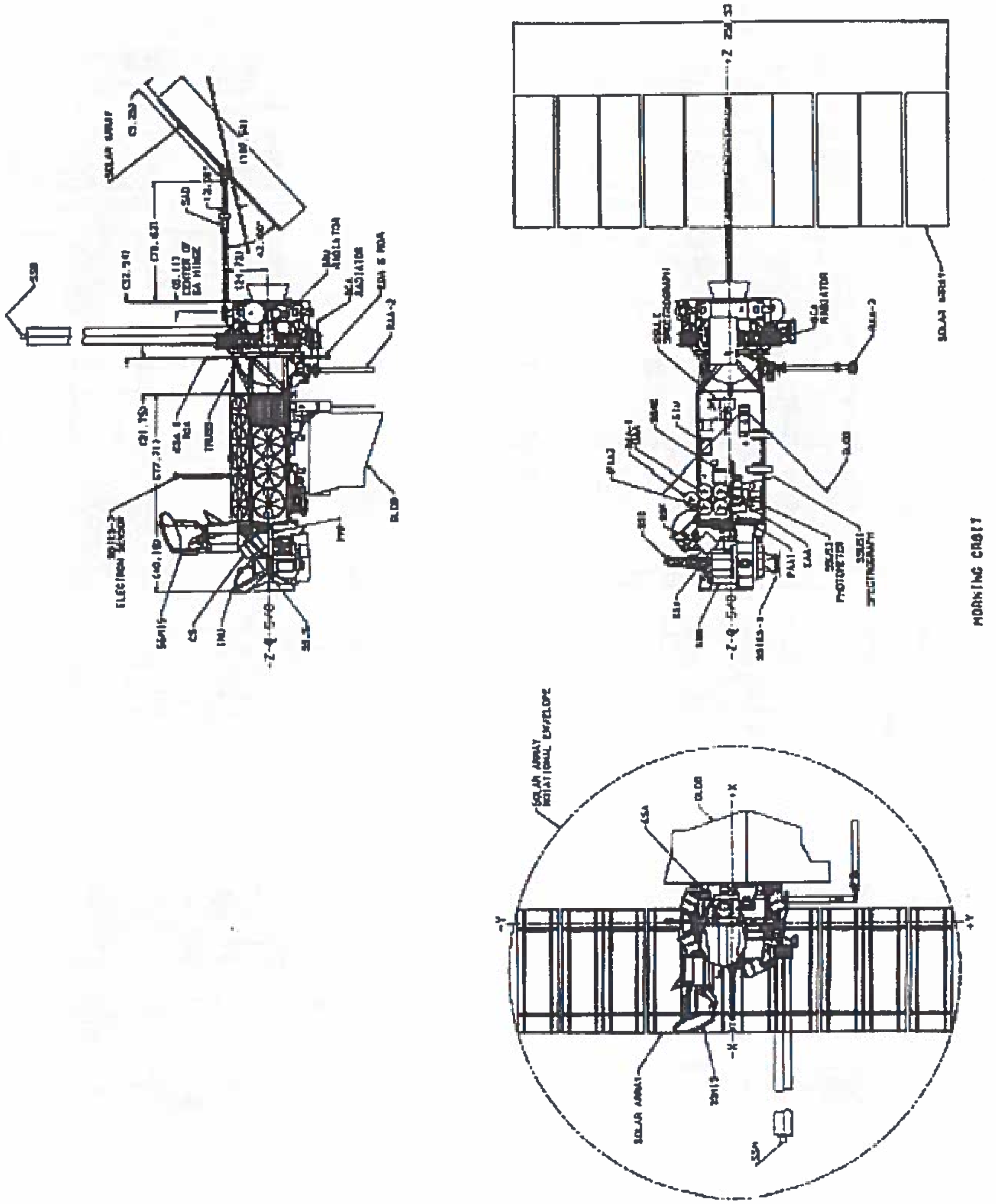
3.1.8 Connector spacing. The location of connectors and spacing of adjacent connectors shall permit access for mate and demate while the sensor is mounted on the spacecraft. For the purpose of design of connector layout, a minimum edge to edge spacing of 0.250 inches is recommended for adjacent connectors. Connector orientations which result in less than the recommended clearance shall be reviewed by the spacecraft contractor prior to implementation, and shall be judged acceptable or unacceptable.

3.1.8.1 Keying of connectors. The use of keyed connectors is encouraged as a means of ensuring that no two connectors are inadvertently mismated. Details of keying technique for individual sensors are contained in the sensor ICDs listed in Appendix A.



Ref LMMS dwg 20025564

Figure 3. DMSP Block 5D-3 (S16-S20) Stowed Configuration



Ref LMMS dwg 20026963

Figure 4. DMSP Block 5D-3 (S16-S20) Orbit Configuration

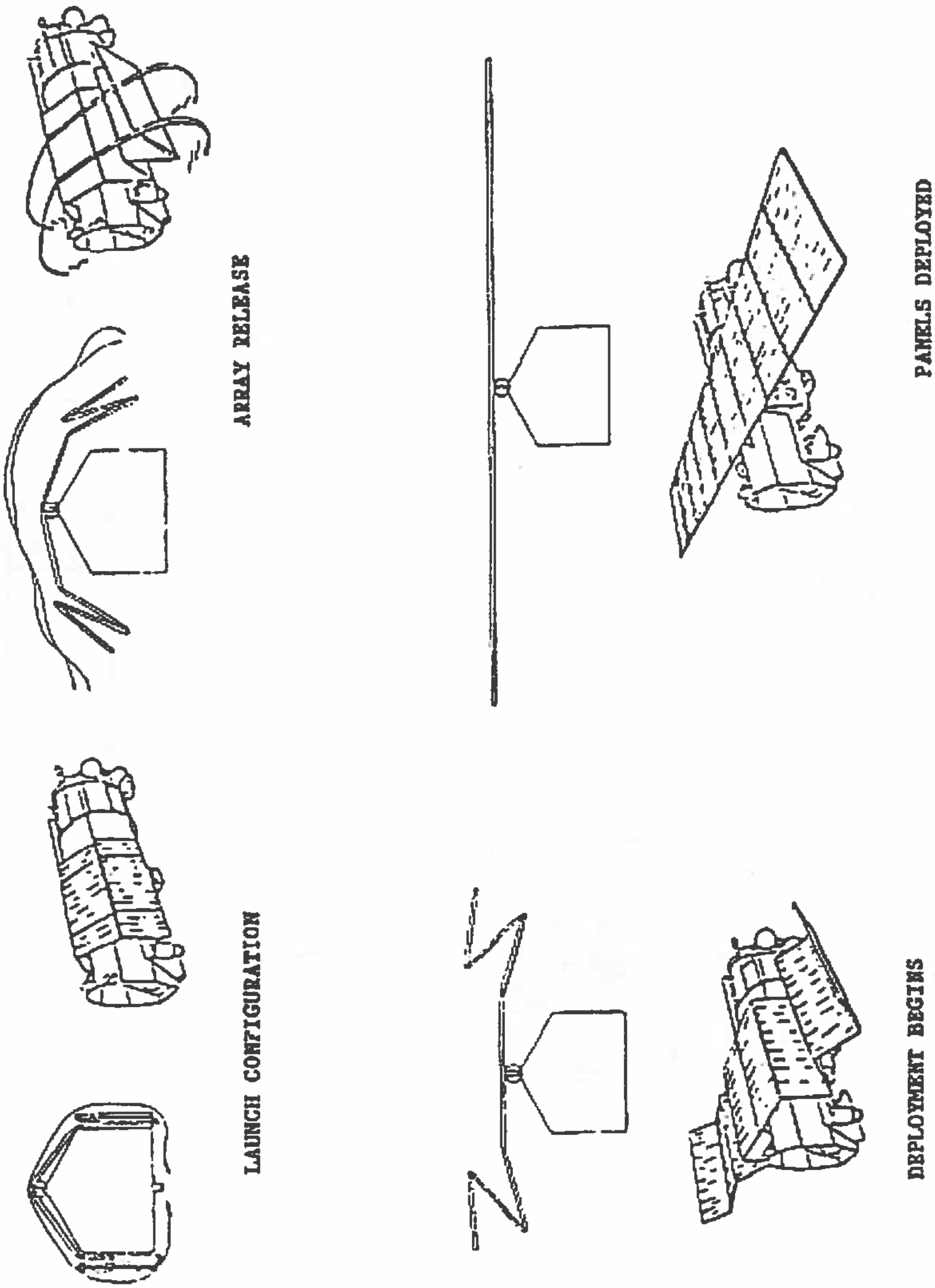


Figure 5. Solar Array Initial Deployment Phase

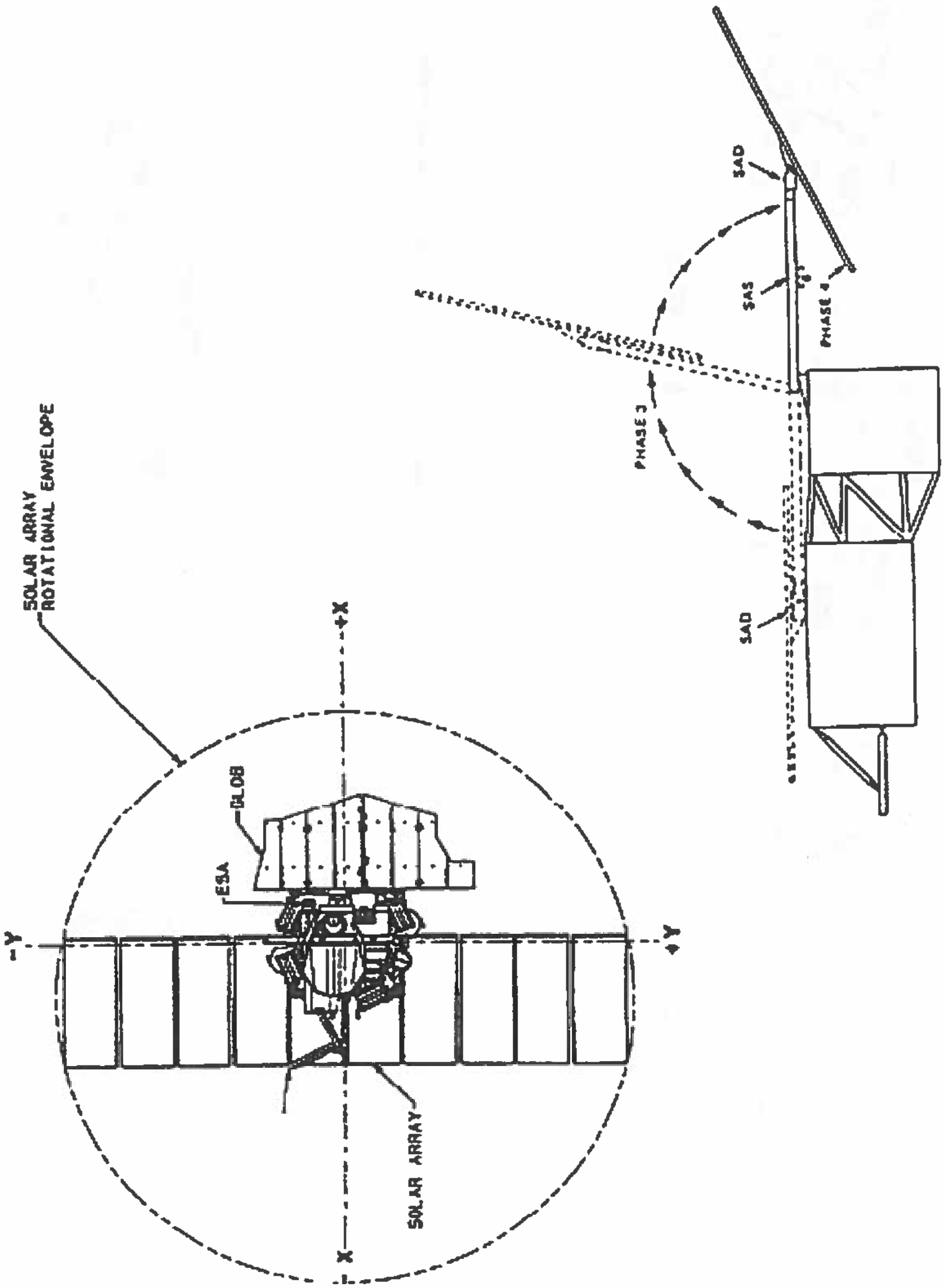
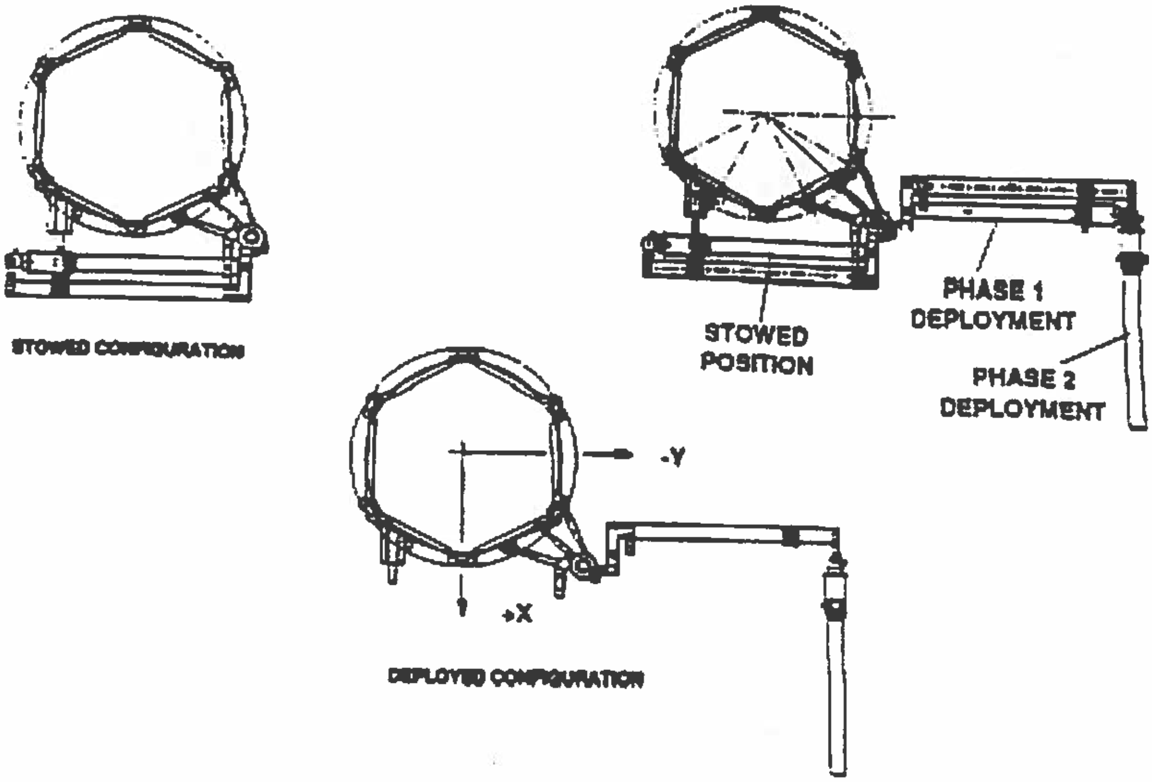
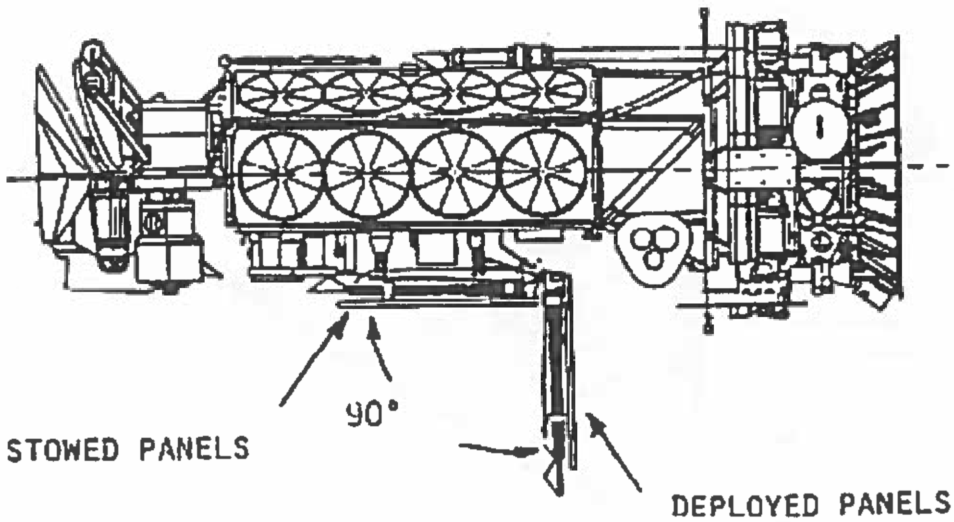


Figure 6. Solar Array Final Deployment Phase

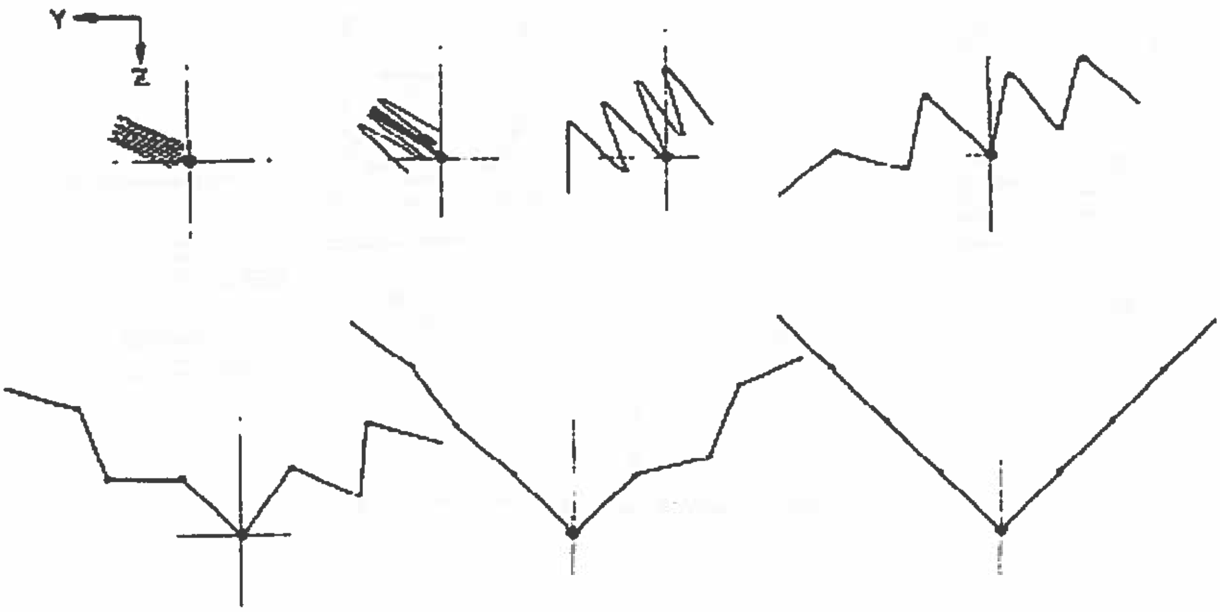


RDS UHF Antenna (RAA-2) Deployment

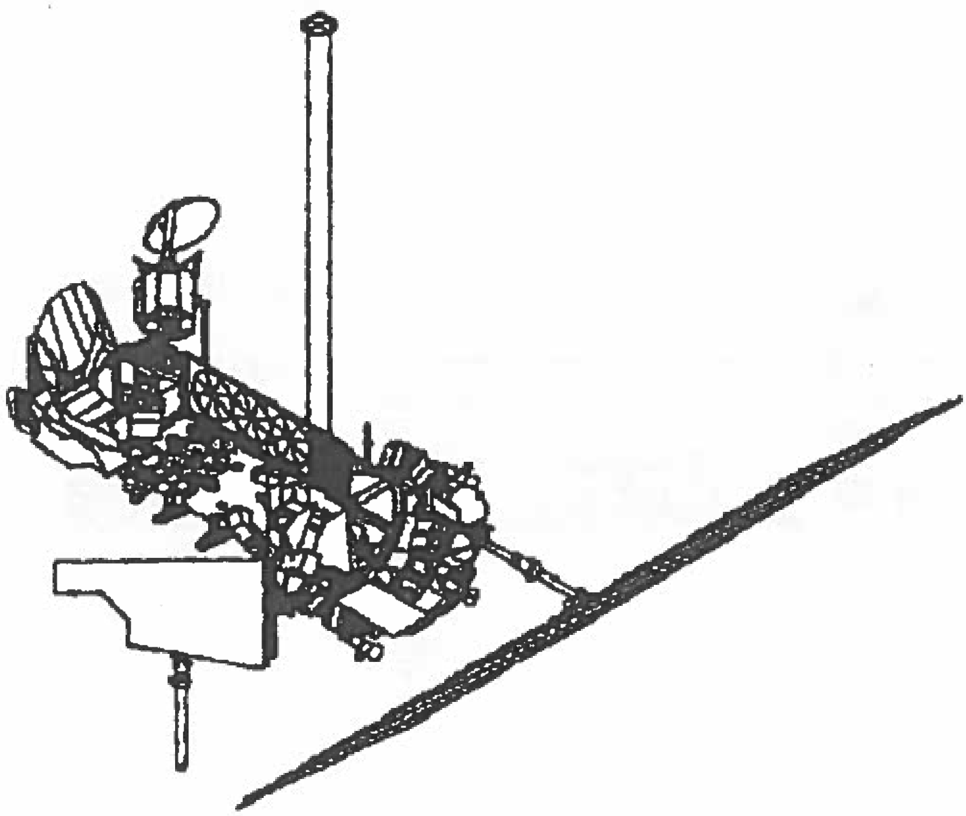


Glare Obstructor Phase 1 Deployment

Figure 7. RDS UHF Antenna Deployment and Glare Obstructor Phase 1 Deployment

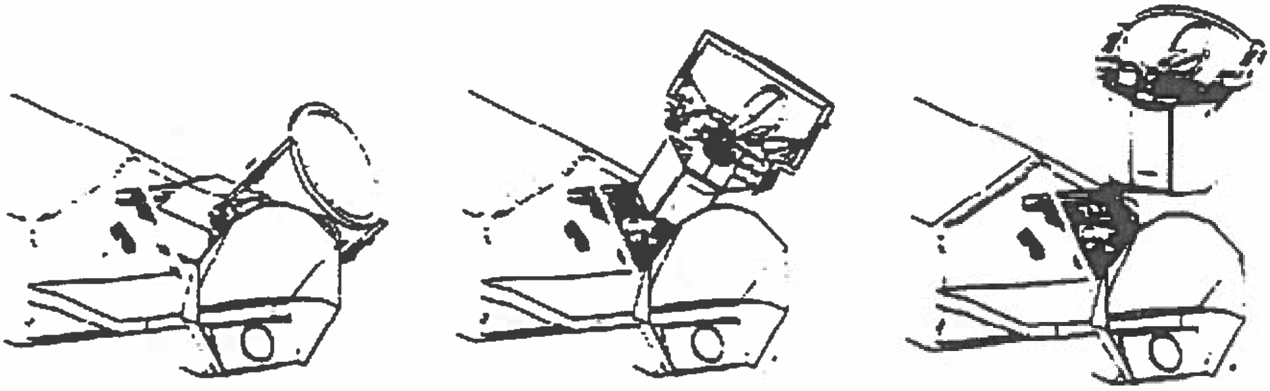


GLOB Phase 2 Deployment

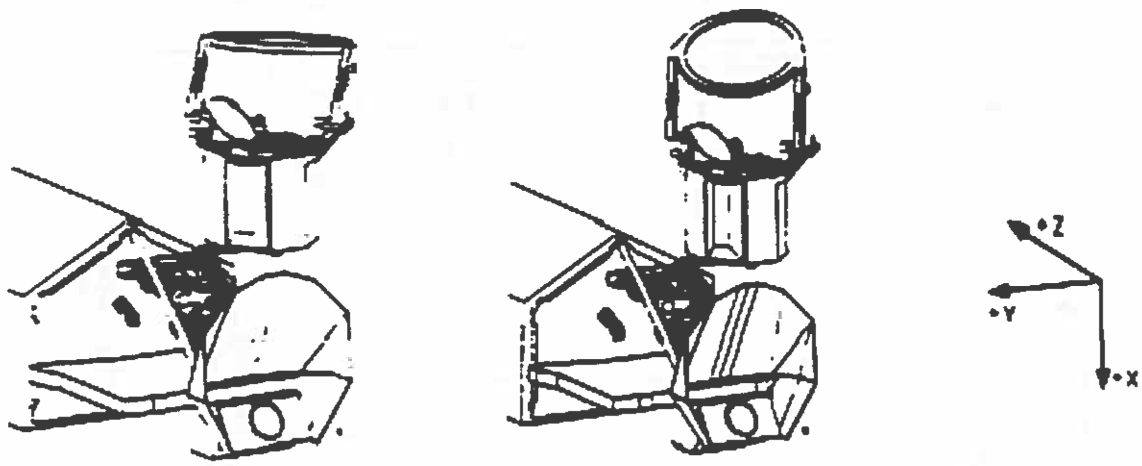


GLOB Fully Deployed

Figure 8. Glare Obstructor Phase 2 Deployment



SSMIS Phase 1 Deployment - Lateral (-Y) Shift with 90 Degree Rotation



SSMIS Phase 2 Deployment - Reflector Deployment

Figure 9. SSMIS Deployment

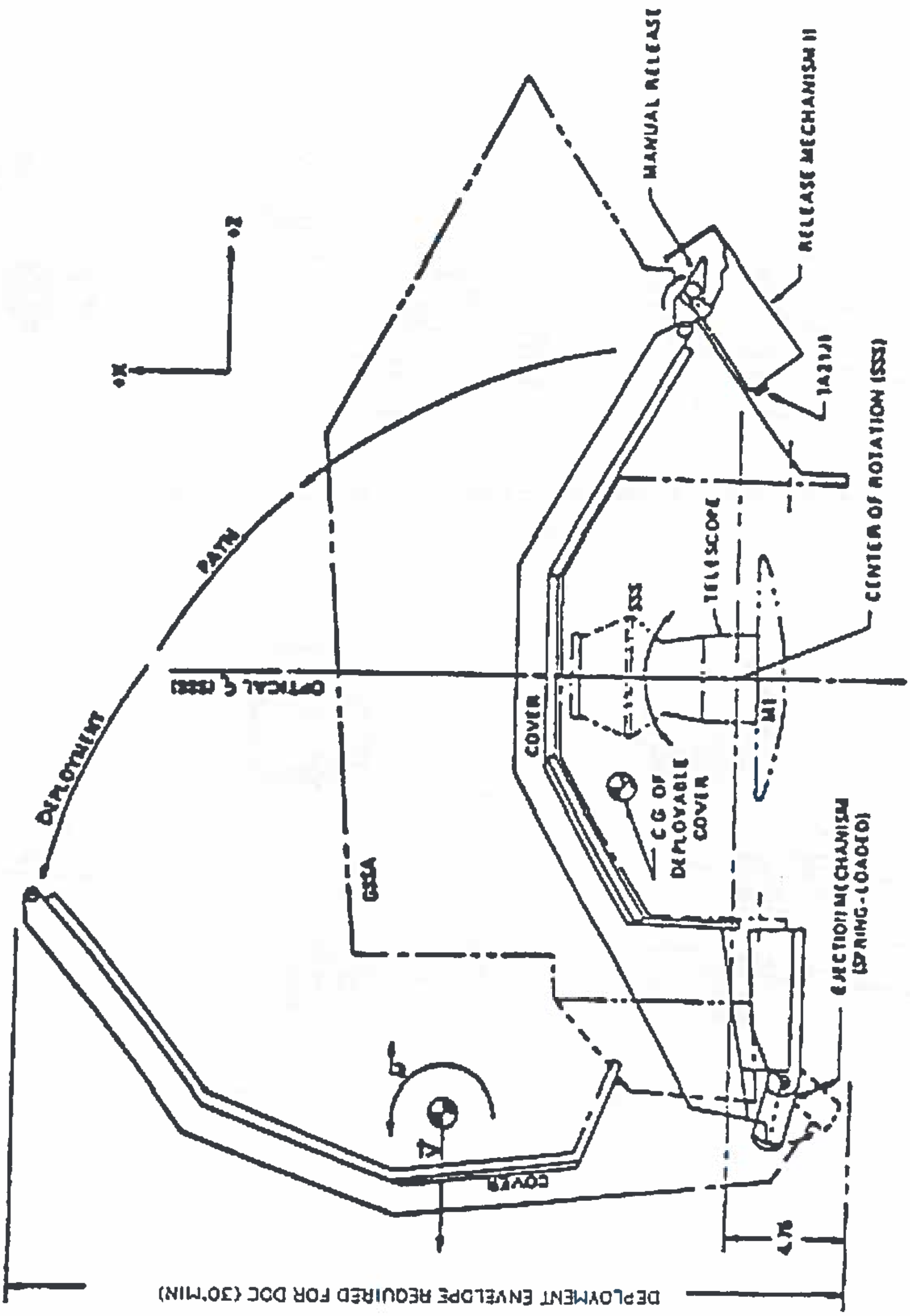
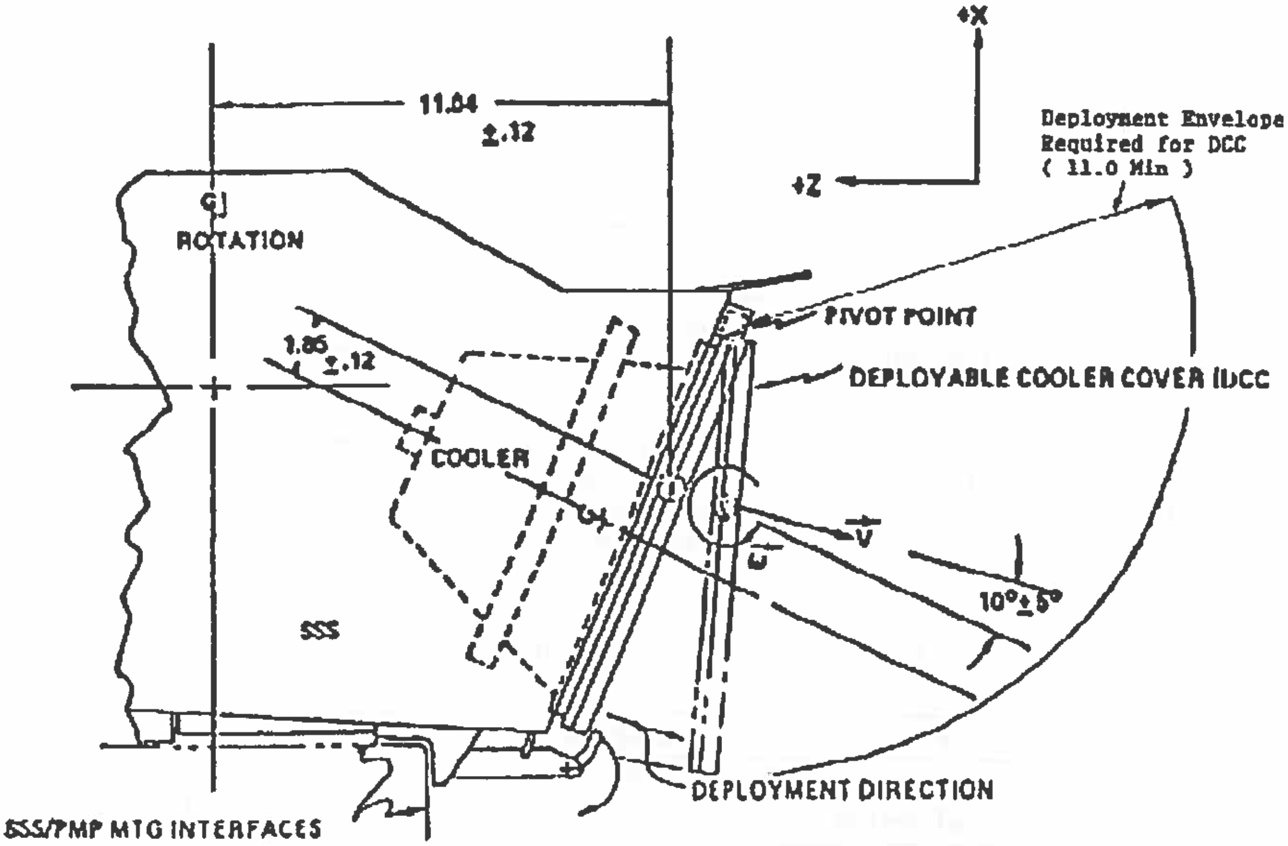


Figure 10. Deployable Optics Cover (DOC) Configuration and Deployment



At the instant the Deployable Cooler Cover (DCC) disengages from the SSS at its pivot point, its C.G. will have a linear velocity (V) in the direction shown and it will have an angular velocity (ω) about its C.G. causing it to tumble away from the spacecraft

Figure 11. DCC Configuration and Deployment

3.2 Spacecraft/OLS and spacecraft/mission sensor electrical interface

3.2.1 General. The spacecraft shall be functionally compatible with the OLS units and the mission sensor units and shall have the capability of supporting all sensors as described in this interface specification. The spacecraft shall provide power for the sensors, and shall provide, collect, store, and transmit the sensor interface signals. Detailed description of the applicable sensor/spacecraft interfaces are contained in each sensor/spacecraft ICD listed in Appendix A.

3.2.2 Interface signal characteristics. The interface signal characteristics shall be as defined in each sensor/spacecraft ICD listed in Appendix A.

3.2.3 Actuator signals. The purpose of the actuator signals are to initiate any caging devices included in the sensors. The specific requirements for actuator signals from the spacecraft to a sensor will be included in the applicable sensor/spacecraft ICD listed in Appendix A. When an actuator signal is provided by the spacecraft, it shall consist of firing pulses from 4.4 to 7.75 amperes for a minimum of 25 milliseconds. The sensor shall provide an input impedance of 1.3 ±0.2 ohms resistive for each caging device that requires an actuator signal. The actuators shall have a maximum no-fire current of 1 ampere and a minimum all-fire current of 4.4 amperes. The SSMIS and SSUSI shall provide an input impedance of 1.05 ±.1 ohms with a minimum all-fire current of 3.5 amperes.

3.2.4 Equipment Status Telemetry (EST). The spacecraft shall accept and process a maximum of 306 EST signals from the sensors allocated as shown in Table I. The number of EST signals used, their characteristics, and functions shall be defined in the individual sensor interface control documents listed in Appendix A.

Table I. Equipment Status Telemetry Allocation

Source	Analog Data	Discrete Data
OLS	89	65
Special Sensors	112	40
Total-All Sensors	201	105

3.2.4.1 Analog data channel input. All analog data channels shall have nominal input signal levels of 0 volts to +5.10 volts with the EST subsystem surviving overrange limits of -15 volts to +15 volts. The source impedance shall be a minimum of 2 kilohms and a maximum of 15 kilohms. The 15k ohm maximum can be exceeded if the defined Programmable Information Processor (PIP) accuracy of ±2 least significant bits is not required.

3.2.4.2 Discrete data channel input. The discrete data channels shall be used for on/off, true/false, or yes/no type data. A logic "1" (low or true or on) shall be -0.1 volt to +1.5 volt. A logic "0" (high or false or off) shall be +3.5 volts to +5.7 volts. The source overrange limits shall be -15 volts to +15 volts. The source impedance shall be a minimum of 2 kilohms and a maximum of 15 kilohms.

3.2.4.3 EST subsystem characteristics. The spacecraft EST subsystem shall impose the following conditions:

a. load - 2 megohms minimum for signals within the data voltage range.

b. data range - 0.0V to +5.1V referenced to spacecraft single point ground.

c. PIP gate failure condition - +5 milliamperes maximum fed back to source.

d. bandwidth - The sampling rate shall vary from 1 sample per 10 seconds to 4 samples per second in the normal 2 and 10 kbit/sec Time Commutated Telemetry Readout (TCTR) modes. Dwell modes are available which provide sampling rates from 32 samples per second to 3600 samples per second. Analog data is DC coupled and is not subject to any filtering other than that provided by the sampling process. The standard sampling rates for each PIP mode are as shown in Table II.

e. quantization - 8 bits per sample.

f. accuracy - ±2 least significant bits (with source impedance 2 to 15 kilohms).

Table II. Standard PIP Sampling Rates

PIP Mode (kbit/sec)	Standard Sampling Rates (Samples/Seconds)	
	Analog	Discrete
2	1/10, 1/1.25	1/10
10	1/2, 4/1	1/2
60	1/2, 4/1	3/1

3.2.5 Power. The spacecraft shall supply a peak output current of 19.6 amperes at +28.0 volts DC and up to 2.0 amperes of current at +5.0 volts DC to the OLS and to the mission sensors with the characteristics described in the subsequent paragraphs. In addition, the spacecraft shall supply a regulated power bus of +10 volts DC to the OLS only, with the characteristics described in section 3.2.5.3.

3.2.5.1 +28 Volts

3.2.5.1.1 Regulated power bus. The spacecraft shall provide a regulated power bus at a level of +28.0 ±0.56 volts exclusive of ripple voltage and transients. The voltage on the +28 volt bus shall not exceed +38 volts, including all ripple and transients. Bus voltage during load current changes up to 6.0 amperes applied at any rate not exceeding 50 milliamperes/millisecond shall not deviate from nominal by more than 200 millivolts for 100 milliseconds maximum.

3.2.5.1.2 Output impedance. The +28 volt regulated bus output impedance shall be as shown in Table III.

Table III. +28 Volt Regulated Bus Output Impedance

Frequency Range	Ohmic Value (Less Than)
DC - 100 Hz	0.03
100 Hz - 1 kHz	0.1
1 kHz - 100 kHz	0.2
100 kHz - 1 MHz	2.0
1 MHz - 10 MHz	20.0

3.2.5.1.3 Voltage ripple. The voltage ripple on the +28 volt bus from the Power Supply Electronics (PSE) shall not exceed 0.25 volt peak-to-peak maximum in the frequency range 10 Hz to 10 MHz.

3.2.5.1.4 Voltage transients. Maximum voltage in the presence of transients on the +28 volt bus shall not exceed the limits specified in Table IV. At 0.0 volts the sensors are in a non-operating state.

Table IV. +28 Volt Bus Voltage Transients

Peak Voltage (Volts)	Maximum Transient Width (milliseconds)
34.0	1
33.5	1.1
29.3	150
26.7	1800
0.0	2000

3.2.5.1.5 Current ripple. The steady state current ripple generated by each mission sensor, except for the SSF, on the 28 volt regulated bus, exclusive of stepping transients, shall not exceed 30 milliamperes peak-to-peak for a bus source impedance of 1 ohm or less. The SSF steady state current ripple may be as high as 400 milliamperes peak-to-peak with a rise and/or fall time greater than 1 ms. The OLS shall be designed to minimize the current ripple imposed on the spacecraft +28 volt power bus. Peak rate of change of load current shall not exceed 50 milliamperes per microsecond with the exception of BBx and OLS power enable turn-on transients. Total OLS current ripple, when measured in a 0-2000 Hz passband shall not exceed 50 milliamperes in RMS value and 700 milliamperes in peak value in air (500 milliamperes in vacuum). Additionally, the peak narrow band (10 Hz) current ripple shall not exceed 100 milliamperes anywhere within the 0-2000 Hz passband.

3.2.5.1.6 Current transients. The rate of load current transients initiated by each mission sensor on the +28 volt regulated bus shall be 20 milliamperes or less per microsecond. During spacecraft power initialization, the spacecraft shall supply peak currents of 2.5 amperes for less than 1 millisecond on the 28 volt regulated bus. The OLS (excluding KG boxes BB1, BB2, BB3, and BB4 shall require a peak transient current of less than 3 amperes from the +28 volt bus for a period of 10 seconds or less for mode changes. The BBxs (KG boxes) may each require peak transient currents of up to 8 amperes each on initial application of power to the +28 volt bus. Transient current required by the BBxs in nominal operation shall not exceed 0.25 amperes. The SSMIS shall be allowed a rate of load current transients of ±30 milliamperes or less per microsecond, with a maximum peak transient current of 735 milliamperes at that rate.

3.2.5.2 +5 Volts

3.2.5.2.1 Regulated power bus. The spacecraft shall normally provide a continuous and uninterrupted regulated power bus at a level of +5.0 ±0.25 volts. The voltage on the +5 volt bus shall not exceed +7.5 volts, including all ripple and transients. If conditions should necessitate this bus having to be interrupted, then the OLS must be OFF prior to this interruption for normal operation of the OLS to occur. During +5 volt power converter switching (primary mode to back-up mode or vice-versa) the +5 volt bus can reduce to 0 volts for up to 0.1 second.

3.2.5.2.2 Output impedance. The +5.0 Volt regulated bus output impedance shall be as shown in Table V.

Table V. +5 Volt Regulated Bus Output Impedance

Frequency Range			Ohmic Value (Less Than)
DC	-	100 kHz	0.2
100 kHz	-	1 MHz	2.0
1 MHz	-	10 MHz	20.0

3.2.5.2.3 Voltage ripple. Voltage ripple on the +5 volt bus shall not exceed 0.25 volt peak-to-peak for frequencies up to 10 MHz.

3.2.5.2.4 Voltage transients. Maximum voltage in the presence of transients on the +5 volt bus shall not exceed the limits specified in Table VI.

Table VI. +5 Volt Bus Voltage Transients

Peak Voltage above Nominal (Volts)	Maximum Transient Width (Microseconds)
0.05	200 to 500
0.10	150 to 200
0.50	100 to 150
1.00	50 to 100
1.50	0 to 50

3.2.5.2.5 Current ripple. Current ripple generated by each mission sensor on the +5.0 volt regulated bus shall be less than 0.5 milliamperes peak-to-peak. The current ripple generated by the OLS on the +5 volt bus, exclusive of mode changing transients, shall not exceed 50 milliamperes peak-to-peak for ripple frequencies up to 100 Hz, and 100 milliamperes peak-to-peak for ripple frequencies from 100 Hz to 10 MHz.

3.2.5.2.6 Current transients. The rise rate of load current transients initiated by each mission sensor on the 5.0 volt regulated bus shall be 1 milliampere or less per microsecond. During spacecraft power initialization, the spacecraft shall supply a peak current of 1 ampere for less than 2 milliseconds on the 5.0 volt regulated bus. The OLS shall require a peak transient current of less than 0.5 amperes from the +5.0 volt bus for a period of 1 second or less for mode changes.

3.2.5.3 +10 Volts

3.2.5.3.1 Regulated power bus. The spacecraft shall provide a regulated power bus at a level of +10.0 ±0.5 volts to the OLS only. The maximum +10V current supplied to the OLS shall be 12 milliamperes. The OLS shall current limit this +10 volt line to 100 milliamperes. The voltage on the +10 volt bus shall not exceed 12 volts, including all ripple and transients. The +10 volt bus shall be used only for powering control signal interface circuits.

3.2.5.3.2 Output impedance. The source impedance of the +10 volt bus shall be as shown in Table VII.

Table VII. +10 Volt Regulated Bus Output Impedance

Frequency Range	Ohmic Value (Less Than)
DC - 200 kHz	0.4
200 kHz - 1 MHz	2.0
1 MHz - 10 MHz	20.0

3.2.5.3.3 Voltage ripple. Voltage ripple on the +10 volt bus shall not exceed 250 millivolts peak-to-peak for frequencies up to 10 MHz.

3.2.5.3.4 <u>Voltage transients.</u> Voltage transients on the +10 volt bus shall not exceed ±1.0 volt nominal-to-peak with a maximum pulse width of 50 microseconds.

3.2.5.3.5 <u>Current ripple.</u> The peak-to-peak amplitude of steady-state load current ripple generated by the OLS shall not exceed 5 percent of the maximum average steady-state current drawn by the OLS. The fundamental frequency of load current ripple shall not exceed 2.5 MHz.

3.2.5.3.6 <u>Current transients.</u> Transient load currents drawn by the OLS shall not exceed 125 percent of the maximum average steady-state current drawn by the OLS and steady-state operation shall be attained within 50 milliseconds from the start of the transient. The rate of change of current during transients shall in no case exceed 20 milliamperes per microsecond.

3.2.5.4 <u>Power application.</u> The sensors shall be designed to withstand the following initialization sequence of spacecraft power:

a. The +28 volt bus will be increased from 0 to +28 volts within 2 seconds. During the initialization sequence the maximum rate of change on the 28 volt bus shall typically be 500 to 1000 volts per second.
b. The +5 volt bus will increase from 0 to +5 volts during the +28 volt bus buildup.
c. The +10 volt Controls Power Converter (CPC) bus will increase from 0 to +10 volts during the +28 volt bus buildup.
d. The nominal +5 volt and +10 volt control signal outputs will be at indeterminate levels (although within the normal 0 to 5 volt and 0 to 10 volt ranges respectively) during the +28 volt bus buildup.

No operational spacecraft mode shall remove +28 volts, +10 volts, or +5 volts from the sensors, except as described in paragraph 3.2.5.2.1 of this IS and paragraph 3.1.2.3.1 of SSMIS ICD 88806.

3.2.6 <u>Grounding requirements.</u> The spacecraft and sensors shall implement a common grounding philosophy. The spacecraft grounding shall accommodate the grounding requirements for the sensors as defined herein and in each sensor/spacecraft ICD listed in Appendix A.

3.2.6.1 <u>Spacecraft signal grounds.</u> The spacecraft provided signal grounds shall be electrically isolated by at least 100k ohm DC from the chassis ground and from power ground. As a design goal, the AC isolation shall be no more than 50 picofarads. Detailed requirements for signal grounds are specified in each sensor/spacecraft ICD listed in Appendix A.

3.2.6.2 <u>Spacecraft power grounds.</u> The spacecraft shall provide power grounds (or returns) for the squib signals and the power interfaces. Squib signals and their returns are provided from the Signal Conditioning Unit on a twisted shielded pair and shall be isolated from all grounds and returns by at least 100 kilohms DC. The spacecraft provided power ground shall be electrically isolated by at least 100 kilohms from the chassis ground and signal returns.

3.2.6.3 <u>Chassis (case) ground.</u> Except for the SSS of the OLS which has the case isolated from the mounting structure, each sensor chassis (case) shall be electrically connected to the spacecraft by an insulation-free metal-to-metal bond of 0.1 ohm or less. All mission sensors shall be designed to provide a chassis ground output pin for use in testing.

3.2.7 Sensor harnessing. Harness responsibility shall be as follows:

a. All harnesses that connect components of a mission sensor system shall be provided by the mission sensor contractor
b. All harnesses that connect components of the OLS shall be provided by the OLS contractor
c. Harness between the OLS or mission sensors and the spacecraft shall be provided by the spacecraft contractor
d. Harness between the OLS and each mission sensor shall be provided by the spacecraft contractor
e. The OLS test harness shall be provided by the spacecraft contractor.

A description of the harness characteristics, connectors, and pin assignments for each sensor/spacecraft functional interface is included in the applicable sensor/spacecraft ICD listed in Appendix A. Each OLS/mission sensor interface signal and its respective return shall be a twisted pair and the cable shall be bundle shielded with the shield terminated at the OLS end. The OLS/mission sensor cable shall have a maximum capacitance of 900 picofarads from any signal line to all other lines and shield.

3.2.8 Electromagnetic compatibility. The spacecraft and sensors shall be designed to minimize emanation of and susceptibility to spurious electromagnetic signals, both radiated and conducted. The spacecraft will have transmitters operating at 400.328 MHz, 400.822 MHz, 2222.5 MHz, 2207.5 MHz, 2237.5 MHz, 2252.5 MHz, 2267.5 MHz and a receiver operating at 1791.748 MHz with a 300 MHz 60 dB bandwidth.

3.2.9 Fusing. The spacecraft Sensor Interface Unit (SIU) fuses power to all mission sensors. Generally, the +28 volt supply to each mission sensor is fused with a 3 ampere fuse, and the +5 volt supply to each mission sensor is fused with a 1 ampere fuse. Exceptions to this standard fusing scheme, as dictated by individual sensor needs, are detailed in the appropriate mission sensor ICD listed in Appendix A.

3.3 OLS/mission sensor electrical interface. Section 3.3 establishes the requirements for the interfaces between the OLS and each of the mission sensors (abbreviated SSPs or SSxs) and for the interfaces between the OLS field test equipment (FTE) and each SSP Aerospace Ground Equipment (AGE). The OLS provides control and data handling for up to 12 (inclusive) SSPs, provided the total requirement for SSP data does not exceed the OLS data handling capacity. The OLS shall be capable of supporting any combination of the SSPs, including the possibility of any or all of them being absent entirely or requiring less support than specified herein. Stated requirements which dictate either the OLS or SSP to provide interface signals shall require that the receiving segment be compatible with the interface signal provided.

3.3.1 SSP generated signals

3.3.1.1 SSP data. Each mission sensor shall provide digital data to the OLS for inclusion with the Stored Data Smooth Mode (SDS), Real-Time Data Smoothed Mode (RDS), or Real-Time Data Mode (RTD) primary data streams. The OLS shall accept a single data signal, designated as (SSxDAT), from each SSP. All SSP output interface signals to the OLS must remain at the zero volt level when the OLS is turned OFF.

3.3.1.1.1 Data content. The mission sensor data signal shall consist of digital Non-Return-to-Zero-Level (NRZ-L) data as defined in Figure 12. Data shall be transferred in phase with the OLS supplied bit clock in bursts of X contiguous bits, at a bit rate of 25600 plus or minus 1 Hz. The 90 percent level (defined in Figure 13.) of all NRZ-L SSP data (SSx DAT) transitions shall occur within 5.5 microseconds after receipt of the 90 percent level (defined in Figure 14) of the positive going edge of the bit clock (SSx BCK) when the loading on the SSxDAT signal is as defined in Figure 22. Each SSP shall upon command from the OLS provide one data block per second. One data block consists of a burst of X contiguous bits, X being the number of bits assigned to each sensor. All SSP data words shall be output from the SSP least significant bit (LSB) first of the first word, with the exception of the SSUSI, which shall output data most significant bit (MSB) first.

3.3.1.1.2 Bit structure. Each SSP shall structure their word length to be 6, 9, 12, 18, or 36 bits in length. In addition each SSP shall incorporate an output buffer or register of sufficient length to allow the OLS to interrogate that buffer or register for an integral number of 36 bit words. For example, a SSP having an output of 6 bits/second will place those 6 bits in a 36 bit (or larger) register. The OLS will output those six bits of data followed by 30 bits which reflect the state of the remainder of the SSP data register.

3.3.1.1.3 Data routing. The spacecraft will route mission sensor data from each of the mission sensor packages to the Special Sensor Processing Unit (SPU) of the OLS.

3.3.1.1.4 OLS signal ground. The OLS shall provide signal grounds for the mission sensor data interface. These OLS provided signal grounds shall be electrically connected within the OLS at the OLS single point ground.

The OLS provided signal grounds shall be isolated by each mission sensor from the mission sensor's chassis, telemetry/spacecraft signal return, and power return by a resistance of at least 20 kilohms shunted by a capacitance of no more than 0.10 microfarads. This isolation shall be maintained by each mission sensor whether powered ON or OFF.

The sum of the currents generated by any mission sensor on the OLS provided signal ground lines shall not exceed 30 milliamperes DC. The peak to peak AC component of this current shall not exceed 15 milliamperes for a total current not to exceed 37.5 milliamperes.

The voltage difference between any two signal ground lines on a single connector that a mission sensor can expect to see shall not be greater than 50 millivolts peak to peak as measured at the OLS SPU connector.

The resistance in the OLS between any of the OLS provided signal grounds on any one mission sensor connector shall not exceed 0.5 ohms as measured at the OLS SPU connector. The resistance between any of these signal ground lines at the OLS SPU connector and the OLS single point ground shall not exceed 1.0 ohms.

The maximum common mode voltage present on the OLS generated signals and the OLS provided signal grounds with respect to the OLS single point ground shall not be greater than ± 1.0 volts.

3.3.1.1.5 Signal characteristics. The mission sensor data signal level is defined as the voltage differential between the signal line and the associated OLS provided signal ground of paragraph 3.3.1.1.4. Additional data signal characteristics are shown in Figure 13. Unless otherwise specified, each SSP data signal shall comply with the characteristics of Figure 13 when operating into the test circuit shown in Figure 22. When the data signal is at the low voltage level, a SSP shall be capable of accepting at least 1.0 milliampere of current from a positive voltage source in the OLS.

3.3.1.2 SSP ON-OFF indicator. Each mission sensor may generate an OLS mission sensor SSxOFF, SSP ON-OFF indicator, interface signal to indicate the transfer status of its data on the SSxDAT signal line. If the mission sensor data on the SSxDAT line is ready to be transferred to the OLS, the mission sensor shall pull the SSxOFF interface signal low (0V nominal) so that the OLS will generate the SSxRED read gate and the SSxBCK bit clock to transfer the data. If the mission sensor data on the SSxDAT line is not ready to be transferred, then the mission sensor will allow the OLS to pull the SSxOFF signal high (3-5V nominal) resulting in both the SSxRED and SSxBCK not being generated for the data transfer. The SSUSI SSxOFF signal remains low when the unit is powered off.

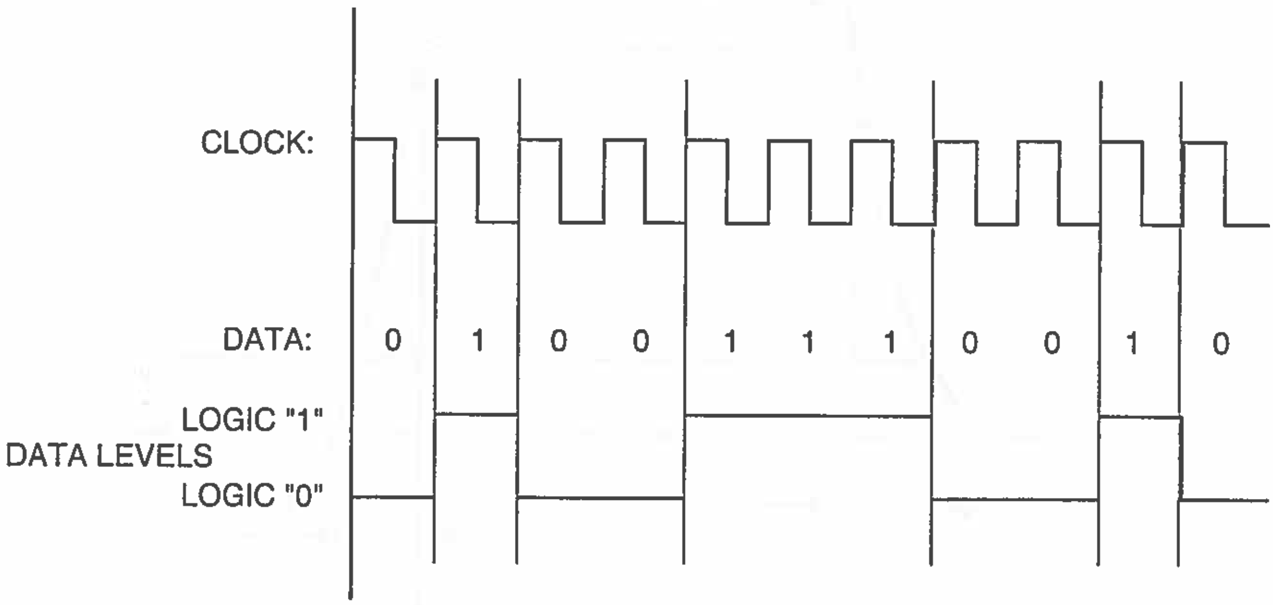
The OLS will format the mission sensor data upon transfer. If the SSxOFF signal remains high however, the OLS will substitute OLS mission sensor OFF Code in place of the mission sensor's data in the mission data format. The OFF Code is a 36 bit word consisting of 35 zeros and a one as defined in the IS-YD-821B data specification. If the mission sensor elects not to use the SSxOFF signal in the manner described above, the mission sensor may permanently tie the SSxOFF signal to SSxOFF Return (OLS signal ground) indicating to the OLS that it should always transfer data from the sensor, at the predetermined one second intervals, even when the sensor is turned off. The OLS will format into the mission sensor data stream whatever the state of the SSxDAT line is at data transfer and the OFF Code will never be inserted into the mission sensor data format. The SSMIS has elected to permanently tie its SSxOFF to OLS signal ground.

The SSF SSxOFF signal is actively controlled by SSF as described above for data transfer. The SSF has both a primary and a redundant OLS interface; therefore one of the two interfaces must be selected for the SSF to actively control the SSxOFF signal for data transfer.

3.3.1.2.1 Data routing. The spacecraft contractor provides the cable harness for each mission sensor. An ON/OFF signal from each mission sensor will be routed to the SPU of the OLS through this harness.

3.3.1.2.2 OLS signal ground. The OLS shall provide a signal ground for the mission sensor ON/OFF Signal. For a description of the OLS provided signal ground refer to paragraph 3.3.1.1.4.

3.3.1.2.3 Signal characteristics. The SSxOFF signal level is defined as the voltage differential between the signal line and the associated OLS provided signal ground of paragraph 3.3.1.2.2. Additional data signal characteristics are shown in Figure 15. Unless otherwise specified, each SSxOFF signal shall comply with the characteristics of Figure 15 when shunted to OLS provided signal ground by an external capacitor of 6800 picofarads at the SSP output. When the data signal is at the low voltage level, a SSP shall be capable of accepting at least 1.0 milliampere of current from a positive source in the OLS.



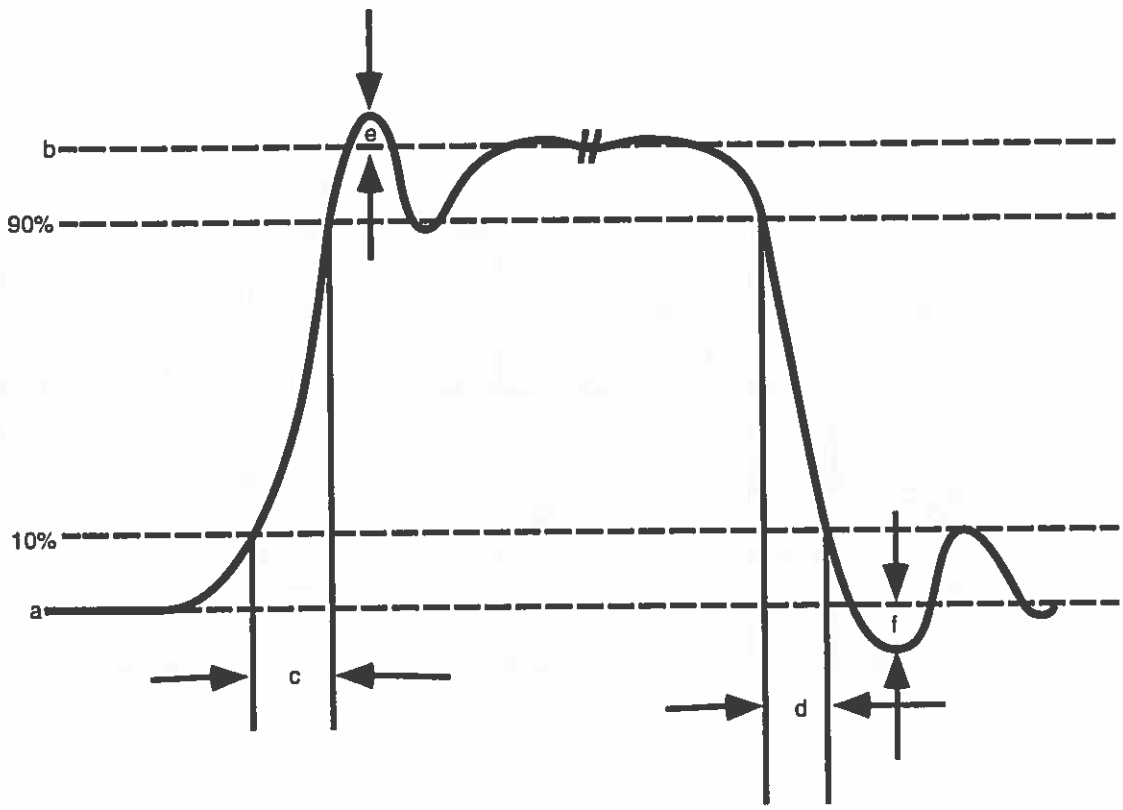
NRZ-L = Non Return to Zero Level (Level changes with clock only when data state changes)

"ONE" = "1" is represented by high level

"ZERO" = "0" is represented by low level

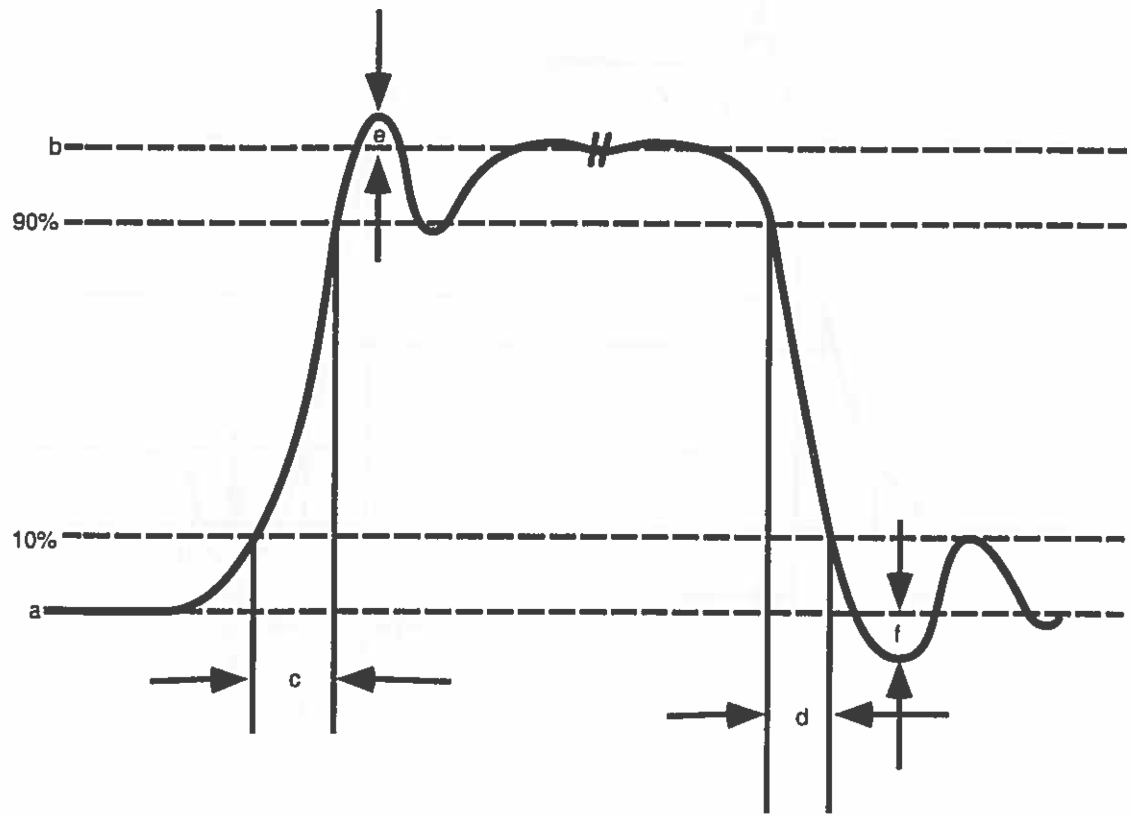
LSB first shifted out of SSP

Figure 12. Digital NRZ-L Data Definition



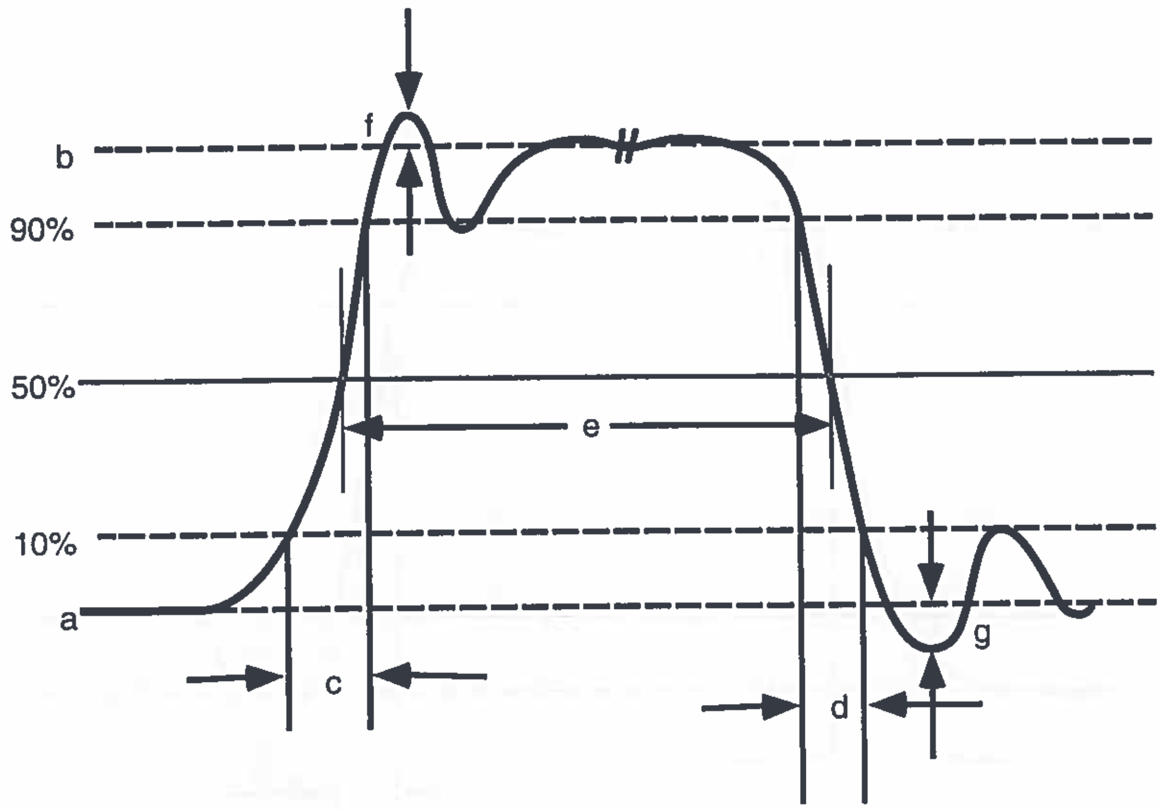
a. Low Level Nominal - Logic "0"	$0.0^{+0.5}_{-0.0}$ Volts
b. High Level Nominal - Logic "1"	$5.0^{+0.5}_{-1.0}$ Volts
c. Rise Time (10% to 90 %)	$t_r \leq 5.5$ μsec
d. Fall Time (90% to 10%)	$t_f \leq 4$ μsec
e. Overshoot Leading Edge	≤ 0.5 Volts
f. Overshoot Trailing Edge	≤ 0.6 Volts

Figure 13. SSP Data (SSxDAT) Waveform Characteristics



a. Low Level Nominal	$0.0^{+0.5}_{-0.0}$ Volts
b. High Level Nominal	$5.0^{+0.5}_{-1.0}$ Volts
c. Rise Time (10% to 90%)	$t_r \leq 5.5$ µsec
d. Fall Time (90% to10%)	$t_f \leq 4$ µsec
e. Overshoot Leading Edge	≤ 0.5 Volts
f. Overshoot Trailing Edge	≤ 0.5 Volts

Figure 14. Waveform Characteristics(SSxREF, SSxBCK, SSxRED, SSxSER, SSxENB)



a.	Low Level Nominal -	$0.0^{+0.5}_{-1.0}$ Volts
b.	High Level Nominal -	$5.0^{+0.5}_{-0.0}$ Volts
c.	Rise Time (10% to 90%)	$t_r \leq 110$ µsec
d.	Fall Time (90% to 10%)	$t_f \leq 80$ µsec
e.	Pulse Width (50% to 50%)	500 ± 50 µsec (SSxMD1 and SSxMD2)
f.	Overshoot Leading Edge	≤0.5 Volts
g.	Overshoot Trailing Edge	≤0.5 Volts

*Note: Only Low Level Nominal requirement is applicable to SSMIS sensor SSdOFF

Figure 15. Waveform Characteristics (SSxPEN, SSxOFF*, SSxMD1, SSxMD2)

3.3.2 OLS generated control signals

3.3.2.1 Power enable. The OLS shall provide a power enable signal designed at SSxPEN to each mission sensor. A high state (+5 volt) shall command the mission sensor to an "ON" condition and a low state (0 volt) shall command the mission sensor to an "OFF" state. These states were chosen so that when the OLS is OFF the mission sensors will also be in an OFF state. The control signal characteristics are listed in Table VIII. The waveform characteristics are shown in Figure 15 when shunted to ground by an external capacitor of 7600 picofarads. The relative timing of the SSxPEN is shown in Figure 16. The SSxPEN is routed to each sensor through the spacecraft harness along with its signal return line SSxPEN RET. This signal return line shall be isolated from all other grounds inside the mission sensor other than OLS signal return lines by 20 kilohms and by no more than 0.10 microfarads. This isolation shall be maintained by each mission sensor whether powered ON or OFF.

Table VIII. Control Signal Characteristics (SSxPEN, SSxENB, SSxSER)

	Low Level	High Level
Voltage Level	0.0 +0.5 -0.0V	5.0 +0.5 -1.0V
Logic Sense	FALSE OFF or "0"	TRUE ON or "1"
Output Impedance	≤2.5k ohms	2.5-4.0k ohms
Signal Current	The source shall be capable of accepting at least 160 μA from the load	The source shall be capable of supplying at least 100 μA to the load

3.3.2.2 Read gate. The OLS shall provide a Read Gate to each mission sensor designated as SSxRED whenever the mission sensor's SSxOFF signal indicates that the mission sensor is ON and ready to send data. This signal shall normally be OFF and shall be pulsed ON during the interval in which the mission sensor data is to be read into the OLS. Each read gate shall be a negative-going pulse with a normal duty cycle of X divided by 25.6 kHz (X being the number of bits assigned to the sensor). SSxRED for any sensor shall not deviate from that described above during or as a result of the OLS Elapsed Time Clock (ETC) update that occurs once daily. The control signal characteristics are listed in Table IX. The waveform characteristics are shown in Figure 14 when shunted by an external capacitor of 470 picofarads to signal return. The relative timing of SSxRED is shown in Figure 16 and Figure 17.

Each read gate shall have a repetition period of 1 second plus or minus 2 milliseconds of jitter and shall change level at the positive going edge of the bit clock. The positive-going and negative-going edges of this pulse shall be coincident with a positive-going edge of the mission sensor reference clock and bit clock to within 10 microseconds as shown in Figure 17.

The SSxRED is routed to each sensor through the spacecraft harness along with its signal return line SSxRED RET. This signal return line shall be isolated from all other mission sensor grounds inside the mission sensor other than the OLS signal return lines by at least 20 kilohms and by no more than 0.10 microfarads.

Table IX. Control Signal Characteristics (SSxRED, SSxBCK, SSxRED, SSxMD1, SSxMD2)

	Low Level	High Level
Voltage Level	0.0 +0.5 -0.0V	5.0 +0.5 -1.0V
Logic Sense	TRUE ON or "1"	FALSE OFF or "0"
Output Impedance	≤2.5k ohms	2.5-4.0k ohms
Signal Current	The source shall be capable of accepting at least 160 µA from the load	The source shall be capable of supplying at least 100 µA to the load

3.3.2.3 Bit clock. The OLS shall provide a gated clock designated as SSxBCK for transferring data from each of the mission sensors. The clock shall be used to provide a transfer of mission sensor data to the OLS when the read gates SSxRED is in its ON state. The bit clock shall be a nominal square wave with a fundamental frequency of 25.6 kHz. The duty cycle of the clock shall be 0.50 ± 0.05. The signal characteristics are given in Table IX. The waveform characteristics are shown in Figure 14 when shunted by an external capacitor of 470 picofarads. The relative timing of SSxBCK is shown in Figure 16 and 17. The SSxBCK is routed to each sensor through the spacecraft harness along with its signal return line SSxBCK RET. The signal return line shall be isolated from all other mission sensor grounds inside the mission sensor other than OLS signal return lines by at least 20 kilohms and no more than 0.10 microfarads. This isolation shall be maintained by each mission sensor whether powered ON or OFF.

3.3.2.4 Reference clock. The OLS shall provide a reference clock designated as SSxREF to each of the mission sensors. This clock may serve as the basic timing reference for operation of the mission sensors. This clock shall be gated with the SSxPEN signal. The reference clock shall be a nominal square wave with a fundamental frequency of 25.6 kHz ±1 Hz. The duty cycle of the clock shall be 0.50 ± 0.05. The control signal characteristics are listed in Table IX. The waveform characteristics are shown in Figure 14. The relative timing of SSxREF is shown in Figures 16 and 17.

The SSxREF signal is routed to each sensor through the spacecraft harness along with its signal return line SSxREF RET. This signal return line shall be isolated from all other mission sensor grounds inside the mission sensor other than OLS signal return lines by at least 20 kilohms and no more than 0.10 microfarads. This isolation shall be maintained by each mission sensor whether powered ON or OFF.

3.3.2.5 Mode 1 select. The OLS shall provide a mode select signal designated SSxMD1 to each of the mission sensors. The mode select signal can be either a level or a pulse depending on mission sensor requirements. The level type Mode 1 select line has two states. The control signal characteristics are listed in Table IX. The waveform characteristics are as shown in Figure 15 when shunted by an external capacitor of 7600 picofarads. The relative timing is shown in Figure 16.

The pulse type Mode 1 select line has control signal characteristics identical to the level type Mode 1 select. The waveform of the pulse has the same characteristics as shown in Figure 16. However, instead of being a level change the SSxMD1 line shall be a negative going pulse of 500 microseconds ± 50 microseconds as measured at the 50 percent points. When the SSxMD1 is in its quiescent state (+5 volt) the mission sensor will remain in its current operating mode. When SSxMD1 is pulsed negative (0 volt), the mission sensor will respond to the command to advance to its next operating mode.

The turn on state of both the level and the pulse type signals is +5 volts. Relative timing of the SSxMD1 is shown in Figure 16. The SSxMD1 signal is routed to each mission sensor through the spacecraft harness along with its return line SSxMD1 RET. This signal return line shall be isolated from other sensor grounds inside the mission sensor other than the OLS signal return lines by at least 20 kilohms and no more than 0.10 microfarads. This isolation shall be maintained by each mission sensor whether powered ON or OFF.

3.3.2.6 Mode 2 select. The OLS shall provide a mode select signal designated SSxMD2 to each of the mission sensors. The SSxMD2 signal is identical in operation to, but independent of, the SSxMD1 signal. See paragraph 3.3.2.5 for a description of the SSxMD1 signal characteristics.

3.3.2.7 Serial command enable. The OLS shall provide a serial command enable line designated SSxENB to be used in conjunction with the SSxREF signal (paragraph 3.3.2.4) and the SSxSER signal (paragraph 3.3.2.8) to transfer an eight bit serial command to a mission sensor.

The control signal characteristics of SSxENB are listed in Table VIII. The waveform characteristics are shown in Figure 14 when shunted by an external capacitor of 470 picofarads. The relative timing of SSxENB is shown in Figure 18. The SSxENB is routed to each mission sensor through the spacecraft harness along with its return signal SSxENB RET. The SSxENB RET shall be isolated from all other mission sensor grounds inside the mission sensor other than OLS signal returns by at least 20 kilohms and not more than 0.10 microfarads. This isolation shall be maintained by each mission sensor whether powered ON or OFF.

3.3.2.8 Serial command word. The OLS shall provide a serial command word designated as SSxSER is used in conjunction with the SSxREF signal (paragraph 3.3.2.4) and the SSxENB signal (paragraph 3.3.2.7) to transfer its contents in the form of a serial command to each mission sensor.

The contents of the eight bit command can be any combination of ones and zeros specified. The control signal characteristics are specified in Table VIII. The waveform characteristics are specified in Figure 14 when shunted by an external capacitor of 470 picofarads. The relative timing of SSxSER and its logic ONE and ZERO states are shown in Figure 18. The SSxSER is an NRZ-L signal that should be transferred by the mission sensor utilizing the SSxREF signal and the SSxENB signal for the transfer. The SSxSER is routed to each mission sensor along with its signal return SSxSER RET. The SSxSER RET shall be isolated from all other mission sensor grounds inside the mission sensor other than OLS signal returns by at least 20 kilohms and not more than 0.10 microfarads. This isolation shall be maintained by each mission sensor whether powered ON or OFF.

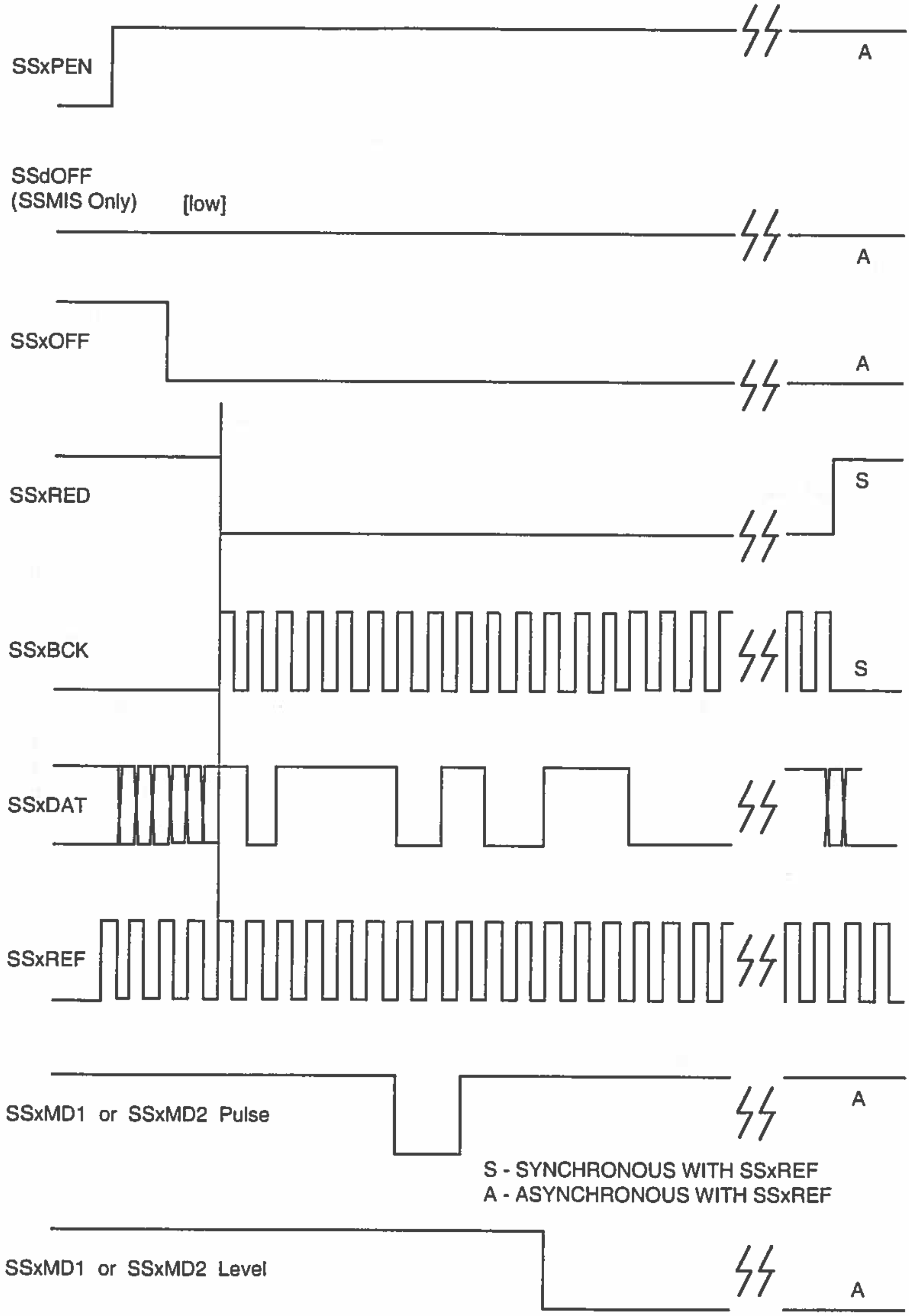


Figure 16. Timing Diagram

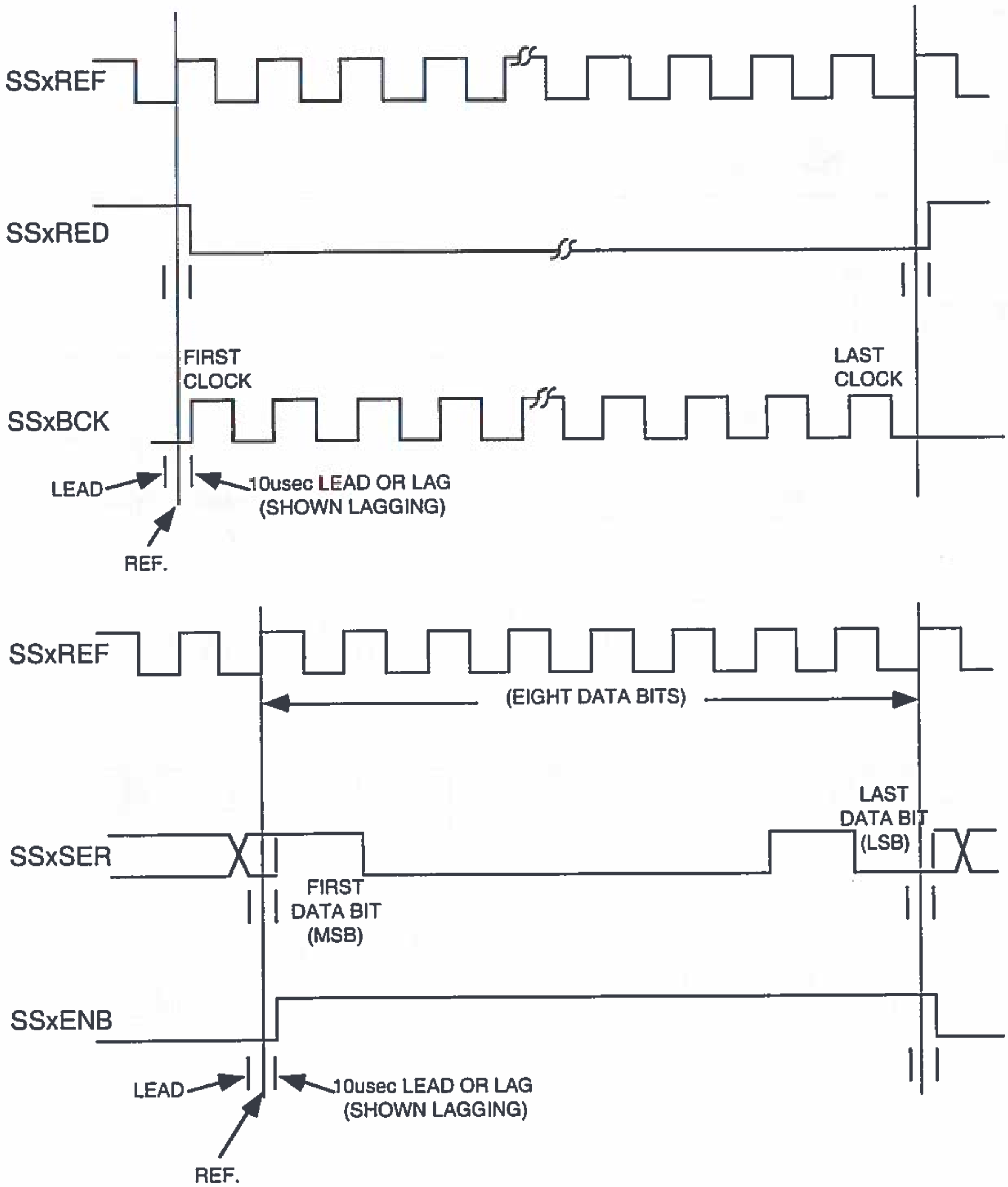
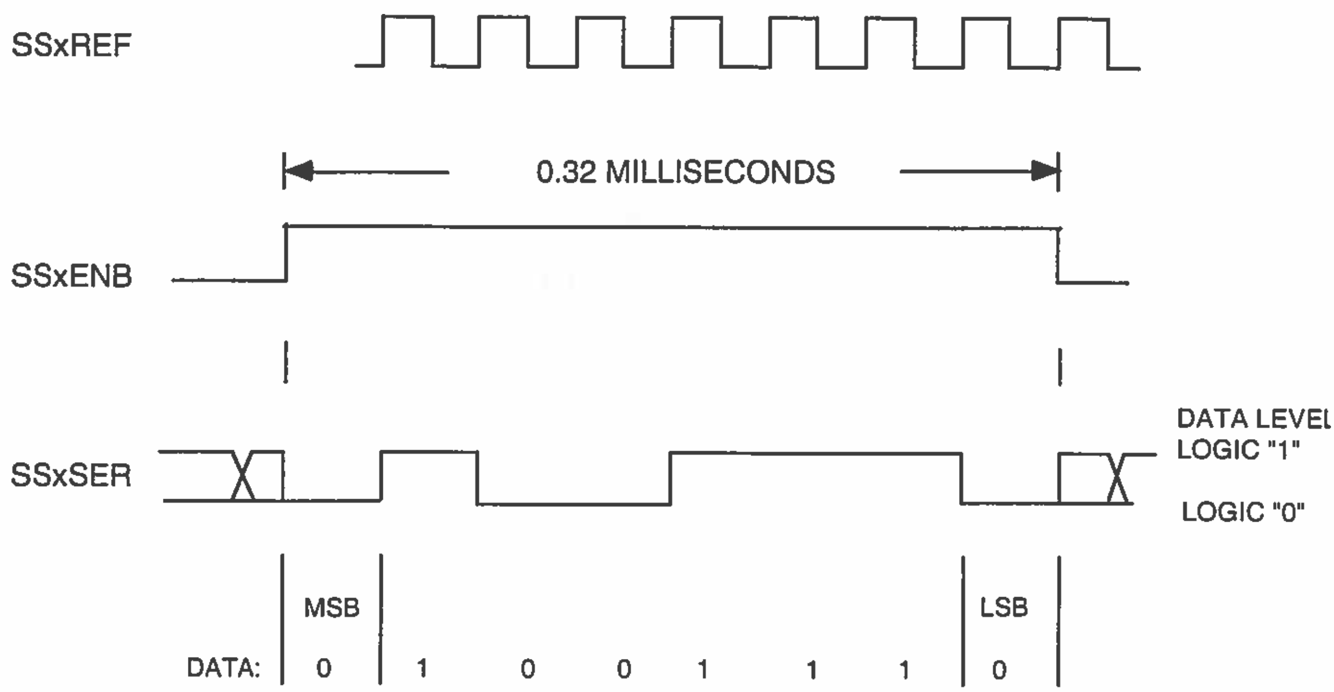


Figure 17. OLS Signal Phase Relations



NRZ-L = Non return to zero level (level changes with clock only when data state changes)

"ONE" = "1" is represented by high level

"ZERO" = "0" is represented by low level

MSB first out

Figure 18. Serial Command Timing

3.3.3 Block transfer. Each OLS, with operational software capability, can provide block transfers of 16 bit words to a mission sensor. The block transfer is via the SSxENB and SSxSER 8 bit serial command interface. The 16 bits are transferred in two 8 bit bytes. The number of 16 bit words transferred as a block can be specified as any number between one and 32,767 inclusive.

3.3.3.1 Block format. For a block transfer of 16 bit words, a fixed format must be used to standardize the process for the OLS and ground system. The format to satisfy the OLS requirements is shown in Figure 19. The format consists of a Control Header, Data and a Block Checksum all of which is transferred to the specified mission sensor.

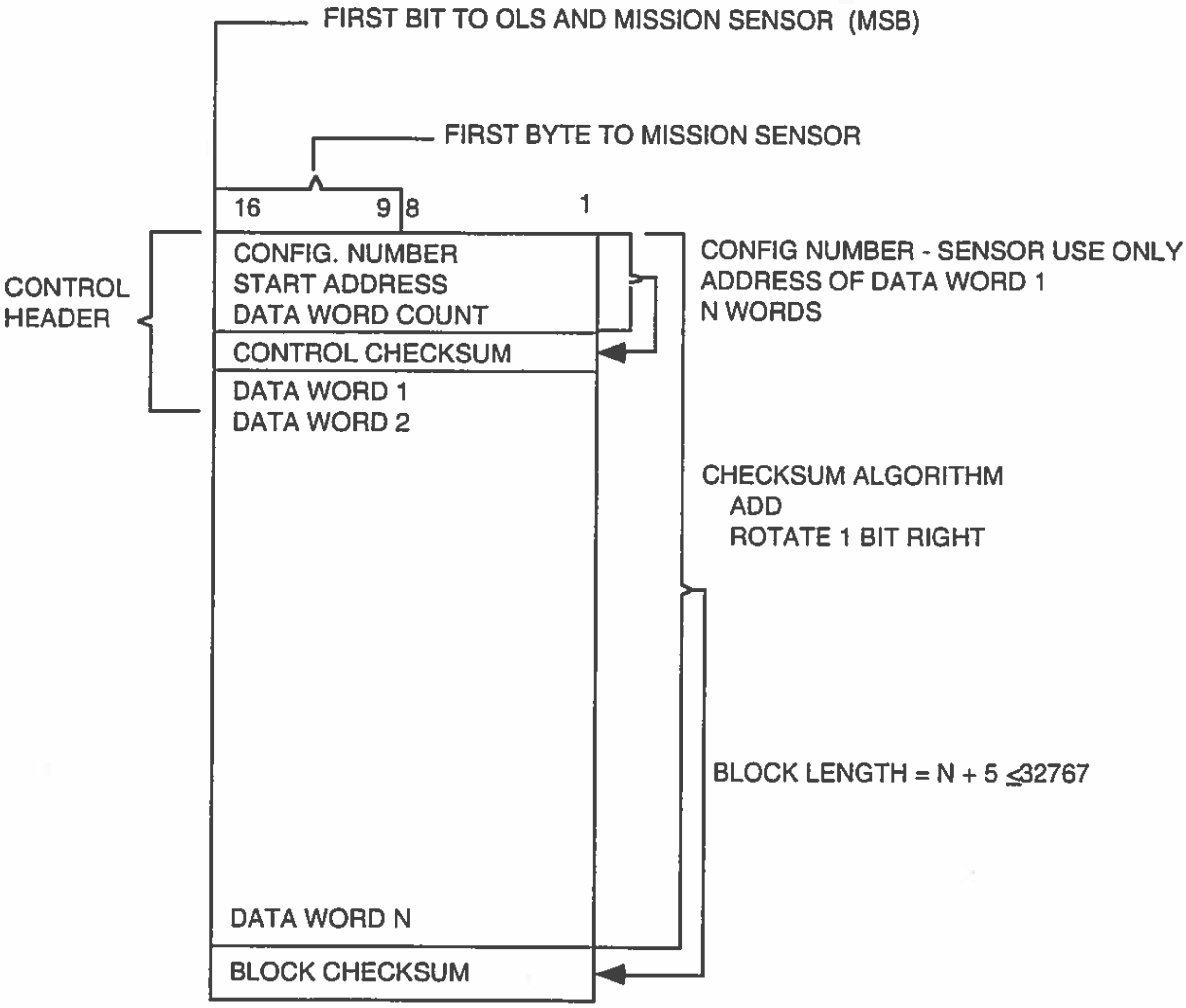
3.3.3.1.1 Control header. The Control Header consists of three control words and a checksum for those three words. The Configuration Number is for use by the mission sensor and will be defined in the mission sensor's ICD referenced in Appendix A. The Starting Address defines a memory word address for the first Data Word of the Block where Data Word 1 is destined to memory "START ADDRESS" and Data Word i is destined to memory "START ADDRESS + i-1." The Data Word Count must specify the number of Data Words in the Block. The checksum algorithm for the Control Header is "Add and Rotate 1 Bit Right".

3.3.3.1.2 Data block. The Data Block consists of 16 bit words. The maximum number of 16 bit words per data block is 32,762 (32,767 less 5 overhead words). The five accounts for the three control words and the two checksums in the Block. The Data Words, if they need further definition, are defined in the mission sensor's ICD referenced in Appendix A.

3.3.3.1.3 Block checksum. The Block Checksum is for the entire Block consisting of the Control Header and Data words. The Block Checksum's algorithm, like the Control Checksum is "Add and Rotate 1 Bit Right".

3.3.3.2 Load timing. The 16 bit words are transferred to the mission sensor as two 8-bit bytes. Leading edges of successive SSxENB gates shall always be separated by at least 0.4 milliseconds. A 16-bit word (2 bytes) shall be transferred at a maximum rate of 1 word every 2.5 milliseconds.

3.3.3.3 Protocol. The protocol, between the OLS and the mission sensor receiving the block transfer, may be tailored for that particular sensor provided that the hardware interface does not change from that specified in this interface specification. The mission sensor may make use of the SSxMD1, SSxMD2 and SSxSER interfaces to ready it for a block transfer, terminate a block transfer, or initialize it after the transfer has taken place. These particulars will appear in the applicable mission sensor ICD listed in Appendix A.



ENTIRE BLOCK TRANSMITTED TO MISSION SENSOR.
OLS RECEIVES BLOCK AS A DATA LOAD (NON-AUTHENTICATED).

Figure 19. Load Block Format

3.3.3.4 Load verification

3.3.3.4.1 OLS originated. The OLS will output unique error codes in its telemetry for the following condition that would prevent a successful block transfer to a mission sensor:

a. OLS computer checksum not equal to the Block Checksum.

If the above error is detected, the OLS will continue to transfer the block load to the mission sensor and place the error code in the OLS System Error Table.

3.3.3.4.2 Mission sensor originated. If the mission sensor utilizes the block load feature, the mission sensor shall provide an EST for the purpose of block load verification. It is recommended that mission sensors have the capability of dumping their memory contents to the ground through the SSxDAT interface upon command. The mission sensor's verification mechanism will be specified in the applicable ICD listed in Appendix A.

3.3.3.5 Load file transmittal. The load file transmittal between contractor's ground facilities shall be an ASCII file consisting of a 512 byte header followed by one or more blocks as described in Section 3.3.3.1. The load blocks shall consist of ASCII hex. The load file header shall include:

a. Program name
b. Revision letter
c. Flight number
d. Sensor name
e. Sensor ID
f. Sensor serial number
g. Number of load block in file
h. File record size
i. Comments

3.3.4 OLS electrical interface circuits. The OLS driver interface circuit shall be as shown in Figure 20. The OLS receiver interface circuit shall be as shown in Figure 21. The SSxDAT test circuit is shown in Figure 22.

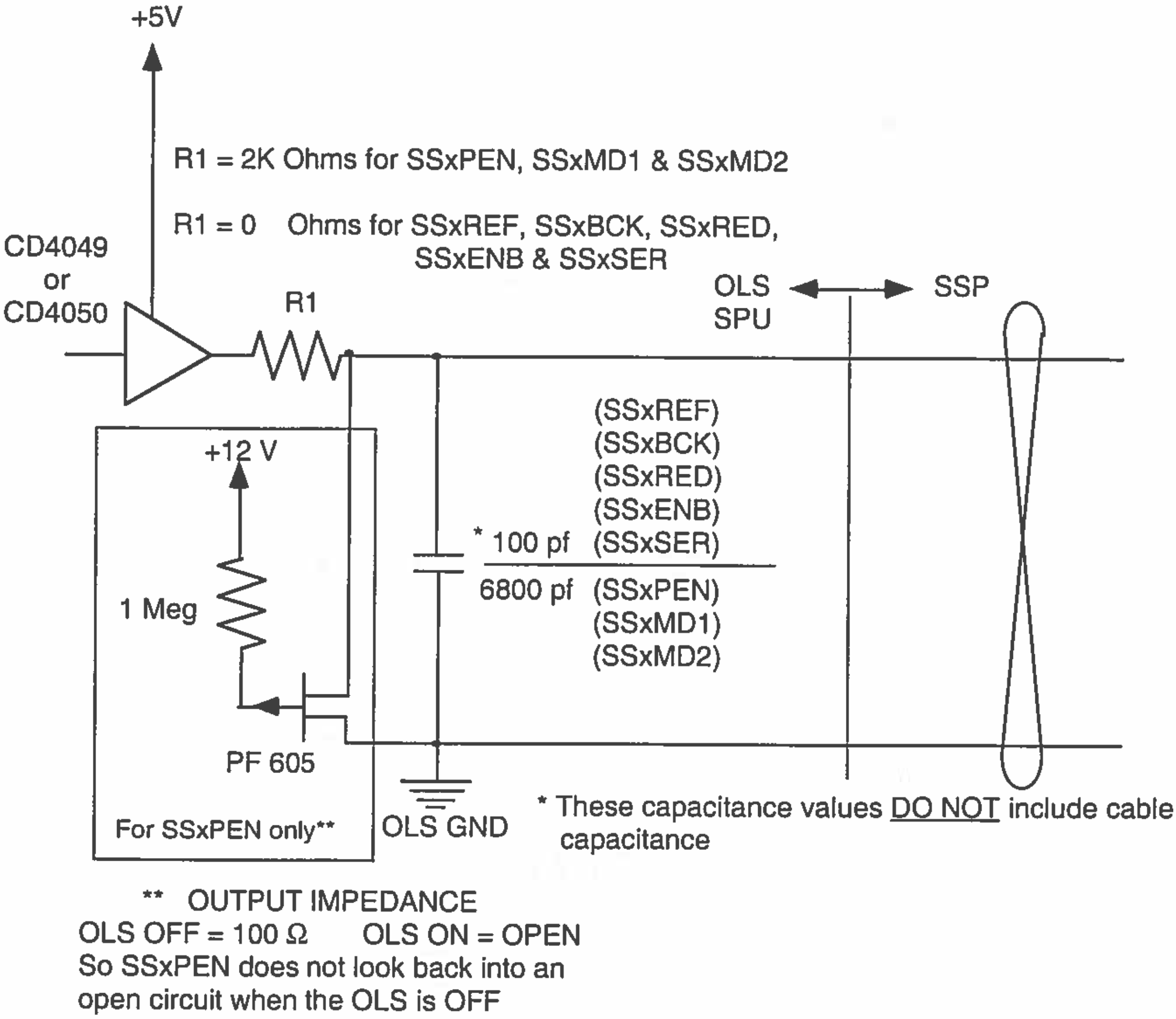


Figure 20. OLS Driver Interface Circuit

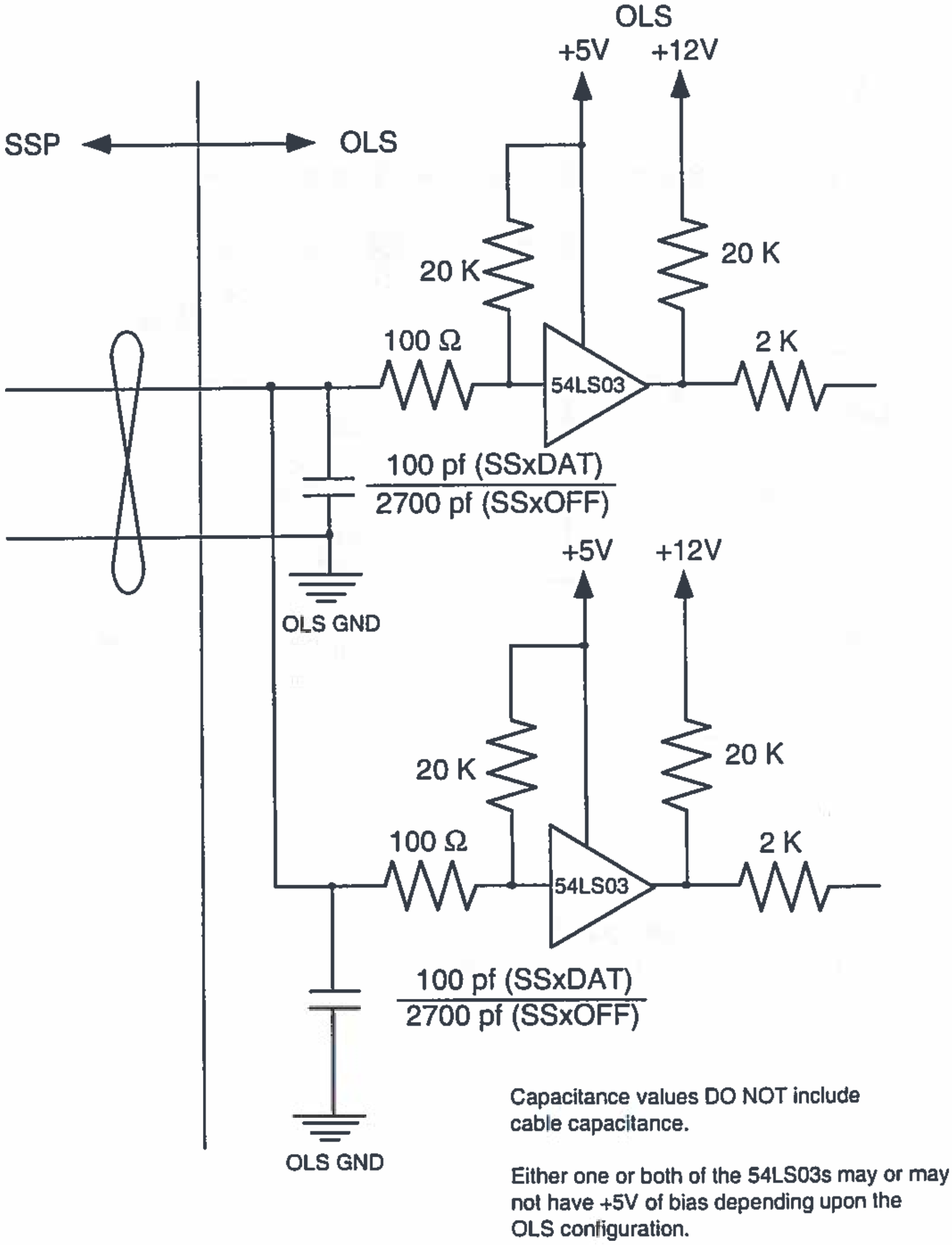
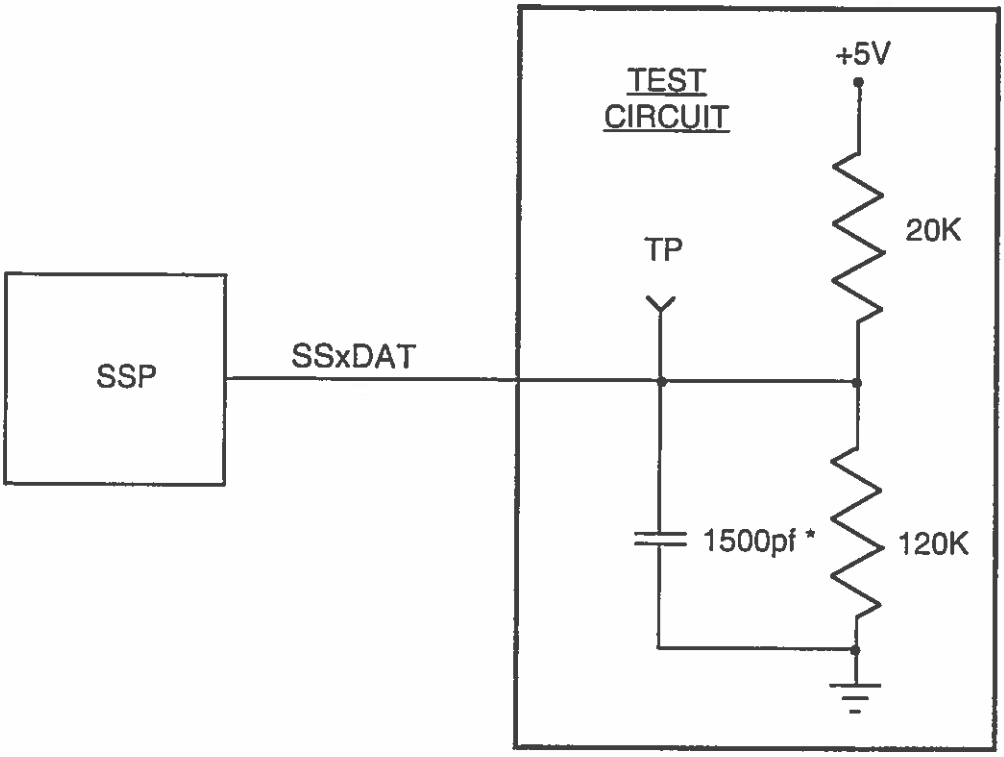


Figure 21. OLS Receiver Interface Circuit



(1) * 1500 picofarads represents the maximum expected capacitance on the SSxDAT signal line. It includes cable capacitance and OLS interface circuit capacitance.

(2) When both sides of the OLS (Normal and Redundant) are powered, the SSP must be able to sink at least 1 milliamperes and still meet the logic "0" level of 0 volts $^{+0.5V}_{-0.0V}$

Figure 22. SSxDAT Test Circuit

3.3.5 <u>SSP cabling and pin assignments.</u> All electrical connections between the OLS and each SSP shall be made by a connector mounted on each SSP. The spacecraft will supply all harnesses required between the OLS and each SSP for SSP data routing, OLS generated control signals, and SSP ON-OFF indicators. Each OLS cable connector shall be a DBM-25S (male on the SSP). Pin assignments shall be as shown in Table X.

Each mission sensor signal to/from the OLS will be twisted with its return. The cable bundle will have one outer shield which will be tied at the OLS end to the SPU chassis.

Table X. SSP Pin Assignments - Standard Interface

Pin	Function	Pin	Function
1	Spare	14	Spare
2	Data	15	On-Off Indicator
3	Data Return	16	On-Off Indicator Return
4	Bit Clock	17	Power Enable
5	Bit Clock Return	18	Power Enable Return
6	Read Gate	19	Mode 1 Select
7	Read Gate Return	20	Mode 1 Select Return
8	Reference Clock	21	Mode 2 Select
9	Reference Clock Return	22	Mode 2 Select Return
10	Serial Command Enable	23	Spare
11	Serial Command Enable Return	24	Spare
12	Serial Command	25	Spare
13	Serial Command Return		

3.3.6 <u>OLS/mission sensor information annex.</u> This section provides basic information on the SSP-OLS data interface.

3.3.6.1 <u>Total SSP data rate and distribution.</u> The OLS has the capability of formatting SSP data within the light smooth (LS) line format or within the thermal smooth (TS) line format. The OLS does not have the capability of formatting the data from one SSP in both LS and TS simultaneously. In the SDS/RDS mode, the LS data line will contain a minimum of 11,124 bits per message mission sensor data and the TS data line will contain a minimum of 12,852 bits per message mission sensor data. In the SDS/RDS mode, the total minimum data rate of 23,976 bits per message is for all mission sensors on a given mission. In the Real Time Data mode (RTD) the total minimum data rate for all mission sensors on a given mission is 24,192 bits per message.

3.3.6.2 Double polling of sensors. The OLS has the capability of formatting X data bits of a SSP in LS and Y bits from the same SSP in TS where Y data bits contiguously follow X data bits and the SSP is addressed twice, The latter capability allows a high data rate SSP to output its data part in LS and the rest in TS. The period between the two consecutive samples should not be less than the period of the OLS bit clock and not greater than the OLS polling period. Thus the SSP would "appear" to be two sensors to the OLS.

3.3.6.3 SSP control signals. Each SSP receives one set of control signals as outlined in this document.

3.3.6.4 SSP data off/invalid signal. The OLS will respond to data off (SSxOFF) by substituting for the SSP data a transition code equal to the bits per second of the original data. The transition code for each 36 bit SSP word is 35 "0" fill bits with a "1" in the LSB position to meet transition density requirements. The OLS will ignore any data on the SSxDAT interface line under these conditions.

3.3.6.5 Time code. The SSP message is formatted once per second by the OLS and contains a 27 bit time code which is the 27 bits of the ETC clock time coincident with the change of state (off to on) of the first read gate to the first SSP (the first SSP whose data is to be included on the LS primary line). The LSB of the time code is 2^{-10} seconds. This ETC clock is the same ETC clock used in the primary data for time coincidence of the OLS scan crossing at nadir. Since the SSP receives a read gate every 1 second $\pm$ 2 milliseconds, and each SSP read gate has a known time delay from the first read gate to the first SSP interrogated for data, the time of receipt of each SSP read gate can be computed and should be constant within approximately 5 milliseconds. Therefore, a SSP can time reference the read gate and establish a relationship between its data samples and the ETC clock (which references orbit position).

3.4 Spacecraft/OLS and spacecraft/mission sensors environmental interface

3.4.1 Thermal

3.4.1.1 General. The spacecraft contractor shall provide thermal control, as required, for each sensor unit for the life of the spacecraft for all sun angles between 0 and 95 degrees (sun angle is defined as the angle between the sun vector and the normal to the spacecraft orbital plane) and for all sensor power dissipations from sensor off-state to sensor on-state. Detailed requirements for sensor power dissipation are defined in each spacecraft/sensor ICD listed in Appendix A. Provisions may be made to terminate thermal control of a sensor which has been declared by the AF to have failed, if such termination results in enhanced system performance. Specific temperature control requirements for each sensor are defined in the applicable sensor/spacecraft ICD listed in Appendix A.

Subject to the field-of-view requirements of this specification, and provided all other requirements of this specification are met, the spacecraft contractor may attach thermal blankets, fins, heaters, or other passive thermal control devices to the sensors. Such attachments shall be defined in the applicable sensor/spacecraft ICD listed in Appendix A.

3.4.1.2 Nodal models. Detailed nodal models may be required for thermal analysis. The need for nodal models, and their description when needed, shall be as defined in each sensor/spacecraft ICD listed in Appendix A.

3.4.2 Contamination. The spacecraft contractor shall provide a design and equipment arrangement which minimizes contamination of the sensor units. Spacecraft and sensors shall be designed to minimize contamination of each other. Outgassing of the propulsion subsystem onto sensors and other sensitive equipment shall be minimized by shielding, protective covers, orientation and arrangement of components, or other means as necessary to meet mission requirements.

3.4.3 Spacecraft test environments. The satellite undergoes testing as described in Section 6. Critical interfaces during these tests are defined in each spacecraft/sensor ICD listed in Appendix A.

3.4.4 Facilities environment. The satellite is subjected to the facilities environments as described in Section 6.

3.4.5 Flight environments. The satellite is subjected to the flight environments as described in Section 6. Critical interfaces during this environment will be defined in the applicable spacecraft/sensor ICD listed in Appendix A.

4.0 QUALITY ASSURANCE PROVISIONS

4.1 General. A Verification Matrix is included in each ICD listed in Appendix A to serve two functions:

a. to provide a basis for establishing that all requirements pertinent to this interface have been addressed.

b. to provide a responsibility matrix to define which of the several organizations participating in the particular interface has responsibility for the performance of the analysis, test, and/or inspection necessary to complete verification of compliance to the requirement.

Methods of verification are defined below:

INSPECTION	(I)	verification is based on visual inspection of such areas as form, fit, and configuration of hardware or of computer programs.
ANALYSIS	(A)	verification is based on studies, calculations, and/or analytical modeling.
TEST	(T)	verification is based on instrumented functional operation and evaluation techniques.

4.2 Test responsibilities. Each ICD listed in Appendix A details the responsibilities for verification of specification compliance. General responsibilities are given below.

4.2.1 Spacecraft contractor. The spacecraft contractor is responsible for:

a. Preparing test plans and procedures for testing of the integrated spacecraft.

b. Performing the testing of the integrated Satellite in accordance with documented test procedures.

c. Providing support, as required, for sensor testing performed by the OLS contractor, and the mission sensor contractor.

4.2.2 OLS contractor. The OLS contractor is responsible for:

a. Providing detailed test procedures and evaluation criteria for testing the OLS to the prime item specification.

b. Performing OLS testing.

c. Performing sensor testing at the spacecraft level, with inputs from the sensor contractor.

d. Providing support for the acceptance and launch test activities.

e. Providing all unique OLS AGE.

4.2.3 Sensor contractor. As applicable to the sensor provided, each sensor contractor is responsible for:

a. Providing detailed test procedures and evaluation criteria for testing the sensor to the prime item specification.

b. Performing sensor testing at the sensor system level.

c. Providing inputs to the OLS contractor to test the sensor at the spacecraft level.

d. Providing support for the acceptance and launch test activities.

e. Providing all unique sensor AGE.

5.0 PREPARATION FOR DELIVERY

This section is not applicable to this specification.

6.0 NOTES

6.1 Organizations. Under the coordination of the identified Space Segment Integrator, who has prime responsibility for the interfaces detailed in Section 1.1, each contractor listed below has collateral responsibility for compliance with the interface requirements, which are applicable to the particular contractor, contained in this specification. The specific responsibilities of each organization, for fulfilling the requirements of this IS, are detailed in Section 4 of this IS and in the applicable ICDs listed in Appendix A.

Space Segment Integrator	Lockheed Martin Missiles and Space P.O. Box 800 Princeton, NJ 08543-0800
Spacecraft Contractor	Lockheed Martin Missiles and Space P.O. Box 800 Princeton, NJ 08543-0800
OLS Contractor	Northrop Grumman Corporation P.O. Box 17320 MS-B290 Baltimore, MD 21203-7320
Mission Sensor Contractors:	
SSF	Sandia National Laboratories Attn: P. Elder, MS 0965 P.O. Box 5800 Albuquerque, NM 87185-0965
SSIES3	Phillips Laboratory/GPSS 29 Randolph Road Hanscom Air Force Base, MA 01731-3010
	University of Texas at Dallas Center for Space Sciences M/S F022 P.O. Box 830688 2601 North Floyd Road Richardson, TX 75080
SSJ5	Phillips Laboratory/GPSS 29 Randolph Road Hanscom Air Force Base, MA 01731-3010
	Amptek, Inc. 6 De Angelo Drive Bedford, MA 01730
SSM	NASA/ GSFC Code 695 Bldg. 2 Room 132 Greenbelt, MD 20771

SSMIS	GenCorp Aerojet Aerojet Electronic Systems Plant 1100 W. Hollyvale Street P.O. Box 296 Azusa, CA 91702-0296
SSULI	Naval Research Laboratory Code 8212 4555 Overlook Avenue S.W. Washington, DC 20375
SSUSI	The Johns Hopkins University Applied Physics Laboratory Johns Hopkins Road Laurel, MD 20723-6099

6.2 Definitions and abbreviations. For the purpose of the spacecraft to sensors interface documentation, the following definitions and/or abbreviations shall apply. Other abbreviations are defined within the documents.

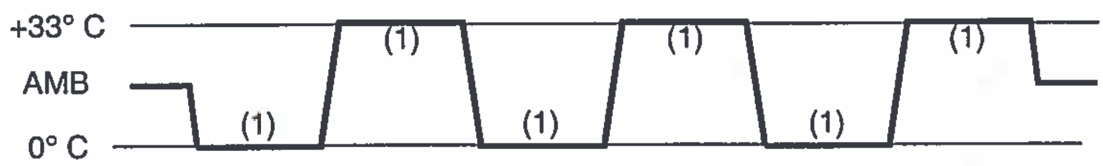
5D-3	-	Block 5D-3 DMSP Satellite
AGE	-	Aerospace Ground Equipment
BBx	-	GFE Encryption Device (BB1, BB2, BB3, BB4)
CCB	-	Configuration Control Board
CIU	-	Controls Interface Unit (S/C)
CPC	-	Controls Power Converter (S/C)
DCC	-	Deployable Cooler Cover (OLS)
DDT	-	Direct Data Transmitter (S/C)
DMSP	-	Defense Meteorological Satellite Program
DOC	-	Deployable Optics Cover (OLS)
EDTX	-	EST Data Transmitter (S/C)
EPH CLK	-	Ephemeris Clock
ESM	-	Equipment Support Module
EST	-	Equipment Status Telemetry
ETC	-	Elapsed Time Clock
FOV	-	Field-of-View
FTE	-	Field Test Equipment
GFE	-	Government Furnished Equipment
GLOB	-	Glare Obstructor
GSSA	-	Glare Suppression System Assembly (OLS)
ICD	-	Interface Control Document
ICN	-	Interface Change Notice
ICWG	-	Interface Control Working Group
ISS	-	Integrated Spacecraft System
IS	-	Interface Specification
LS	-	Light Smooth
LSB	-	Least Significant Bit
MSB	-	Most Significant Bit
NRZ-L	-	Non-Return-to-Zero-Level
OLS	-	Operational Linescan System
OSU	-	Output Switching Unit (OLS)
PC	-	Power Converter (S/C)
PDTX	-	Primary Data Transmitter (PDT1, PDT2) (S/C)
PIP	-	Programmable Information Processor (S/C)
PMO	-	Program Management Office (CWD)
PMP	-	Precision Mounting Platform
PRx	-	Primary Tape Recorder (PR1, PR2, PR3, PR4) (OLS)
PSE	-	Power Supply Electronics (S/C)
PSU	-	Power Supply Unit (OLS)
RDS	-	Real Time Data Smoothed Mode
REU	-	RDS Encryption Unit
RSS	-	Reaction Support Structure
RTD	-	Real-Time Data Mode - the direct data mode
S/C	-	Spacecraft
SCN	-	Specification Change Notice
SCU	-	Signal Conditioning Unit (S/C)
SDF	-	Stored Data Fine Mode - one of the OLS record modes
SDS	-	Stored Data Smooth Mode - one of the OLS record modes
SPO	-	System Program Office
SPS	-	Signal Processing Subsystem (OLS)

SPU	-	Special Sensor Processing Unit (OLS)
SSP	-	Special Sensor Unit (P is a general form) - also known as Mission Sensor Unit
SSS	-	Sensor Scanning Subsystem (OLS)
STE	-	Special Test Equipment (OLS)
TBD	-	To be determined
TCTR	-	Time Commutated Telemetry Readout
TIM	-	Technical Interchange Meeting
TML	-	Total Mass Loss
TS	-	Thermal Smooth
USAF	-	United States Air Force
VCM	-	Volatile Condensable Material
X,Y,Z	-	Reference axes of geodetic local vertical
X_p, Y_p, Z_p	-	Primary alignment reference axes - also alignment coordinate axes of SSS
X_r, Y_r, Z_r	-	Secondary references axes - also special sensor alignment coordinate axes
x_m, y_m, z_m	-	Mounting or mechanical axes

6.3. Test environments. The spacecraft test environments are described herein for informational purposes only. Sensor contractors shall review these environments and document with the DMSP SPO any conditions at variance with or considered potentially in excess of environments stated in the applicable sensor segment specification. Where applicable sensor segment specifications conflict with these environments, the sensor segment specifications shall take precedence. It shall be the responsibility of the DMSP SPO to resolve any such conflicts.

6.3.1 Thermal test. The satellite thermal test environment conditions at ambient pressure are shown in Figure 23. Functional verification of the spacecraft/sensor interface is performed at each temperature plateau during one or more cycles. The rate of temperature change will be 10°C/hour maximum. Tolerance on the temperature extremes is ±2°C.

The thermal chamber relative humidity will be 60% maximum. Temperature transitions shall be accomplished in a manner that prohibits condensation on any equipment under test.



NOTE: (1) = Active Test

Figure 23. Thermal Test Profile

6.3.2 Orbit confidence test. The satellite will be subjected to a vacuum of less than 10^{-5} Torr and programmed for a daily 14 orbit mission profile. System Performance Tests will be conducted in a vacuum before and after the orbit tests. The complete orbit confidence test, including the System Performance Tests, will consist of 21 days in vacuum. During the test, the satellite will be operated at nominal orbit temperature by controlling vacuum chamber walls to -70°C minimum. The satellite will be operated using simulated solar array power. Power load levels will be varied as in orbit by commanding satellite equipment on and off. The satellite subsystems, including sensor packages, will be monitored during the test.

6.3.3 Acoustic vibration test. The integrated spacecraft, configured for launch, will be subjected to an acoustic vibration test in accordance with the levels and exposure times as shown in Table XI. Equipment operating during launch mode is operated and monitored during this test.

Table XI. Acoustic Test Sound Pressure Levels

1/3 Octave Band Center Frequency (Hz)	1/3 Octave Band Sound Pressure Level (dB) Acceptance	Test Tolerance*
40	117.5	±3 dB
50	120	±3 dB
63	122	±3 dB
80	124	±3 dB
100	126	±3 dB
125	127.5	±3 dB
160	129	±3 dB
200	130	±3 dB
250	130.8	±3 dB
315	130.5	±3 dB
400	130	±3 dB
500	129	±3 dB
630	128	±3 dB
800	127	±15 dB
1000	125	±15 dB
1250	122.5	±15 dB
1600	119.5	±15 dB
2000	116	±15 dB
2500	113	±15 dB
3150	110	±15 dB
4000	106	±15 dB
5000	104	±15 dB
6300	101	±15 dB
8000	97.5	±15 dB
10000	95	±15 dB
Overall SPL	139.7	±3 dB

Notes: (1) Test Duration 1.0 Minute
(2) SPL Reference - 0.0002 dyne/cm^2

*Tolerance - difference between specification value and room average sound pressure level for each 1/3 octave band and for the overall level.

6.3.4 Pyro shock test. The satellite will be subjected to pyro shocks generated by all integrated spacecraft ordnance devices. Live separation ordnance will be used. Sensor input and spacecraft equipment responses will be measured. A system performance test will be conducted to verify that no functional degradation of the system results from these shocks.

6.3.5 EMI test. The EMI test is designed to test pertinent spacecraft parameters in the presence of a self-induced worst case environment. The exposure consists of spacecraft induced radiations in the UHF band at 400 MHz and in the S-band region produced by the spacecraft transmitters through their normal network and antennae and any other interferers on the spacecraft.

6.3.6 Degaussing. The spacecraft Precision Mounting Platform (PMP), Equipment Support Module (ESM), and Reaction Support Structure (RSS) will be degaussed in each axis with a 50 Gauss maximum field. The field will be cycled positively and negatively 25 times decrementing from 50 Gauss to 0.25 Gauss.

6.4 Facilities environment. The spacecraft integration and test area at the spacecraft contractor's facility will conform to the requirements of a Class 10,000 clean room. Temperature is controlled to 75 ±10°F and humidity to 45 ±10%. During those periods requiring spacecraft location outside this integration and test area, the spacecraft will be draped with a polyethylene cover and local purge applied as required.

6.5 Transport environments

6.5.1 Factory to prelaunch processing facility. The 5D-3 spacecraft is shipped in a dedicated container. The container is equipped with shock mounting and is provided with a continuous nitrogen purge. Temperature, humidity, and load information are recorded.

6.5.2 Prelaunch processing facility to pad. The integrated 5D-3 spacecraft is transported to the launch site in the shipping container. Temperature, humidity, and load information are recorded continuously.

6.6 Flight environment. The anticipated flight-induced environments for the spacecraft are described in the following subparagraphs for informational purposes only. Sensor contractors shall review these environments and document with the DMSP SPO any conditions at variance with or considered potentially in excess of environments stated in the applicable sensor segment specification. Where applicable sensor segment specifications conflict with these environments, the sensor segment specifications shall take precedence. It shall be the responsibility of the DMSP SPO to resolve any such conflicts.

6.6.1 Launch and ascent environment

6.6.1.1 Acceleration. The maximum flight accelerations for a standard core Titan II configuration will be less than 10 G's. Variations from this environment shall be documented in the applicable sensor segment specification.

6.6.1.2 Fairing temperature. The inside metallic wall temperature of the fairing visible to the sensors shall not exceed 302 degrees Fahrenheit and an emissivity of 0.1. The inside surface of the fairing covered by acoustic blankets and visible to the sensors shall not exceed 130 degrees Fahrenheit and shall not have an emissivity in excess of 0.86.

The maximum dispersed free molecular heating rate at payload fairing jettison and thereafter shall be less than 322 BTU per ft^2 per hour.

6.6.1.3 Contamination. The Fairing forward adapter internal surfaces, the forward surfaces of the conical adapter, and all of the Launch Vehicle equipment within this volume will be cleaned to MIL-STD-1246A (Product Cleanliness Levels and Contamination Control Program, 18 Aug 67) Level 500G at VAFB. All other surfaces will be visually clean. The Fairing will be cleaned of both absorbed and surface molecular contamination which would outgas below the maximum expected internal temperature. Materials used inside the Fairing will not produce greater than 0.2 percent VCM (by weight) nor TML of 2 percent or more from satellite installation on the launch vehicle until satellite separation.

6.6.1.4 Fairing environment. Payload fairing air shall be supplied continuously after encapsulation. There shall be a goal of class 1000 air at the top of the spacecraft, with a requirement for class 10000 or better air at the inlet to the payload fairing. Inlet air temperature shall be controlled at 60 to 65 degrees F, with relative humidity of 50 $\pm$ 5 percent. Refer to applicable sensor ICD listed in Appendix A for purge requirements.

6.6.1.5 Pressure. Payload fairing internal pressure decay during ascent shall not exceed 0.5 PSI/second.

The maximum sound pressure fluctuations in any frequency band at any time during flight are the levels shown in Table XI in the column headed Acceptance.

6.6.2 On orbit environment

6.6.2.1 Orbit characteristics. The normal Block 5D-3 orbit is a nominally circular, sun-synchronous, 450 nautical mile altitude orbit.

6.6.2.2 Radiation. The radiation environment is the combined effects of the environments shown in Table XII, Omnidirectional Proton Flux, Table XIII, Omnidirectional Electron Flux, and Table XIV, Solar Flare Proton Environment for any Three Year or Greater Time Period Between 1977 and 1983. These environments are to be taken as isotropic and omnidirectional and represent the incident spectra on the external surfaces of the spacecraft.

Table XV, Peak Fluxes - Electrons, and Table XVI, Peak Fluxes - Protons show the peak electron and proton flux levels versus particle energy expected in the nominal orbit. These peak fluxes occur over a longitude range from 60°E to 111°W and a latitude range from 15°N to 55°S.

Table XII. Omnidirectional Proton Flux
(protons/cm^2 - day)

Energy Ranges (MeV) E1- E2	Avg. Flux Above E1 (Per Day)	Avg. Integral Flux In Band E1 - E2 (Per Day)
1.00 - 2.00	1.60E 08	6.19E 07
2.00 - 3.00	9.78E 07	2.01E 07
3.00 - 4.00	7.77E 07	1.26E 07
4.00 - 5.00	6.51E 07	8.29E 06
5.00 - 6.00	5.68E 07	5.70E 06
6.00 - 7.00	5.11E 07	3.34E 06
7.00 - 8.00	4.78E 07	2.80E 06
8.00 - 9.00	4.50E 07	2.37E 06
9.00 - 10.00	4.26E 07	2.04E 06
10.00 - 11.00	4.06E 07	1.17E 06
11.00 - 12.00	3.94E 07	1.10E 06
12.00 - 13.00	3.83E 07	1.03E 06
13.00 - 14.00	3.73E 07	9.71E 05
14.00 - 15.00	3.63E 07	9.16E 05
15.00 - 20.00	3.54E 07	3.93E 06
20.00 - 25.00	3.14E 07	1.99E 06
25.00 - 30.00	2.94E 07	1.79E 06
30.00 - 35.00	2.77E 07	1.40E 06
35.00 - 40.00	2.63E 07	1.31E 06
40.00 - 45.00	2.49E 07	1.23E 06
45.00 - 50.00	2.37E 07	1.16E 06
50.00 - 60.00	2.25E 07	2.28E 06
60.00 - 70.00	2.03E 07	2.02E 06
70.00 - 80.00	1.82E 07	1.80E 06
80.00 - 90.00	1.64E 07	1.60E 06
90.00 - 100.00	1.48E 07	1.43E 06
100.00 - 110.00	1.34E 07	1.35E 06
110.00 - 120.00	1.21E 07	1.21E 06
120.00 - 130.00	1.09E 07	1.08E 06
130.00 - 140.00	9.78E 06	9.66E 05
140.00 - 150 00	8.81E 06	8.67E 05
150.00 - 200.00	7.95E 06	3.17E 06
200.00 - 250.00	4.77E 06	1.86E 06
250.00 - 300.00	2.92E 06	1.12E 06
300.00 - Infinity	1.79E 06	1.79E 06

Table XIII. Omnidirectional Electron Flux
(electrons/cm^2 - day)

Energy Ranges (MeV) E1 - E2	Avg. Flux Above E1 (Per Day)	Avg. Integral Flux In Band E1 - E2 (Per Day)
0.05 - 0.25	6.04E 10	5.02E 10
0.25 - 0.50	1.02E 10	7.25E 09
0.50 - 0.75	2.96E 09	1.31E 09
0.75 - 1.00	1.65E 09	5.95E 08
1.00 - 1.25	1.06E 09	3.13E 08
1.25 - 1.50	7.42E 08	2.14E 08
1.50 - 1.75	5.28E 08	1.44E 08
1.75 - 2.00	3.84E 08	1.03E 08
2.00 - 2.25	2.81E 08	6.74E 07
2.25 - 2.50	2.13E 08	5.07E 07
2.50 - 2.75	1.63E 08	4.22E 07
2.75 - 3.00	1.21E 08	2.94E 07
3.00 - 3.25	9.12E 07	1.86E 07
3.25 - 3.50	7.26E 07	1.41E 07
3.50 - 3.75	5.85E 07	1.11E 07
3.75 - 4.00	4.74E 07	8.90E 06
4.00 - 4.25	3.84E 07	1.12E 07
4.25 - 4.50	2.72E 07	7.94E 06
4.50 - 4.75	1.93E 07	5.61E 06
4.75 - 5.00	1.37E 07	3.97E 06
5.00 - 5.25	9.68E 06	4.43E 06
5.25 - 5.50	5.25E 06	2.40E 06
5.50 - 5.75	2.85E 06	1.30E 06
5.75 - 6.00	1.55E 06	7.07E 05
6.00 - 6.25	8.40E 05	5.88E 05
6.25 - 6.50	2.53E 05	1.77E 05
6.50 - 6.75	7.52E 04	5.39E 04
6.75 - 7.00	2.13E 04	2.13E 04
7.00 - 7.25	0.0	0.0
7.25	0.0	

Table XIV. Solar Flare Proton Environment (protons/cm^2) for any Three Year or Greater Time Period Between 1977 - 1983

Flare Proton Energy (E) (MeV)	Flux Above Energy (E)
0.10	2.0×10^{11}
0.50	1.3×10^{11}
1.00	1.0×10^{11}
5.00	4.0×10^{10}
10.00	2.0×10^{10}
20.00	9.0×10^{9}
50.00	2.0×10^{9}
100.00	4.0×10^{8}
200.00	3.5×10^{7}

Table XV. Peak Fluxes - Electrons

Particle Energy (E) (MeV)	Peak Flux (E) per cm^2/sec
0.05	1.1×10^{7}
0.50	3.7×10^{5}
1.00	1.8×10^{5}
2.00	5.7×10^{4}
3.00	2.3×10^{4}
4.00	1.0×10^{4}
5.00	3.0×10^{3}

Table XVI. Peak Fluxes - Protons

Particle Energy (E) (MeV)	Peak Flux (E) per cm^2/sec
10^{-4}	1.1×10^6
0.10	5.2×10^5
2.00	1.8×10^4
5.00	7.8×10^3
10	7.2×10^3
20	6.2×10^3
30	5.5×10^3
50	4.4×10^3
100	2.5×10^3

10.0 APPENDIX I. OLS AND MISSION SENSORS

10.1 Scope. This appendix lists the DMSP Block 5D-3 (S16-S20) sensors and references their appropriate ICDs.

10.2 Sensors and ICDs. Table XVII lists the S16-S20 OLS and mission sensors and references their applicable ICDs.

Table XVII. DMSP Block 5D-3 (S16-S20) OLS and Mission Sensors

Sensor	Name	ICD
OLS	Operational Line Scan System	ICD-88802
SSM	Triaxial Fluxgate Magnetometer	ICD-88803
SSIES3	Plasma Monitoring System	ICD-88804
SSJ5	Electron and Ion Spectrometer	ICD-88805
SSMIS	Microwave Imager Sounder	ICD-88806
SSF	classified	ICD-88807
SSULI	Ultraviolet Limb Imager	ICD-88809
SSUSI	Spectrographic Imager	ICD-88810

10.3 Interface documentation. The relationship of the ICDs listed in Table XVII to the other requirements documents for the DMSP Space Segment are described in document ICD-88801.