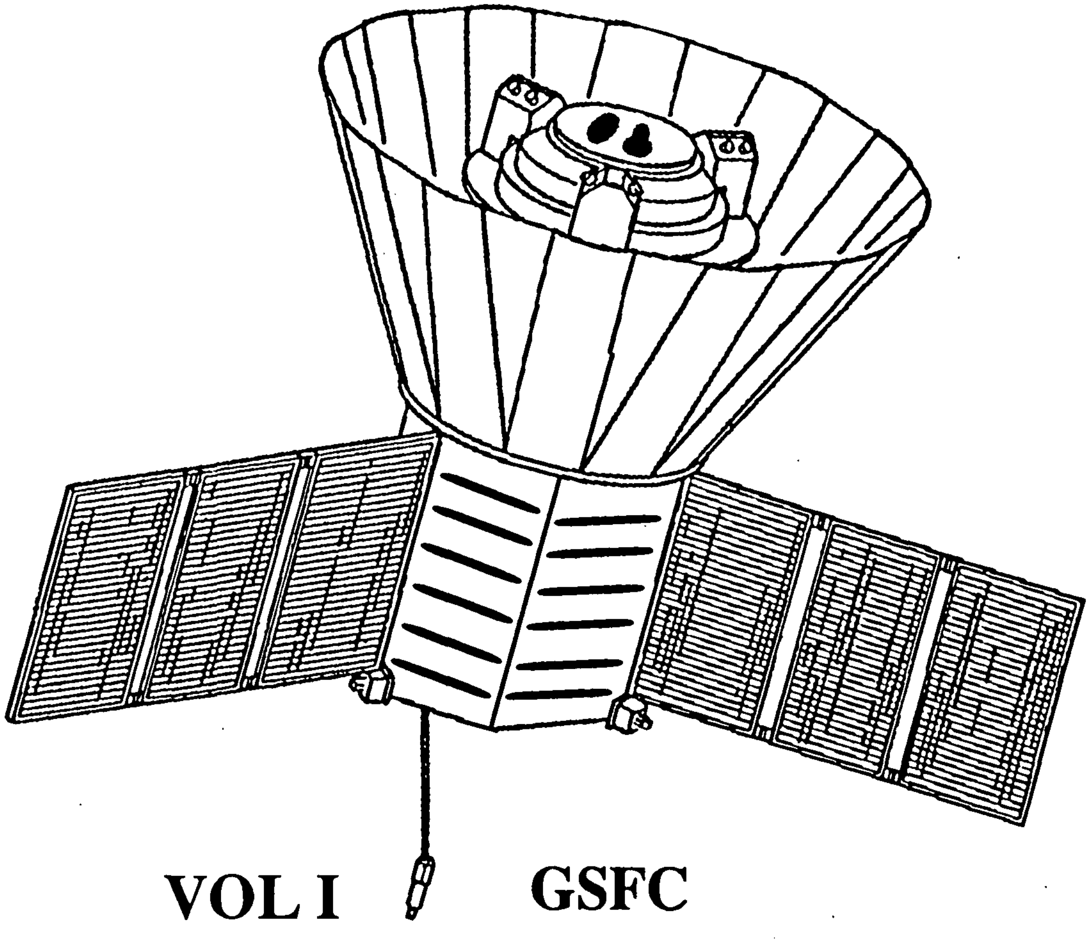
ON ORBIT ENGINEERING PERFORMANCE

COBE



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AGENDA
COBE ON-ORBIT ENGINEERING PERFORMANCE
GSFC, BUILDING 3 AUDITORIUM
MARCH 7 & 8, 1990

TOPIC	PRESENTER
INTRODUCTION	R. MATTSON D. McCARTHY
COBE OVERVIEW	D. McCARTHY A. FRAGOMENI
QUALITY ASSURANCE	A. HARPER
STRUCTURE	
SYSTEM	J. WOODS
ANALYSES	O. SHEINMAN
DEPLOYABLES	
SOLAR ARRAY	S. NGO
TRF	A. STEWART
ORDNANCE/ANTENNA BOOM	M. PHAN
THERMAL	R. CHALMERS

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POWER	
SYSTEM AND SOLAR ARRAYS	J. JERMAKIAN
PSE	D. MANZER
BATTERIES	S. TILLER
C&DH, TAPE RECORDERS, SCU, HARNESS	M. FLANEGAN
ATTITUDE CONTROL	H. HOFFMAN
SYSTEM	~~T. FLATELY~~
ACE	W. SQUILLARI
WHEELS, GYROS, MMA	M. FEMIANO
EARTH SCANNERS, SUN SENSORS	P. NEWMAN
ANALYSES	S. PLACANICA
COMMUNICATIONS	
SYSTEM	J. ROGERS
USO	D. ZILLIG
TRANSPONDERS	G. KRONMILLER
ANTENNAE	B. JACKSON
DEWAR	S. VOLZ
CONTAMINATION	R. BARNEY
INSTRUMENTS OVERVIEW	E. YOUNG
INSTRUMENT MECHANISMS	M. RYSCHKEWITSCH

FIRAS	M. ROBERTO
DIRBE	L. LINSTROM
DMR	R. MILLS
OPERATIONS AND GROUND SYSTEMS	J. WOLFGANG
ORBIT AND ATTITUDE DETERMINATION	R. DeFAZIO

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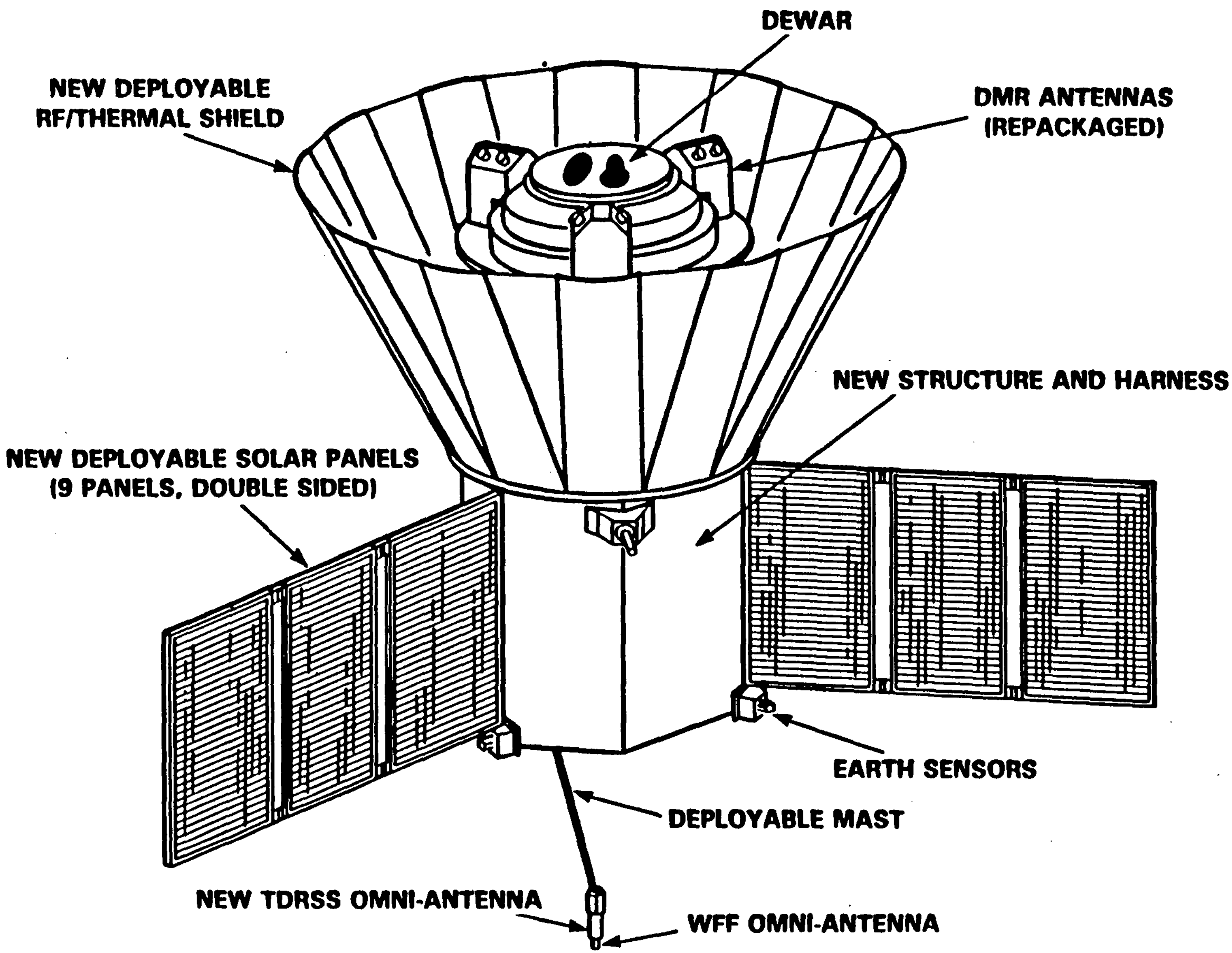
COBE/DELTA CONFIGURATION



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SIGNIFICANT EVENTS

- COBE WAS SUCCESSFULLY LAUNCHED ON NOVEMBER 18, 1989 AT 0634 A.M. PST.
- DELTA LAUNCH VEHICLE PERFORMED FLAWLESSLY:
  - 900.5 KM X 899.3 KM
  - 99.03° INCLINATION
- OBSERVATORY DEPLOYMENTS OCCURRED AS PLANNED.
- DEWAR COVER SUCCESSFULLY DEPLOYED ON DAY 4, AS PLANNED, AND CRYOGEN TEMPERATURE CURRENTLY AT 1.41°K.
- ALL THREE INSTRUMENTS OPERATING AND ACQUIRING SCIENCE DATA.
- POCC/NETWORK SUPPORT HAS BEEN EXCELLENT.

11/30/89

FIRST TEN DAYS IN THE LIFE OF COBE

DAY		HIGHLIGHTS
1	11/18/89	o LAUNCH o ALL MAJOR ASCENT, ACQUISITION, SEPARATION, AND INITIALIZATION ACTIVITIES PERFORMED.
2	11/19/89	o ROLL TO 2 DEGREES PERFORMED TO ANALYZE HOT COVER/COOL MAINSHELL CONDITION. RETURNED TO 0 DEGREES TO MINIMIZE CLAMP BAND COOLING. o DMR FULL POWER UP AND SCIENCE DATA MODE WITH FREQUENT CALIBRATIONS o DEWAR INTERNAL COOLDOWN/PUMPDOWN BEGINS.
3	11/20/89	o FIRAS POWER UP AND CHECKOUT o DIRBE CHECKOUT
4	11/21/89	o DEWAR COVER EJECTION o FIRST SKY OBSERVATIONS BY DIRBE AND FIRAS o ACS B GYRO FAILURE--B GYRO COMMANDED OFF--CUT GYRO OUT OF LOOP

FIRST TEN DAYS IN THE LIFE OF COBE (CONTINUED)

DAY		HIGHLIGHTS
5	11/22/89	o FIRAS MTM UNLATCHED AND SUCCESSFULLY PLACED IN SCANNING MODE. o FIRAS EXPERIENCES END OF TRAVEL HITS. o INSTRUMENTS COLLECTING SCIENCE DATA.
6	11/23/89	o RECONFIGURED ACS TO A GYRO FAILURE TOLERANT MODE. o INSTRUMENTS COLLECTING SCIENCE DATA.
7	11/24/89	o FIRAS NULLING OF LOW FREQUENCY CHANNELS USING ICAL. o GYRO ANALYSIS SHOWS S/C SPIN-UP IS SAFE. SPIN-UP SCHEDULED FOR DAYS 8, 9, AND 10. o INSTRUMENTS COLLECTING SCIENCE DATA.

FIRST TEN DAYS IN THE LIFE OF COBE (CONTINUED)

DAY		HIGHLIGHTS
8	11/25/89	o S/C SPIN-UP, TO 0.4 RPM SUCCESSFULLY PERFORMED. o DEWAR COOLS BELOW 1.6 K SPEC. o INSTRUMENTS COLLECTING SCIENCE DATA.
9	11/26/89	o S/C SPIN-UP TO 0.6 RPM SUCCESSFULLY PERFORMED. o INSTRUMENTS COLLECTING SCIENCE DATA.
10	11/27/89	o S/C SPIN-UP TO 0.8 RPM (FINAL MISSION RATE) SUCCESSFULLY PERFORMED. o INSTRUMENTS COLLECTING SCIENCE DATA.

ON ORBIT ANOMALIE SUMMARY

PROBLEMS/ISSUES	PROGRAMMATIC IMPACT	ACTION	ESTABLISHED/ COMPLETED DATE
B GYRO FAILED TO OPERATE ON DAY 4 OF THE MISSION.	ADDITIONAL FAILURES OF THE REMAINING GYROS (A & C) COULD RESULT IN ATTITUDE INSTABILITY AND DEWAR POINTING INTO SUN.	- ~~GRYO A&C CROSS STRAPPING REMOVED PROVIDING SAFE OPERATION SHOULD SECOND GYRO FAIL; CONING INCREASED ABOUT X-AXIS, BUT WITHIN SPEC.~~	COMPLETED
		- CODE 712 IS ANALYZING CURRENT CONFIGURATION, INCLUDING POTENTIAL FAILURES.	DECEMBER, 1989
		- GYRO COMMITTEE, CHAIRED BY HENRY PRICE, REVIEWING FAILURE TO REPORT BACK TO PROJECT WITH FINDINGS AND RECOMMENDATIONS.	DECEMBER, 1989
FIRAS MIRROR TRANSPORT MECHANISM EXPERIENCING "END OF TRAVEL HITS", PRIMARILY IN RADIATION BELT ENVIRONMENT (SAA AND VAB).	-LONG TERM: THERMAL INPUTS RESULT IN SOME LOSS OF DEWAR LIFETIME. -SHORT TERM: EXTENDS DIRBE DETECTOR ANNEALING SCHEDULE RESULTING IN SOME LOSS OF DATA.	COMMITTEE CHAIRED BY JOHN PYLE (CODE 710) TASKED TO REVIEW PROBLEM AND REPORT BACK TO PROJECT WITH RECOMMENDATIONS.	DECEMBER 8, 1989

BACKGROUND AND SYSTEM OVERVIEW

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BACKGROUND

MARCH 1976	COBE STUDY TEAM FORMED AT GODDARD
JULY 1977	START OF DEFINITION PHASE
JULY 1982	HEADQUARTERS APPROVAL FOR COBE DEVELOPMENT PHASE (LAUNCH JANUARY 1989)
NOVEMBER 1982	COBE/STS SCHEDULE ACCELERATION--LAUNCH FALL 1988
NOVEMBER 1984	COMPLETED CRITICAL DESIGN REVIEW OF OBSERVATORY (SHUTTLE LAUNCH)
JANUARY 1986	SHUTTLE 51L ACCIDENT
FEBRUARY 1986	INITIATED ALTERNATE LAUNCH MODE STUDY
NOVEMBER 1986	HEADQUARTERS APPROVES LAUNCH OF COBE ON A DELTA 5920; LAUNCH SCHEDULED FOR FEBRUARY 1989.
MAY 1987	COMPLETED CRITICAL DESIGN REVIEW OF OBSERVATORY (DELTA LAUNCH)
NOVEMBER 1987	INSTRUMENT TECHNICAL PROBLEMS (CHOPPER AND MIRROR TRANSPORT MECHANISM) IMPACT INSTRUMENT DELIVERY SCHEDULE AND LAUNCH DATE
DECEMBER 1987	COMPLETED INTEGRATION AND TESTING ON COBE/SHUTTLE MOCK-UP.
FEBRUARY 1988	DECISION TO DELETE FUTURE INSTRUMENT TESTING IN TEST DEWAR AND PROCEED DIRECTLY TO FLIGHT DEWAR; LAUNCH RESCHEDULED TO MAY 1989.
MARCH 1989	OBSERVATORY ACCEPTANCE TESTS STARTED.

COBE MISSION

- PERFORM A DEFINITIVE EXPLORATION OF THE DIFFUSE COSMIC BACKGROUND RADIATION AND PROVIDE A FULL SKY MAP OF THE BACKGROUND RADIATION
- LAUNCH BY STS FROM VANDENBERG LAUNCH SITE, CALENDAR YEAR 1989
- ONE YEAR MISSION LIFE
- EXECUTED IN-HOUSE AT THE GSFC WITH MAJOR SUBSYSTEM PROCUREMENTS

GSFC

INSTRUMENT PERFORMANCE REQUIREMENTS

FIRAS

- MEASURE THE SPECTRUM OF THE COSMIC BACKGROUND RADIATION OVER THE WAVELENGTH RANGE 0.1 TO 10 MM

- PROVIDE AN ABSOLUTE DETERMINATION OF THE SPECTRUM TO AN ACCURACY AND RMS NOISE LEVEL OF 10^{-13} W/CM^{2}SR, FOR EACH 7 DEGREE PIXEL IN THE SKY IN THE BAND FROM 0.5 TO 5 MM

- SPECTRUM RESOLUTION WILL BE 5% OR 0.2 CM^{-1}, WHICHEVER IS LARGER

DMR

- MEASURE THE LARGE-ANGULAR-SCALE INTENSITY DISTRIBUTION OF THE COSMIC BACKGROUND AT 32, 53 AND 90 GHz.

- SEARCH FOR ANISOTROPY IN THE BACKGROUND RADIATION WITH AN RMS NOISE LEVEL OF 1 PART IN 10^4 (0.3 MK) OR SMALLER, IN EACH 7 DEGREE PIXEL

DIRBE

- MEASURE THE SPECTRUM AND ANGULAR DISTRIBUTION OF THE BACKGROUND IN 10 DISCRETE PHOTOMETRIC SPECTRAL BANDS BETWEEN 1 AND 300 MICRONS

- BANDS BETWEEN 1 AND 4 MICRONS WILL MEASURE THE LINEAR POLARIZATION OF THE BACKGROUND AS WELL AS THE INTENSITY

- IN EACH 1 DEGREE PIXEL, PROVIDE AN ABSOLUTE DETERMINATION OF THE INTENSITY IN EACH BAND TO AN RMS NOISE LEVEL OF 10^{-13} W/CM^2 SR OR 1% OF THE ASTROPHYSICAL BACKGROUND (WHICHEVER IS LARGER)

MISSION LIFETIME REQUIREMENTS

LIFETIME REQUIREMENT DERIVED FROM:

- o VIEW NO LESS THAN 80 PERCENT OF THE CELESTIAL SPHERE FOR FIRAS AND DIRBE
- o VIEW NO LESS THAN 95 PERCENT OF THE CELESTIAL SPHERE FOR DMR
- o ACHIEVE SPECIFIED SENSITIVITIES
- o OBSERVE 50 PERCENT OF THE SKY AT A RANGE OF SOLAR ELONGATIONS FROM 64 TO 124 DEGREES WITH DMR AND DIRBE

THEREFORE:

- o MINIMUM MISSION OPERATIONAL LIFETIME FOR FIRAS AND DIRBE IS 6 MONTHS (PLANNED LIFETIME OF 12 MONTHS)
- o MINIMUM MISSION LIFETIME FOR DMR IS 12 MONTHS

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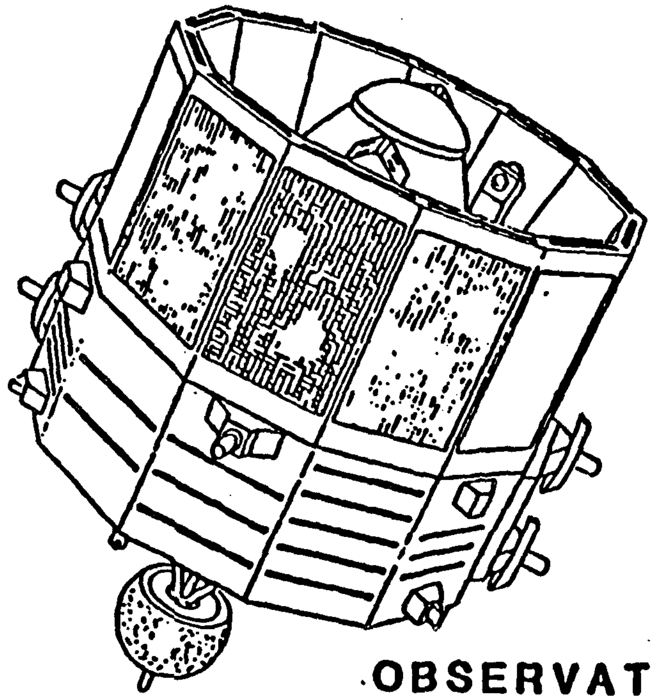
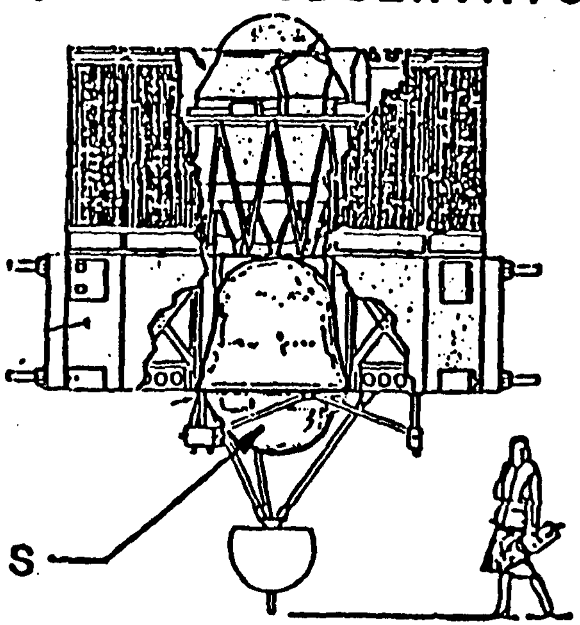
SUMMARY OF COBE MISSION CHARACTERISTICS

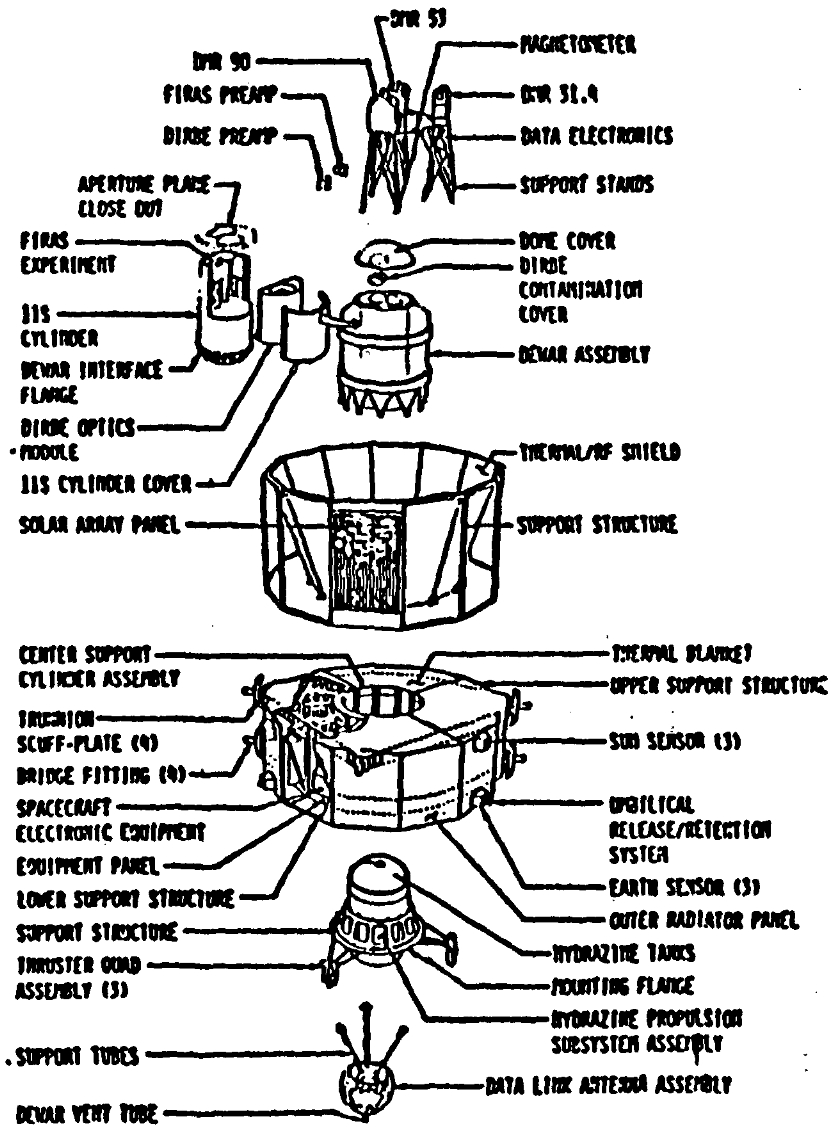
LAUNCH DATE	FY 1988
MISSION LIFE	1 YEAR PLANNED (CURRENT DEWAR LIFETIME; > 12 MONTHS)
ORBIT	900 KM CIRCULAR, SUN SYNCHRONOUS, 6 AM/6 PM NODE CROSSING PARK ORBIT TO MISSION ORBIT TRANSFER--LESS THAN 1 WEEK
LAUNCH VEHICLE	SHUTTLE, WESTERN SPACE AND MISSILE CENTER
OBSERVATORY	STM/PROTOFLIGHT
WEIGHT (BOL)	10,000 LBS (11,500 LBS CONTROL WEIGHT SUBMITTED TO JSC)
LENGTH/DIAMETER	18/15 FT.
INSTRUMENTS	DIFFERENTIAL MICROWAVE RADIOMETER (DMR), FAR INFRARED ABSOLUTE SPECTROPHOTOMETER (FIRAS), DIFFUSE INFRARED BACKGROUND EXPERIMENT (DIRBE),
DEWAR	IRAS DESIGN MODIFIED FOR COBE
PROPULSION	MONOPROPELLANT HYDRAZINE
ATTITUDE CONTROL	ZERO MOMENTUM/3-AXIS STABILIZED
POWER	SOLAR ARRAY/BATTERIES DIRECT ENERGY TRANSFER
THERMAL	PASSIVE CONTROL
DATA RATE	4096 BPS (REAL-TIME AND RECORD), 655 KBPS PLAYBACK
GROUND SYSTEM	
OPERATION AND CONTROL	MULTI-SATELLITE OPERATIONS CONTROL CENTER
FORWARD DATA LINK	COMMAND/TDRSS-MA (UP TO 2 HOURS PER DAY)
RETURN DATA LINK	TELEMETRY/TDRSS-MA (UP TO 2 HOURS PER DAY)
SCIENCE DATA LINK	DIRECT-TO-GROUND RECEIVING STATION AT THE GSFC
SCIENCE DATA PROCESSING	COBE SCIENCE DATA ROOM



OBSERVATORY



HPS



EXPLODED VIEW OF OBSERVATORY

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BACKGROUND

- o AFTER THE 51L ACCIDENT, THE COBE PROJECT REVIEWED USE OF ELV's AND ALTERNATE LAUNCH SCENARIOS TO ACCOMPLISH COBE MISSION
- o DECISION MADE TO DESIGN COBE FOR A DELTA 3920A WITH AN 8' DIA. FAIRING; MAINTAINS COMPATIBILITY WITH OTHER ELV's. REFERENCE THE AUGUST 7, 1986 PRESENTATION TO NASA HEADQUARTERS (S. KELLER, ET AL)
- o BASELINED A SUN SYNCHRONOUS POLAR ORBIT FROM VAFB
- o SPACECRAFT LAUNCH READINESS IS 29 MONTHS AFTER HEADQUARTERS DIRECTION; PROJECT INTERPRETS POP 86-2M LETTER AS DIRECTION
- o ACCELERATED SCHEDULE NECESSITATES A SKUNKWORKS MODE OF OPERATION
  - DMR REPACKAGING
  - SPACECRAFT REDESIGN

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PROGRAMMATIC CONSIDERATIONS

- o WILL DESIGN A SPACECRAFT COMPATIBLE WITH A DELTA 3920A
- o SCHEDULE - TIGHT AT 29 MONTHS
- o DESIGNING FOR DELTA ALLOWS COBE TO BE LAUNCHED ON ANY VEHICLE GREATER THAN DELTA
- o WILL REQUIRE TDRS SINGLE ACCESS (S/A) USE 5 TIMES/DAY FOR 20 MINUTES EACH
- o DELTA WILL PROVIDE A QUALIFIED PAYLOAD ATTACH FITTING PLUS THE NECESSARY PYROTECHNICS AND FIRING CIRCUITS
- o MAKES AVAILABLE THE CURRENT COBE PRIMARY STRUCTURE ($5M), HYDRAZINE PROPULSION SYSTEM ($5M), ELECTRONICALLY SWITCHED ANTENNA ($3M)

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REQUIRED CENTER COMMITMENT

- A 'SKUNKWORKS' OPERATION (I.E., A DEDICATED, COLLOCATED ENGINEERING AND SUPPORT TEAM UNDER ONE ROOF).
- ADDITIONAL 4,000 SQUARE FEET OF SPACE AVAILABLE FOR THE ENTIRE COLLOCATED TEAM IN BUILDING 7/10/15 COMPLEX.
- PROCUREMENTS WILL BE PROCESSED IMMEDIATELY.
- NO LIMIT ON OVERTIME AND/OR COMPENSATORY TIME, AS NECESSARY TO MAINTAIN SCHEDULE.
- PROJECT EXPEDITORS TO TRACK PROCUREMENTS, HARDWARE, AND DOCUMENTATION.
- INSTITUTIONAL PRIORITY.
- DOCUMENTATION AND REPORTING WILL BE STREAMLINED.
- DOLLARS AND PROCUREMENT MANAGEMENT CONTROLLED AT THE PROJECT OFFICE LEVEL.

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COBE EXPENDABLE LAUNCH VEHICAL (ELV) ANALYSIS FOR CIRCULAR POLAR ORBIT (CPO)*

LAUNCH VEHICLE	CAPABILITY	PAYLOAD DIAMETER	COMMENTS	**DISCRIMINATOR
SHUTTLE	11,500 LBS. TO CPO XFER HYDRAZINE PROPULSION SUBSYSTEM (HPS) CIRCULARIZE	180"	BASELINE	AVAILABILITY AT WTR
DELTA 3920	4,800 LBS. TO CPO	100"	NEW LIGHT WEIGHT STRUCTURE, NEW HARNESS, MINIMAL REDUNDANCY, DEPLOYABLES	MOST SIGNIFICANT IMPACT TO PROJECT, DELTA AVAILABILITY AT WTR
ATLAS CENTAUR	9,000 LBS. TO CPO	144"	MODIFY EXISTING STRUCTURE NEW HARNESS	AT PRESENT THERE IS NO A/C CAPABILITY AT WTR
TITAN 34D	12,000 LBS. TO CPO XFER ORBIT, CIRCULARIZE WITH EXISTING HPS	111"	NEW COBE STRUCTURE, NEW HARNESS, DEPLOYABLES	LAUNCH FACILITY AVAILABILITY
TITAN 34D7	15,000 LBS. TO CPO XFER ORBIT, CIRCULARIZE WITH EXISTING HPS	180"	MODIFY EXISTING STRUCTURE, (MOST CLOSELY EMULATES SHUTTLE)	VEHICLE COST
ARIANE (42/44 LP)	9,000 TO 12,000 LBS. TO CPO	144"	MODIFY EXISTING STRUCTURE, NEW HARNESS	VEHICLE COST

* 900KM, 99° INCLINATION

** ALL ELV'S REQUIRE NEW LOAD PATH AND RE-QUAL OF STRUCTURE

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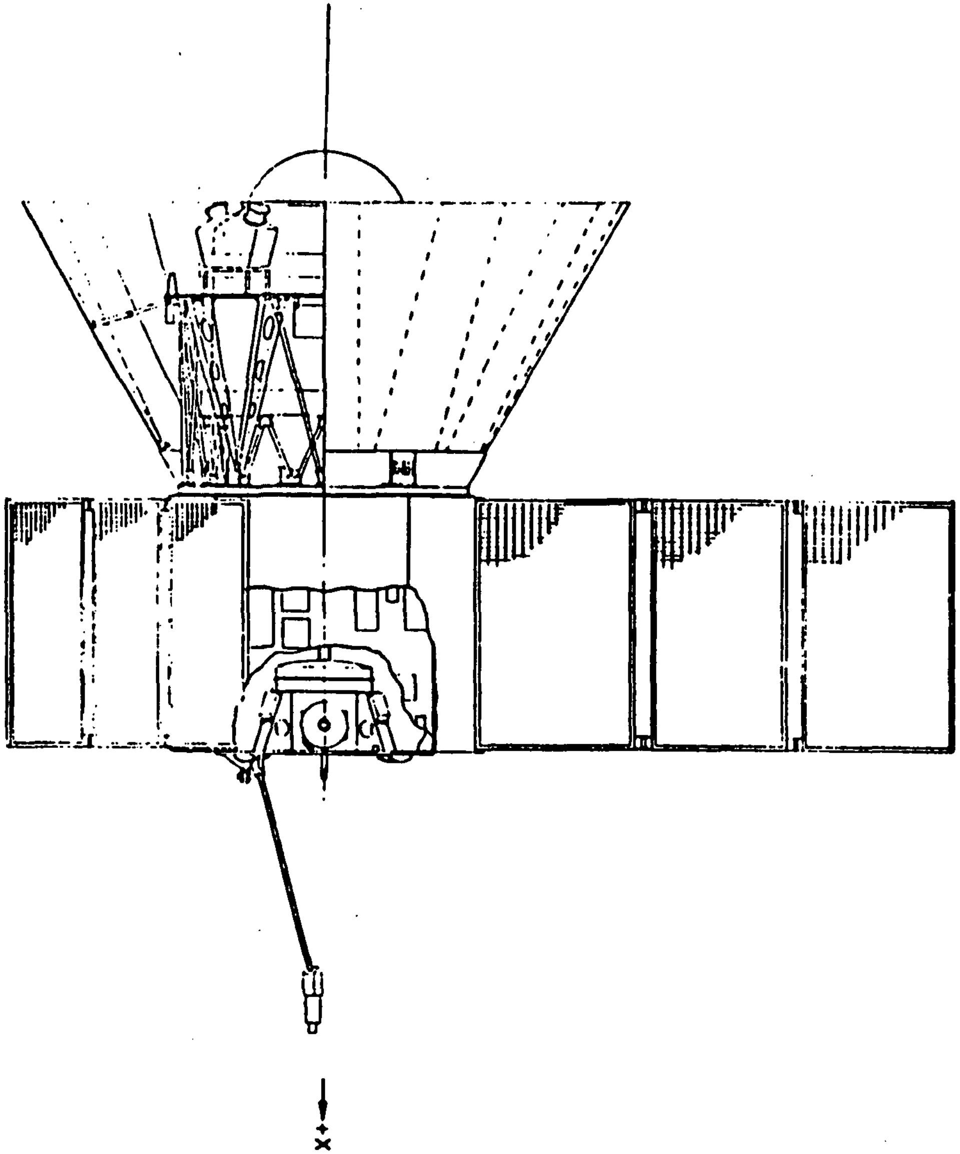
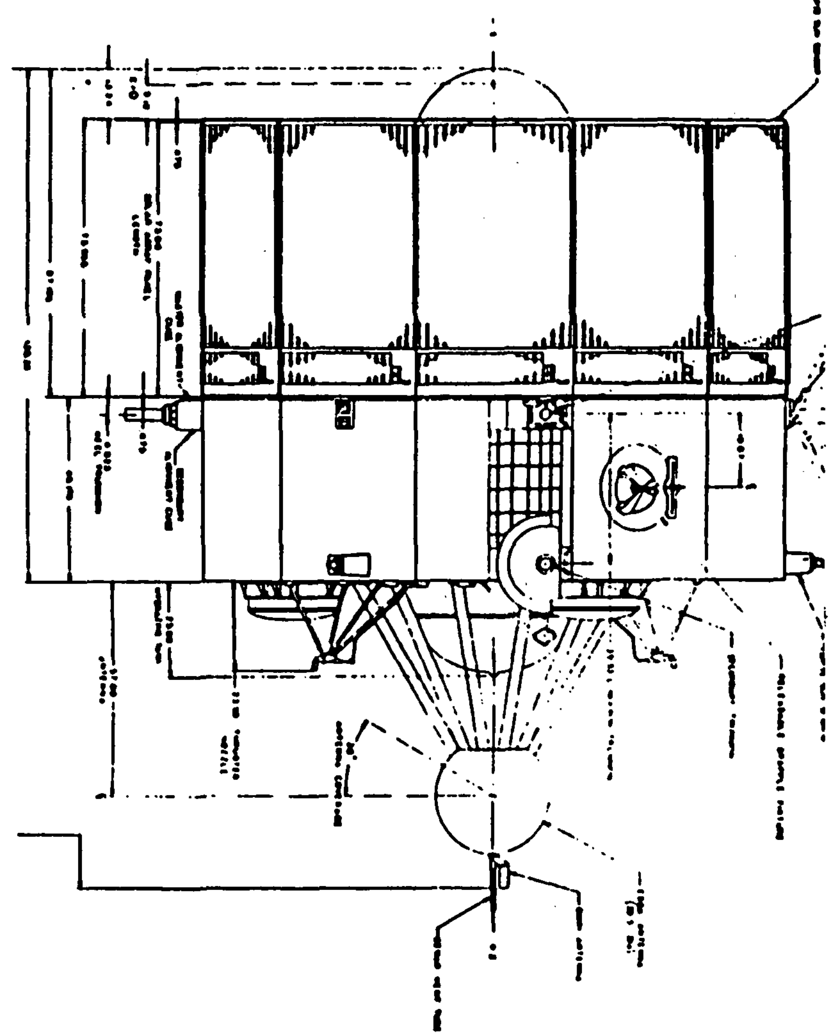


COBE DELTA



COBE STS

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SUMMARY OF SUBSYSTEM CHANGES REQUIRED FOR A DELTA LAUNCH

SUBSYSTEM	CHANGE	HERITAGE
PROPULSION SYSTEM	PROPULSION SYSTEM IS NOT REQUIRED	N/A
DIRBE	NO CHANGE	N/A
FIRAS	NO CHANGE	N/A
DEWAR	MOUNT DMR RECEIVERS TO DEWAR GIRTH RING	IRAS
COMMAND/DATA HANDLING	NO CHANGE	N/A
STRUCTURE	NEW LIGHTWEIGHT PRIMARY STRUCTURE	DELTA CLASS SPACECRAFT SUCH AS SMM, IUE, ISEE, ETC.
	NEW DEPLOYABLE RF/THERMAL SHIELD	- SMM HINGE MECHANISM - ATS-6 ANTENNA (LMSC) - BLOCK 5D SUN SHIELD (RCA)
ACS	ELIMINATE TRANSFER ORBIT ELECTRONICS	N/A
	BLANKING FOR SENSORS	LANDSAT D/D'

SUMMARY OF SUBSYSTEM CHANGES REQUIRED FOR A DELTA LAUNCH--CONTINUED

SUBSYSTEM	CHANGE	HERITAGE
COMMUNICATIONS	ELIMINATE ELECTRONICALLY SWITCHING SPHERICAL ARRAY ANTENNA (ESSA)	N/A
	NEW OMNI TO COMMUNICATE WITH TDRS WITH DEPLOYABLE MAST	ISEE SPACECRAFT DE SPACECRAFT
POWER	NEW DEPLOYABLE SOLAR ARRAYS	ATS-6 SMM HINGE
	EXCHANGE 50AMP-HR BATTERIES FOR 20 AMP-HR BATTERIES	"OFF THE SHELF" HARDWARE
	POWER SUPPLY ELECTRONICS WEIGHT REDUCTIONS & POWER INCREASE	N/A
THERMAL	RECONFIGURATION REQUIRES ADDITIONAL ANALYSES	DELTA CLASS SPACECRAFT ISEE, IUE, SMM, ETC.
DIFFERENTIAL MICRO-WAVE RADIOMETERS	REWORK 31GHZ RECEIVER AND REPACKAGE 53 AND 90 GHZ RECEIVERS TO FIT WITHIN 8' DIAMETER FAIRING	MAKES USE OF EXISTING RECEIVER COMPONENTS
ELECTRICAL	NEW ELECTRICAL HARNESS	COBE HARNESS FOR STS; EXISTING BOXES WITH DEFINED INTERFACES
	SCU - PYRO CIRCUITS	

WEIGHT STATUS

SUBSYSTEM	STS LAUNCH WEIGHT (LBS.)	DELTA LAUNCH WEIGHT (LBS.)	REASON FOR CHANGE
PROPULSION	2,064	0	NOT REQUIRED FOR A DELTA LAUNCH
COMMAND/DATA HANDLING	175	175	
ATTITUDE CONTROL	475	362	ONE MOMENTUM WHEEL/NO GYRO SHELF
TRANSPONDERS/ANTENNAS	206	42	REMOVE ELECTRONICALLY DESPUN ANTENNA/ELECTRONICS & REPLACE WITH OMNI
ELECTRICAL (HARNESS)	489	285	SHORTER HARNESS RUNS
POWER	603	382	2 20-AMP/HR BATTERIES FOR 2 50-AMP/HR AND MODS TO PSE & SHUNT DISSIPATORS
THERMAL & BALANCE WEIGHTS	211	80	SMALLER SPACECRAFT & NO BALANCE WEIGHTS
STRUCTURE	3,803	1,000	NEW STRUCTURE DESIGN
DEWAR	1,426	1,360	AUX. VAC. SYSTEM & CLAMP BAND THERMAL SHIELD NOT REQUIRED FOR A DELTA LAUNCH
FIRAS/DIRBE/CRYO OPT. ASSBLY.	790	790	
DMR	352	352	BASED ON STS ACTUALS
TOTAL	10,594	4,828	

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COBE/DELTA WEIGHT ANALYSIS

ACTUAL WEIGHT (EXISTING HARDWARE)	2,642 LBS.	(55%)
DERIVED WEIGHT (80% IS STRUCTURE)	1,315 LBS.	(27%)
ESTIMATED WEIGHTS	871 LBS.	(18%)
TOTAL SPACECRAFT WEIGHT	4,828 LBS.	(100%)
DELTA 3920A PERFORMANCE (95% PCS, 900 KM SUN SYNC.)	5,125 LBS.	
MARGIN	297 LBS.	
PERCENT MARGIN ON ESTIMATED AND DERIVED WEIGHT		14%

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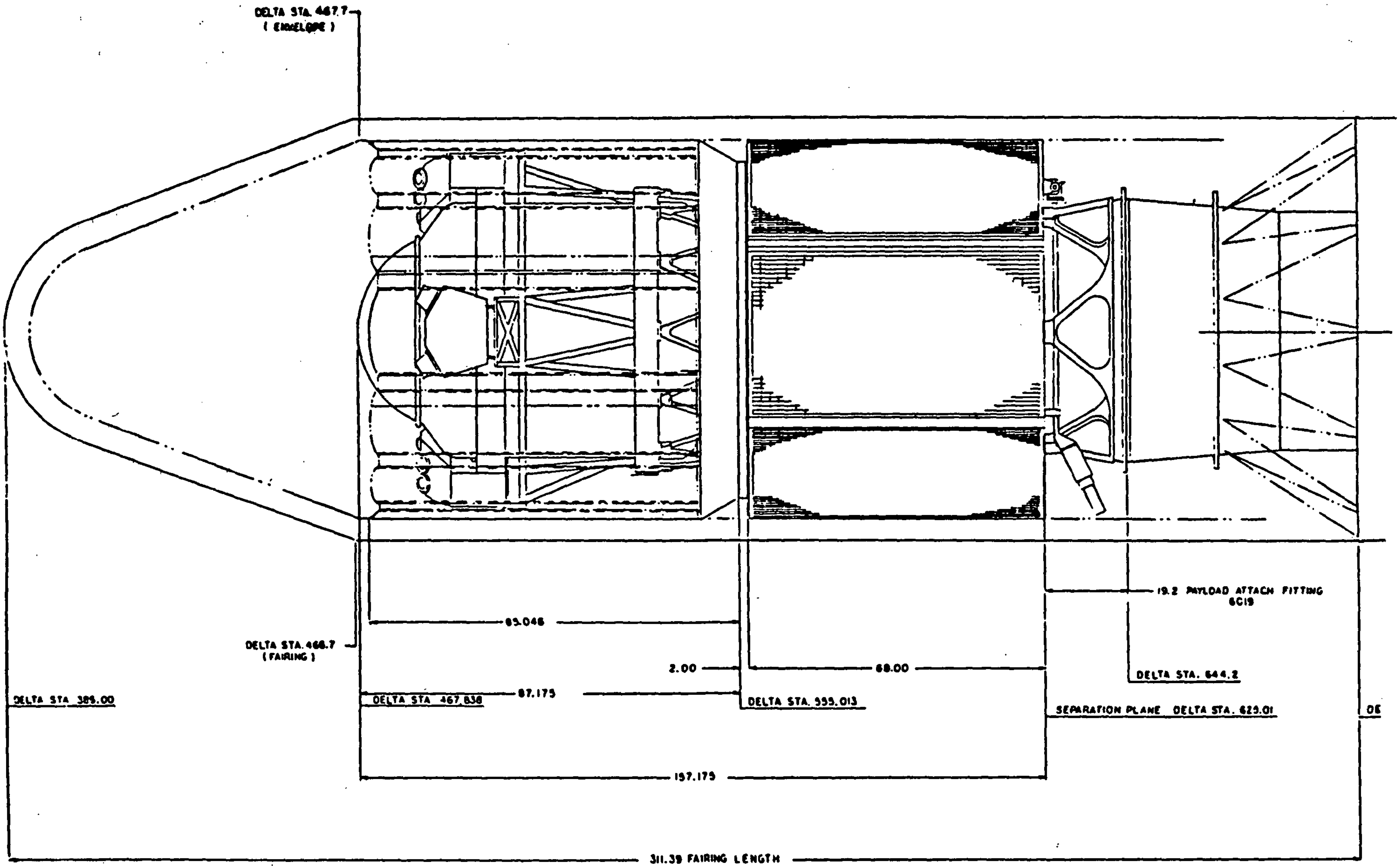

LAUNCH CONFIGURATION - SIDE VIEW

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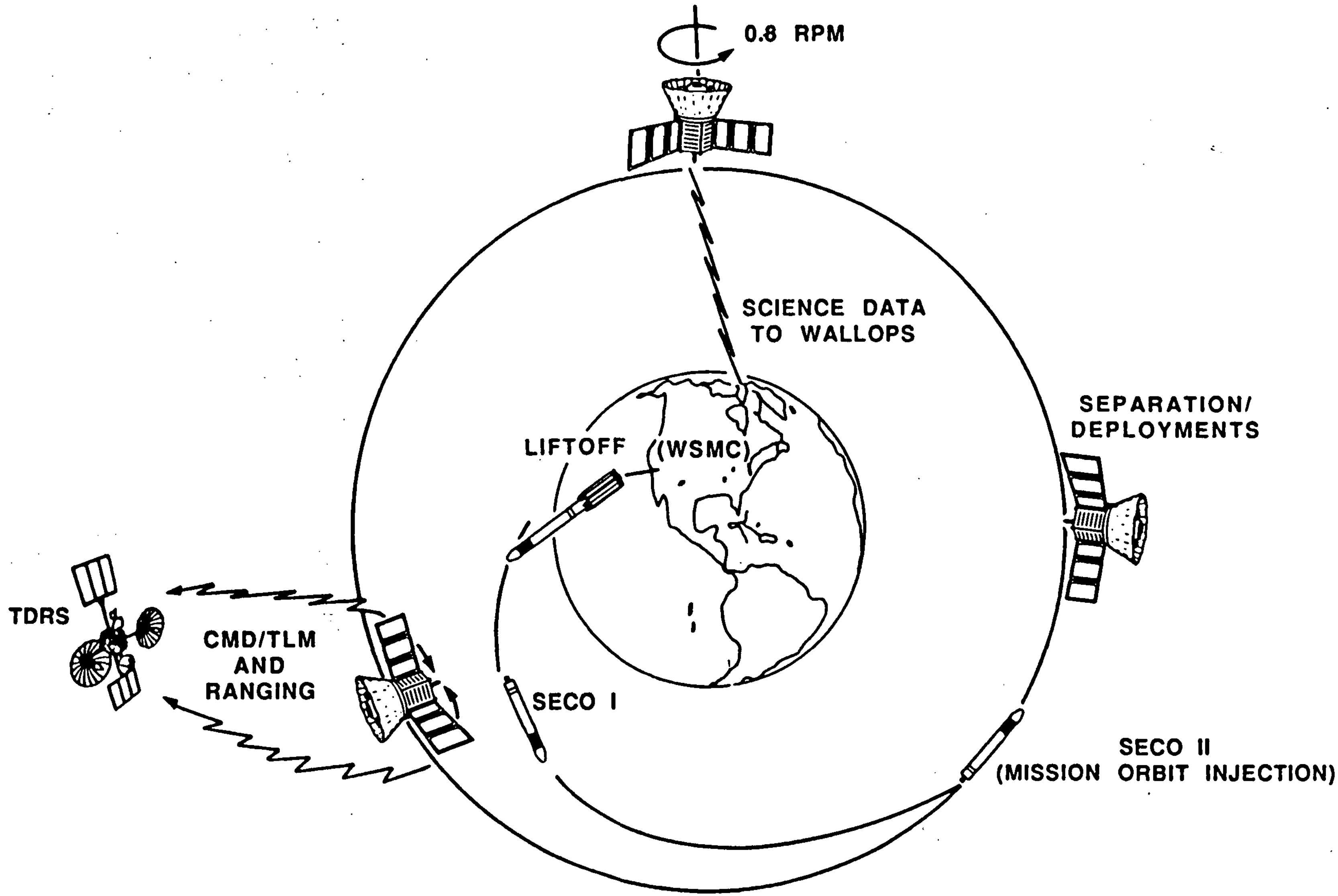
COBE/DELTA LAUNCH PROFILE

1. DELTA ORIENTS COBE WITH PROPER ATTITUDE AND RATES (ZERO SPIN RATE) AFTER SOLAR ARRAY AND THERMAL SHIELD DEPLOYMENT
2. COBE RELEASE FROM DELTA INITIATES TORQUERS, MOMENTUM WHEELS, ANTENNA DEPLOYMENT
3. RUN UP MOMENTUM WHEEL TO SPACECRAFT ROTATION OF 0.3 RPM
4. CAPTURE IN LOCAL VERTICAL WITH CONTROL TORQUERS
5. COMMAND-TO-MISSION MODE
6. SPIN UP SPACECRAFT TO 0.8 RPM
7. ASSESS PERFORMANCE
8. CHECK PITCH-BACK MANEUVER AND ROLL MANEUVER
9. SLOW SPACECRAFT SPIN RATE TO 0.5 RPM
10. PITCH BACK FOR DEWAR COVER EJECTION
11. EJECT DEWAR COVER
12. RETURN TO MISSION MODE
13. SPIN UP SPACECRAFT TO 0.8 RPM
14. START SCIENCE OBSERVATIONS



COBE - Delta Launch Profile

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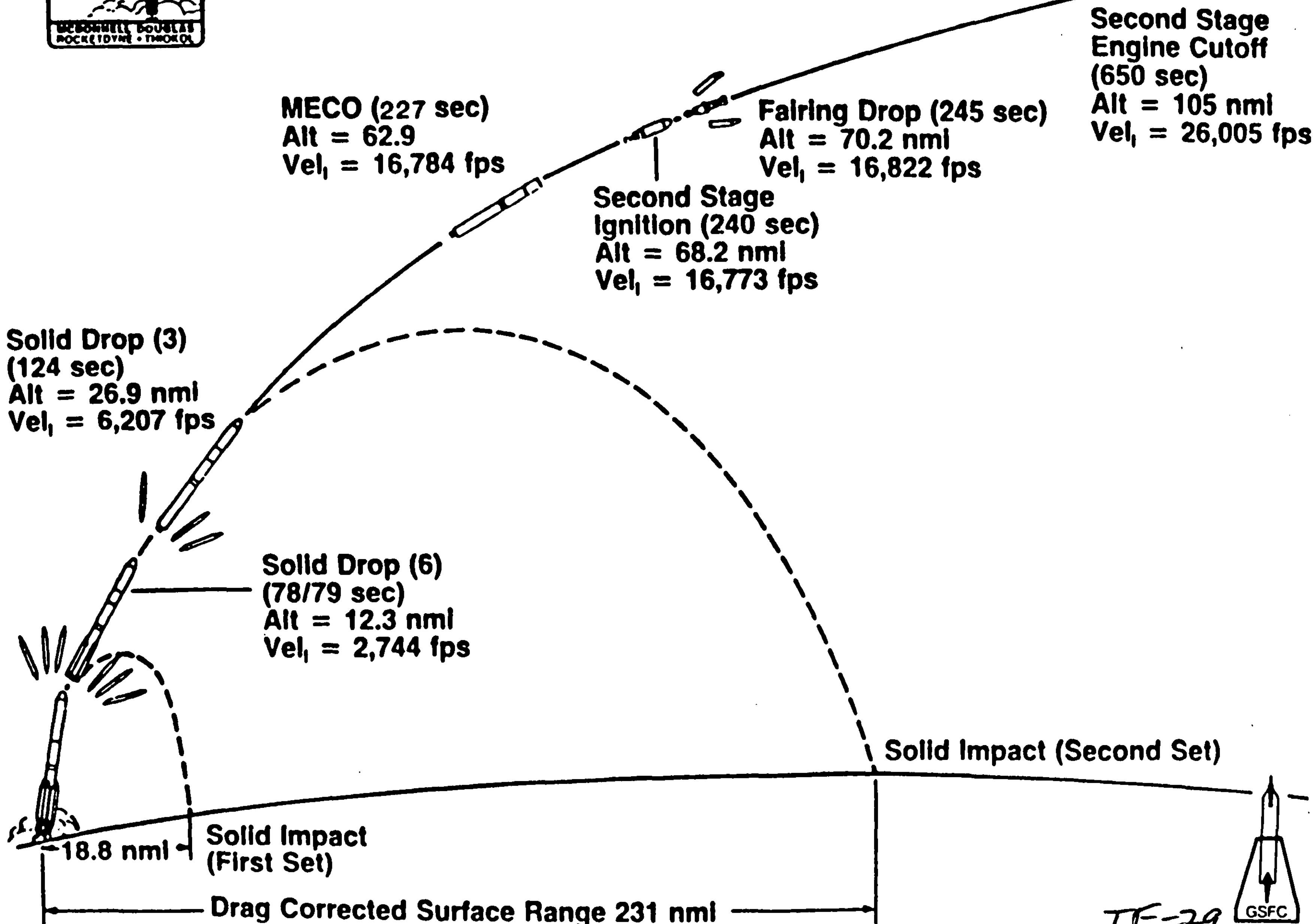
COBE MISSION
TYPICAL BOOST PROFILE



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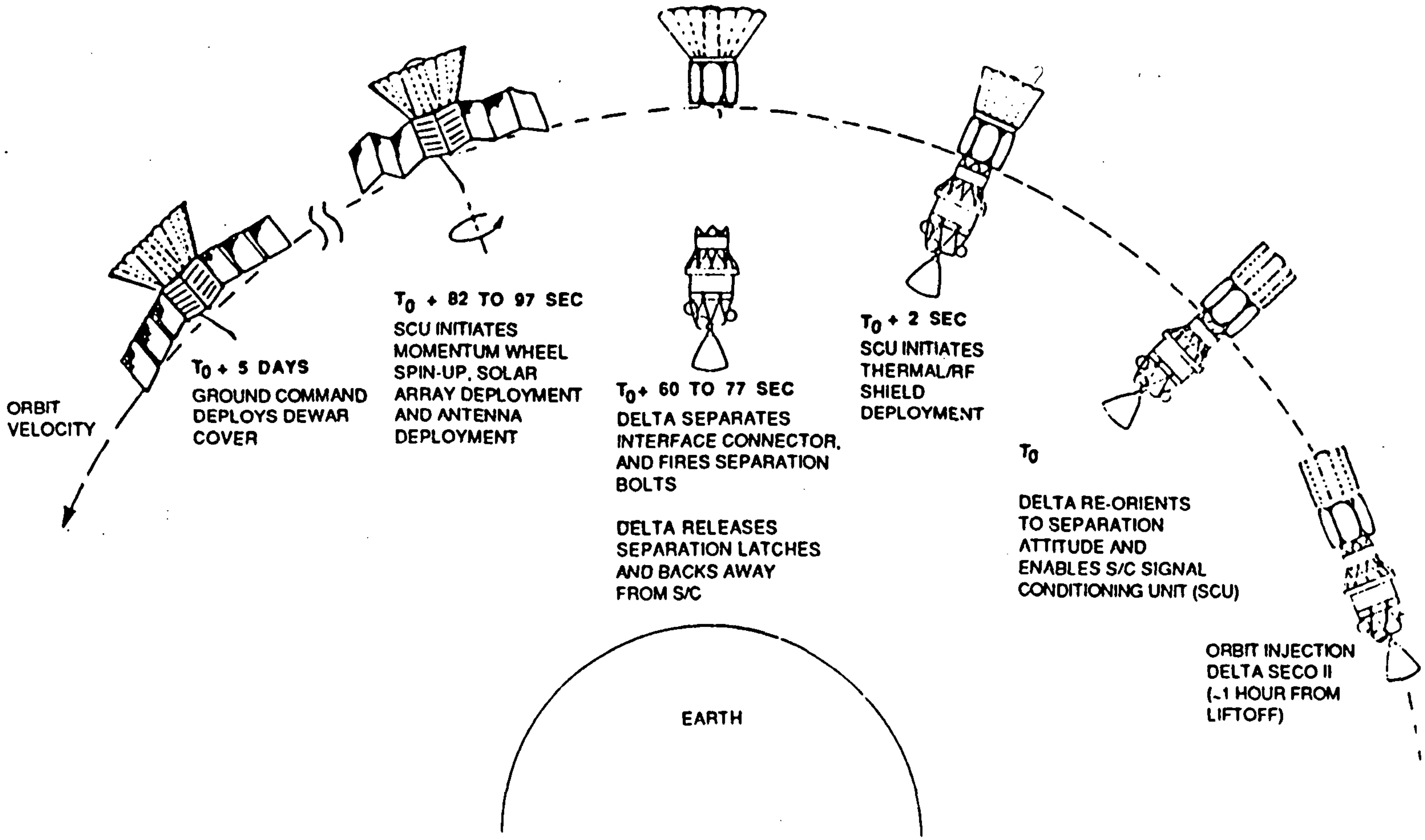

COBE On-Orbit Deployment

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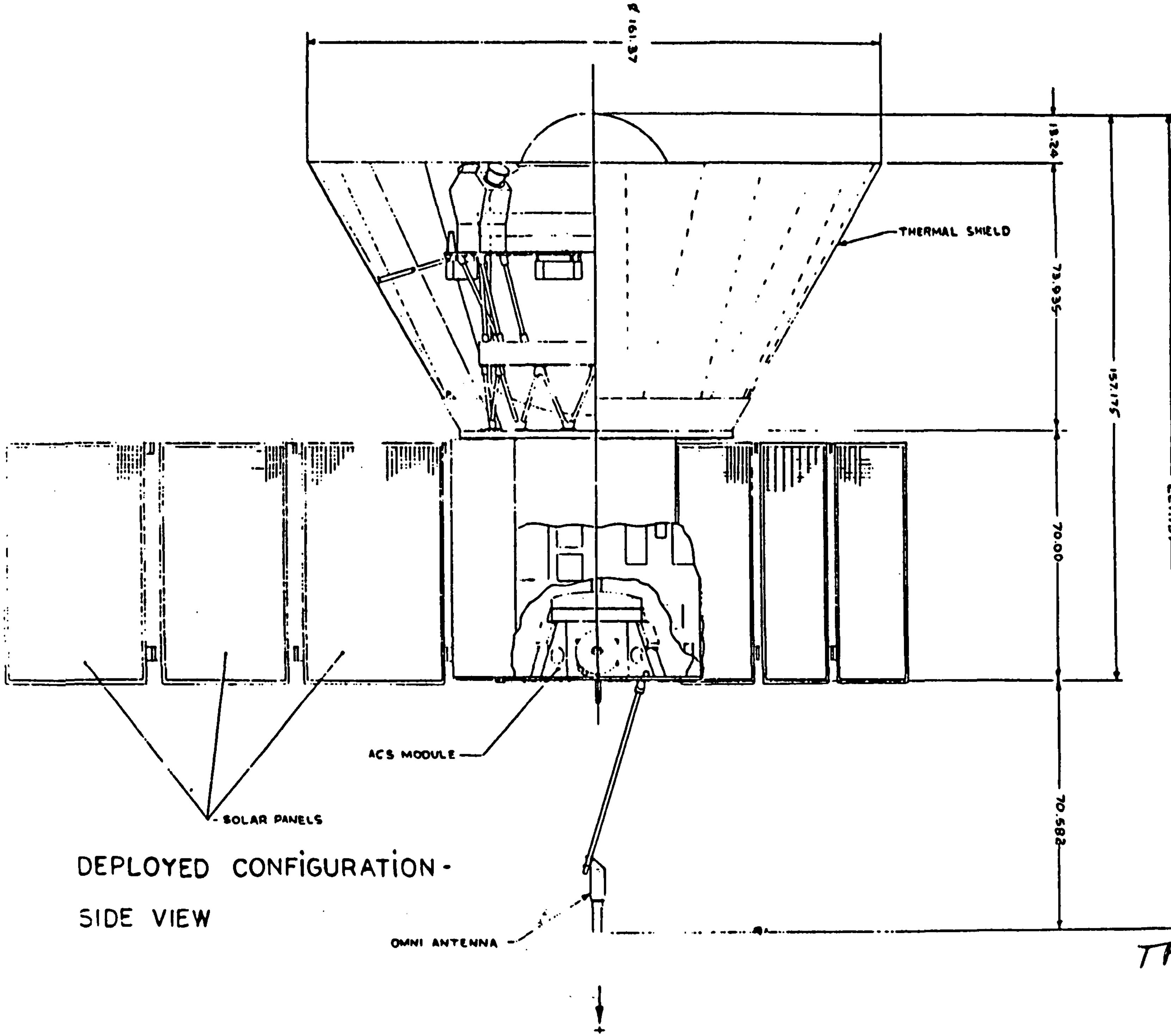

DEPLOYED CONFIGURATION - SIDE VIEW

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COBE SYSTEM REQUIREMENTS

1. DEFINED BY: COBE-SP-401-1004-01, REV. B, AUGUST 1988, "COBE SYSTEM PERFORMANCE SPECIFICATION FOR A DELTA LAUNCH"

2. COMPLIANCE WITH: COBE-PV-401-1004-01, NOVEMBER 1989, "COBE PERFORMANCE VERIFICATION MATRIX"

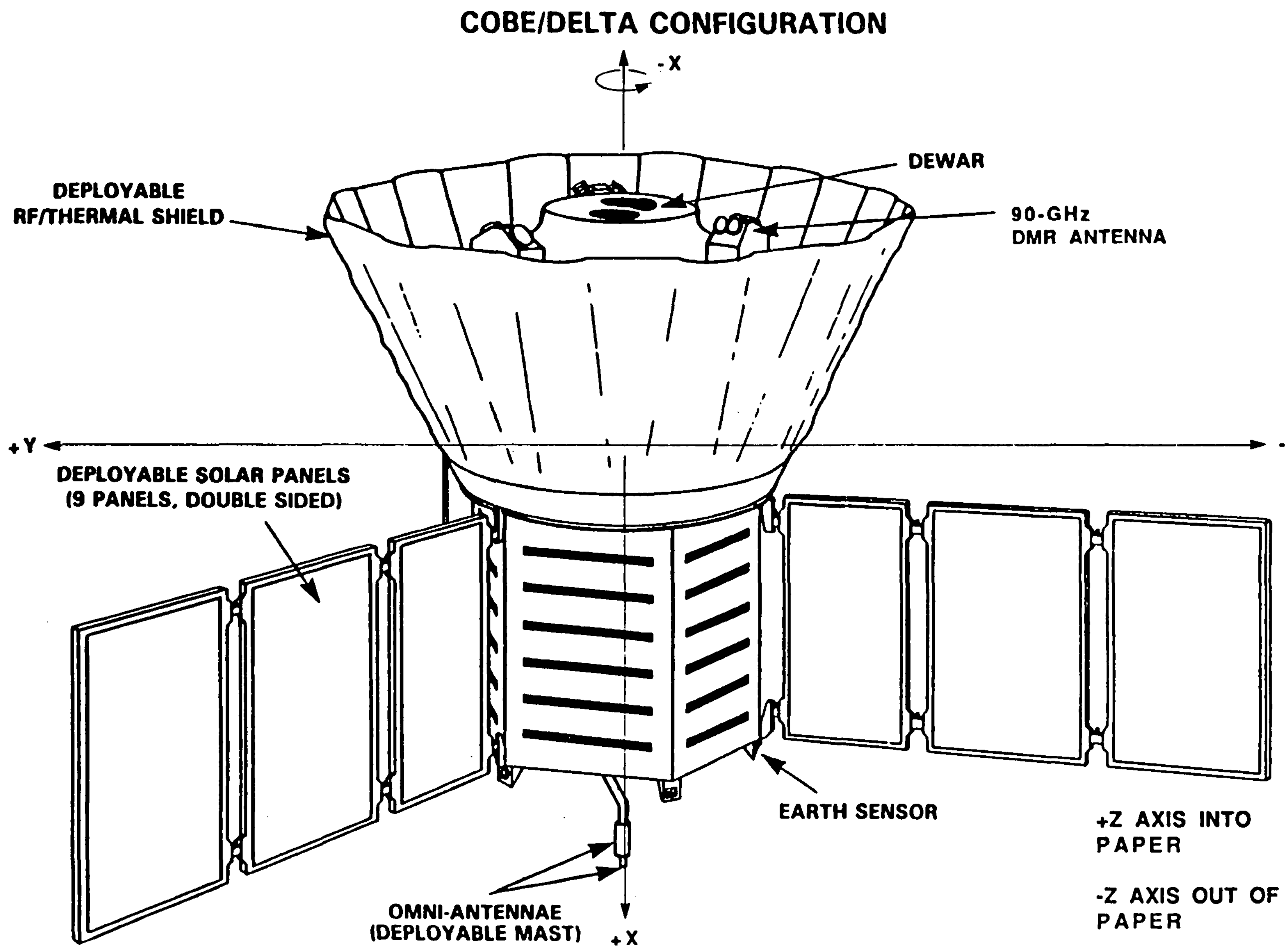

Figure 3-1. COBE Body Fixed Reference Frame

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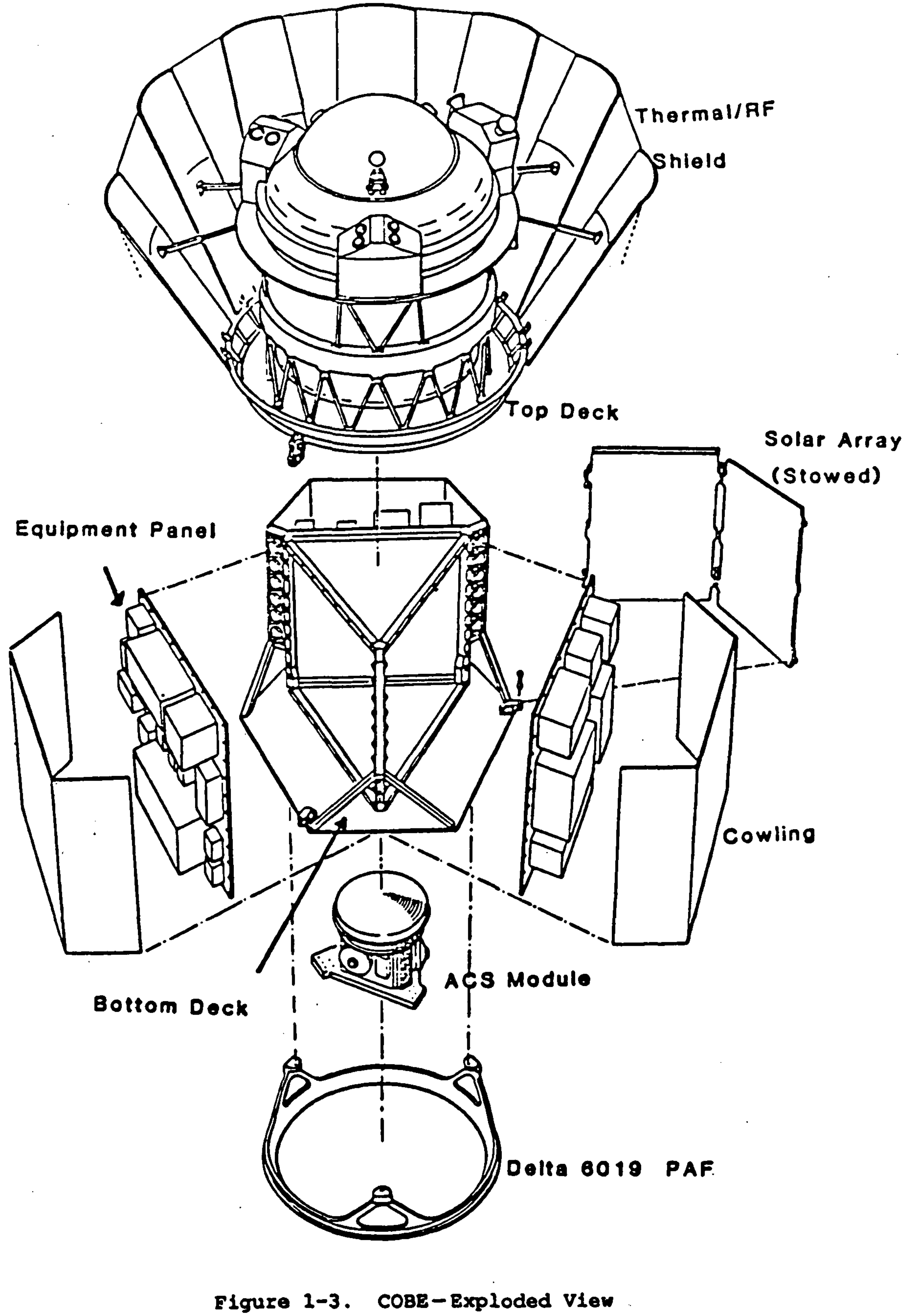

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Figure 1-3. COBE—Exploded View

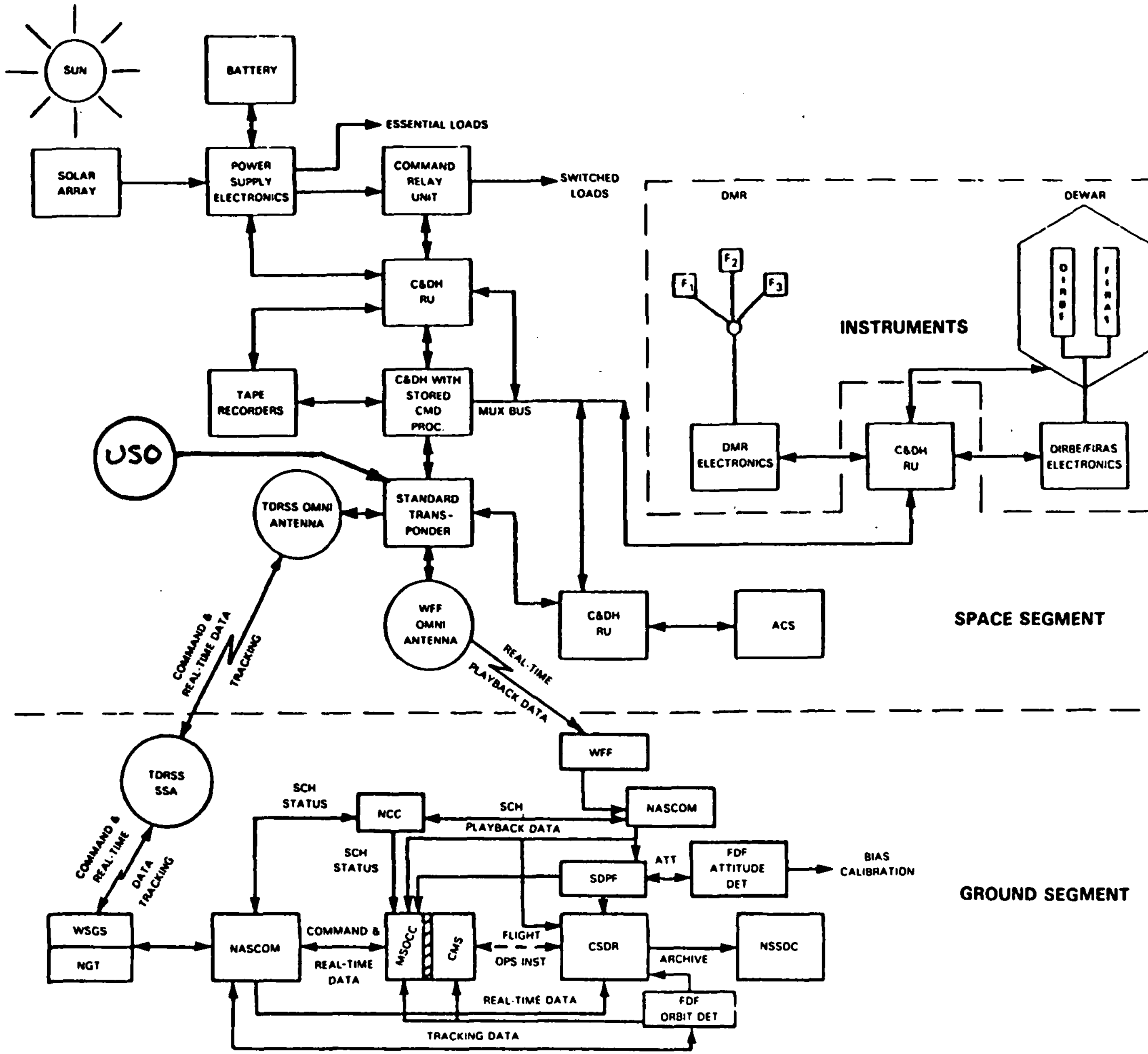

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Figure 1-1. COBE System Diagram

- PROVIDE A 3 AXIS CONTROLLED ROTATING PLATFORM IN THE MISSION MODE TO ENABLE THE FIRAS AND DIRBE TO VIEW NO LESS THAN 80 PERCENT OF THE COSMIC BACKGROUND RADIATION DATA OVER 80 PERCENT OF THE CELESTIAL SPHERE WITHIN A SIX-MONTH PERIOD AND 95 PERCENT OF THE CELESTIAL SPHERE WITHIN A 12-MONTH PERIOD FOR DMR.

- UTILIZE THE TDRSS SSA LINKS FOR COMMUNICATIONS TO AND FROM THE POCC AND GSFC.

- STORE DATA DURING THE MISSION ORBIT AND DUMP THIS DATA ONCE A DAY DIRECTLY TO THE WALLOPS FLIGHT FACILITY GROUND STATION.

- OPERATE AT A NOMINAL MISSION CIRCULAR ALTITUDE OF 900 +9KM, -26KM, SUNSYNCHRONOUS, 0600 OR 1800 HOURS LOCAL MEAN ASCENDING NODAL CROSSING TIME, 99.03° $\pm$ 0.03° INCLINATION, AND MAINTAIN THE NODE CROSSING TIME WITHIN -1/2 HOUR TO +3/4 HOURS FOR A YEAR.

- PROVIDE A HELIUM BATH TEMPERATURE OF 1.6K OR LESS FOR THE OPTICAL PORTIONS OF THE DIRBE AND FIRAS INSTRUMENTS, ALSO REFERRED TO AS THE CRYOGENIC OPTICAL ASSEMBLY (COA) WHEN COMBINED WITH THE INSTRUMENT INTERFACE STRUCTURE (IIS).

- PASSIVELY COOL THE 2 HIGHER FREQUENCY DMR'S TO 140K OR LESS AND MAINTAIN THE LOWER FREQUENCY DMR AT 300$\pm$5K.

- SPIN AT A RATE OF 0.815$\pm$0.015 RPM ABOUT AN AXIS OFFSET FROM AND NEARLY PARALLEL TO THE OBSERVATORY GEOMETRIC X AXIS.

- SPACECRAFT SPIN RATES ABOUT THE -X AXIS SHALL MAINTAIN SAFE THERMAL AND ENERGY BALANCE OPERATION.
- MAINTAIN THE SPIN VECTOR/SUN LINE ANGLE AT 94.0° WITH A ± 1 DEGREE MAXIMUM ERROR.
- KEEP THE SPIN AXIS POINTED GENERALLY AWAY FROM THE EARTH AND WITHIN 6° with a ± 1 DEGREE MAXIMUM ERROR.
- KEEP THE SPIN AXIS POINTED GENERALLY AWAY FROM THE EARTH AND WITH 6° MAXIMUM ERROR OF THE SUN-EARTH NADIR PLANE.
- CAPABILITY TO VARY THE ANGLE BETWEEN THE SPIN AXIS AND SUN VECTOR FROM 90° in 1/2° INCREMENTS.
- CAPABILITY TO VARY THE PITCH ANGLE (BACK ONLY) IN THE ORBIT PLANE FROM 0° TO 30° IN 1° INCREMENTS, AND REMAIN AT ANY STEP FOR UP TO 3 DAYS.
- PROVIDE FOR TIME TAGGED ATTITUDE RECONSTRUCTION (ON THE GROUND) TO AN ACCURACY OF ± 1.0 DEGREES FOR DIRBE AND $\pm 1.0°$ (3σ) RANDOM VARIATION AND 0.2° PEAK PERIODIC VARIATION FOR FIRAS AND DMR, IN EACH INSTRUMENT LINE OF SIGHT REFERENCED TO INERTIAL COORDINATES.
- PROVIDE FOR ATTITUDE RECONSTRUCTION USING COMBINED DIRBE AND SPACECRAFT DATA TO AN ACCURACY OF 3.5 ARC MIN (1σ).

- PROVIDE TIME TAGGED TELEMETRY DATA WITH A RESOLUTION OF ± 1 MILLISECOND AND AN ACCURACY OF ± 10 MILLISECONDS.

- THE OBSERVATORY SHALL BE REDUNDANT WHERE FEASIBLE SO THAT NO SINGLE POINT FAILURE SHALL CAUSE MISSION FAILURE.

- THE UNAMBIGUOUS STATUS OF THE OBSERVATORY SHALL BE TELE-METERED WITHIN 128 MINOR FRAMES HAVING A 32.0 SEC. REPETITION PERIOD FOR THE SCIENCE FORMAT AT 4096 BPS (EXCEPTIONS FOR MULTIPLEXED TELEMETRY FOR INSTRUMENTS AND SCU SHALL BE PERMITTED); FOR EACH COMMAND THAT CHANGES STATUS OR PARAMETERS, THERE SHALL BE AN UNAMBIGUOUS TELEMETERED STATUS CHANGE.

- ALL COMMANDS WHICH CAN LEAD TO FAILURES SHALL BE AT LEAST A TWO-BIT CHANGE FROM ALL OTHER COMMANDS. CRITICAL COMMANDS ARE DEFINED TO BE ANY COMMAND WHICH COULD CAUSE MISSION DEGRADATION OR COULD RESULT IN HAZARDS; ALL CRITICAL FUNCTIONS MUST BE CONTROLLED BY AT LEAST TWO COMMANDS.

- INSTRUMENT SCIENCE OPERATION SHALL NOT BE PERFORMED UNTIL AFTER ARRIVAL AT THE MISSION ORBIT AND CHECKOUT OF THE OBSERVATORY HAS BEEN COMPLETED.

- THE MAXIMUM PERMISSIBLE DOPPLER SHIFT CAUSED BY THE ERROR IN PREDICTING THE OBSERVATORY VELOCITY SHALL BE LESS THAN ± 700 HZ OUT OF A 2287.5 MHZ TRANSMITTER FREQUENCY.

- THE MAXIMUM TIMING ERROR FOR PREDICTING THE OBSERVATORY'S POSITION AT THE TIME OF INTENDED TDRSS ACQUISITION SHALL BE LESS THAN $\pm$9 SECONDS.

- THE DELTA TARGETED ORBIT IS AS FOLLOWS

  (1) ORBIT ALTITUDE--900KM (+9KM, -26 KM), CIRCULAR
  (2) INCLINATION--99.03° $\pm$ 0.03°
  (3) SUNSYNCHRONOUS--0600, -15 MINUTES + 30 MINUTES OR 1800 HOURS, -15 MINUTES +30 MINUTES LOCAL MEAN ASCENDING NODE CROSSING TIME
  (4) LAUNCH--ANY DAY OF YEAR

- ALL SOLAR ANGLES FORWARD OF THE COBE Y-Z PLANE SHALL BE MINIMIZED WHILE ON THE DELTA AND DURING THE MISSION LIFETIME TO PREVENT UNNECESSARY HEATING OF THE DEWAR COVER AND MAINSHELL. NO ILLUMINATION OF THE MAINSHELL SHALL OCCUR AFTER DEWAR COVER EJECTION.

- <u>RELIABILITY AND SINGLE POINT FAILURES (SPF)</u>--MINIMIZE SINGLE POINT FAILURES--NO SPF SHALL CAUSE A MISSION FAILURE. MAXIMIZE OPERATIONAL WORKAROUNDS.

  (SEE VERIFICATION MATRIX FOR LIST OF SPF'S AND RATIONALE FOR ACCEPTANCE)

- <u>CONTAMINATION</u>--DMR RECEIVER THROAT 300A
  DIRBE PRIMARY MIRROR 100A
  DIRBE OTHER SURFACES 500A
  FIRAS SKYHORN CALILB. 200A
  FIRAS OTHER SURFACES 300A

- AUTONOMOUS FOR UP TO 20 HOURS

  - ONE MOMENTUM WHEEL (LATER ADDED SECOND WHEEL)
  - DEWAR OVER-TEMP SENSING
  - ACE POWER SUPPLY SWITCH-OVER (AUTO)
  - NON-ESSENTIAL LOAD REMOVAL
  - DIRBE SHUTTER CLOSING
  - LOSE ONE ACS LOOP AND MAINTAIN CONTROL

- MAXIMIZE THE USE OF FAIL-SAFE DESIGNS AND PROVIDE THE NECESSARY INHIBITS.

- PREVENT THE GENERATION OF UNWANTED OUTPUT SIGNALS AND PREVENT DEGRADATION OF PERFORMANCE OF ASSOCIATED EQUIPMENT DUE TO POWER FAILURE, INTERNAL CIRCUIT FAILURE, COMPONENT FAILURE, NOISE, RADIO INTERFERENCE, ELECTRICAL TRANSIENTS, OR ENVIRONMENTS.

- <u>COMPONENT SHELF LIFE</u>--COMPONENTS SHALL BE DESIGNED FOR A <u>MINIMUM SHELF LIFE OF</u> 4 YEARS.

- <u>RADIATION ENVIRONMENT</u>
  - ELECTRONIC PARTS SHALL WITHSTAND A MINIMUM TOTAL DOSAGE OF ELECTRON AND PROTON BOMBARDMENT OF 4 X 10^3 RADS WITH 3/32" WALL THICKNESS. (TOP DECK 5 X 10^3 RADS.)

- <u>ELECTROMAGNETIC COMPATIBILITY</u>--OBSERVATORY SHALL BE DESIGNED FOR ELECTROMAGNETIC SELF COMPATIBILITY AND FOR ELECTROMAGNETIC COMPATIBILITY WITH THE INSTRUMENTS AND THE DELTA FOR ALL PHASES OF THE MISSION. MIL-STD-461 SHALL BE USED.

- <u>MAGNETIC REQUIREMENT</u>
  - D-C MAGNETIC FIELD PRODUCED BY ANY COMPONENT
    - $\leq$ 0.2 AM^2 DIPOLE MOMENT
    - $\leq$ 0.1 AM^2 DIPOLE MOMENT AFTER 50 X 10^{-4} TESTS DEPERM
  - INDIVIDUAL SHIELDING OF BOXES

- <u>FRACTURE CONTROL</u>--EACH DESIGN WILL PRECLUDE FAILURE OF ANY ATTACHMENT BOLTS AND CONTAINMENT OF ANY HARDWARE $>$0.03 POUNDS.

- <u>MICROPHONICS</u>
  - FIRAS MTM $<$ 3.3 X 10^{-12} F^3 G^2/HZ
  - DMR $<$ .01 GRMS/HZ (10 HZ TO 1000 HZ)

- <u>CORONA DISCHARGE</u>--CAN BE NEITHER THE SOURCE OF OR SUSCEPTIBLE TO.

- <u>CHARGING/DISCHARGING</u>--SURFACE MATERIAL AND FINISHES SHALL BE SELECTED TO MINIMIZE THE EFFECTS OF CHARGING AND DISCHARGING. (SHOULD HAVE BEEN BETTER DEFINED.)

VERIFICATION

- o DESIGN QUALIFICATION BY ANALYSIS AND/OR TEST
- o COMPONENT/SUBSYSTEM ACCEPTANCE TESTS
  - FUNCTIONAL
  - EMC
  - VIB
  - T/V
- o OBSERVATORY ACCEPTANCE TESTS
  - FUNCTIONAL
  - EMC
  - ACOUSTIC/RANDOM
  - TV AND TB
- o 200 TROUBLE-FREE TEST HOURS
  - 100 HOURS FOLLOWING OBSERVATORY EMC
  - 100 HOURS AFTER OBSERVATORY ACCEPTANCE TESTS
- o TREND DATA COLLECTION FOR SELECTED PARAMETERS

VERIFICATION (CONTINUED)

- o VERIFICATION MATRIX 1 TO 1 WITH SYSTEM SPEC.
- o ETC MET WEEKLY TO REVIEW/APPROVE ALL OBSERVATORY LEVEL MALFUNCTIONS/PROBLEMS, WAIVERS/DEVIATIONS, AND TEST PLANS AND PROCEDURES.

ANALYSES REQUIRED

- o STANDARD STRUCTURAL ANALYSES
- o STANDARD THERMAL ANALYSES
- o FMECA'S
  - TOTAL FLIGHT SYSTEM
  - ANY GSE DIRECTLY INTERFACING WITH FLIGHT OBSERVATORY
- o SYSTEM SAFETY ANALYSES
- o OPERATIONAL HAZARDS ANALYSES (OHA) FOR ALL MAJOR TESTS
- o CIRCUIT ANALYSES:
  - PARTS STRESS ANALYSES
  - WORST CASE ANALYSES
  - "SPICE"

REQUIREMENT	ASSESSMENT	PERFORMANCE SHORTFALL	WAIVER RATIONALE

II. SUMMARY OF EXCEPTIONS TO REQUIREMENTS

A qualitative assessment of each subrequirement has been made. A status/resolution summary of items rated as "unacceptable" and "acceptable" follows:

3.3 Mission Performance and Operational Requirements

3.3.1 Mission Orbit

REQUIREMENT	ASSESSMENT	PERFORMANCE SHORTFALL	WAIVER RATIONALE
o Maintain the spin vector/ sun line angle at 94.0° within $\pm$ 1 degree.	Valid only during sunlit periods.	This angle will experience transients up to 4° during maximum shadow time in eclipse season.	Acceptable. Total time in shadow is small compared to overall mission time. Resulting angles can be accomodated in processing of science data.
o The observatory shall be redundant where feasible so that no single point failure shall cause mission failure. and **Reliability and Single Point Failure** (From Section 8.2) o Each mission critical failure mode shall be investigated using design analysis, historical failure data, failure mode effects and criticality analysis (FMECA), and past experience.	Meets Requirement by FMECA Analyses and Structural Tests, except for agreed-to single point failures listed as performance shortfalls.	**Agreed-to Single Point Failures:** 1. RF Transfer Switches 2. Dewar Plumbing and Valves (V4 failure, Porous Plug) 3. Unfused Portions of Essential Bus 4. X-Axis Gyro in Eclipse (Recoverable - 2 ea on) 5. ACS Power Supply in Eclipse (Recoverable) 6. ACE in Eclipse (Recoverable) 7. PSE Current Measuring Shunt Failure 8. MTM Mechanical Failure 9. DIRBE Shutter Mechanical Failure 10. DIRBE Chopper Mechanical Failure 11. FIRAS Excal Mechanical Failure 12. Loss of Battery Continued...	The listed single point failures have been agreed to during PDR's, CDR's, documentation, and other reviews without CCR's. Reference: Memo from J. Turtil to distribution dated October 22, 1989; Subject: Certification of COBE Single Point Failures.

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REQUIREMENT	ASSESSMENT	PERFORMANCE SHORTFALL	WAIVER RATIONALE
Reliability and Single Point Failure (Continued)		Agreed-to Single Point Failures (Con't) 13. Failure of Deployables (Pyros are Redundant) RF/Thermal Shield Solar Array Panels Dewar Cover Antenna Boom 14. Structure 15. Dewar 16. Resistor Failure in PSE Charge/Shunt Drive could result in out-of Spec Performance.	
3.3.1 Mission Orbit (Continued)			
o For each command that changes status or parameters, there shall be an unambiguous telemetered status change.		Requirement met except for approximately 1% of commands. Example: pyro FIRE command.	Acceptable. The changes resulting from these commands can be inferred from other telemetered data.
o The maximum timing error for predicting the Observatory's position at the time of intended TDRSS acquisition shall be less than ± 9 seconds.		Effects of helium venting may require more frequent updates.	Acceptable as indicated.

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REQUIREMENT	ASSESSMENT	PERFORMANCE SHORTFALL	WAIVER RATIONALE
3.4 Instrument Performance Requirements			
3.4.1 Scientific Observation Requirement			
3.4.1.1 DIRBE Measure the spectrum and angular distribution of the diffuse infrared background radiation (Lambda I $_{\text{Lambda}}$) to a sensitivity of 10^{-13} watts per square centimeter steradian, or 1% of the zodiacal scattering, or emmission, whichever is greater, at wavelengths from 1 to 300 micrometers for each 0.7 square degree field of view.		Project CCR #725: Band 9 (120-200 microns) factor of 3 less sensitive than requirement.	Accepted by Project CCB. Replacement of detector is severe programmatic impact. Although sensitivity is not met, data can provide meaningful science. Reference: Letter from NASA HQ/Code E to NASA GSFC/Code 100 dated November 14, 1989; Subject Waiver of Level 1 Science Requirements for COBE DIRBE Band 9.
	DIRBE Band 8 heater failed.	Inability to closely regulate temperature of detector and anneal detector following radiation impact.	Accepted for flight by Project. Plan to overbias detector in orbit to reduce effects of nuclear radiation. Reference: Information presented by J. Mather at the COBE Spacecraft Flight Readiness Review on November 11, 1989: "DIRBE Band 8 Heater Anomaly."

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REQUIREMENT	ASSESSMENT	PERFORMANCE SHORTFALL	WAIVER RATIONALE
3.5.1.2 Deployables			
3.5.1.2.1 Thermal/RF Shield			
o Shield Temperatures: Survival: -70 to 200°C Operational: -30 to 50°C Internal: ≤ 240K	T/RF shield cannot be exposed to +200°C without degradation of honeycomb bond.	None is expected. The 200° C requirement is a carryover from STS; a maximum temperature of less than 65° C is predicted.	Acceptable as indicated.
3.5.3 Attitude Control			
o The Observatory's orientation shall preclude the Sun from illuminating the dewar cover or main shell for extended periods. No illumination of the main shell shall occur in mission mode.	Requirement not satisfied in vertical mode.	Not a mission impact since vertical mode is planned only during the first few orbits.	Acceptable. Vertical mode operations occur prior to dewar cover ejection.
3.5.5.1 Instrument Operating Temperatures			
o FIRAS detectors shall be maintained below 1.6K.	Cannot be maintained below 1.6K if Dewar cryogen tank is at 1.6K. (Ref. Section 3.5.2 in Section III)	None is expected. The Dewar cryogen tank is predicted to run at 1.45 k in mission orbit. FIRAS requirement will be met with this condition.	Acceptable as indicated.

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REQUIREMENT	ASSESSMENT	PERFORMANCE SHORTFALL	WAIVER RATIONALE
3.5.8 Electrical			
o The signal and power returns shall be isolated from each other in each electronic assembly.	Frequency Standards do not meet requirement.	None observed.	Existing flight hardware design used. Acceptable performance noted in test of Observatory.
o The power returns shall be isolated from the chassis of each electronic assembly.	Frequency Standards do not meet requirement.	None observed.	Existing flight hardware design used. Acceptable performance noted in test of Observatory.
o Except for the instrument electronics, the signal returns shall be isolated from the chassis of each electronic assembly.	Requirements are generally met inside the assembly, but signal grounds are tied to chassis outside the assembly.	Performance actually improved.	The original requirement was incorrect.
o The frame shall not be used to return primary current to the power supply.	Frequency Standards and Momentum Wheel Electronics Assemblies do not meet requirement.	Possible increase in system noise signature.	Existing flight hardware design used. MWEA potential defect minimized by shortening current path to PSE. Acceptable performance noted during Observatory testing.

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REQUIREMENT	ASSESSMENT	PERFORMANCE SHORTFALL	WAIVER RATIONALE
3.5.8 Electrical (Continued)			
o Fusing shall be accomplished by means of redundant fuses (in parallel), with each fuse capable of carrying at least 150% of maximum steady state current.	Redundant fusing is not used for all Power Supply Electronics (PSE) internal circuits. Ref: Memo from D. Manzer to COBE Observatory Manager, "PSE Fuses for Flight and SPF Justification," dated November 7, 1989.	Battery charger circuit failure.	Acceptable. Can lose 4 of the 12 battery charger circuits without mission impact.
		Shunt dissipator circuit failure.	Acceptable. Normal dissipation can be accomodated with 15 of the 24 shunt circuits.
		Boost converter circuit failure.	Acceptable. Normal spacecraft operations can be supported by 2 of the 3 boost converters.
COBE ALIGNMENT REQUIREMENTS SEE SECTION III (From TABLE 3-6)			
DIG SUN SENSOR: Stability 0.05° EARTH SCANNER: Stability 0.05°		Waivers granted to DSS's and ESA's for alignment stability in excess of requirement (as result of static load tests). Project CCR #729.	Approved by Project CCB. Out-of specification condition is minor and is not expected to affect mission performance.
DMR Heads: Placement 0.5°		DMR 31 GHz Head does not meet placement requirement (tilt and horn pair plane).	Acceptable as is. Errors are not considered large enough to merit repositioning. Agreement by Project and Principal Investigator.

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REQUIREMENT	ASSESSMENT	PERFORMANCE SHORTFALL	WAIVER RATIONALE
4.5 Dewar GSE			
o Maintain a vacuum on the dewar while the dewar is on the Delta prior to launch.	Pump must be disconnected at 12 hours prior to launch.	Reduction in Dewar helium lifetime as a result of launch delays.	Acceptable based on launch abort/turnaround plan to minimize impact. After 3 launch attempts (3 days), Dewar topoff and pumpdown will be required.
5.9 Observatory Pre-Flight Acceptance Tests			
o Prior to launch, the Observatory and its elements must successfully pass 200 trouble-free hours of operation.	Ref. "Verification Matrix for the COBE Spacecraft", dated September 28, 1989. (Appendix A to COBE-SP-750-1702-02, <u>Verification Plan and Specification for COBE.)</u>	Items not expected to meet 200-hour trouble-free requirement prior to launch: Tape Recorder 1 Tape Recorder 2 Instrument Telemetry Unit 1	Acceptable to Project. Changes made to Tape Recorders and Instrument Telemetry Unit were minor and operation of the Spacecraft for the sole purpose of accumulating trouble-free time was concluded to be inappropriate.

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REQUIREMENT	ASSESSMENT	PERFORMANCE SHORTFALL	WAIVER RATIONALE
8.2 **Reliability and Single Point Failures**			
o Redundant circuits shall be routed through separate connectors and wire bundles	Generally not true for spacecraft electrical assemblies.		Implemented where deemed necessary (e.g. pyro circuits).
8.7 **Corona Suppression**			
Electrical and electronic equipment shall be designed so that it is neither the source of, nor is susceptible to, corona discharge.		Diplexer #2 is known to be susceptible to corona discharge.	Acceptable. Transmitter will be "off" for launch; transmitter #2 to be turned on no earlier than 1 hour after launch
8.13 **Electrical Connectors**			
o Separate connectors shall be provided for the functions of power, data/commands, and telemetry.	Generally true for power vs. data/commands and telemetry; but, generally not true for data/commands vs telemetry.	None observed.	Requirement was implemented where deemed necessary (e.g. instrument detectors). Generally not implemented because of practical limitation.
o All connectors shall be keyed; connectors on the same black box having the same shell size shall be keyed differently.	Connector keying was not implemented in component (block box) and harness design.	Mistakes during integration and test leading to degradation of flight hardware.	Used flight hardware already designed. Design of flight harnesses minimizes mismate possibilities. Emphasis placed on procedures and training of personnel to insure proper mating of external harnesses and plugs.

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8.14 Magnetic Compatibility

REQUIREMENT	ASSESSMENT	PERFORMANCE SHORTFALL	WAIVER RATIONALE
o The DC magnetic field produced by any component (except for torquer bars), including any of its operating modes, shall not exceed 0.2 am^2 dipole moment following its manufacture, 0.3 am^2 after exposure to a 5×10^{-4} tesla magnetic field and a 0.1 am^2 moment after a 50×10^{-4} tesla deperm.	With minor exception component testing was deferred to Observatory level of test.	None. System requirements are acceptable.	Acceptable as indicated.
o The assembled Observatory shall not exceed 3.0 am^2 dipole moment following its manufacture, 5.0 am^2 after exposure to a 5×10^{-14} tesla magnetic field and a 2.0 am^2 moment after a 50×10^{-4} tesla deperm.	Magnetic requirement relaxed based on analysis by Code 712. ("Residual Dipole Effect Upon the COBE Attitude Control Subsystem," memo by S. Placanica dated October 13, 1988) Waiver for test tolerance based on relaxed requirement.	Negligible affect on control system.	Acceptable based on Observatory testing.

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REQUIREMENT	ASSESSMENT	PERFORMANCE SHORTFALL	WAIVER RATIONALE
o The Observatory and each of its components shall not produce magnetic fields due to internal current flows in excess of 0.5 and 0.05 am^2, respectively.	Magnetic requirement relaxed based on analysis by Code 712.	Negligible affect on control system.	Acceptable based on Observatory testing.
8.19 Microphonics			
Mechanical noise generated by any component within the observatory shall not cause degraded performance of any subsystem or instrument.		Instrument interference is noted when either Momentum Wheel is above 2000 RPM.	Both wheels can be used in complementary manner to control Observatory spin without either being above 2000 RPM. In addition, in orbit ajustments for optimizing microphonic effects are possible.

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LESSONS LEARNED

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CODE 400 RECOMMENDATIONS FOR FUTURE PROGRAMS

1. INSUFFICIENT PARALLEL ANALYSES OF CRITICAL SUBSYSTEMS
2. LACK OF ONE-ON-ONE PEER REVIEW BY GOVERNMENT TECHNICAL PERSONNEL
3. LACK OF ONE-ON-ONE PEER REVIEW BY CONTRACTOR TECHNICAL PERSONNEL
4. FAILURE TO RIGOROUSLY CONTROL PLANNED TEST PROGRAM
5. INSUFFICIENT ANALYSES OF LIFE-SENSITIVE COMPONENTS AND ASSOCIATED LIFE TESTING RATIONALE
6. INSUFFICIENT MAINTENANCE OF FUNCTIONAL TEST LOGS
7. INSUFFICIENT TROUBLE-FREE FUNCTIONAL OPERATING TIME OF CRITICAL SUBSYSTEMS PRIOR TO LAUNCH
8. INAPPROPRIATE IMPLEMENTATION OF REDUNDANCY
9. INSUFFICIENT TREND ANALYSIS DURING TEST PROGRAM.
10. INSUFFICIENT WORST-CASE ANALYSES

COBE LESSONS LEARNED--ENGINEERING

- CONTAMINATION CONTROL; A COST AND SCHEDULE DRIVER; DEFINE REAL REQUIREMENTS

  --ESTABLISH BUDGET EARLY BY SYSTEMS ENGINEERS AND PI'S
  --PLAN ON BLACK BOX BAKEOUTS AS PART OF BOX-LEVEL ACCEPTANCE TESTS.

- HI-FIDELITY HARDWARE; BUILD TWO OF ALL PRIMARY STRUCTURES

- MECHANISMS; THE KEY TECHNOLOGY PROBLEM

- TEST DEWARS; NEED LOTS, SOME VIBRATABLE; DESIGN LIFETIMES INTO THE VIBRATABLE ONES

- COLD STRUCTURAL QUALIFICATION

- INSTRUMENT COST AND SCHEDULE DOUBLED FROM INITIAL ESTIMATES; WHY? SIGNIFICANTLY UNDERESTIMATED THE JOB; CRYOGENICS; RETEST RATES; CONTAMINATION CONTROL; TEST HARNESSES AND BLACK BOXES

- NOT BEING ABLE TO FORESEE THAT TEST CONDITIONS WOULD EXCEED THE IN-ORBIT ENVIRONMENT, E.G. HONEYCOMB IN TV/TB AND EARTH SCANNERS IN TV AND TB--COULD SEE CHAMBER WALLS--TOO WARM.

- INSTRUMENT TEST DEWAR HAD TO BE REDESIGNED DUE TO FATIGUE (STRESS CORROSION) AND HIGH "G" LOADS DURING VIBRATION TEST.
- NEED MORE EMPHASIS EARLY-ON REGARDING STRUCTURAL TOOLING AND MANUFACTURING SIMULATORS AND GSE.
- FACTOR "G" NEGATION INTO THE DESIGN EARLY.
- NEED MORE SYSTEM ANALYSES, E.G. FMECA'S, WORST-CASE ANALYSIS, PARTS STRESS ANALYSES, AND "SPICE" BEFORE WE START CUTTING HARDWARE; I.E. START EARLY.
- ANALYSIS OF TEST DATA--COBE INSTRUMENTS REQUIRED UP TO TWO WEEKS TO ANALYZE DATA. THIS PRECLUDES LEGITIMATE REAL-TIME PASS/FAIL CRITERIA.
- PACKAGING DESIGN
  - STS DESIGNED--TOO COMPLEX, TOO LARGE--PCB'S, DIFFICULT TO HANDLE, COULD CAUSE THERMAL PROBLEMS.
  - ELECTRONIC BOXES THAT WEIGH OVER 100 POUNDS.
  - DESIGN TO MAKE SURE GROUNDING, EMI, THERMAL, SHIELDING, STRUCTURE WERE CONSIDERED AND CAN IT BE EASILY REPAIRED AND TESTED.

- MORE SYSTEMS DESIGN
  - BLACK BOXES TOO COMPLEX AND HEAVY
  - BETTER POWER DISTRIBUTION AND CONTROL
  - GSE, HANDLING AND TRANSPORTATION MUST BE CONSIDERED DURING INITIAL DESIGN.
  - S/C COOLING DURING GROUND OPERATIONS
  - EASIER MECHANICAL INTERFACE WITH LAUNCH VEHICLE

- MISSION OPERATIONS
  - START DETAILS VERY EARLY
  - OPERATING HANDBOOKS FOR EACH BLACK BOX/SUBSYSTEM
  - ISOLATION OF CRITICAL COMMANDS
  - LAUNCH AND EARLY ORBIT OPS VERY CRITICAL

- TUTORIALS AND PROJECT CONDUCTED DESIGN REVIEWS
  - START TUTORIALS EARLY ON DESIGN CRITERIA AND OPERATIONS.
  - CONDUCT MORE PROJECT-CHAIRED REVIEWS STARTING WITH CONCEPTS, PRELIMINARY DESIGN, ETC.
  - MANUFACTURING, FABRICATION AND ASSEMBLY REVIEWS
  - TEST REVIEWS

- CRYOGENICS--REQUIRES MUCH MORE ENGINEERING DESIGN TO BUILD HARDWARE AT ROOM TEMPERATURE AND THEN FUNCTION AT CRYOGENIC TEMPERATURE. TAKES MORE TIME TO TEST. RETEST RATES APPROACHED 100 PERCENT--SCHEDULE DRIVER.

- ADHESIVES--THERE IS NOT A UNIFORM UNDERSTANDING OR TABULATION OF PROPER ADHESIVES AT THEIR REQUIRED SERVICE TEMPERATURES.

- PARTS--MUST BE CONCERNED WHEN CHOOSING PARTS TO SELECT THOSE WITH THE HIGHEST RADIATION TOLERANCE. DR. TRAINOR'S LECTURE ON SINGLE EVENT UPSETS MUST BE HEARD EARLY-ON BY ALL.

- FASTENERS--FAILURE TO ANALYZE FOR STRENGTH AND ACCOUNT FOR STRESS CORROSION/CRACKING.

- EARLIER VERIFICATION OF INTERFACES AND PERFORMANCE ON ACTUAL HARDWARE, E.G., EVEN BREADBOARDS AND ENGINEERING MODELS.

- NOT ENOUGH ATTENTION PAID EARLY ENOUGH TO THE DETAILS OF TESTING AND THE SUBTLETIES THERE-OF. EXAMPLES:

  - THE TV-TB THERMAL SHIELD (TEST) TEMPERATURES WOULD EXCEED THE ON-ORBIT AND DESIGN LIMITS.

  - THE EARTH SCANNER WOULD EXCEED THE ON-ORBIT AND DESIGN LIMITS AND EMI TYPE PROBLEMS INDUCED BY THE FACILITY IN THE SCREEN ROOM AND SES.

- TOOK FOREVER AND A DAY TO POWER UP AND VERIFY THE STATUS OF THE OBSERVATORY. NEED A MORE FLEXIBLE C&DH SYSTEM. GROUND TEST REQUIREMENTS SHOULD HAVE BEEN FACTORED INTO DESIGN OF THE C&DH SYSTEM TO ENABLE US TO SPEED UP THE SAMPLING RATES AND CONCENTRATE GREATER PORTIONS OF THE FORMAT IN SPECIFIC AREAS AND FOR SPECIFIC TIMES.

- ONE-ON-ONE PEER REVIEW BY TECHNICAL PERSONNEL--LOOK AT THE DRAWINGS, NOT VU-GRAPHS.

- PARALLEL ANALYSES OF CRITICAL SUBSYSTEM (APL REVIEW OF ACS)

- RIGOROUSLY CONTROL PLANNED TEST PROGRAM

- ANALYSES OF LIFE-SENSITIVE COMPONENTS AND ASSOCIATED LIFE TESTING RATIONALE

- MAINTENANCE OF FUNCTIONAL TEST LOGS

- TROUBLE-FREE FUNCTIONAL OPERATING TIME OF CRITICAL SUBSYSTEMS PRIOR TO LAUNCH

- IMPLEMENTATION OF REDUNDANCY AND OPERATIONAL WORKAROUNDS

- TREND ANALYSIS DURING TEST PROGRAM--START AT BLACK BOX

- o EMI/EMC

  - WITH SENSITIVE INSTRUMENTS, WE ARE ATTEMPTING TO MEASURE THAT WHICH WE DO NOT WANT ON THE SACECRAFT; I.E. "NOISE.
  - DID NOT WORRY ENOUGH ABOUT THIS EARLY-ON.
  - DEFINITIVE INTERNAL BOX LEVEL AND SYSTEM LEVEL DESIGN CRITERIA. EQUIPMENT LAYOUT, SHIELDING, HARNESSING, GROUNDING, ETC.
  - BOX-LEVEL TESTS PER MIL-STD-461 NOT GOOD ENOUGH. TOO BROAD A SPECTRUM AND FREQUENCIES WERE NOT MODULATED.
  - REVIEW ECAC DATA EARLY-ON AND DETERMINE ON-ORBIT FREQUENCY SPECTRA, MODULATION AND FIELD STRENGTHS.

  - CONDUCT INTENSIVE/REALISTIC BOX-LEVEL EMI TESTS.
  - SPECIFICALLY DESIGN OBSERVATORY-LEVEL EMI TEST TO VERIFY:

    - SELF COMPATIBILITY
    - TRANSPORTATION ENVIRONMENT
    - COMPATIBILITY WITH LAUNCH VEHICLE
    - CAPABILITY TO SURVIVE IN PRESENCE OF GROUND SOURCES

  -- DO THE MISSION AT L-2

- o RANGE AND LAUNCH OPERATIONS--START EARLY

- o TEST PLANS/PROCEDURES MUST BE APPROVED BY THE PROJECT OFFICE.

- o DON'T EVER GET COMPLACENT, ESPECIALLY WITH EXISTING DESIGNS/HARDWARE.

LAST MINUTE SUGGESTIONS:

- SIT DOWN WITH CONTRACTORS & EXPLAIN RULES
- EXPAND OPERATING TEMP. LIMITS OF BLACK BOXES:
  - COBE 0 TO 40°C
  - DOD MUCH WIDER
- PAINT - DIFFICULT TO SATISFY THERMAL, CONTAMINATION & CHARGING.
- THERMAL CONDUCTORS, LOOKED AT:
  - CHOTHERM
  - GREASE
  - NUSIL
  - NUSIL w/ TEFLON SHEET
- WIRE - EVALUATED 8 TYPES TO SATISFY HARNESSING & CONTAMINATION CONTROL
- GROUNDING
  - KEEP CURRENTS OUT OF CHASSIS
  - USE STRUCTURE AS SIGNAL RETURN
  - ROUTING VERY CRITICAL CLOSE TO DETECTORS
  - PAY ATTENTION TO TEST EQUIP. & FACILITY GNDNG.
- CIRCUIT ANALYSES
  - POLARIZED PARTS IN BACKWARDS
  - IMPROPER DERATING.

TF-63A

COBE LESSONS LEARNED--ORGANIZATION AND MANAGEMENT

- OPTIMUM SITUATION--COLLOCATE ENTIRE PROJECT TEAM IN ONE BUILDING.
- EXPAND SYSTEMS ENGINEERING
  - BROADER BASE: ELECTRICAL, MECHANICAL, CONTROLS AND ANALYSTS TO DO FMECA'S, ETC.
  - ALL SYSTEMS ENGINEERS MUST REPORT TO THE SYSTEM MANAGER.
- ALL MONEY MUST BE CONTROLLED AND DISTRIBUTED BY PROJECT OFFICE IN ACCORDANCE WITH WBS.
- CENTRALIZE ALL PROCUREMENT IN THE PROJECT OFFICE.
- ONE CCB RUN BY THE PROJECT OFFICE--RIGOROUSLY CONTROL ALL CHANGES.
- ALL DRAWINGS, SPECS, DOCUMENTS, CHANGE ORDERS, ETC. MUST BE RELEASED BY SINGLE PERSON IN PROJECT OFFICE. ALL DOCUMENTS APPROVED BY PROJECT OFFICE.
- GSFC SORELY NEEDS PRODUCTION CONTROL.
- BRANCHES/DIVISIONS MUST ADVISE THE PROJECT OFFICE AND CONSENT TO CONTROL OF THE PROJECT OFFICE.

TF-6
64

COBE LESSONS LEARNED--ORGANIZATION AND MANAGEMENT (CONTINUED)

- BRANCHES MUST BE RESPONSIBLE FOR DELIVERABLES TIED TO THE WBS. ONE INDIVIDUAL MUST BE RESPONSIBLE FOR A DELIVERABLE. WBS TO LEVEL 5.
- PROJECT OFFICE AND BRANCH SHOULD CO-SIGN SUB-SYSTEM MANAGER KSO'S AND PERFORMANCE APPRAISALS.
- MORE DETAILED SOWS FOR BOTH CONTRACTORS AND BRANCHES.
- CLOSED LOOP SYSTEM TO FOLLOW UP ON ALL ACTION ITEMS.
- CM, CERT LOGS, R&QA SHOULD BE IN PLACE AND NOT DEVELOPED ALONG THE WAY.
- COBE INSTRUMENT TEAM--MATRIX ORGANIZATION BUILDS UP ULCERS, TEARS DOWN SCHEDULES, AND INCREASES COST. DETAILED INSTRUMENT DEVELOPMENT PLAN (IDP) NEEDED VERY EARLY AND TIED TO WBS.
- I&T MANAGER PART OF THE PROJECT OFFICE.
- FORMAL/WEEKLY STATUS REVIEWS WITH EACH SUBSYSTEM MANAGER.
- SHORT FALL OF CRITICAL SKILLS

COBE ORGANIZATION

TF-66

COBE IS PERFORMED IN THE IN-HOUSE MODE TO STRENGTHEN AND PRESERVE THIS CAPABILITY AT THE GSFC

- COVERS A MAJORITY OF THE ENGINEERING DIRECTORATE TECHNOLOGIES
  - INSTRUMENT DESIGN AND DEVELOPMENT AND DATA HANDLING
  - SPACECRAFT DESIGN AND DEVELOPMENT
  - OBSERVATORY INTEGRATION AND TEST
- PROVIDES HIGH TECHNOLOGY WORK
  - INFRARED DETECTORS
  - CRYOGENICS
- PROVIDES HANDS ON EXPERIENCE
  - NEW HIRES
  - TRAINING TOOL

GSFC

TF-67

CUSTOMER PERSPECTIVE--COBE PROJECT OFFICE

SYSTEM DEFINITION--CONFIRM THE OBJECTIVE

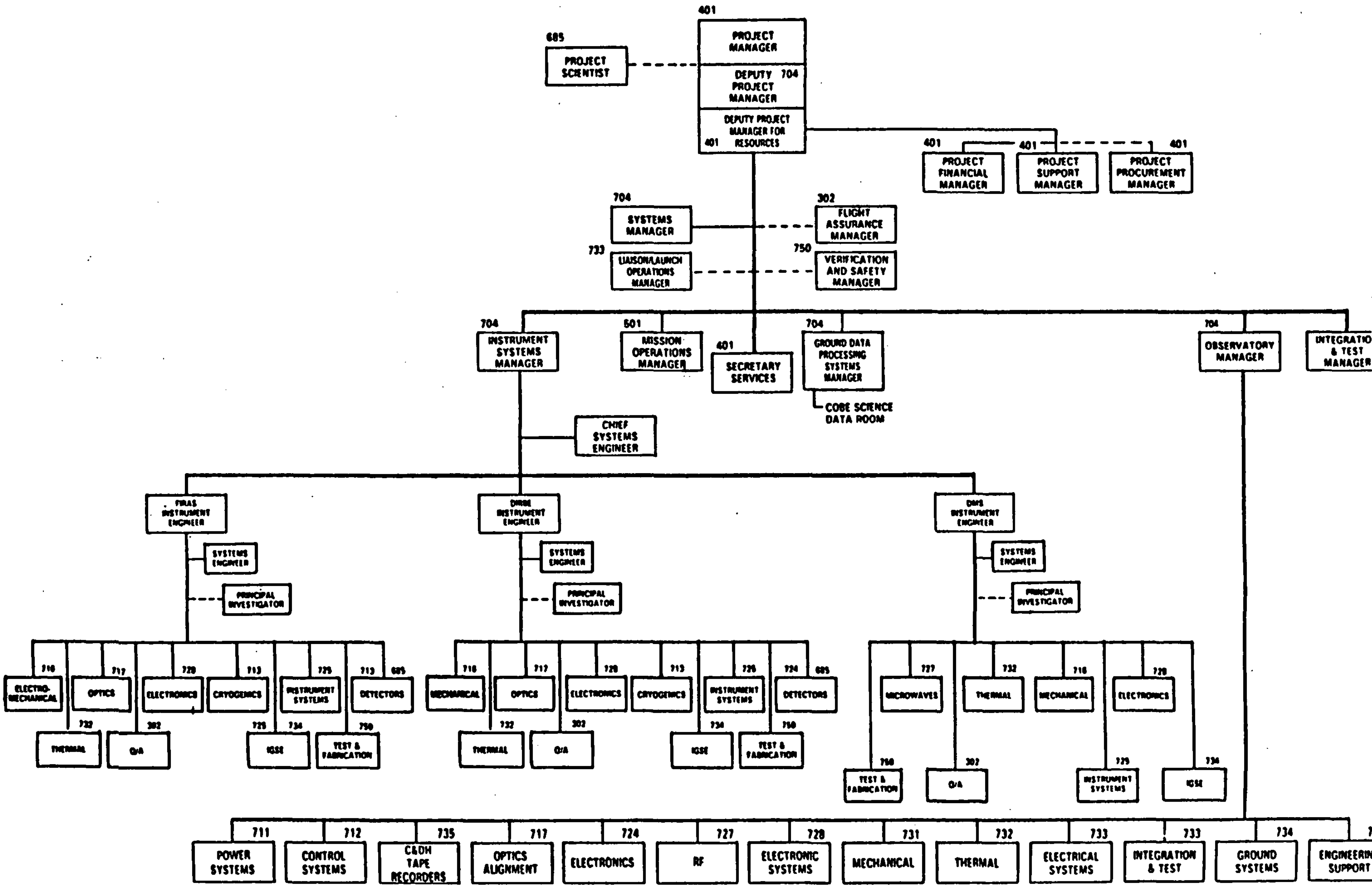
- LEVEL I PROJECT REQUIREMENTS (PROJECT PLAN)
- OBSERVATORY LEVEL SPECIFICATION (CODE 401/701)
- SUBSYSTEM SPECIFICATIONS--CO-SIGNED (CODE 401/704/700)
- STATEMENT OF WORK--CO-SIGNED (CODE 401/704/700)
- INTERFACE CONTROL DOCUMENTS--CO-SIGNED (CODE 401/704/700)

PLANS/STUDIES--ESTABLISH BOUNDARIES AND THEIR COMPLIANCE (CODE 400/704)

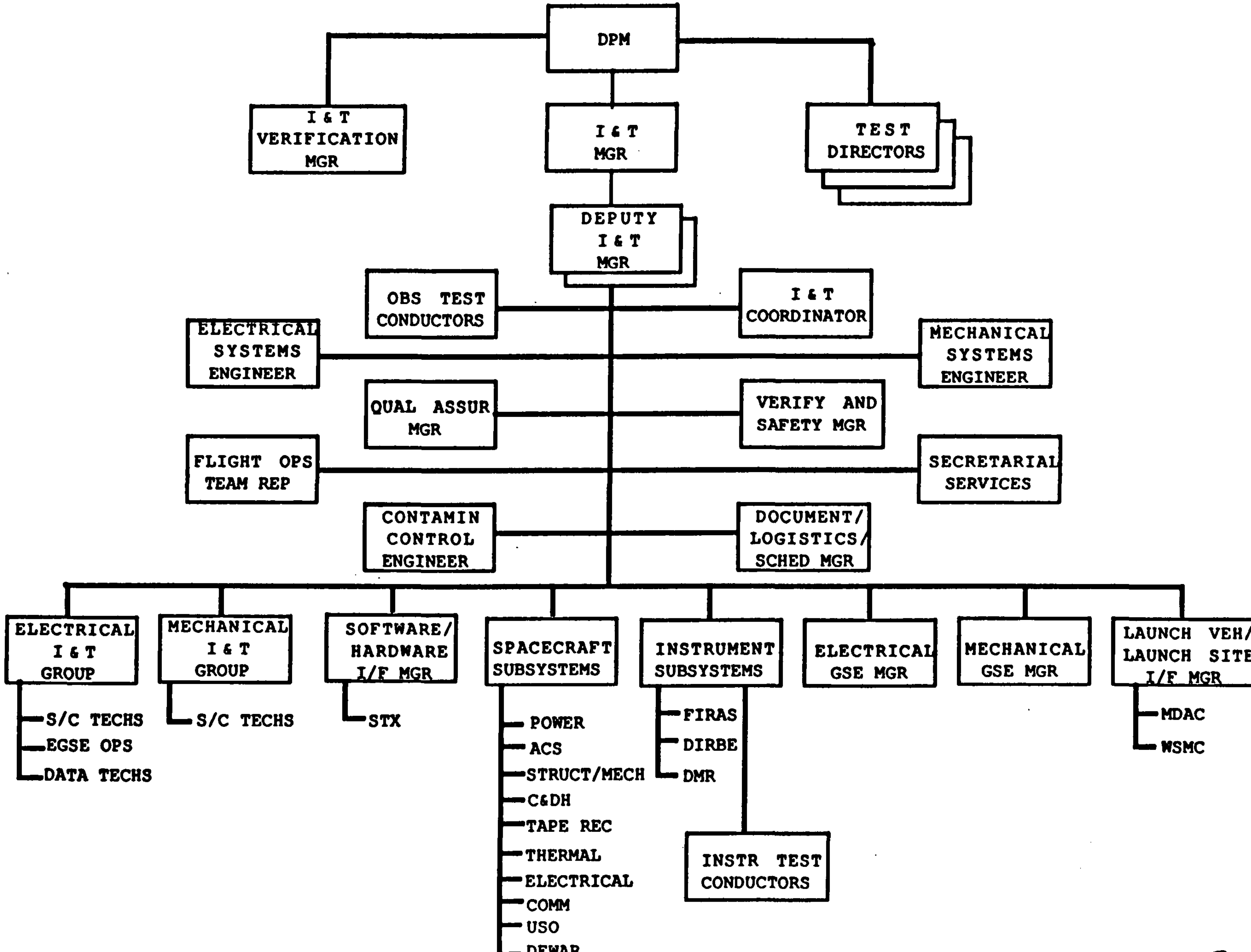
ENGINEERING EXECUTION--PARTITION THE SYSTEM (CODE 401/704)

TRADE-OFFS--OPTIMIZATION (CODE 401/704)

TF-68


401
PROJECT MANAGER
685
PROJECT SCIENTIST
DEPUTY PROJECT MANAGER 704
DEPUTY PROJECT MANAGER FOR RESOURCES 401
401
PROJECT FINANCIAL MANAGER
401
PROJECT SUPPORT MANAGER
401
PROJECT PROCUREMENT MANAGER
704
SYSTEMS MANAGER
302
FLIGHT ASSURANCE MANAGER
733
LIAISON/LAUNCH OPERATIONS MANAGER
750
VERIFICATION AND SAFETY MANAGER
704
INSTRUMENT SYSTEMS MANAGER
501
MISSION OPERATIONS MANAGER
401
SECRETARY SERVICES
704
GROUND DATA PROCESSING SYSTEMS MANAGER
COBE SCIENCE DATA ROOM
704
OBSERVATORY MANAGER
INTEGRATIO & TEST MANAGER
CHIEF SYSTEMS ENGINEER
FIRAS INSTRUMENT ENGINEER
DIRBE INSTRUMENT ENGINEER
DMR INSTRUMENT ENGINEER
SYSTEMS ENGINEER
PRINCIPAL INVESTIGATOR
ELECTRO-MECHANICAL
OPTICS
ELECTRONICS
CRYOGENICS
INSTRUMENT SYSTEMS
DETECTORS
THERMAL
Q/A
IGSE
TEST & FABRICATION
MECHANICAL
MICROWAVES
711
POWER SYSTEMS
712
CONTROL SYSTEMS
735
C&DH TAPE RECORDERS
717
OPTICS ALIGNMENT
724
ELECTRONICS
727
RF
728
ELECTRONIC SYSTEMS
731
MECHANICAL
732
THERMAL
733
ELECTRICAL SYSTEMS
733
INTEGRATION & TEST
734
GROUND SYSTEMS
750
ENGINEERING SUPPORT

TF-69


DPM
I & T VERIFICATION MGR
I & T MGR
TEST DIRECTORS
DEPUTY I & T MGR
OBS TEST CONDUCTORS
I & T COORDINATOR
ELECTRICAL SYSTEMS ENGINEER
MECHANICAL SYSTEMS ENGINEER
QUAL ASSUR MGR
VERIFY AND SAFETY MGR
FLIGHT OPS TEAM REP
SECRETARIAL SERVICES
CONTAMIN CONTROL ENGINEER
DOCUMENT/ LOGISTICS/ SCHED MGR
ELECTRICAL I & T GROUP
S/C TECHS
EGSE OPS
DATA TECHS
MECHANICAL I & T GROUP
S/C TECHS
SOFTWARE/ HARDWARE I/F MGR
STX
SPACECRAFT SUBSYSTEMS
POWER
ACS
STRUCT/MECH
C&DH
TAPE REC
THERMAL
ELECTRICAL
COMM
USO
DEWAR
INSTRUMENT SUBSYSTEMS
FIRAS
DIRBE
DMR
INSTR TEST CONDUCTORS
ELECTRICAL GSE MGR
MECHANICAL GSE MGR
LAUNCH VEH/ LAUNCH SITE I/F MGR
MDAC
WSMC

TF-70

IN-HOUSE SUBSYSTEM PROCUREMENT MODE

GSFC PERFORMS THE ROLE OF THE "SYSTEMS CONTRACTOR"

- INSTRUMENTS: BUY PARTS/COMPONENTS
  DESIGN/FABRICATE/INTEGRATE/TEST IN-HOUSE
- SPACECRAFT: SYSTEM DESIGN IN-HOUSE, BUY COMPONENTS (EARTH SCANNERS, BATTERIES, REACTION WHEELS, ETC.,) AND MAJOR SUBSYSTEMS (PROPULSION, DEWAR, ETC.)
- GROUND SYSTEM: SYSTEM DESIGN IN-HOUSE, BUY HARDWARE/SOFTWARE
- INTEGRATION AND TEST: IN-HOUSE/CONTRACTOR SUPPORT
- LAUNCH/MISSION OPS: IN-HOUSE/CONTRACTOR SUPPORT

GSFC

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SOURCE BOARD

- o SPECS/SOW PREPARED BY FUNCTIONAL ORGANIZATION AND APPROVED BY PROJECT PRIOR TO SUBMITTAL TO SEB.
- o SYSTEMS MANAGER MEMBER OF ALL SEB'S, OTHER PROJECT PERSONNEL CONSULTANTS.
- o SEB CHAIRPERSON--NO SET POLICY
- o OBSERVATORY MANAGER PART OF NEGOTIATION TEAM
- o ALL SPEC/SOW CHANGES APPROVED BY CCB.

TF-72

DOCUMENTATION CONTROL AND METHOD OF CHANGE

- PROJECT APPROVES AND MAINTAINS TOP LEVEL DOCUMENTS EXAMPLES:
  - COBE SYSTEMS PERFORMANCE SPECIFICATION: COBE-SR-401-1004-01
  - COBE CONFIGURATION MANAGEMENT PLAN: PL-401-1001-01
  - COBE VERIFICATION AND TEST PLAN: COBE-PL-730-1702-01

- PROJECT APPROVES ALL INSTRUMENT AND SPACECRAFT SUBSYSTEM SPECIFICATIONS/ICD'S. DOCUMENTS NORMALLY MAINTAINED AT BRANCH LEVEL.

- COBE CONFIGURATION MANAGEMENT PLAN ESTABLISHES THE BASIC FRAMEWORK FOR CONTROL OF CHANGES
  - CLASS I CHANGES (PROJECT APPROVAL REQUIRED)
  - CLASS II CHANGES (PROJECT REPRESENTATIVE INVOLVED)

- PROJECT APPROVES (OR DISAPPROVES) AND MAINTAINS ALL CLASS I CHANGES
  - CONFIGURATION BOARD MEETS WEEKLY TO DISPOSITION CHANGES
  - DISPOSITION EMERGENCY CHANGE REQUESTS IN "REAL-TIME"

- PROJECT REPRESENTATIVES PARTICIPATE IN AND REVIEW ALL CLASS II CHANGE ACTIONS

GSFC

DOCUMENTATION

ALL PROCURED ITEMS ARE DELIVERED WITH ACCEPTANCE DATA PACKAGE.

ALL IN-HOUSE HARDWARE IS DELIVERED WITH CERT LOGS.

OBSERVATORY IS INTEGRATED VIA APPROVED (I&T MANAGER & Q/A) WORK ORDERS.

OBSERVATORY IS TESTED WITH APPROVED WORK ORDERS AND APPROVED TEST PLAN (DPM) AND TEST PROCEDURES (OBSERVATORY MANAGER AND/OR INSTRUMENT MANAGER)

ALL PROBLEMS ARE DOCUMENTED BY Q/A (REFERENCE AGAINST WORK ORDER) AND REVIEWED BY OBSERVATORY AND/OR INSTRUMENT MANAGERS.

PROBLEM RECORDS THAT BECOME MALFUNCTION REPORTS ARE REVIEWED BY ENVIRONMENTAL TEST COMMITTEE--SIGNED OFF BY Q/A AND OBSERVATORY OR INSTRUMENT MANAGER AND SYSTEMS MANAGER.

TEST CONDUCTOR LOG SHOWS WHAT PROCEDURES WERE RUN FOR EACH OBSERVATORY TEST.

DOCUMENTATION (CONTINUED)

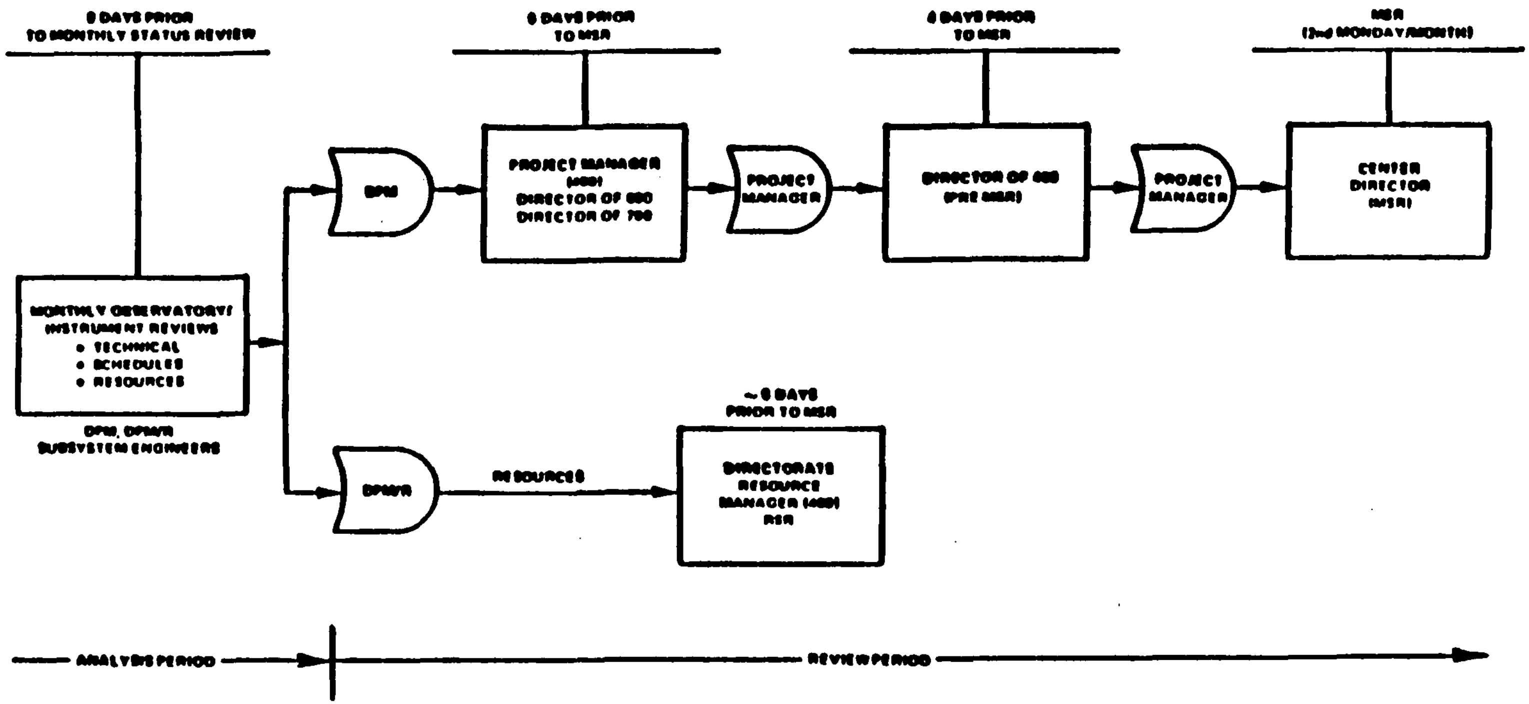
ACTUAL SUBSYSTEM VALUES ARE ARCHIVED IN THE COMPUTER FILE (FILED BY DATE).

Q/A MAINTAINS LIST OF ALL PROBLEMS/MALFUNCTIONS AGAINST EACH TEST.

Q/A MAINTAINS SHEETS SUBMITTED WITHIN 24-HOURS OF A TEST.

COBE INSTRUMENTS HAVE REQUIRED APPROXIMATELY 2 WEEKS TO ANALYZE DATA. PRECLUDES A LEGITIMATE REAL-TIME PASS/FAIL CRITERIA.

COBE MONTHLY REPORTING SCHEDULE



TF-76

COBE PROJECT DOLLAR/MANPOWER/SCHEDULE REPORTING SYSTEM (CONT.)

- CAUSE AND IMPACT OF VARIANCES WORKED MONTHLY WITH SUBSYSTEM MANAGERS. ESTIMATE AT COMPLETION (EAC) DETERMINED AND CORRECTIVE ACTION PLANNED

- PROJECT SUMMARIZES DATA AT THE SPACECRAFT, DEWAR, AND INDIVIDUAL INSTRUMENT LEVEL AND PRESENTS MONTHLY TO "DIRECTORS OF"

GSFC

COBE PROJECT DOLLAR/MANPOWER/SCHEDULE

REPORTING SYSTEM

- o DETAILED SCHEDULES, DOLLAR AND MANPOWER PLANS ESTABLISHED AT THE SUBSYSTEM LEVEL:

  - -- COST IDENTIFIED AT THE BOX AND TASK LEVEL FOR THE SPACECRAFT SUBSYSTEMS; THE COMPONENT AND TASK LEVEL FOR THE INSTRUMENTS.

  - -- MANPOWER REQUIREMENTS AND ORGANIZATIONAL RESPONSIBILITY IDENTIFIED TO SAME LEVEL AS COST.

  - -- PERT SCHEDULES DEVELOPED FOR EACH SPACECRAFT SUBSYSTEM AND INSTRUMENT:

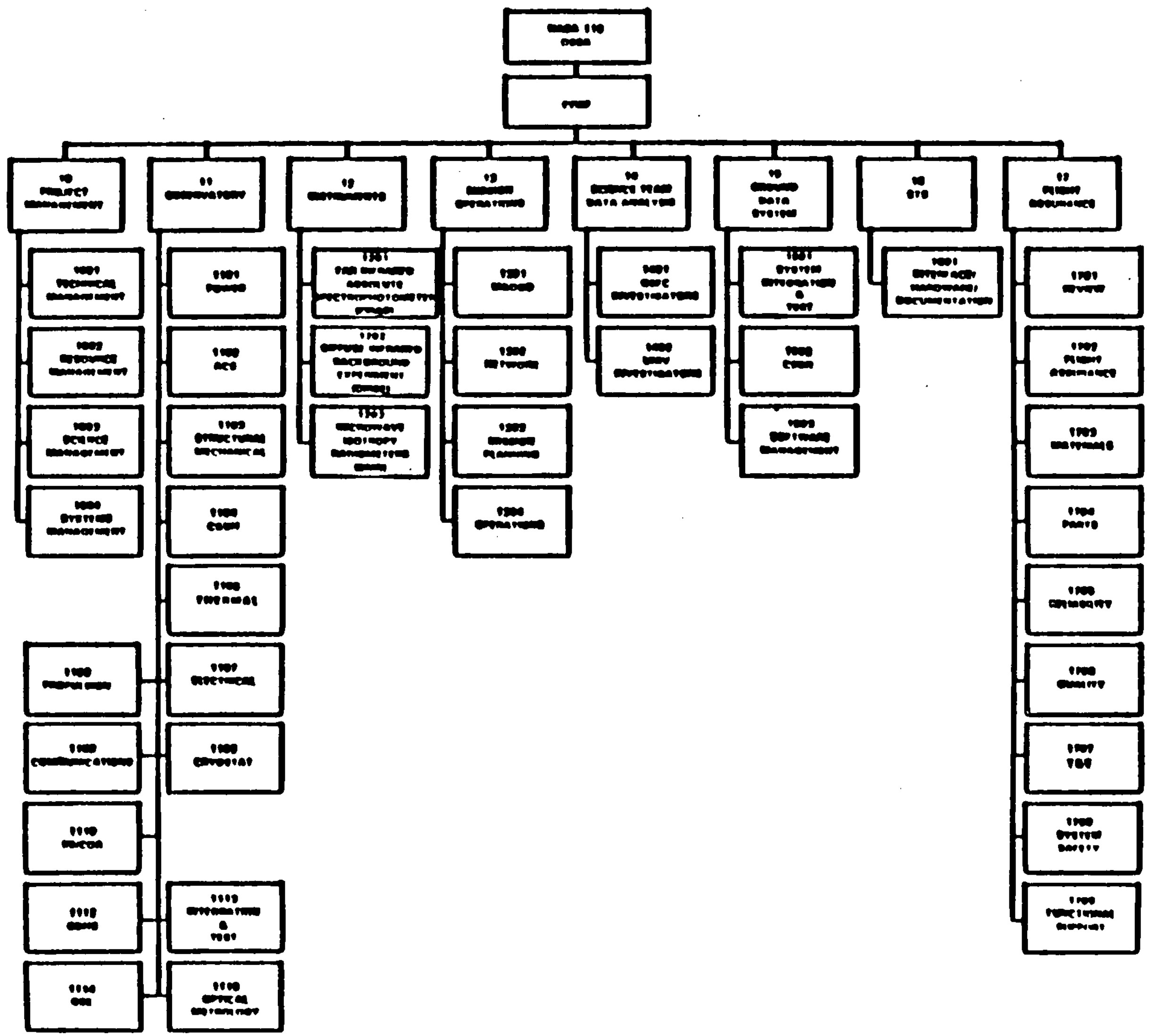
    - -- SPACECRAFT 17 NETS (2000 NODES)
    - -- INSTRUMENTS 3 NETS (250 NODES EACH)

- o DATA ACCUMULATED FROM SUPPORTING ORGANIZATIONS AND SUMMARIZED IN PROJECT OFFICE AT THE SUBSYSTEM LEVEL.

- o TECHNICAL PROGRESS/PROBLEMS REVIEWED WITH SUBSYSTEM MANAGER DAILY/WEEKLY/MONTHLY.

TF-78

PROJECT WORK BREAKDOWN STRUCTURE FOR COBE



TF-79

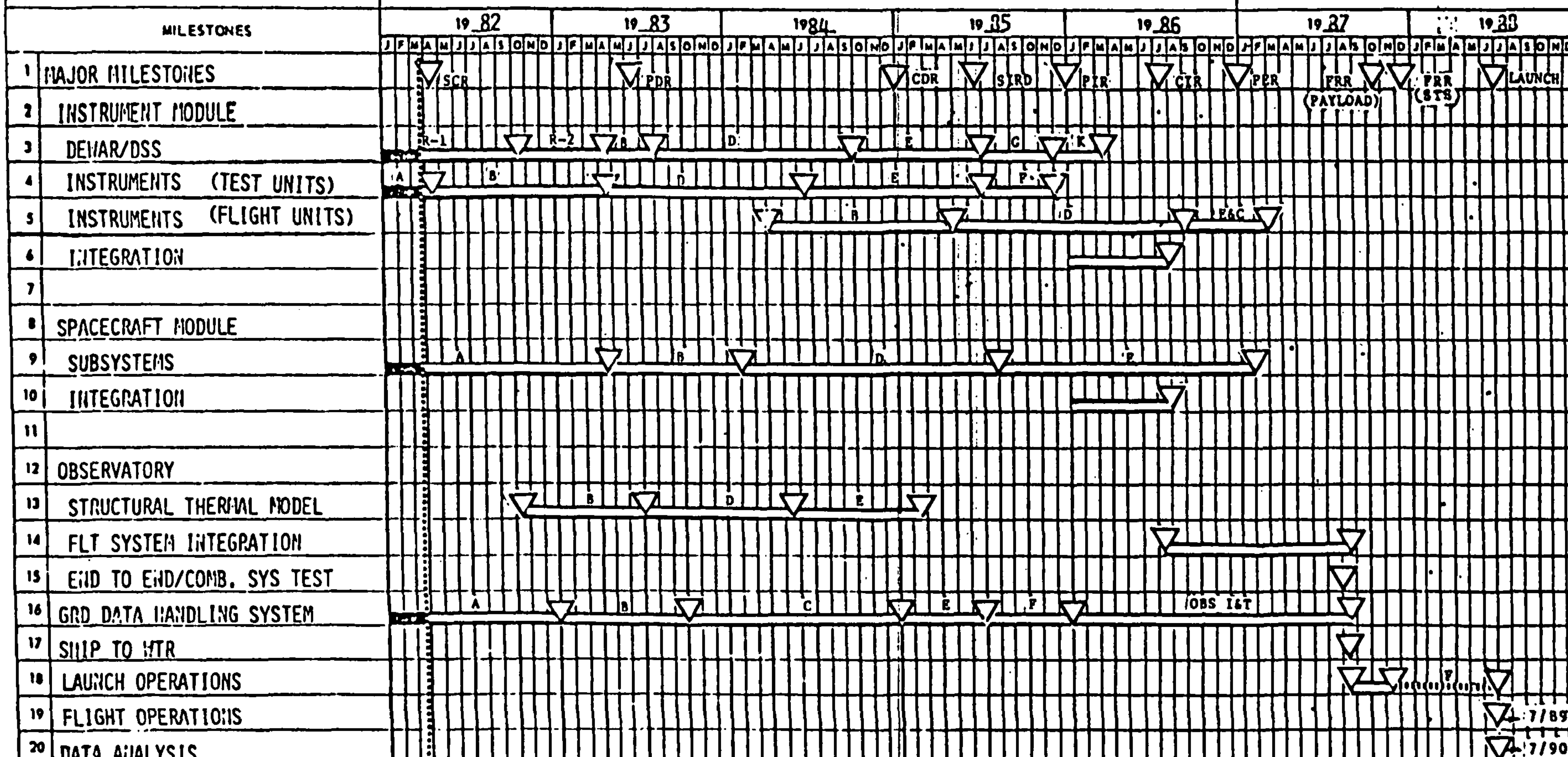
CENTER
RESPONSIBILITY:
APPROVAL
ACCOMPLISHMENT

MILESTONE SCHEDULE

PROJECT COBE PROJECT

1 LEVEL

ORIG. SCHED. APPR. (DATE)
LAST SCHED. CHG. (INITIALS)
STATUS AS OF 3-29-82 (DATE) (INITIALS)



NOTES:

SCR - SYSTEM CONCEPT REVIEW
PDR - PRELIMINARY DESIGN REVIEW
CDR - CRITICAL DESIGN REVIEW
SIRD - SUPPORT & INSTRUMENTATION REQUIREMENTS DOCUMENT
PIR - PRE-INTEGRATION REVIEW
CIR - CARGO INTEGRATION REVIEW
PER - PRE ENVIRONMENTAL REVIEW

FRR - FLIGHT READINESS REVIEW
R-1 - RFP PREPS
R-2 - RFP CYCLE
A - DEFINITION
B - DESIGN
C - CALIBRATION

D - DEVELOPMENT
E - TEST
F - CONTINGENCY
G - SET-UP AT GSFC DEWAR

K - I&T INST TEST UNIT

FF-80

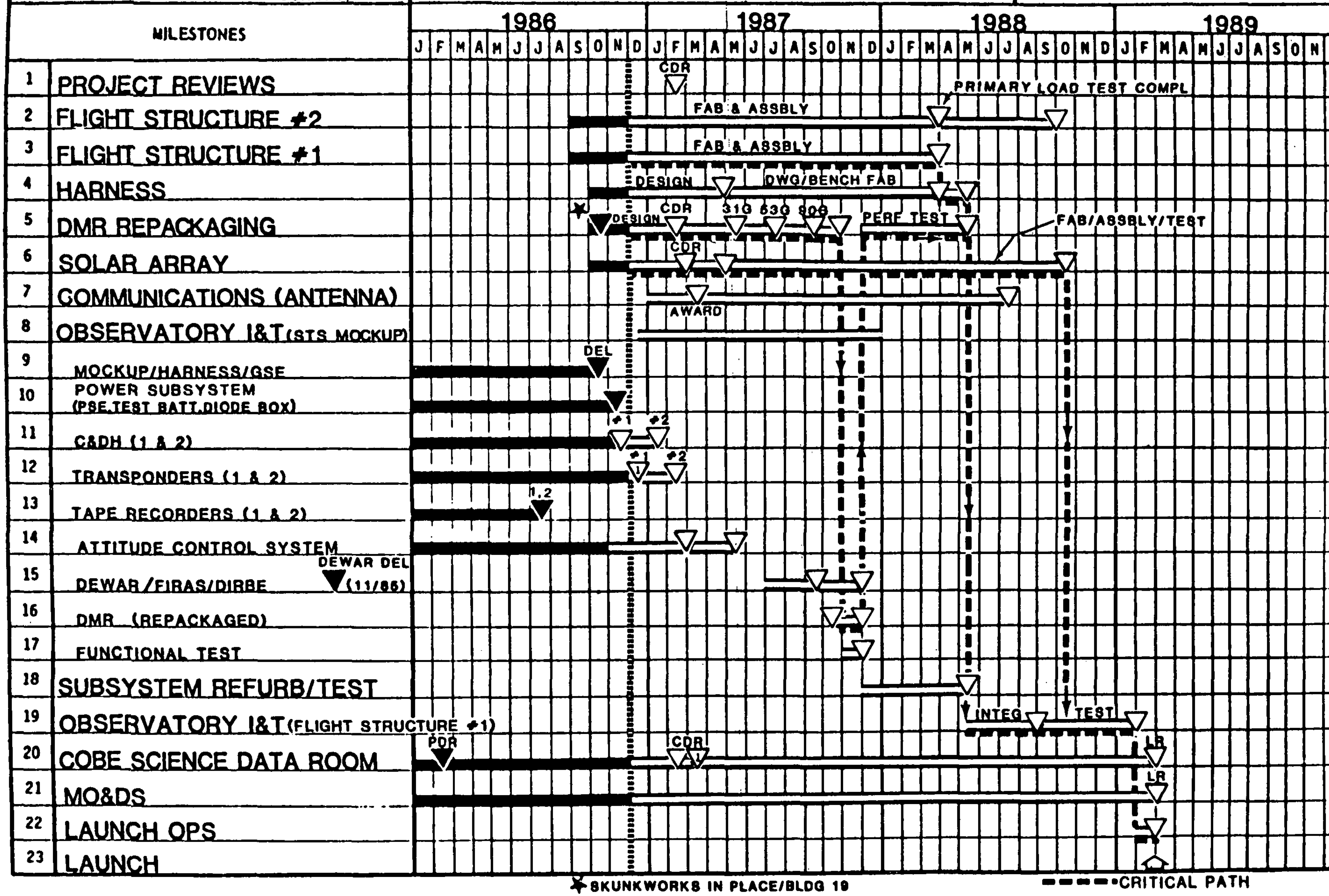
RESPONSIBILITY:

APPROVAL R.A. MATTSON

ACCOMPLISHMENT D.K. MCCARTHY

COBE/DELTA SCHEDULE

STATUS AS OF 11/30/86 (DATE)



TF-81

SCHEDULE RESPONSIBILITY Rose A. Mattson

R.A. MATTSON

STATUS RESPONSIBILITY D.K. McCarthy

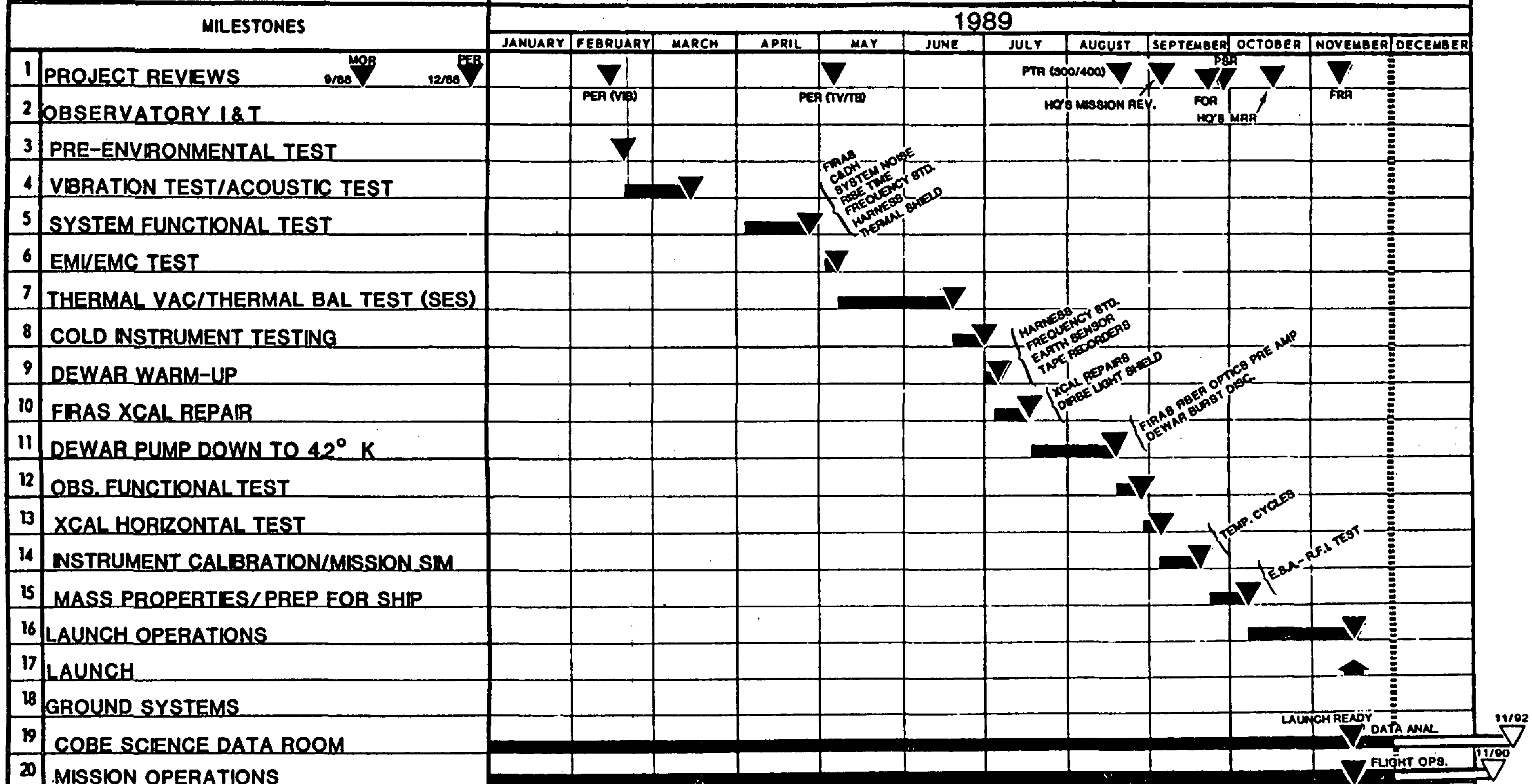
D.K. MCCARTHY

COBE MASTER SCHEDULE

ORIGINAL SCHEDULE APPROVAL 9/30/88

LAST SCHEDULE CHANGE 11/6/89

STATUS AS OF 11/30/89 (DATE)



NOTES:

MOR - MISSION OPERATIONS REVIEW

PER - PRE ENVIRONMENTAL REVIEW

PTR - POST TEST REVIEW

FOR - FLIGHT OPERATIONS REVIEW

PSR - PRE SHIP REVIEW

MRR - MISSION READINESS REVIEW

FRR - FLIGHT READINESS REVIEW

TF-82

COBE ASSURANCE MANAGEMENT

LESSONS LEARNED

March 7, 1990

AH-1

GENERAL

- TEAM SPIRIT, COMMUNICATION
- SKILLED PEOPLE WHO TAKE PRIDE IN THEIR WORK, THINK ABOUT WHAT THEY'RE DOING
- SKELETON CREWS
- OFFICE OF FLIGHT ASSURANCE TECHNICAL EXPERTS
- TEAM SPIRIT

PARTS

- PARTS LISTS ON ELECTRONIC DATA BASE, UP TO DATE
- AS BUILT PARTS LISTS
- ECO REVIEW FOR EEE PARTS CHANGES
- RADIATION TESTING - SINGLE EVENT UPSET REVIEW
- REVIEW FOR APPLICATION
- PART PROGRAM COSTS
- BOX RELIABILITY

MATERIALS

- FASTENER TESTING AND CONTROL IMPLEMENTED
- MATERIALS REVIEWED FOR APPLICATION
- CONTAMINATION VS ESD CONTROL
- PYROTECHNICS
- CONTAMINATION

CONFIGURATION MANAGEMENT

- CLOSED LOOP SYSTEM WORKED WELL
- CERT LOGS WORKED WELL
- BETTER CONTROL OF AS-BUILT, ORIGINAL DRAWINGS - SPECIFICALLY HARNESS MILLIOHM DRAWINGS
- QA NOT INCLUDED IN ALL ASPECTS OF MECHANICAL DRAWING AND ECO REVIEW

EARLY PLANNING AND DESIGN

- PEER REVIEWS OF DESIGNS IMPLEMENTED
- REALISTIC DRAWING REQUIREMENTS
- DESIGNING WITH GROUND TEST AND ENVIRONMENTS IN MIND
- QA REVIEW OF TASK ORDERS IMPLEMENTED
- QA REVIEW OF DESIGN CHANGES

MANUFACTURING AND PROCESSING

- LIMITED LIFE ITEMS TRACKED
- CERT LOGS IMPLEMENTED
- ENVIRONMENTAL TEST COMMITEE FORMED
- CONTAMINATION CONTROL
- SURVEILLANCE OF CONTRACTOR

INTEGRATION AND TEST

- PROCEDURES SHOULD STAND ALONE, WHEN POSSIBLE, HAVE MORE THAN ONE PERSON QUALIFIED TO RUN PROCEDURE

- ESD CONTROL - YES, GLOVES AND WRISTATS ARE A BAD IDEA, NEEDS MORE ATTENTION FOR FUTURE PROGRAMS

- MATE/DEMATE - DIFFICULT TO DETERMINE NUMBER OF MATES/DEMATES WITH COBE LOG, LOG SHOULD BE LISTED PER CONNECTOR, NOT PER DAY.

- DAILY I&T MEETINGS VERY USEFUL

- WALKTHROUGHS VERY USEFUL

- CONTROL OF UNSCHEDULED TESTS

- WORK ORDER AUTHORIZATIONS IMPLEMENTED

- TEST SOFTWARE AND HARDWARE CONTROL

- CONSISTENT FUNCTIONAL/PERFORMANCE TESTS - WHEN WE HAD THEM, THEY WERE GREAT, WHEN WE DIDN'T TIME AND EFFORT WAS WASTED

- TESTS WERE RUN AS CLOSE TO FLIGHT CONFIGURATION AS POSSIBLE. TEST BOX CONFIGURATION CONTROL - SOME SURPRISES WHEN TEST BOX OR HARNESS NOT AS EXPECTED

- CLOSED LOOP NONCONFORMANCE CONTROL SYSTEM IMPLEMENTED

- FACILITY QA INITIATED

- STAFFING, INADEUATE AT TIMES FOR 24 HOUR OPERATIONS

LAUNCH SITE

- o AVOID NEW, UNREHEARSED TESTS CLOSE TO LAUNCH DAY
- o ADEQUATE STAFFING FOR 24 HOUR TESTING
- o COMMUNICATIONS NET CHECKOUT AND REHEARSAL VERY USEFUL
- o MORE INTERFACE BETWEEN LAUNCH VEHICLE AND SPACECRAFT PERSONNEL FOR I&T PLANNING
- o MAINTAIN THE STATUS QUO

POST LAUNCH

- SPACECRAFT ORBITAL ANOMALY REPORT (SOAR) SYSTEM UTILIZED
- MISSION DATA CENTER
- BONDED STORES OF FLIGHT SPARES USEFUL FOR POST LAUNCH TROUBLESHOOTING

NARRATIVE

GENERAL

1. TEAM SPIRIT

The team spirit over came many adversities.

2. RESPONSIBLE PERSONNEL

One thinking person is better than 100 procedures. Procedures are very necessary, but occasionally, there'll be a mistake in one or it won't explain something fully. COBE personnel never hesitated to report a problem.

3. MANAGEMENT SUPPORT

An assurance management program doesn't work as well when management doesn't support it. COBE management was very responsive to perforance assurance requirements.

4. SKELETON CREW

The in-house COBE assurance mangement team was much smaller than that for a comparable out-of-house spacecraft. Some bending of the rules was required for the sanity and sleep of those involved. However, the spirit of GSFC performance assurance requirements was never compromised, and the entire COBE team became involved in the program. Total Quality Management is the buzzword today from NASA headquarters. The implementation of that idea was seen on COBE and other GSFC programs.

5. OFFICE OF FLIGHT ASSURANCE TECHNICAL EXPERTS

One thing COBE personnel made good use of was the technical expertise of Code 300 personnel. Many people see us only as watchdogs but the Code 300 groups contain experts in materials and EEE parts, experienced designers, and personnel experienced in test and evaluation.

PARTS CONTROL

1. UP TO DATE PARTS LIST, ON ELECTRONIC DATABASE

The parts lists must be kept up to date and on a computer data base. COBE was unable to keep the electronic data base up to date. As a result, most part searches were done by hand, using recipe cards, parts lists, and ECO's. This, of course, was very labor intensive.

Some of our lists were in good shape, but for others, to be really be sure of the as-built configuration, we had to go back to the manufacturer. In one case, we almost replaced a part because our lists showed it to be a bad date code. That particular part had already been replaced at the manufacturer, but they had failed to up date the as-built parts list with the correct date codes.

2. AS-BUILT PARTS LISTS

As-built parts lists must reflect manufacturer, d/c and, ideally, board serial number. Not all of our lists had this, but those that did saved eons of time researching parts data. The new NHB handbooks require this information specifically, and it's worth the effort to make sure you get it. If the contractor is being funded to maintain the as-built list, be sure to include the requirement to perform special parts searches, not just those related to GIDEP. NASA TWX Alerts are one example of non-GIDEP parts alerts.

3. ECO REVIEW FOR EEE PARTS CHANAGES

COBE implemented parts engineer review of ECO's for EEE parts changes. A logical follow on to parts list review.

4. RADIATION TESTING

Radiation testing was included in the COBE parts program. Single Event Upsets and Latchups should be reviewed.

5. REVIEW FOR APPLICATION

EEE parts should be reviewed for application as well as Perferred Parts status. The peer reviews of each design would be an excellent way to double check applications. A parts engineer should be a member of the review team. Stress analyses and worst case analyses also are places where parts ratings can be taken into account. Some poor applications on COBE were discovered by stress analysis and parts were changed early in design. Another application, where a part did not meet full derating criteria, was not discovered until peer reviews performed very late in the program. Although this part was acceptable as is, it illustrated the need for early peer reviews.

6. PARTS PROGRAM COSTS

We've learned that a Grade 2 parts program is expensive. It is worth a little effort on the design engineer's part to check for the availability of standard parts. Besides their proven reliability, they save expensive NSPAR reviews and qual programs.

Even after the boxes are built, parts costs continue. There are Alert searches and random part failures that require research and testing. Although costs level out after hardware builds are complete, they are still incurred up to launch day. In fact, COBE had a small amount of parts testing performed after launch.

Tracking of parts program costs is essential. COBE closely tracked parts test status and charge backs. A number of large charge back errors were noted - charges from other programs were put on the COBE books. Schedule tracking also kept parts moving through the screening process.

7. BOX RELIABILITY

It at all possible, avoid using the same date code in redundant boxes. If a part lot is found to be defective, the effect will be lessoned if we use many different lots in our boxes.

MATERIALS CONTROL

1. FASTENER CONTROL

We now realize that counterfeit and substandard fasteners exist in US stocks. COBE went back and tested stock fasteners and implemented a fastener program on new buys. We did discover cracked, porous, and soft fasteners during our receiving inspection of transporter fasteners. This demonstrated that ground support equipment must be considered in the fastener program.

2. MATERIALS REVIEW

Materials must be reviewed for application. Some COBE applications were severe, with cryogenic temperatures and contamination cosnstraints. The fact that we had so few materials problems is owed to the close work with Code 313 in determining materials usage.

3. ESD VS CONTAMINATION

Consider contamination aspects when evaluating ESD control materials.

4. PYROTECHNICS

Nray of pyrotechnic devices was utilized to verify charge fill of flight devices.

5. CONTAMINATION

Strict adherance to contamination requirements in materials usage eliminated many potential problems for the spacecraft.

One problem we did suffer from was Caprolactam contamination from the antistatic blankets that were used. By the time an alert was issued for the material, RCAS 2400, COBE had been covered with it for months. Most of the contamination was eliminated during thermal vacuum, but some remained and turned up as a white residue on the spacecraft after shipment. RCAS was replaced with Llumalloy.

CONFIGURATION MANAGEMENT

1. CLOSED LOOP SYSTEM

The use of a closed loop system worked well for COBE. No CCR was closed until the work was completed and inspected.

2. CERT LOGS

Again, cert logs are invaluable for recording and verifying as-built configuration.

3. CONTROL OF DRAWINGS

We did not control harness drawings with milliohm measurements very well. As a result, the drawings became torn, some were lost and the rest were just hard to dig through.

4. QA REVIEW OF DRAWINGS

All major contractors include QA in the review of drawings. We did not review mechanical drawings on a regular basis. No specific problems were encountered, but a QA review enables us to do such things as check for materials applications, verify fracture control implementation and make sure all the other proper people have reviewed the drawings.

EARLY PLANNING AND DESIGN

1. PEER REVIEWS OF DESIGNS IMPLEMENTED

Peer reviews of deployables and some electronics systems provided detailed insight into the designs. The deployables reviews, in particular did an excellent job of action item followup.

2. REALISTIC DRAWING REQUIREMENTS

Avoid too tight tolerances. We had an excessive reject rate on COBE mechanical parts, and most of the problems written up were bought use-as-is. Also, we were forced by schedule to accept some items that we could have reworked. In the future more pressure should be put to bear on the machine shops to build parts to print. The Institutional Assurance group is setting up a trend program to identify areas of concern.

3. DESIGNING WITH GROUND TEST AND ENVIRONMENTS IN MIND

In some cases, the 1g environment "helped" our system, masking design problems. Special 0 g tests for the XCAL mechanism exposed design faults not apparent at 1 g.

In other cases, the 1g environment hurt our system. The loading on the honeycomb DMR ring during t/v was going to be much more severe than in flight loading. Some scrambling and testing was performed close to t/v to ensure that no damage would be incurred.

4. QA REVIEW OF TASK ORDERS

QA review of Task Orders and Purchase Requests was performed to ensure that proper quality requriements were imposed. Most COBE contractors imposed proper requirements on themselves, but we occasionally ran into problems where QA review was missed and contractors did not plan for the cost of quality assurance. On other programs, we've also had contractors refuse to perform certain quality tasks because that clause has been left off the Task Order. Bottom line, great when we did it , shakey when we forgot.

5. CONTROL OF DESIGN CHANGES AND QA REVIEW

Tight control of design changes throughout the design and manufacturing process is necessary. This is the only way to accurately track the as-built configuration. Also, QA should be part of drawing review. This would enable us to review for materials and parts applications, fracture control implementation, etc.

MANUFACTURING AND PROCESSING

1. LIMITED LIFE

Limited life items were tracked. Because of unscheduled tests and activities, systems are often run more than originally planned. Tracking the limited life articles on COBE made it easy for us to feel comfortable about how much margin we had in system lifetimes.

2. CERT LOGS

Cert logs provided an excellent source of test and configuration information. Many many times, we were able to go back to cert logs and find out what cleaning procedure had been used on a box, or to verify the date of a test. Many people find them a bother to fill out, but they are the only piece of documentation that travels with the hardware throughout it's life. Engineer's notebooks may contain the same information, but different people that work on a box will have different notebooks, and these will not travel with the hardware.

Some cert logs were lost during the course of the program. Cert logs should be as carefully tracked as flight hardware.

3. ENVIRONMENTAL TEST COMMITEE

This was a special committee formed on COBE to ensure that the verification plan was executed on all subsystems. In addition to tracking testing, the committee acted as a peer review group for test plans and procedures.

4. CONTAMINATION

From the beginning of the hardware build to launch, contamination control should be considered. There are some schools that believe that hardware can be built in an uncontrolled environment and then cleaned to required cleanliness levels. Some hardware, such as harnesses, can never be brought back to acceptable levels once contaminated. The COBE program at GSFC maintained c.c. early on. Care taken in keep hardware clean helpd us during thermal vacuum preparation and of course, elimintaed flight problems.

At least two failures on COBE were traced to contamination introduced during manufacturing. Two boxes in the C&DH system had small metallic slivers introduced during the conformal coating process. These failures were not discovered until quite late in the program. Had the contractor maintained closer surveillance of the boards and environment, these problems may have been eliminated.

5. SURVEILLANCE OF CONTRACTORS

We can't be a at a contractor's plant all the time, and we don't really want to be. However, whenever possible, visits to contractors and potential contractors should be made to get a feel for their performance assurance program. Review of paperwork alone is not effective. You have to get out on the floor and see the equipment being used, see if the technicians are certified, verify if they are using the wonderful procedures they submitted in their proposal, etc.

INTEGRATION AND TEST

1. PROCEDURES

Procedures: a necessary evil. Procedures are there to remind a skilled technician or engineer what to do. They're not there to teach us what to do. They tell us what equipment we need, they provide QA with a checklist so that we can aid in the verification of the procedure. They provide a forum for the retention of data. They also provide a mechanism for review; one person is not planning and performing a test, a team of people are involved. The tests that worked best in our mind were the ones with the most clear procedures. The set up went quickly, the pass fail criteria were there, etc. Generally we ran into fewer surprises when working with a proc that was well thought out.

Whenever possible, more than one person should be qualified to run a procedure. That way, when one person is absent, I&T doesn't grind to a halt. A couple of our subsystems were undermanned during the test period.

A word on pass/fail criteria. More than once, we were burned when we didn't believe the pass/fail criteria in our procedures. An open circuit in the pyro harnesses went undetected for months, even though the data showed a clear failure. As QA people, we should not accept a quick, on the spot evaluation such as "it must be in the test box". Very often it <u>is</u> just a set up anomaly, but all test failures must be written up and evaluated as nonconformances.

Before beginning a test verify that all the necessary test equipment, break out boxes, etc. are available. If the procedures lists these right up front, it will be easy to just go through the list and check things out. The engineers and technicians sometimes had to spend some of their scheduled test time hunting up equipment.

2. ESD CONTROL

Yes, gloves and wrist stats are not always compatible with ESD control. The MTM watchdog box was damaged by ESD, possibly during a period when no wrist stat was being used at all. Everyone knows that this is important. COBE did not suffer any serious ESD failures, but after the BBXRT incident, we now realize that our procedures allowed for many of the same hazards. The Code 700 ESD working group should deal with this problem.

3. CONNECTORS

Mate/Demate logs should list mates per connector, not per day. We kept a daily log and it was almost impossible to tell how many times a particular connector had been mated or demated. GSFC does have log sheets in the new format. Although we'll end up with many sheets in the end, if we start at the beginning of a program and make up new sheets as we connect new connectors, the implementation won't be too difficult.

4. I&T MEETINGS

Daily I&T meetings were a good forum for planning test activities.

5. WALKTHROUGHS

At designated points through the program, COBE instituted walkthroughs. A representative from each subsystem was invited to come and review the hardware. In general, COBE was found to be in good shape, but we did discover discrepancies that would have caused problems later.

6. UNSCHEDULED TESTS

Times were always hectic on COBE, but we feel, in performance assurance, that we should have been more cautious of our approval of unscheduled tests. Although no damage ever occurred to any instrument or subsystem, the potential was there. We did have tests that wasted time because they were designed in haste. Time was spent investigating "problems" that were only misinterpretations of data. Sometimes too many parameters were changed at once, making interpretation of data difficult. Also, because these invariably occurred in the off hours, the people running the tests were not always as familiar with the system as the test designer. If problems occurred within the procedure, and mistakes were easy to make in a proc written in an hour, a lot of time was spent trying to get the thing to run.

7. WORK ORDER AUTHORIZATIONS (WOA'S)

Work Order Authorizations worked well.

GOOD - They made verification of work done and by whom very easy
- They allowed for control and review of work to be performed.
- QA was include in review, allowed for proper QA requirements, notified QA for scheduling of manpower, allowed us to close the loop on CCR actions.

BAD - Too many jobs were included on some WOAs. Work should be completed within one work period, otherwise, we found that a WOA could linger in the cleanroom for months. Occasionally, they got lost.
- Occasionally too many groups in the cleanroom at once. QA requirements for witnessing were sometimes missed. Sometimes QA had to act as a traffic cop in the test area. This could be avoided if the I&T group has the luxury of it's own traffic cop to manage the changing flow of work.

8. TEST SOFTWARE AND HARDWARE CONTROL

Control of test software and hardware does not have to be quite as rigorous as flight. However, we did run into problems because of our loose controls.

Failures in test harnesses cost us some time during I&T. Good commercial quality, if not flight level, should be something to strive for.

Design and configuration control of test equipment should not be taken lightly. Although we never damaged flight equipment, we did blow out a small piece of test equipment because miscommunication in a test box design. An evening's work would have been saved if the drawings had been reviewed with more care. Another time, we thought there was a short in the pryo harnessing. After several hours, it was discovered that a test turnaround plug was not wired as expected.

Once test equipment is working, it changing the configuration should not be considered trivial.

These examples are also reasons why safe to mate tests are so important.

A similar argument goes for test software. After a few bloopers on "improved" test software, we began to perform closer reviews of the procedures with the engineers. Going through the proc with the engineer step by step didn't take too much time, and we did ferret out some problems that way. For tests of critical functions, or procedures that could cause damage to the spacecraft, configuration control should be imposed.

9. CONSISTENCY

Consistency between performance tests that are run again and again through the life of the system are very important. When we had them, they provided a concise view of the system's health. When we didn't, we wasted time and effort trying to trend a system whose test parameters kept changing.

10. MISSION CONFIGURATION

Tests that were run as close to mission configuration as possible gave us the best insight into in-flight system performance. A concrete example of this was the use of flight arming plugs, not flight like. We learned a few times that what you think is exactly like flight is not always, by design or workmanship. It was worth our peace of mind to test with the real thing.

One add on to this is the requirement for control of small flight pieces like fuse plugs and arming connectors. These units go on and off the spacecraft and can be mishandled if not immediately put in a flight storage area.

11. NONCONFORMANCE REPORTING

A closed loop problem and malfunction reporting system was implemented for COBE. No nonconformance was closed until the work was completed. Some other systems close NR's when the corrective action is determined. Out of sight, out of mind, the corrective action can be forgotten.

12. FACILTY QA

Qaulity assurance for facilty operations such as thermal vacuum and vibration was attempted for COBE. Review of test procedures, verification of test equipment and set up were some of the things that we tried. For future programs, we recommend that facility groups be included in the full mission assurance program.

13. STAFFING

Can be summed up as long hours, tired people, mistakes are possible.

Special consideration could be taken for off shifts. Because of manpower problems, we sometimes had personnel on night shifts who were not quite as familiar with the systems as the day shift. Operational mistakes were made. And back to the unscheduled test theme, QA should be very wary of unscheduled tests during off hours when full staff is not present for review of procedures.

LAUNCH SITE

1. AVOID NEW, UNREHEARSED TESTS CLOSE TO DATE OF LAUNCH.

We had a test planned that would check critical dewar valves one last time before launch. Although this test was reviewed carefully a number of times before its execution, a glitch occurred. The test harness, which had been used on the spacecraft for at least a year, had turnarounds that we were unaware of. The test failed on the first attempt. It was the day before launch, and every hour counted. The problem was discovered and corrected, and the test completed in time. This may not always be the case. It also may not always be the case that the engineers retain cool heads. To start unscheduled troubleshooting or test activities with the clock ticking away can cause people to make serious judgement errors.

2. ADEQUATE STAFFING

Because of lack of travel funding, or attrition, inadequate staffing was often a problem. Personnel were forced to work long hours. In cases where there was only one representative from a subsystem or instrument, that individual would work late into the night. Concentrated effort is impossible over such a long day.

3. COMMUNICATION CHECKOUT

Check out the communications net and rehearse protocols before launch day. We discovered that there were many faults in the communication system during rehearsals. Channel by channel checkouts ensured that the net was up and running by launch day.

4. INTERFACE BETWEEN LAUNCH VEHICLE AND SPACECRAFT PERSONNEL

In one famous incident, COBE I&T made a request to the Delta shift manager to stop fairing work. Although that request was accepted, the crew in the tower continued work. The ensuing miscommunication caused the COBE to be exposed in the tower for some four hours. In another, the COBE FAM was told at a Delta staff that COBE personnel could safe the spacecraft before pad closure. At the same time, a call came in from the pad that the crew was being ejected.

At the beginning of operations, we could have used a little more understanding between Delta and COBE personnel on who to go to in event of a schedule change.

More interface for I&T meetings to understand launch vehicle schedules as they mesh with COBE would have been useful. Those kind of integrated schedules were instituted later in the game and they were helpful.

Also, special spacecraft safety restrictions should be made clear to vehicle personnel. Contamination was a big item that was discussed many months before our arrival to the launch site. Restrictions against flash photography, Xray, and cleaning solutions, could also have been included.

5. MAINTAIN THE STATUS QUO

As work level plateaued, daily I&T meetings became erratic. The I&T meetings allowed different groups to schedule tests efficiently. They also provided a forum for reviewing complex test procedures and making sure all participating groups were prepared and understood their roles in the test. Without good meetings, personnel were unsure at times when a test was to be run.

POST LAUNCH

1. SPACECRAFT ORBITAL ANOMALY REPORTS (SOAR'S)

COBE has implemented the GSFC SOAR system. This adds COBE data to the ever growing database for GSFC spacecraft.

2. MISSION DATA CENTER

COBE's idea to have a mission data center was a good one. Although the execution became difficlt with limited manpower, it was useful to hae a single location for storage of tapes, data packages test reports and drawings.

3. FLIGHT STOCKS

Keep critical flight stocks available in case of in-orbit failure. Spare boards and parts were used for troubleshooting of the Rate Measurement Assembly failure and the Firas MTM anomalies.

AH 26

COBE
COSMIC BACKGROUND EXPLORER

COBE ACS

OVERVIEW

H. C. HOFFMAN
GUIDANCE AND CONTROL BRANCH
CODE:712 ACS-1

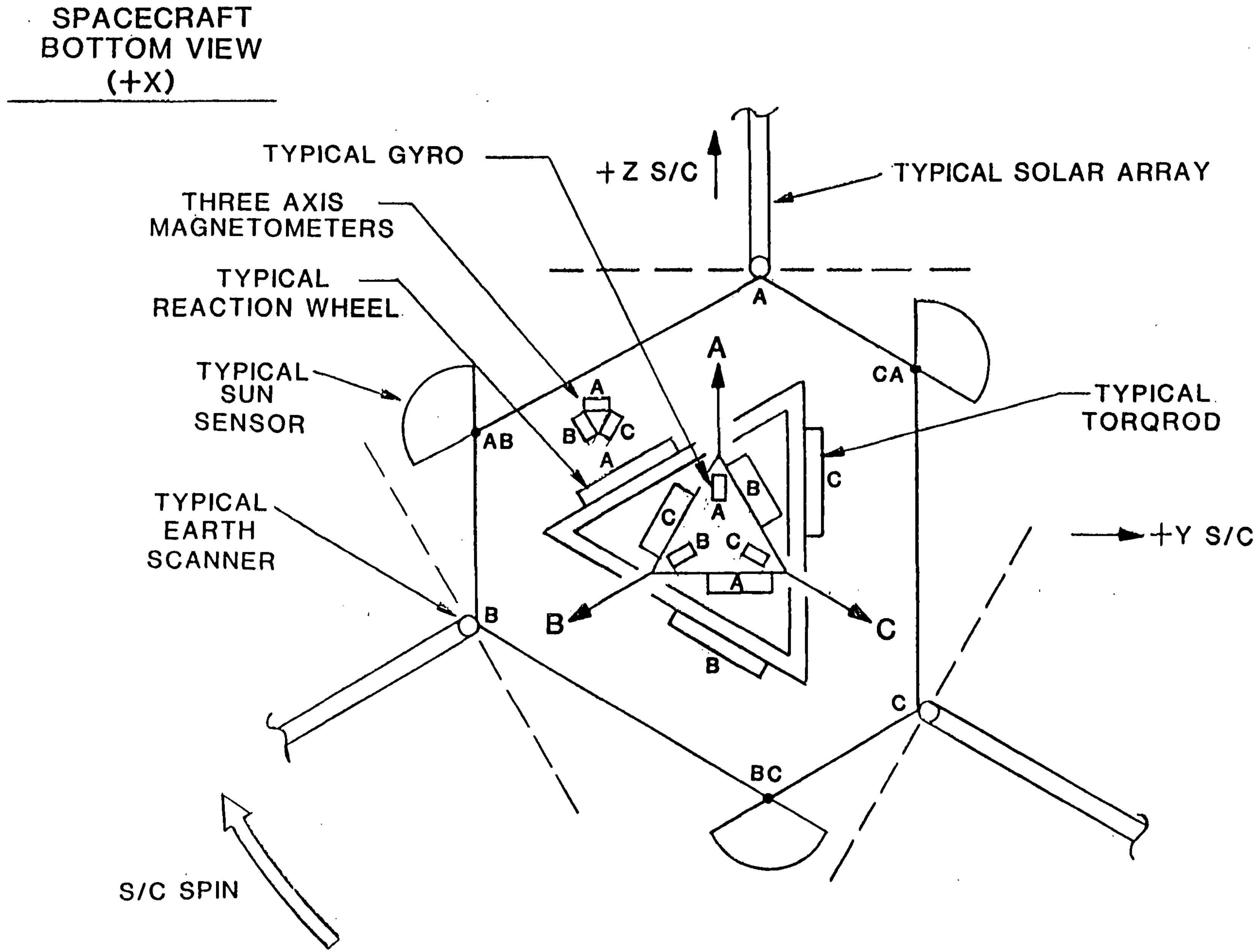
DESIGN PHILOSOPHY

- DESIGN BACKWARDS
- ALL SENSORS AND MUSCLES MOUNTED CROOKED
- IN ORBIT/TRANSFER/BOOST
- MULTIPLE SIMULATIONS
- KEEP ANALYSIS ON-LINE THRU LAUNCH

ACS CONCEPT

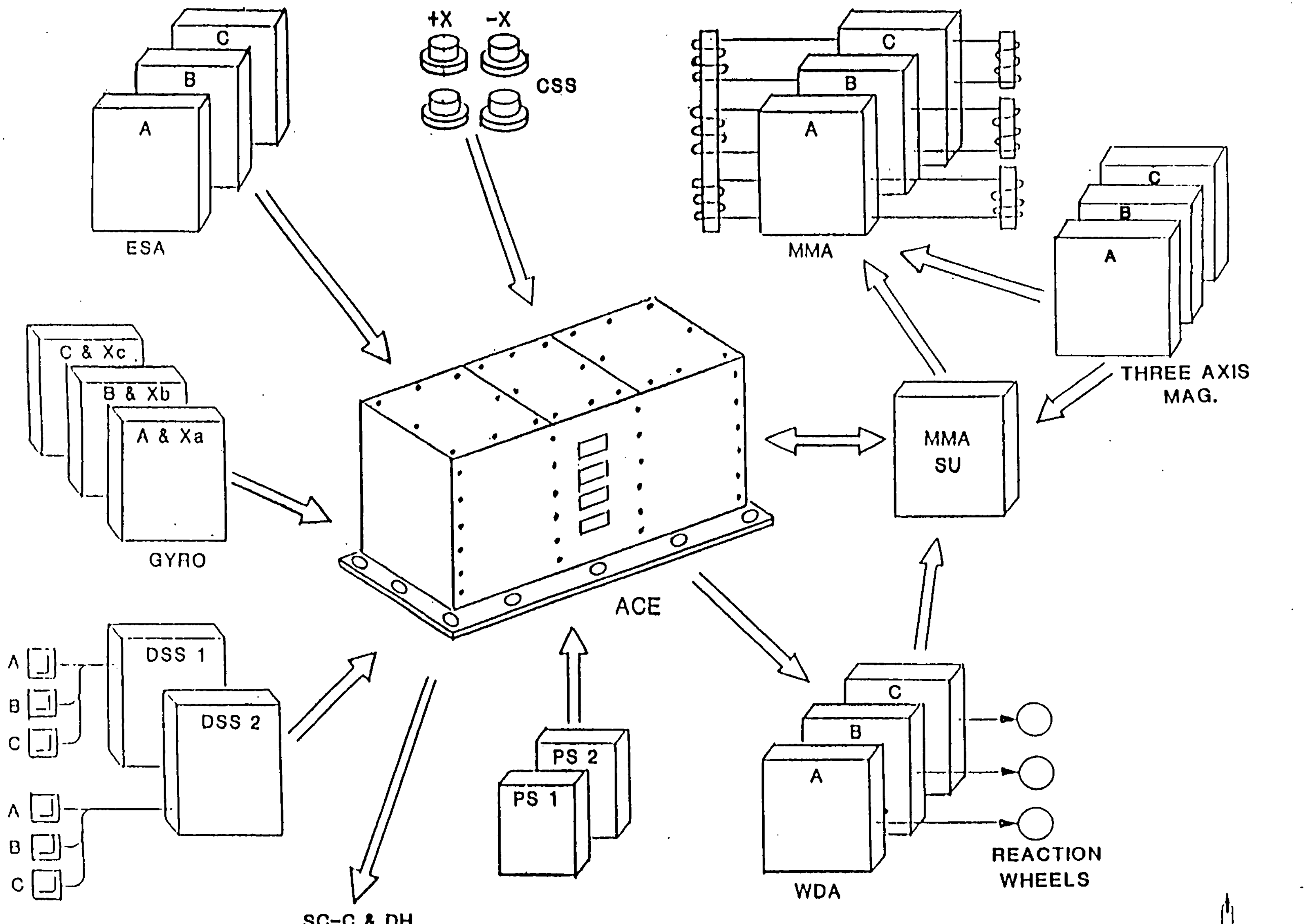
- 3 AXIS SYSTEM THAT SPINS
- BASICALLY ANALOG
- A, B AND C AXES CONTROL PITCH AND ROLL
- YAW AXIS SPINS

ACS-3



BASIC COBE GEOMETRY

ACS-4



COBE ACS
COMPONENT INTERFACE

ACS-5

PERFORMANCE

- LAUNCHED NOVEMBER 18, 1989
- CONTROL SYSTEM PERFECT (EARLY ORBITS)
- NO UNSCRIPTED COMMANDS REQUIRED
- GYRO FAILED AFTER ONLY 4 DAYS
- UNEXPECTED ECLIPSES JANUARY 26, 1990

LESSONS LEARNED/EARLY ORBIT PERFORMANCE

- ESA EMI PROBLEMS ——————————————— P. NEWMAN
- GYRO FAILURE/FOLLOW-ON ANALYSIS ——— S. PLACANICA
- ECLIPSE ——————————————————— M. FEMIANO
- SUN SENSOR PERFORMANCE —————————— T. FLATLEY

ACS-7

ACS-8

COBE

EARTH SENSOR

EMI

P.A.NEWMAN 712

ACS-9

COBE

EARTH SENSOR

EMI PROBLEM

(RADIATED SUSCEPTIBILITY)

AND ITS SOLUTION

Phillip A. Newman Jr.

NASA Goddard Space Flight Center

CODE 712.2

Greenbelt, MD 20771

ACS-10

Introduction

The COBE Earth Sensor is a microprocessor controlled instrument that operates in the 14-16 micron spectral region. The microprocessor analyzes a sampled earth signal, determines the earth center and outputs an error signal, proportional to the angular distance from the nadir, that is used to control the spacecraft. The microprocessor also commutates the brushless dc motor and controls its speed. Figure 1 shows a block diagram of the COBE Earth Sensor. The sensor met all of the NASA performance requirements during acceptance test and the four flight units have performed flawlessly in flight.

However the COBE Earth Sensor was found to be extremely sensitive to modulated RF, especially in the frequency range of UHF/L band radar (400 to 450 MHz), at field strengths stronger than 1V/m. It was determined that there were a number of operational radar systems that could produce fields of 3V/m or greater at the COBE orbit. A solution was found in a Faraday cage that completely shielded the Earth Sensor up to 10V/m and did not interfere with the optical performance.

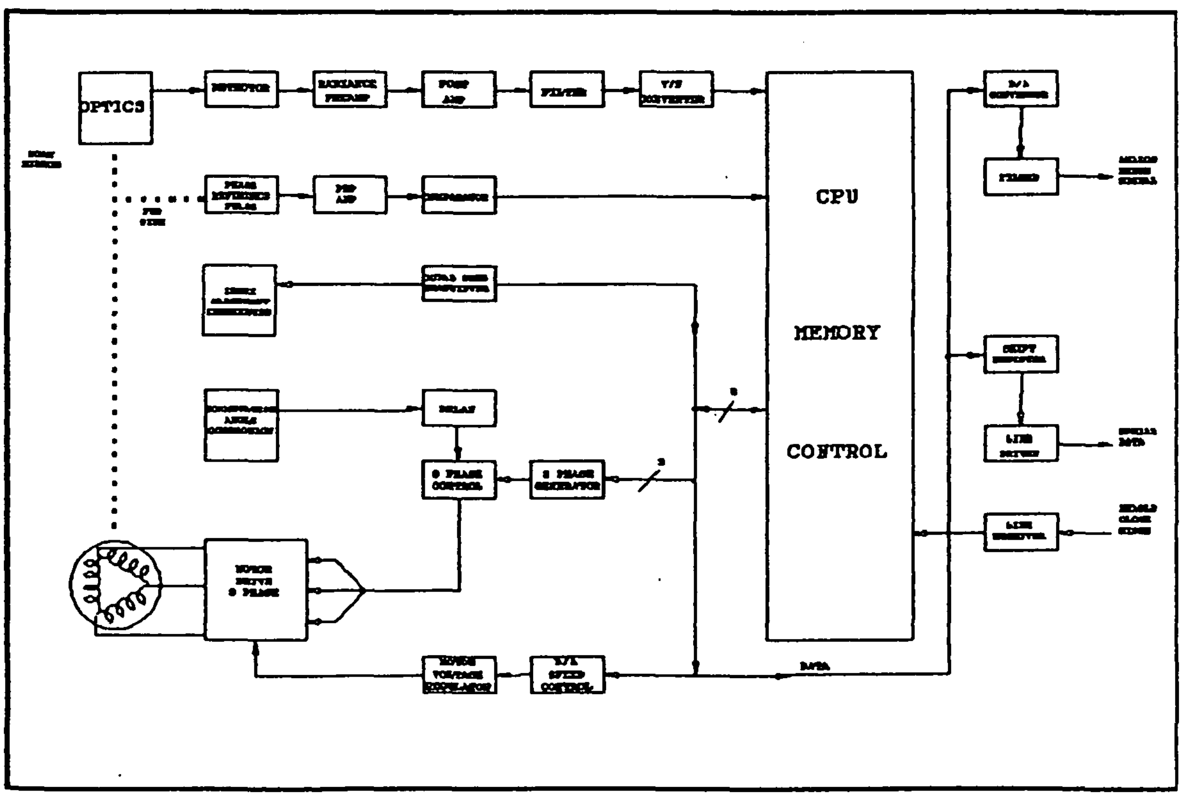


Figure 1 COBE Earth Sensor Block Diagram

Chronology

The Earth Sensor Qualification Model underwent a complete EMI test as required by the specification and MILSTD 461A. The test in question, RS-03 radiated susceptibility, was passed to the spec limit of 2V/m. MILSTD 461A does not require modulation of the RF although later versions do.

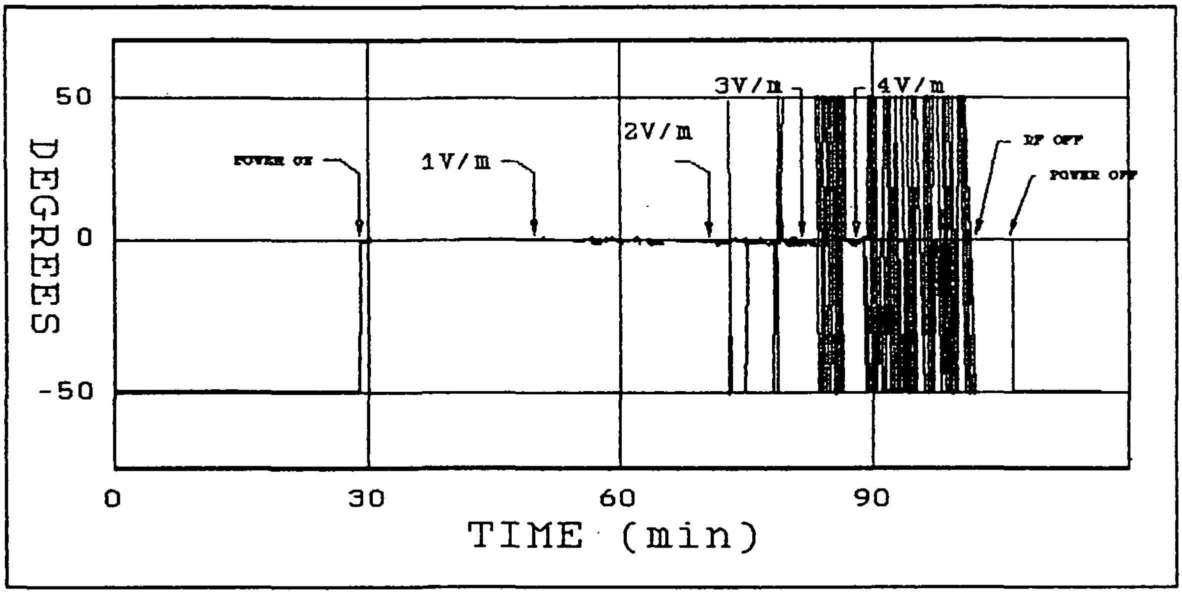


Figure 2 Earth Sensor Error Signal during S/C RF Susceptibility Test

During a spacecraft level EMI susceptibility test in May of 1989, where the RF was modulated at 30 Hz and 100.1 Hz in order to accommodate one of the experiments, it was found that the Earth Sensor head closest to the antenna gave spurious error signals that eventually saturated and even locked up the sensor. The test was conducted without an earth target.

At first the concern was with high power radars at the launch site causing a lock-up in a sensor that would result in a hard failure. An investigation indicated that component damage was unlikely at field strengths less than a microwave oven. By mid-July it became clear that there were a number of radars at undisclosed locations that had high enough field strengths to cause serious operational problems. Later, additional systems were disclosed that could produce even higher field strengths.

A component test was conducted on the qualification model Earth Sensor in early September of 1989. It was discovered that the modulated rf was being detected by the sensor head. The differentiated signal could be seen riding on top of the analog earth sensor video signal (figure 3).

ACS-12

A second spacecraft level EMI test was conducted in the last week of September of 1989 where each of the three flight sensor heads were positioned, in turn, in close proximity to the antenna. All heads had the same problem to varying degrees. Several approaches to shielding the sensor head were tried with some success. A crude partial Faraday cage seemed the most promising.

A tiger team was formed to solve the problem. The first meeting at Barnes Engineering in Shelton, CT produced a cage design and outlined a critical test plan to assure that it would solve the problem without impacting the ESA performance.

The first unit was fabricated over a weekend using 5 mil Be-Cu wire. The EMI test showed no evidence of the modulated RF in the video at levels of 10V/m and greater. Subsequent optical tests were all successful. This first qual model broke a wire during acoustic test. It was determined that the wire had a substantial number of defects (voids) that made it unsuitable for this application. The unit was reworked with a new source of 4 mil Be-Cu wire and passed all of the mechanical testing without failure.

Four flight units were completed and tested by October 26, 1989 and were shipped to Vandenberg AFB for installation on the spacecraft prior to the closing of the Delta shroud.

No evidence of any earth sensor anomaly has been observed since launch.

<u>Description of the Problem</u>

The analog circuits of the earth sensor are designed to optimize the signal to noise characteristics of the earth signal. A sun signal is clipped at about +1V. The ac coupled earth signal normally rides between a dc level of -4 to -1 volt for a 250°K earth.

The modulation of the RF is detected as a differentiated signal riding on top of the earth signal (figure 2). Since it is not ordinarily at an exact multiple of the scanner signal it moves with respect to the earth pulse at each repetition. At low levels the only noticeable effect is when the modulation signal moves over an edge. This has the effect of changing the apparent position of the edge slightly and thus the error signal appears noisy. The real problem occurs as the RF level increases. When the modulation signal reaches the clip level and the lower power supply rail, the error signal starts to go between ±50 degrees, a saturated error signal. As the level gets higher the differentiated RF modulation signal begins to broaden and flatten at its peak and raise the dc level of the signal. At this point the error signal usually goes to either ± saturation and stays there.

AS-13

On at least three occasions an earth sensor has been seen to 'lock up'; that is the error signal remained anomalous even after the RF was removed.

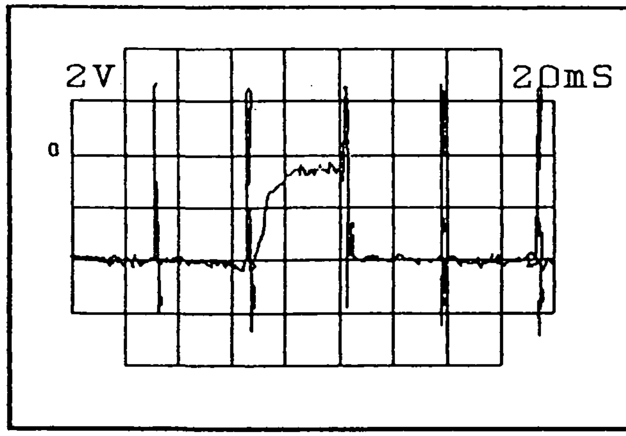


Figure 3 COBE ESA Video signal vs Time. 403MHz/30Hz/1ms

Description of the Solution

A Faraday cage was built that consisted of fine wires (4 mil Be-Cu) strung and pre-tensioned between upper and lower Be-Cu plates (Figures 4 and 5). The upper plate was joined to the lower plate by a member located at the anti-nadir position in the blanking region of the sensor. The lower plate is fastened to a brass collar which is electrically and thermally tied to the sensor by a section of RF finger strip located around its inner circumference. The collar is mechanically fastened to the mounting feet of the sensor head through stand off studs.

The 4 mil wires are located 0.8 apart on a circle that is somewhat larger in diameter than the sensor. A major concern was the effect of these wires on the optical performance of the sensor due to obscuration, heating or the reflection of solar energy.

A critical test was conducted at Barnes Engineering where an engineering model Faraday cage was fabricated with the wires configured so that they could be heated up under controlled conditions and observed by the earth sensor against a liquid nitrogen background. Half of the wires (consecutive) were heated at a time while the video signal was observed for any modulation. In this test the wires were heated to red heat (≈1200°C) without any observable effect.

Another test involved the possible sensing of solar radiation reflected off the wires. A 1 solar constant collimated infrared source was used at Barnes to stimulate the sensor from every angle; again without observable effect.

A third optical test concerned itself with the margin between the edge of the field of view and the Be-Cu plates. This was conducted in the near field with a hot soldering iron and showed that a substantial margin existed.

The EMI test on the Faraday Cage swept the RF signal, modulated at 30 Hz with a 1 millisecond pulse width, from 10 KHz to 2.6 GHz with vertical and horizontal polarizations at a field strength of

ACS-14

10 V/m. For this test only the analog video was observed. Except for one narrow frequency band at 900 MHz no evidence of the modulation signal could be detected.

At this point all of properties of the Faraday cage were known and found to be satisfactory with the exception of its ability to survive launch loads. Since the fine wires of the Faraday cage are very delicate and can easily be damaged in handling, a modification of the qual model grounding circuitry was accomplished as a possible backup position. The qual model earth sensor head was modified to tie the chassis ground and signal grounds together as close to the detector as possible and to cut the self test LED leads and also ground them at the same place. An EMI test showed that this did not improve the susceptibility at all.

The final qual model and flight model Faraday cages underwent acoustic and random vibration tests to anticipated launch load levels without failure.

Conclusion

The Faraday cage, as designed and fabricated by a joint team of engineers and technicians from the GSFC and Barnes Engineering as well as some of their contractors, meets all of the requirements for suppressing the potential susceptibility of the earth sensor to high power radar signals while not impacting, to any degree, its performance.

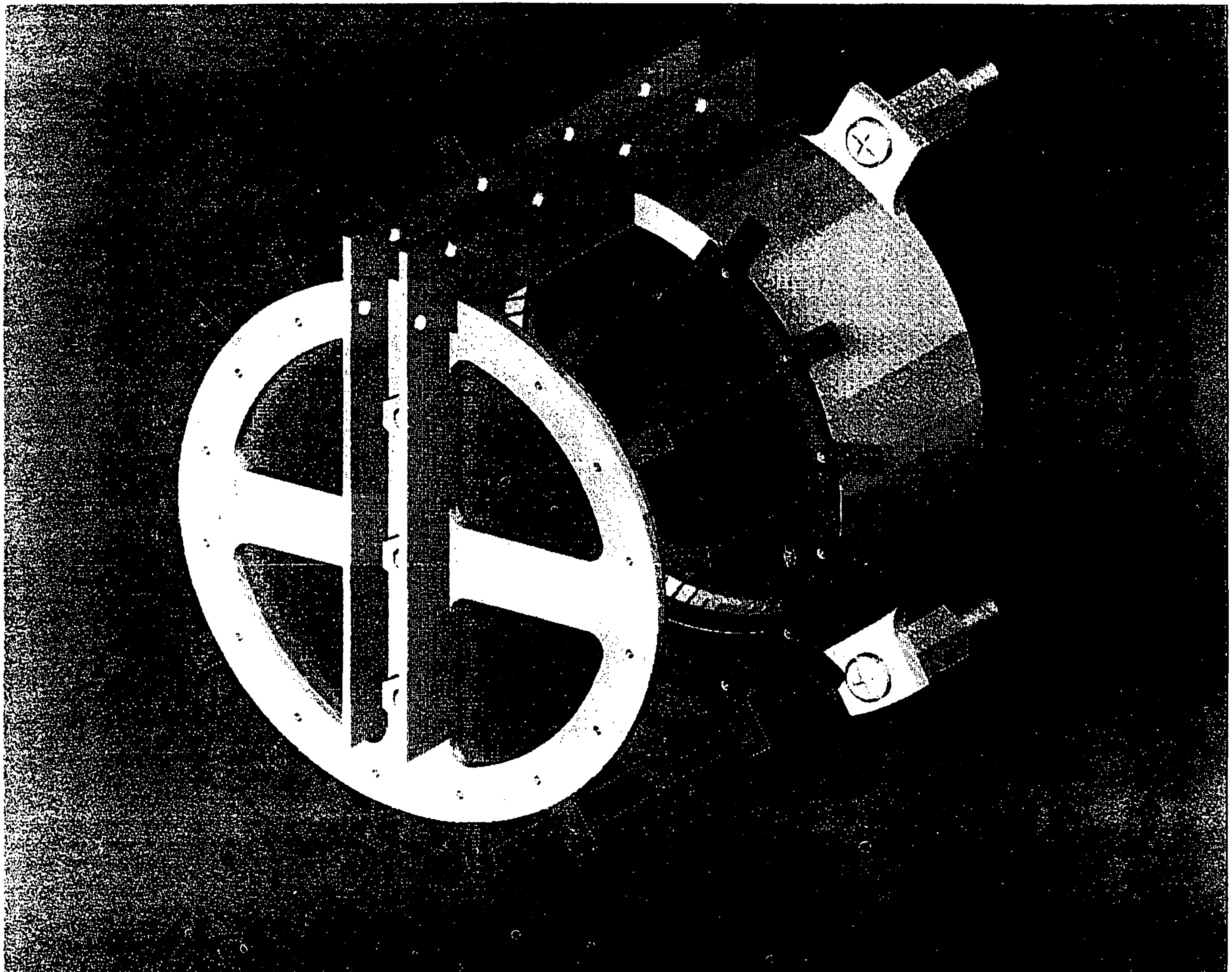


FIGURE 4

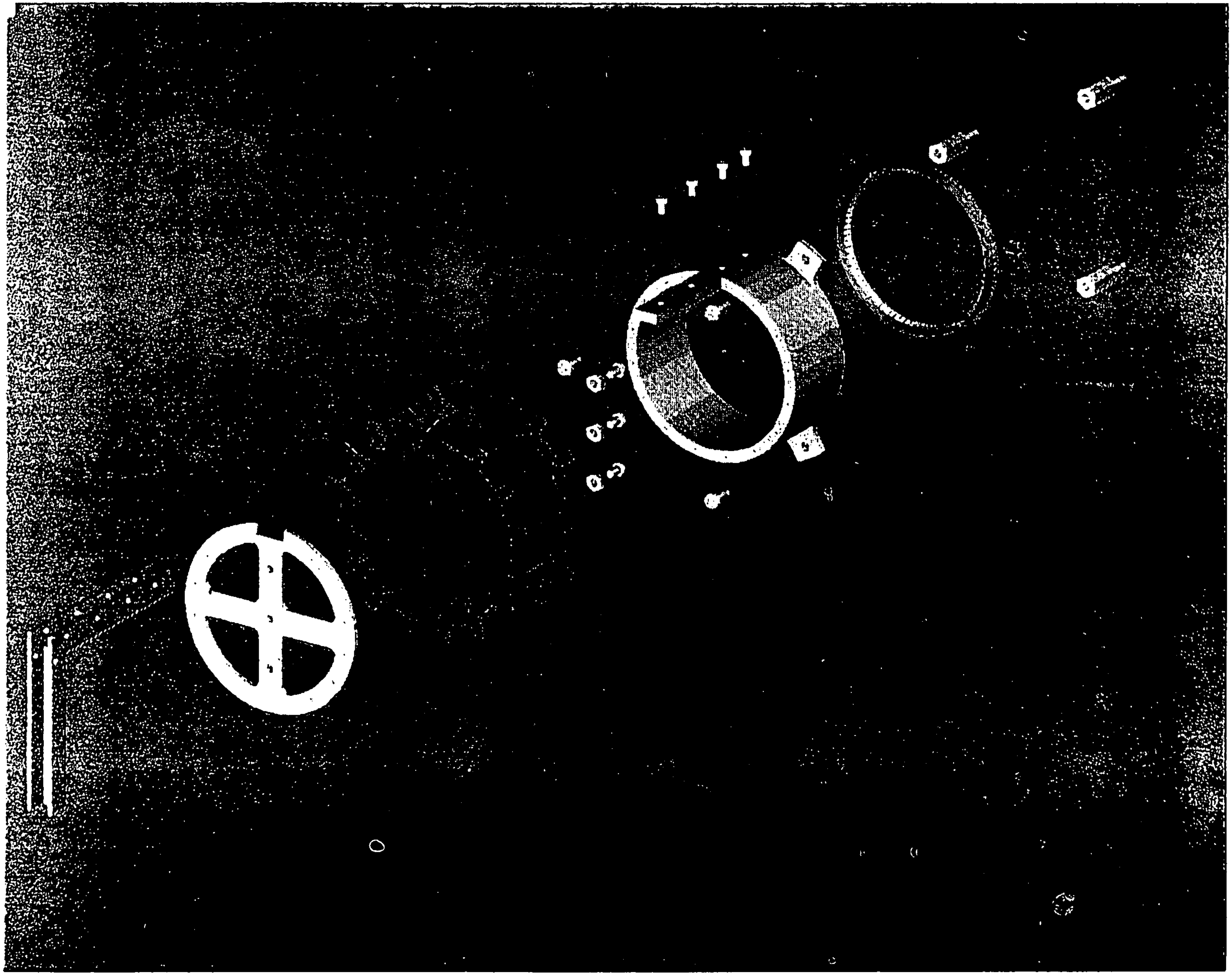


FIGURE 5

GYRO FAILURE/FOLLOW-ON ANALYSIS

S. PLACANICA

ACS-17

GYRO FAILURE

- November 22, 1989 05:30 Z (Day 326 - 4 days after launch) B-axis transverse gyro failed
- Operated in this failure mode until 07:23 Z when the B = – (A + C) cross-strapping command was given
- Performance during failure mode

  Sun elevation angle range: –1.1° to –6.3°

  Pitch angle range: 1.5° to 7.2°
- Following the cross-strapping command, the ACS resumed nominal pointing performance
- Current mode as of Day 327 11:37 Z

  Gyro B off

  No cross-strapping

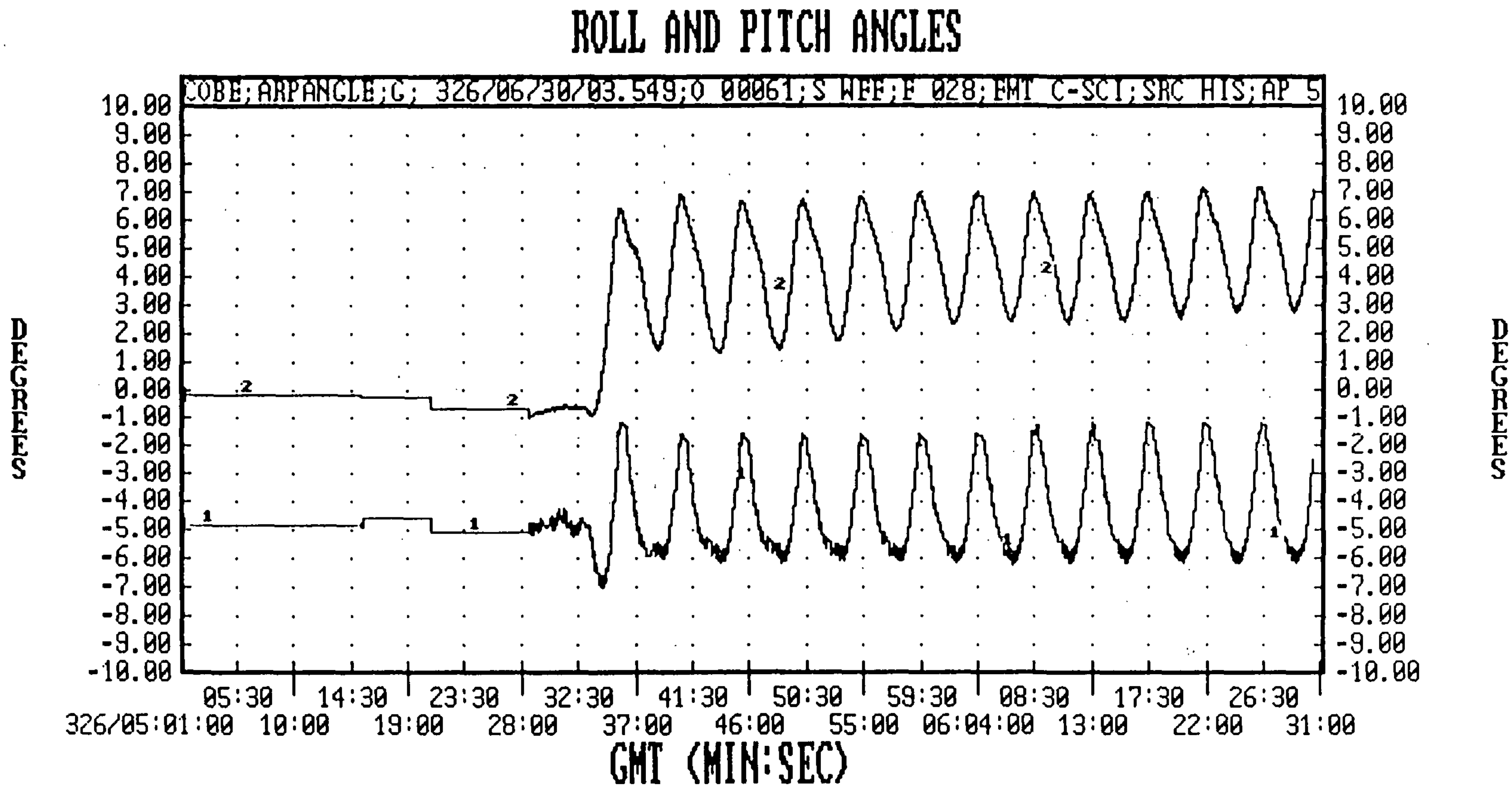
  No orbit rate stripping

B-AXIS GYRO FAILURE CHARACTERISTICS

- Motor current dropped from 75 to 22 milliamps
- Zero pulse counts
- Analog rate registered negative full scale (zero volts)
- Gyro baseplate temperature increased from 24°C to 39°C
- 1.3 amp drop in the essential bus current following the removal of 28 volt power to gyro

GYRO FAILURE

- Northrop, the gyro manufacturer, experienced a similar failure during a gyro test two weeks before the COBE in-flight failure
- Appears to have been an overloading of the 22 volt regulator
- 57 different electronic parts could have caused the failure
- The COBE Gyro Review Committee concluded that the failure was a random electronic part failure that overloaded the regulator



ITEM	MNEMONIC	VALUE	FLAGS	BAD QUALITY	OFF SCALE	SCALE SIDE
1	AROLLANG	-2.34		4	0	LEFT
2	APTCHANG	7.22	RH	4	0_	RIGHT

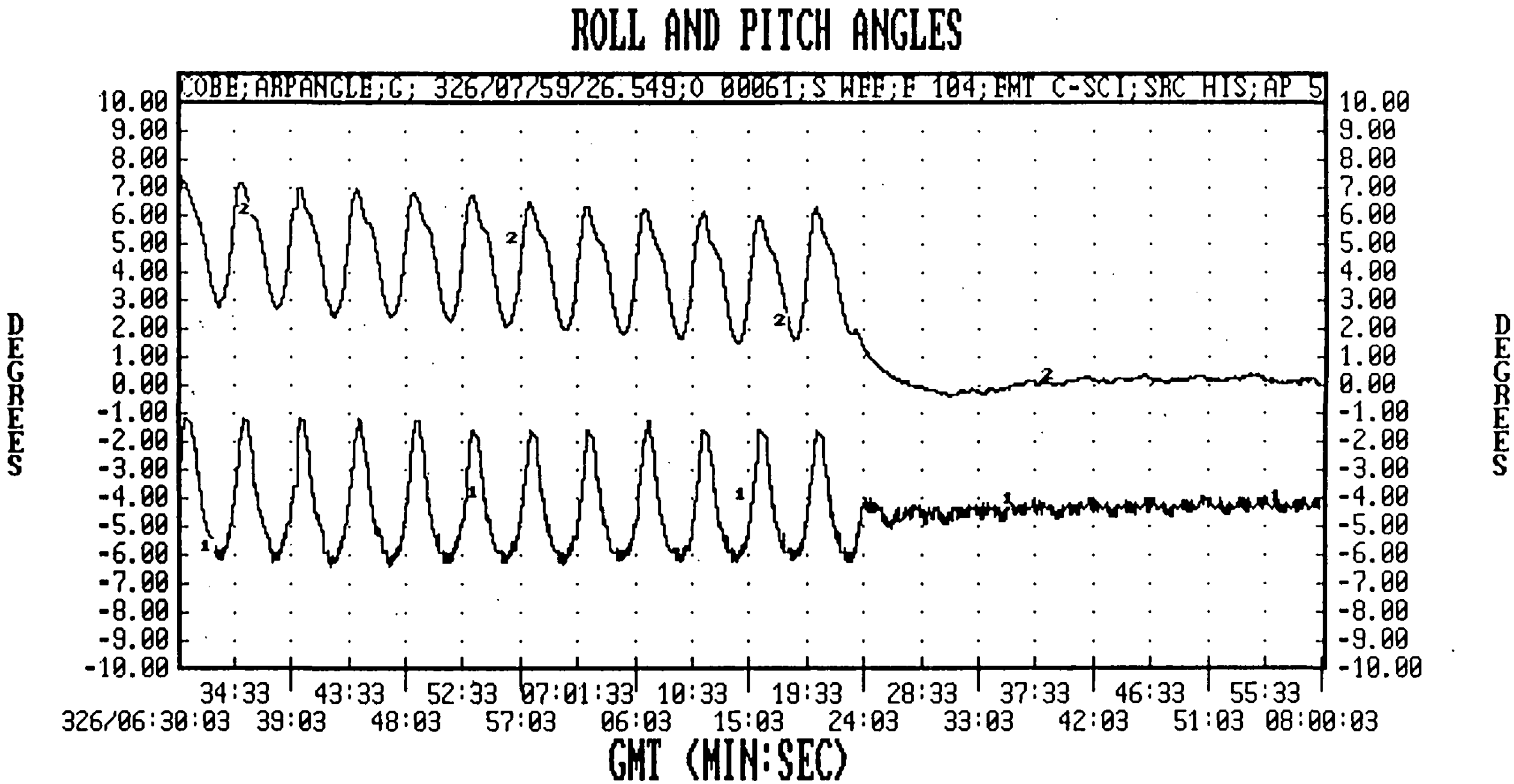
RUN-TIME COMMANDS

CTRL-F6: Activate Page
CTRL-F7: Restart Plot
CTRL-F8: Stop Plot
ALT-F1: ISC Terminal
ALT-F4: Setup Screen

ALT-F10: Quit

Gyro B Failure

ACS-21



ITEM	MNEMONIC	VALUE	FLAGS	BAD QUALITY	OFF SCALE	SCALE SIDE
1	AROLLANG	-4.09		5	0	LEFT
2	APTCHANG	.04	RH	5	0	RIGHT

RUN-TIME COMMANDS

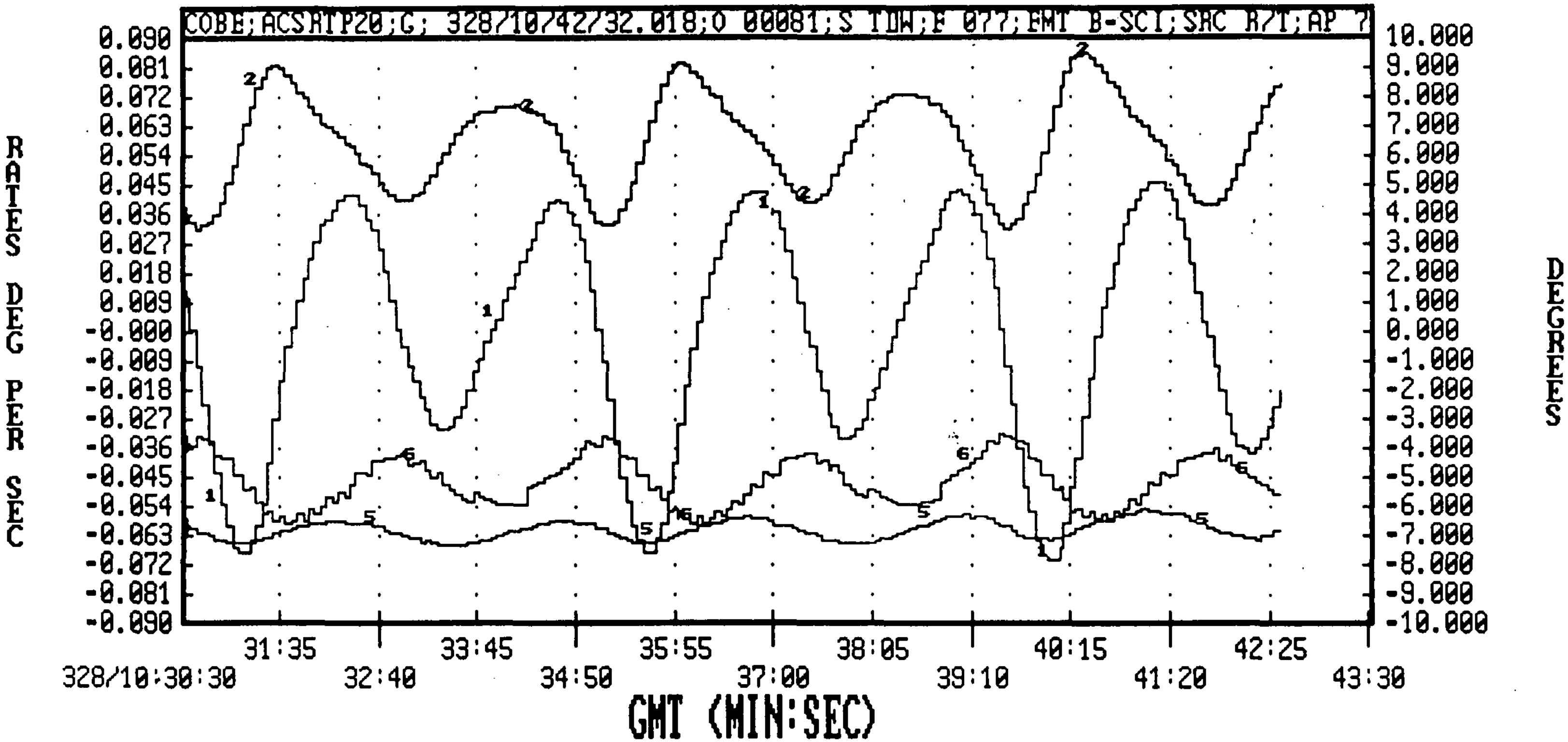
CTRL-F6: Activate Page
CTRL-F7: Restart Plot
CTRL-F8: Stop Plot
ALT-F1: ISC Terminal
ALT-F4: Setup Screen

ALT-F10: Quit

Initiation of B = – (A + C) Cross-strapping

ACS-22

ROLL AND PITCH ANGLES



ITEM	MNEMONIC	VALUE	FLAGS	BAD QUALITY	OFF SCALE	SCALE SIDE
1	AGROLLRT	-.010		0	0	LEFT
2	AGPTCHRT	.079		0	0	LEFT
5	APTCHANG	-6.690	YL	0	5	RIGHT
6	AROLLANG	-5.732		0	15	RIGHT

RUN-TIME COMMANDS

CTRL-F6: Activate Page
CTRL-F7: Restart Plot
CTRL-F8: Stop Plot
ALT-F1: ISC Terminal
ALT-F4: Setup Screen

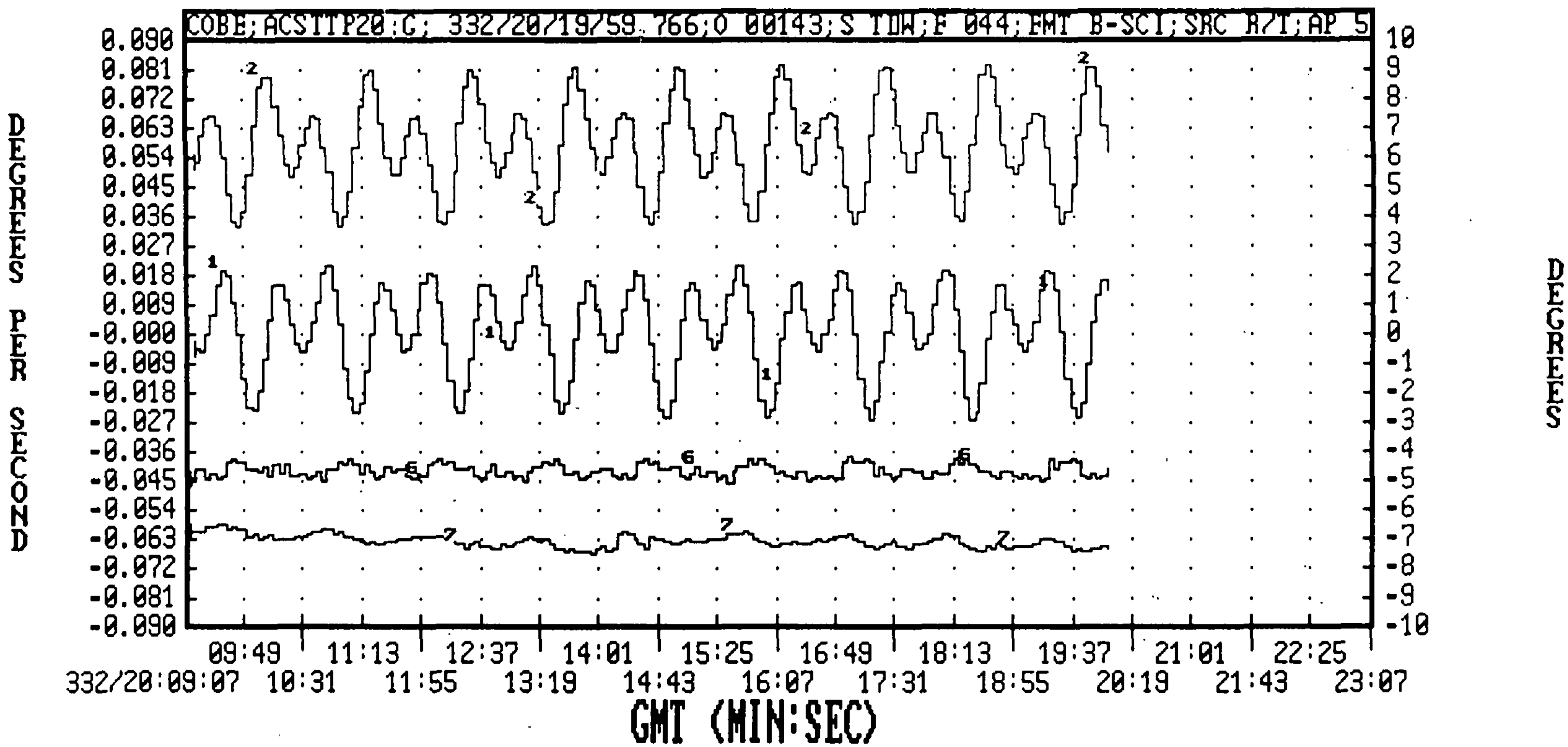
ALT-F10: Quit

Spin Rate = – 0.2 rpm, Gyro B Off,

No Cross-strapping, No Orbit Rate Stripping

ACS-23

ROLL AND PITCH ANGLES



ITEM	MNEMONIC	VALUE	FLAGS	BAD QUALITY	OFF SCALE	SCALE SIDE
1	AGROLLRT	.011	S	0	0	LEFT
2	AGPTCHRT	.053	S	0	0	LEFT
6	AROLLANG	-4.79220	S	0	0	RIGHT
7	APTCHANG	-7.17118	SRL	0	0	RIGHT

RUN-TIME COMMANDS

CTRL-F6: Activate Page
CTRL-F7: Restart Plot
CTRL-F8: Stop Plot
ALT-F1: ISC Terminal
ALT-F4: Setup Screen

ALT-F10: Quit

Spin Rate = – 0.815 rpm, Gyro B Off,

No Cross-strapping, No Orbit Rate Stripping

ACS-24

FOLLOW-ON ANALYSIS -- MISSION MODE

- HIFIJR simulations investigated risk of an additional component failure during mission mode operation
- Failure modes: Gyro, earth scanner, torque rod, reaction wheel
- Matrix of 45 runs
- Worst case simulations result in a roll angle of 8.6° into the sun

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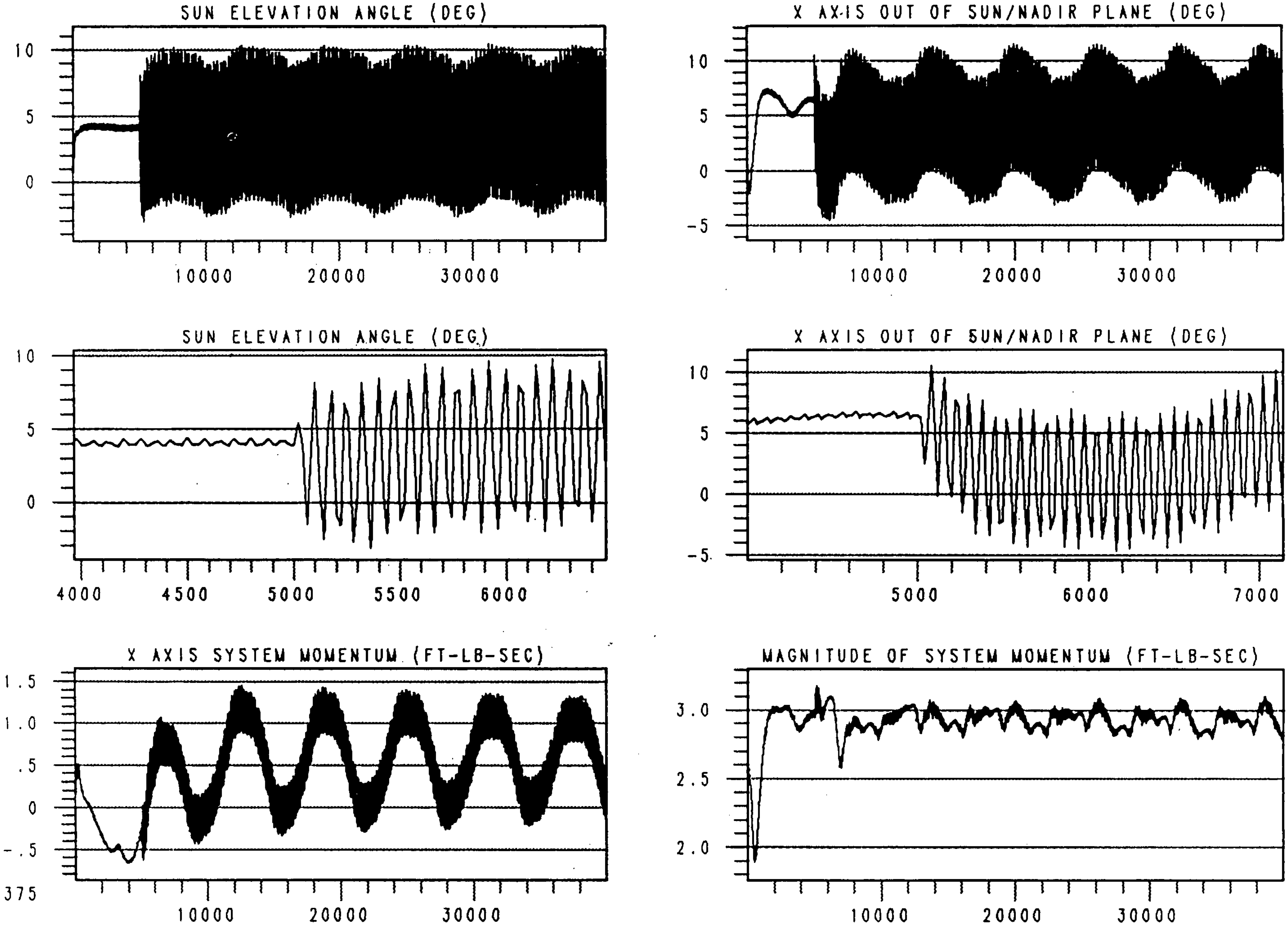
MISSION MODE
WORST CASE GYRO A FAILURES

	Gyro A Failure <u>High Positive</u>	Gyro A Failure <u>High Negative</u>
Roll angle range	–2.81° to 10.40°	–0.94° to 9.37°
Roll angle transient	–3.15°	–1.20°
Avg roll angle	3.82°	4.14°
Pitch angle range	–3.21° to 11.50°	–2.15° to 9.80°
Avg pitch angle	4.19°	3.82°

File: COBEHI.BIN;375 COBE MISSION MODE The time is 09:21:48.11 30-NOV-1989



GYRO B OFF; ALL STRIPPING OFF; GYRO A FAIL HIGH POSITIVE AT 5000 S

ACS-27

MISSION MODE
WORST CASE GYRO C FAILURES

	Gyro C Failure High Positive	Gyro C Failure High Negative
Roll angle range	–0.39° to 7.83°	–0.24° to 7.84°
Roll angle transient	–0.70°	–0.70°
Avg roll angle	3.67°	3.77°
Pitch angle range	–1.46° to 13.21°	–1.64° to 12.68°
Avg pitch angle	5.68°	5.23°

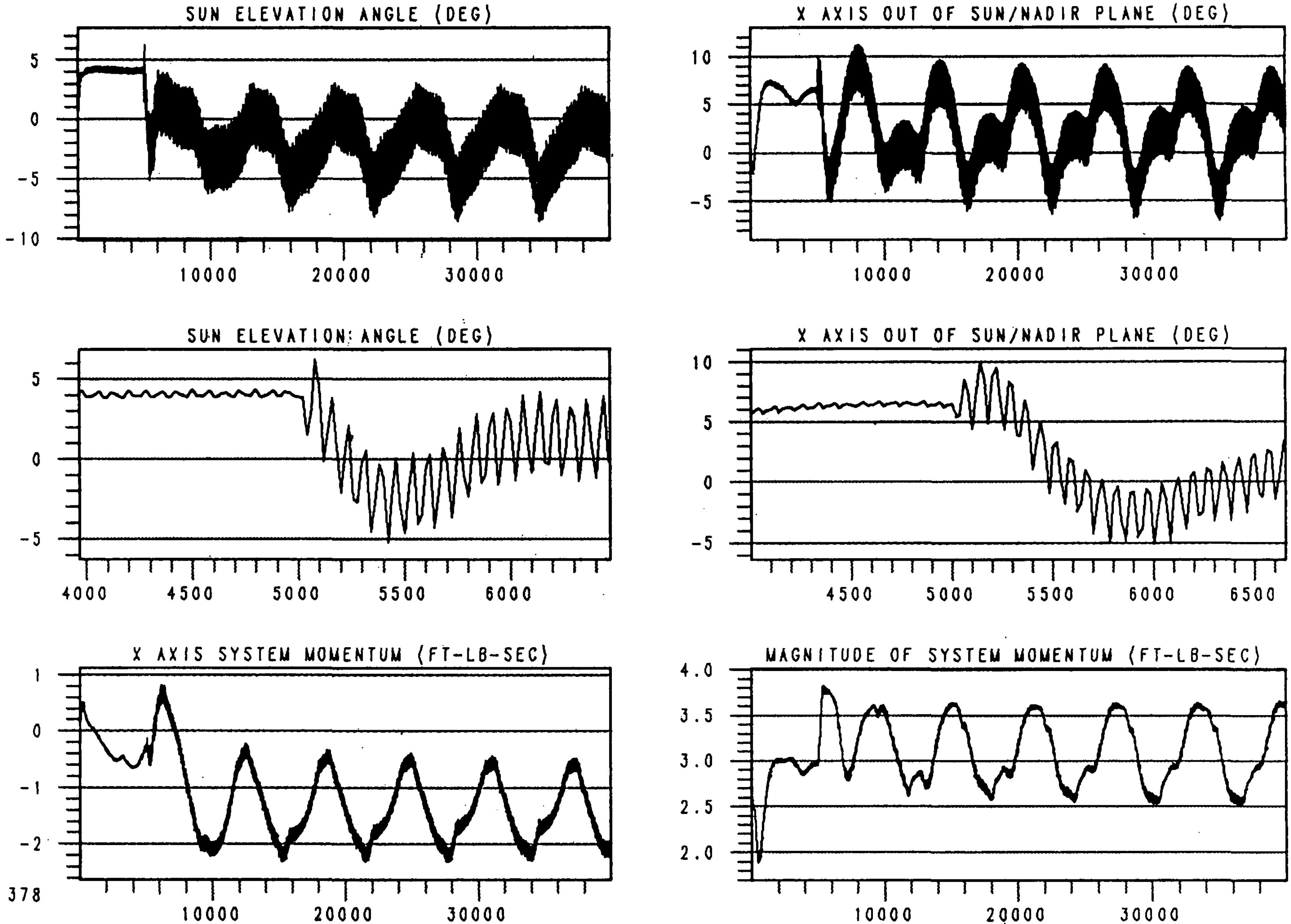
ACS-28

MISSION MODE
WORST CASE SCANNER FAILURES

	Scanner A Failure <u>High Positive</u>	Scanner C Failure <u>High Positive</u>
Roll angle range	–0.66° to 8.39°	–8.62° to 3.39°
Roll angle transient	–1.52°	–5.24°
Avg roll angle	4.09°	–1.91°
Pitch angle range	1.23° to 13.63°	–7.12° to 11.12°
Avg pitch angle	7.26°	2.70°

A-CS-29

File: COBEHI.BIN;378 COBE MISSION MODE The time is 09:49:24.72 30-NOV-1989



RUN 378

GYRO B OFF; ALL STRIPPING OFF; SCANNER C FAIL HIGH NEGATIVE AT 5000 S

ACS-30

FOLLOW-ON ANALYSIS -- ECLIPSE SEASON

- Predicted performance during deepest eclipse for current configuration of B-axis gyro off and no orbit rate stripping

  <u>Sun elevation angle</u>

  Range: 2.98° to 10.71°

  Average: 4.69°

  <u>Pitch angle</u>

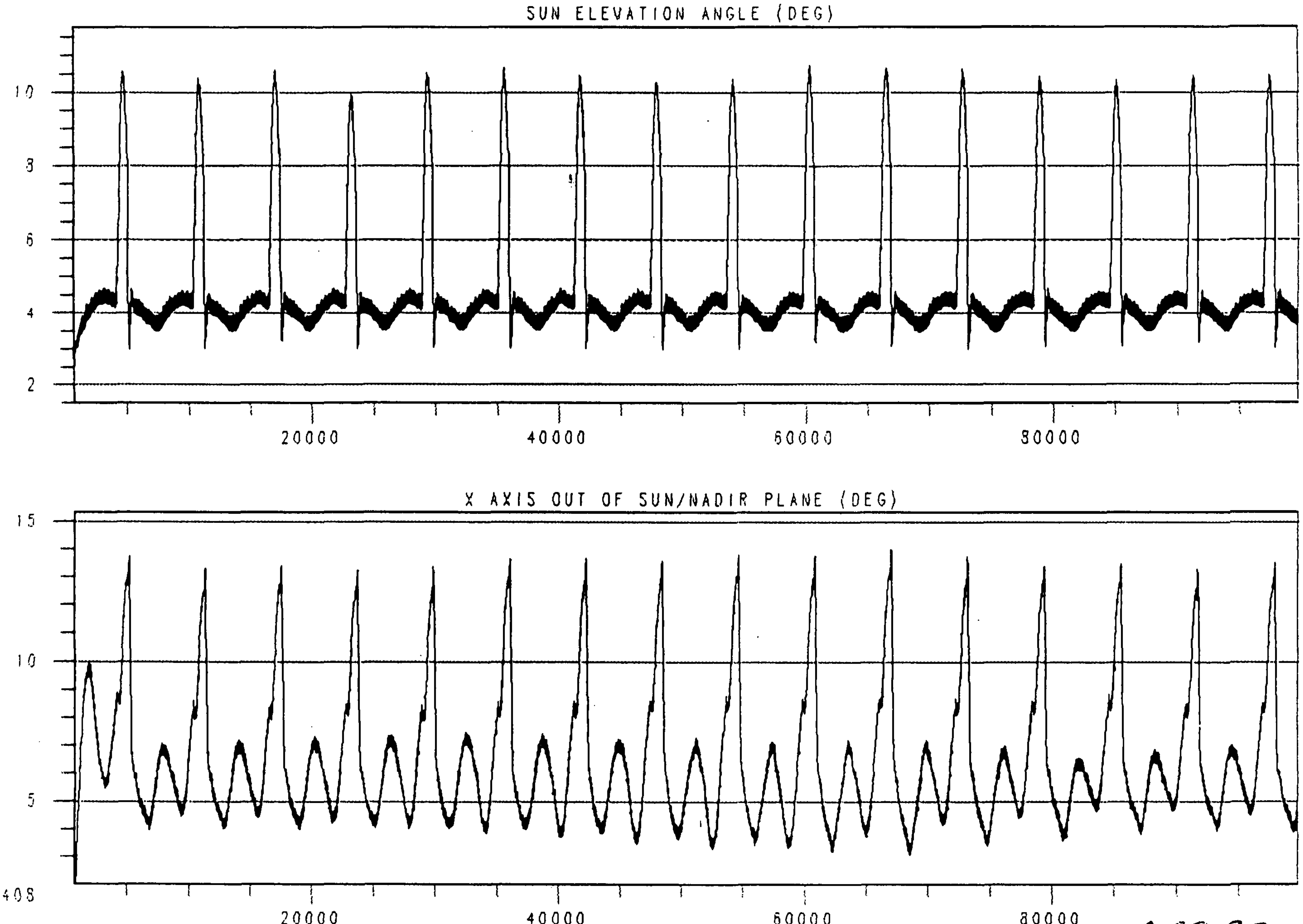
  Range : 3.06° to 13.99°

  Average: 6.44 °

- Summer solstice

- HIFIJR simulation number 408

File: COBEHI.BIN;408 COBE MISSION MODE The time is 14:03:52.26 12-DEC-1989



RUN 408

GYRO B OFF: ALL STRIPPING OFF: SUN DEC=+23.4 DEG

ACS-32

PITCH BACK MANEUVERS

- HIFIJR Simulation run 440 -- 30° pitch back maneuver under the current configuration
- Predicted performance at 30° pitch back

  Sun elevation angle

  Range: 4.27° to 5.39°

  Average: 4.67°

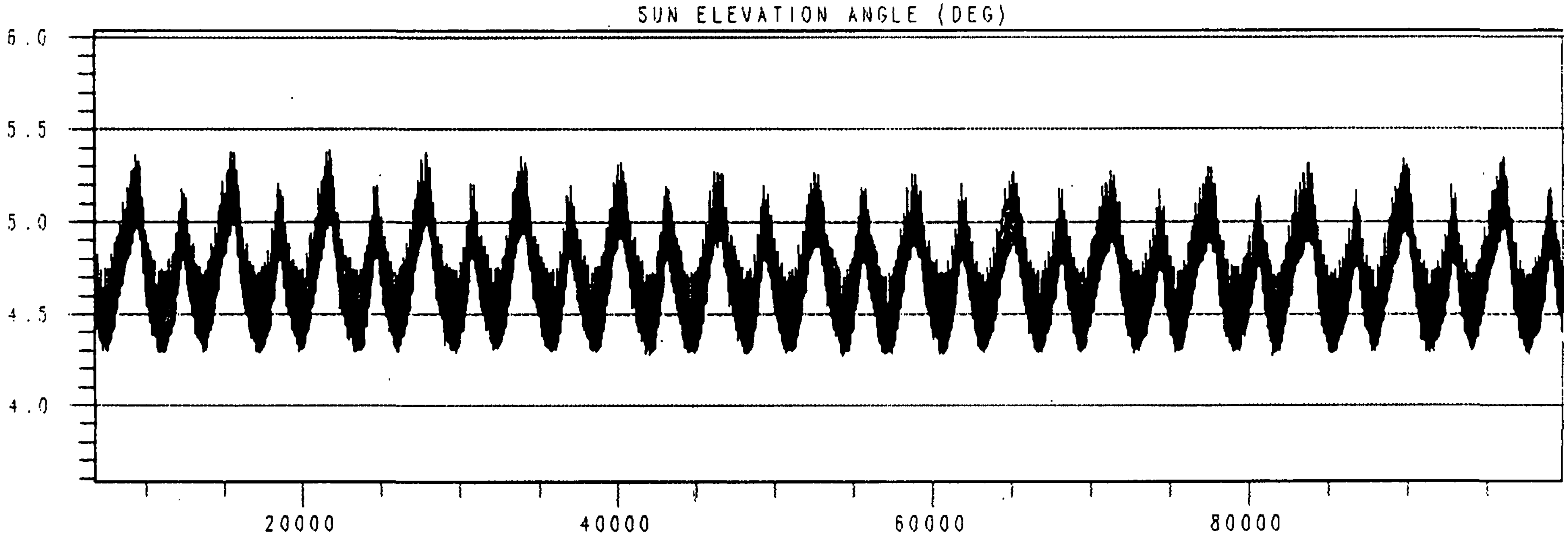
  Pitch Angle

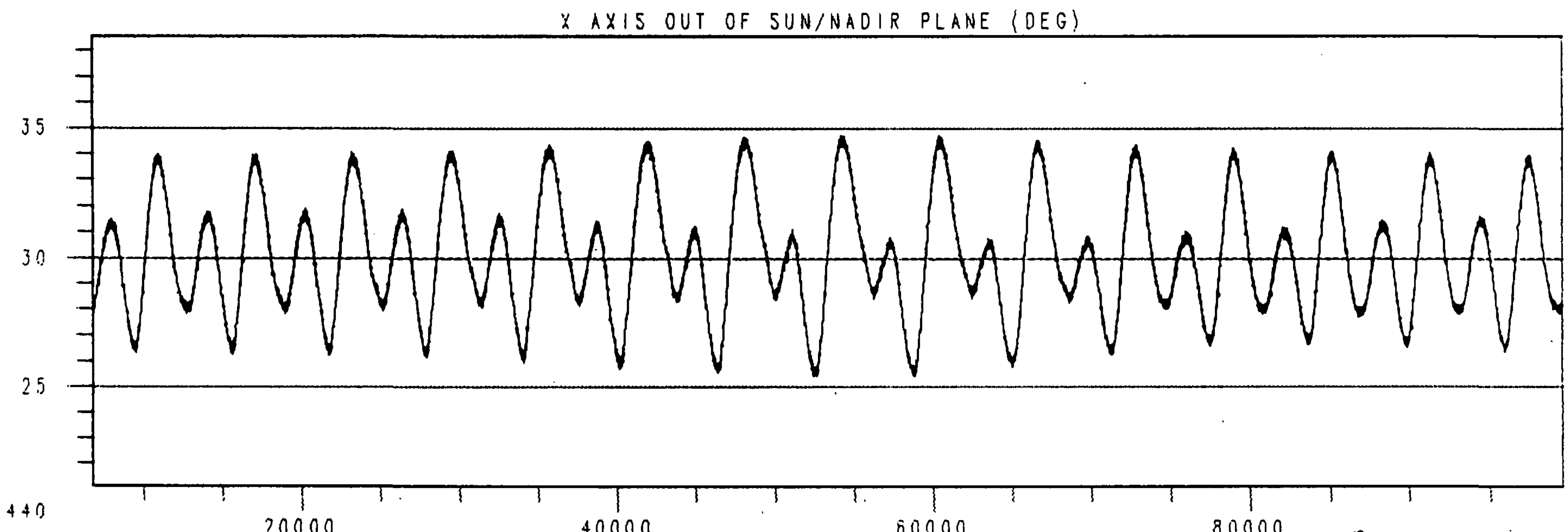
  Range: 25.34° to 34.70°

  Average: 30.00°

- No pitch back maneuvers are allowed during the two month eclipse season
- Pitch back maneuver is performed through the use of the three orbit rate stripping commands

File: COBEHI.BIN;440 COBE MISSION MODE The time is 10:53:42.35 29-JAN-1990





RUN 440

GYRO B OFF; PITCH BACK 30 DEG USING ALL STRIPPING TERMS; SUN DEC=+19 DEG

ACS-34

PITCH BACK RISKS

- HIFIJR simulations studied risk of second failure during 30° pitch back
- Matrix of 21 runs
- Worst case simulations show large pitch excursions and up to 2.5° roll into the sun

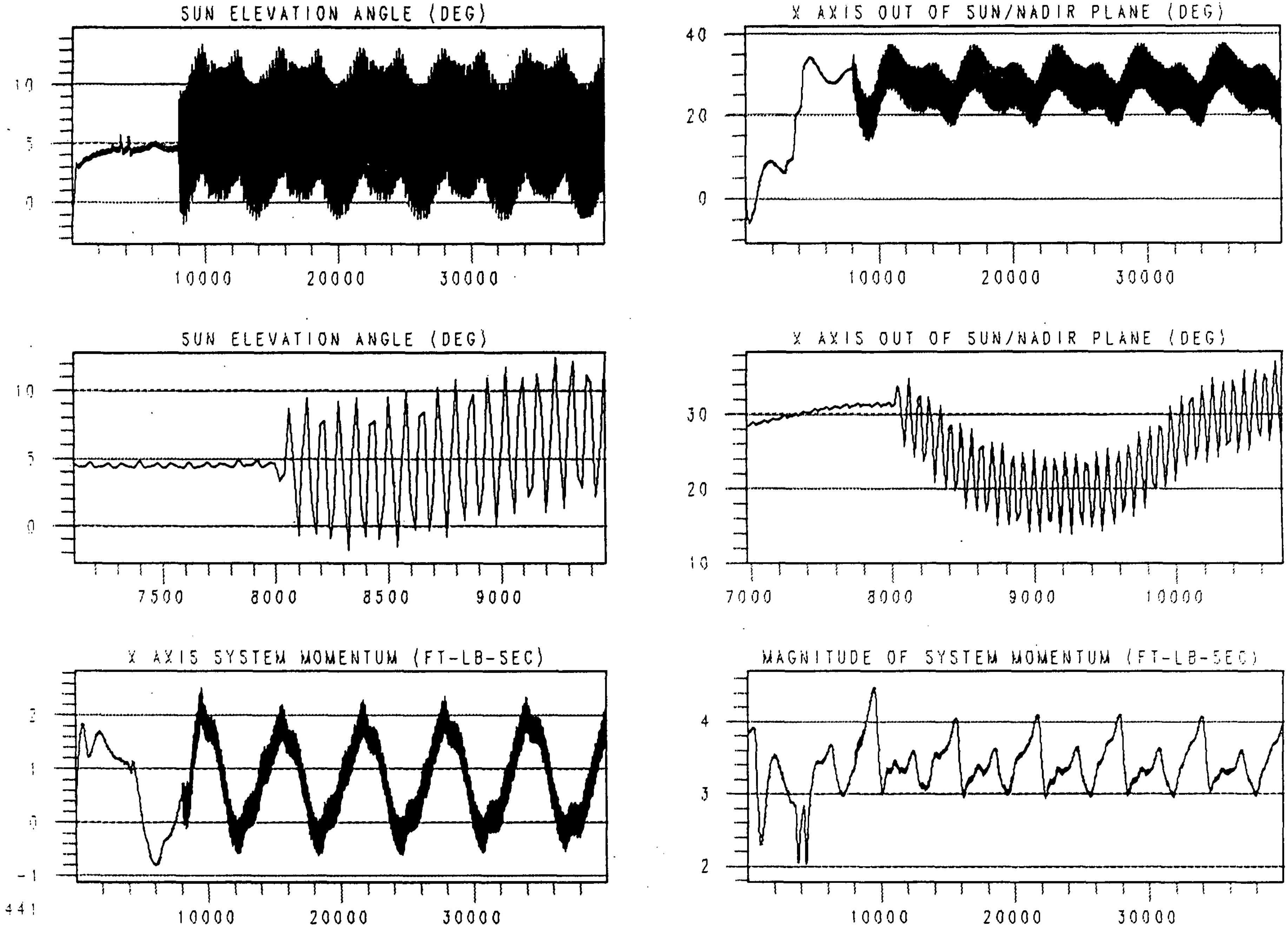
ACS-35

30° PITCH BACK
WORST CASE GYRO FAILURES

	Gyro A Failure High Positive	Gyro A Failure High Negative
Roll angle range	–1.45° to 13.20°	6.47° to 11.23°
Roll angle transient	–1.90°	–0.07°
Avg roll angle	5.89°	5.92°
Pitch angle range	17.11° to 37.48°	18.32 to 36.26°
Avg pitch angle	27.23°	27.17°

ACS-36

File: COBEHI.BIN;441 COBE MISSION MODE The time is 11:45:55.10 30-JAN-1990



RUN 441
GYRO B OFF: PITCH BACK 30 DEG: GYRO A FAIL HIGH POSITIVE AT 8000 S: SUN DEC=+19 DEG

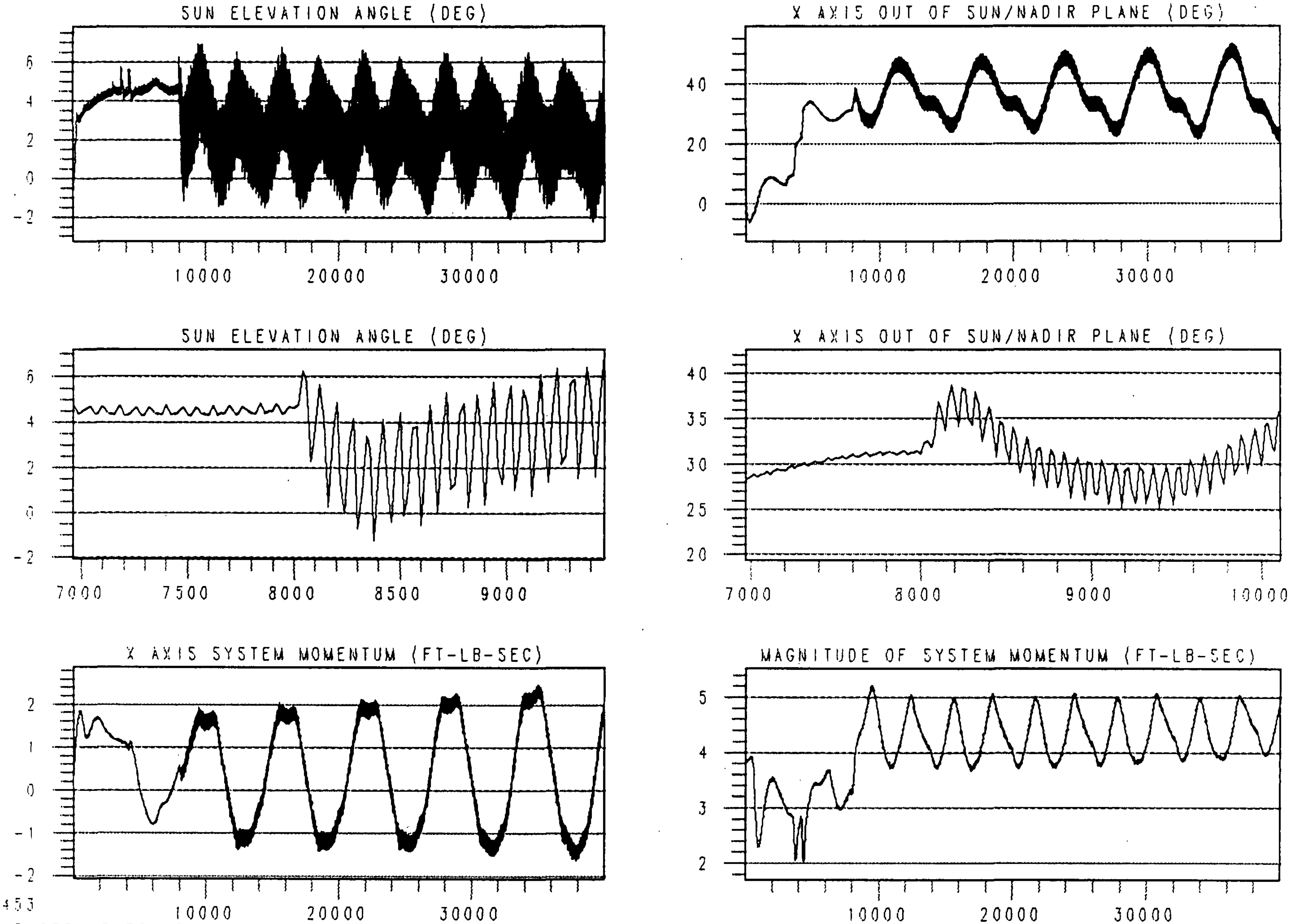
ACS-37

30° PITCH BACK
WORST CASE SCANNER FAILURES

	Scanner C Failure High Positive	Scanner C Failure Off
Roll angle range	–2.31° to 6.75°	0.86° to 6.70°
Roll angle transient	–1.25°	3.91°
Avg roll angle	2.40°	4.54°
Pitch angle range	20.44° to 53.26°	31.50° to 82.29°
Avg pitch angle	36.57°	48.04°

ACS-38

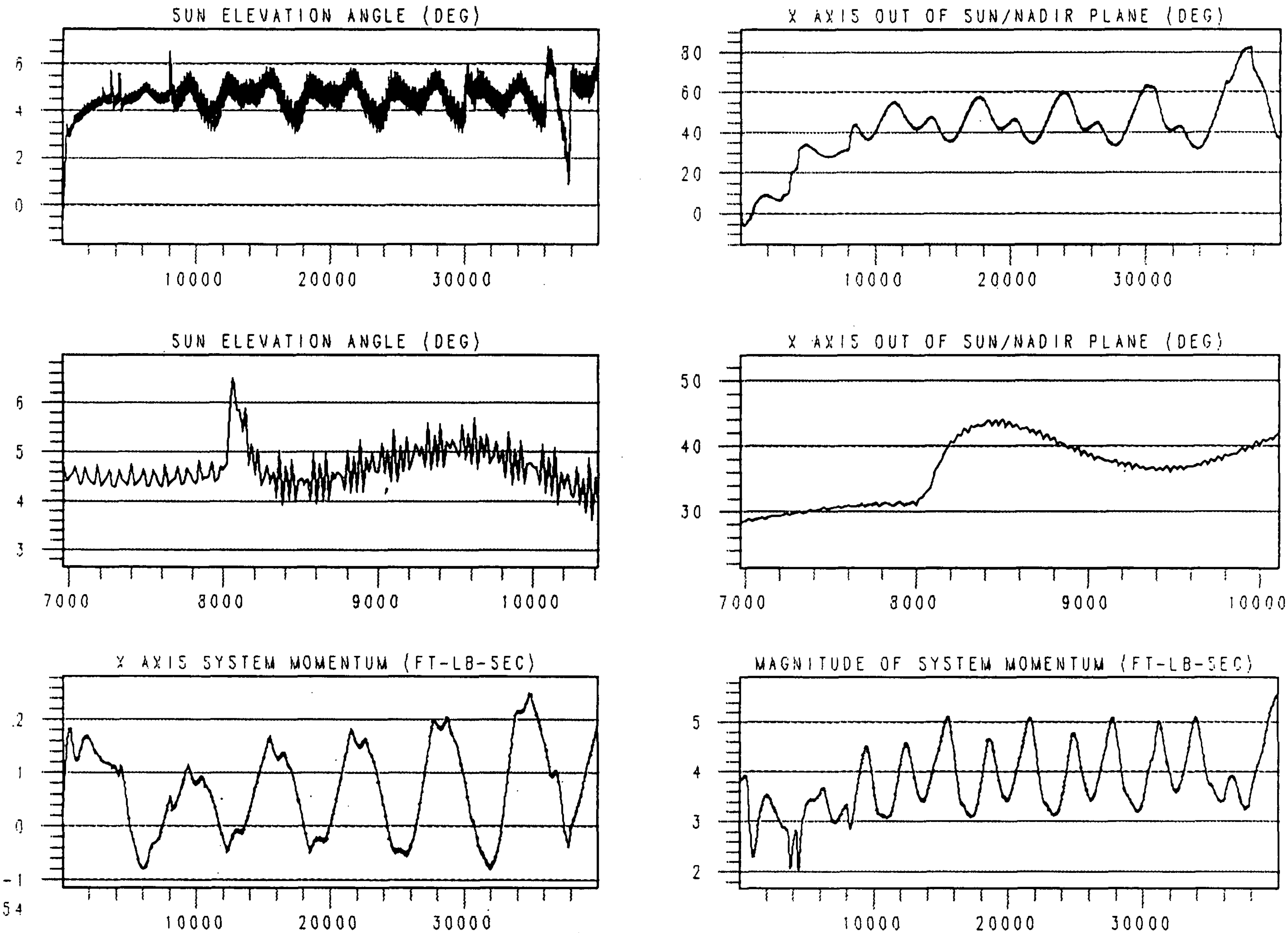
File: COBEHI.BIN:453 COBE MISSION MODE The time is 09:57:27.43 1-FEB-1990



RUN 453
GYRO B OFF; PITCH BACK 30 DEG; SCANNER C FAIL HIGH POSITIVE AT 8000 S; SUN DEC=+19 DEG

ACS-39

File: COBEHI.BIN;454 COBE MISSION MODE The time is 10:06:14.24 1-FEB-1990



RUN 454

GYRO B OFF; PITCH BACK 30 DEG; SCANNER C FAIL OFF AT 8000 S; SUN DEC=+19 DEG

ACS-40

ACS-41

December 19, 1989

TO: 700/Chief Engineer

FROM: 303/Flight Assurance Manager, COBE

SUBJECT: COBE Gyro Review Commitee Action Items

In regard to Action Item #4, Code 311 reviewed the gyro assembly parts list for any radiation sensitive device. Their conclusion is that there could have been no degradation due to total dose radiation. There simply was not enough time in orbit. And while COBE did not specifically test for Single Event Phenomenon, i.e. latchup, no parts that were particularly sensitive to this failure mode were found.

In addition to the formal action items, Code 303 also took action to test residual RMA transistors found in the COBE stock. These transistors were of the manufacturer and type of the one that failed at Northrup. Parts were subjected to xray, PIND, and wire bond pull tests. While 3 of 10 parts failed PIND testing, only organic fibers were found inside. Xrays were nominal and the wire bonds passed with flying colors.

Northrup has identified some 40 EEE parts that could have caused our RMA failure. Right now, they're working on getting the screening data for those parts. I'm still interested in these data. Although our 3868 transistors came out with a clean bill of health, I was surprised by the PIND failures. Did the failures manage to slip through the Northrup system, or was this a decision based on concrete test results?

Abigail Harper

Abigail Harper

cc: R. Baumann/300
W. Kneval/303
M. Femiano/712.3

ACS-472

In Reply Refer To:
FG:7-6634(6042)JFF:sb

NORTHROP

Precision Products Division
Electronics Systems Group
Northrop Corporation
100 Morse Street
Norwood, Massachusetts 02062
Telephone 617 762-5300

December 22, 1989

NASA
Goddard Space Flight Center
Greenbelt, MD 20777

Attention: M. Femiano, COBE Project Office

Gentlemen:

Reference: COBE RMA Failure Analysis Support

Subject: Report of Findings to Date

Enclosed is one copy each of the following data:

1. List of components whose failure could overload the +22 VDC regulator.

2. Test reports for the screening of
   A. JANTXV2N3868S transistor
   B. JANTXV2N2222A transistor
   C. JANTXV1N4148-1 diode
   D. JANTXV1N5806 diode

3. Part Analysis Report for a JTX2N3868 transistor which failed during assembly testing of another product. This item was from a different lot date code. Note that this was JANTX vs JANTXV used in COBE.

The specific cause of the RMA failure could not be determined given the timeframe or data available.

Very truly yours,

James F. FitzGerald
Senior Systems Contracts Administrator

Enclosure: (1), (2), (3) as noted - 1 each

COMPONENTS WHOSE FAILURE CAN OVERLOAD +22VDC REGULATOR

(NOTE: RESISTOR FAILURES ARE NOT INCLUDED)

ACS-43

REF DES	TYPE	LOCATION
C5, C6	CAPICITOR, TANT	~~RBE~~/OSC/WHL SPLY BOARD
Q15, Q16	TRANSISTOR, 2N2222A	"
Y1	CRYSTAL	"
U2	μCKT, 54L04	"
U3	μCKT, 54L73	"
U4	μCKT, 54L93	"
U5	μCKT, 54L74	"
CR6-CR13	DIODE, 1N4148	"
Q9-Q12, Q17-Q20	TRANSISTOR, 2N2222A	"
Q13, Q14, Q21, Q22	TRANSISTOR, 2N3868	"
C8, C12	CAPACITOR, CER.	"
T3, T4	TRANSFORMER	TRANSFORMER BOARD
CR5-CR8	DIODE, 1N5806	"
C3, C4	CAPACITOR, TANT.	"
MP1	GYRO	CHASSIS
C5, C6, C11, C12	CAPACITOR, CER.	POWER SUPPLY BOARD
C12, C13	CAPACITOR, CER.	TELEMETRY BOARD
U4-U7	μCKT, LM108A	"
U3	μCKT, AD581	"
A2-A6	μCKT, LM108A	DRE BOARD
C18, C20	CAPACITOR, CER.	DRE BOARD

End (1)

Jan. 2, 1990

ACS-44

TO: 700/Chief Engineer Henry Price

FROM: 712/ Michael Femiano

SUBJECT: COBE Gyro Review Committee Action Items

Henry -

I have today received from Northrop the 3 items which I requested, in order to close out their investigation of the COBE gyro failure. These are :

(1) A list of the 57 electronic parts whose failure could have overloaded the 22 volt regulator. These include various transistors, capacitors,diodes and microcircuits, distributed across all 6 boards in the Rate Measuring Assembly.

(2) Test reports for the screening of 4 of these parts:
- (a) 2N3868 transistor
- (b) 2N2222A transistor
- (c) 1N4148-1 diode
- (d) 1N5806 diode

It was a 2N3868 transistor which failed in the classified Northrop program. The screening data for the COBE build shows that 40 2N3868 devices were screened by Assurance Technology Corporation for Northrop and passed successfully. This included the Particle Impact Noise Detection (PIND test) per MIL-STD-750 Method 2052. There were no failures in that test.

(3) The failure analysis report on the 2N3868 transistor failure in the classified program. The report contains no surprises, indicating the failure to be a collector-emmiter short circuit due to a contaminating gold flake. Polaroid photographs had already be made available to GSFC (you have them).

As far as their assessment of the cause for COBE's gyro failure, Northrop concludes "the specific cause of the RMA failure could not be determined given the timeframe or data available".

This concludes Northrop's failure investigation. I have attached copies of the above information.

ACS-45

Michael Femiano

Michael Femiano

cc: Henry Hoffman/712
Al Sherman /710
Abigail Harper /303
Bob Baumann /300

ACS-46

January 4, 1990

TO: 401/COBE Project Manager

FROM: 700/Chief Engineer

SUBJECT: COBE Gyro Review Committee Findings

The review committee has considered many aspects of the COBE B Gyro failure and come to the conclusion that the failure was a random part failure that shorted a regulator and caused the gyro to malfunction.

Northrop, the gyro manufacturer, has identified 57 parts that could have caused the failure. One part considered highly suspect was a switching transistor in the motor drive circuit because of a similar failure in unit test at Northrop that was identical to the in-orbit failure. All transistors of the same type in the COBE residual stock were tested and came out with a clean bill of health.

Radiation effects were discounted because of the short time in orbit. Review of the worst case analysis revealed that all parts were properly derated and conservatively utilized. No additional clues were uncovered by a total S/C data review at the time of the malfunction. EMI/EMC test data was also revealed and no anomalies were uncovered.

All the above is the basis for the conclusion that a random part failure shorted the power system.

Henry Price

Henry W. Price

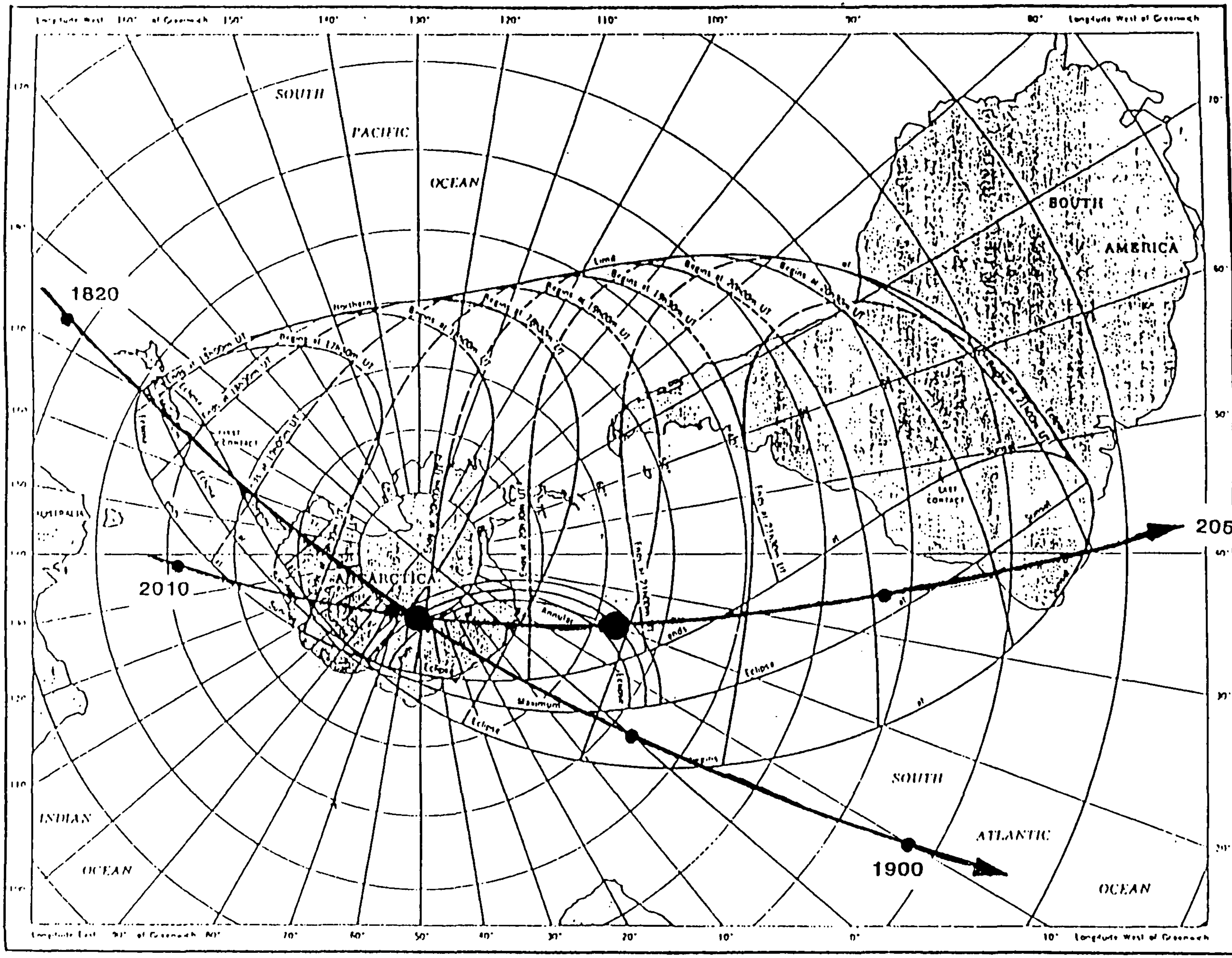
cc: A. Harper/300
R. Freeman/400
T. Huber/700
H. Price/700
J. Turtil/704
A. Sherman/710
H. Hoffman/712
M. Femiano/712
J. Wilson/401/EER
B. Martin/Swales
D. Gilman/HQ/EZD

ECLIPSE

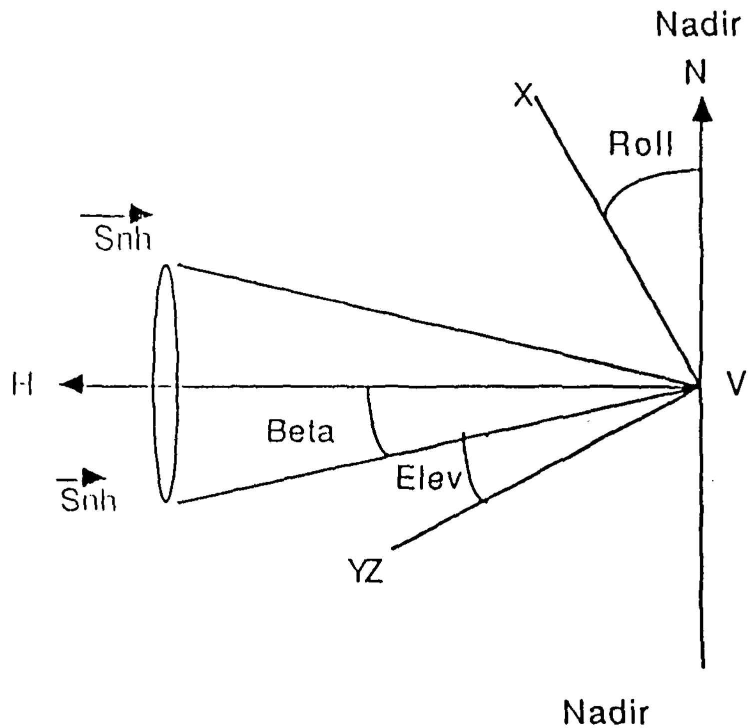
ACS-407

M. FEMIANO

ANNULAR SOLAR ECLIPSE OF 1990 JANUARY 26



ACS-48



Sun Cone about Orbit Momentum H

Half Cone Angle 18.63 - 9.0 = 9.63

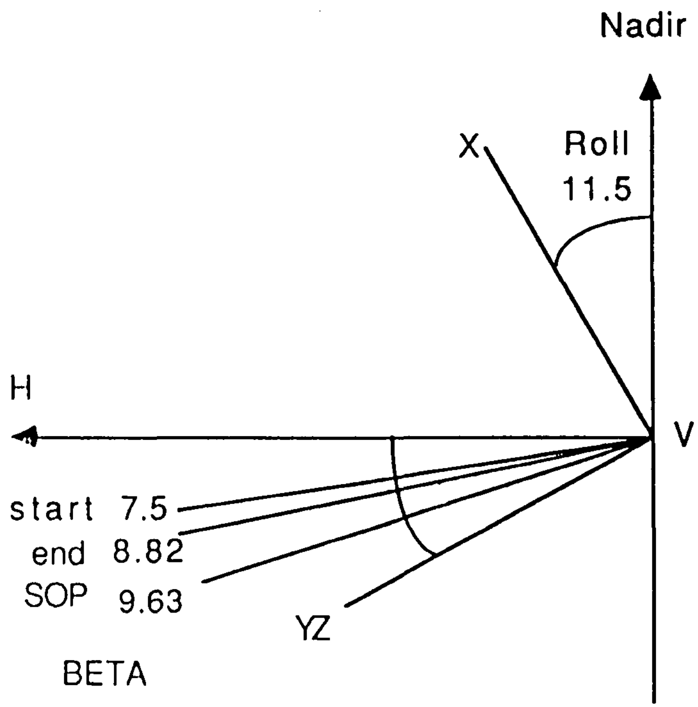
$\vec{Snh} = \bar{S}$ projection in N,H plane

Beta from H to $\vec{Snh}$ in NH plane

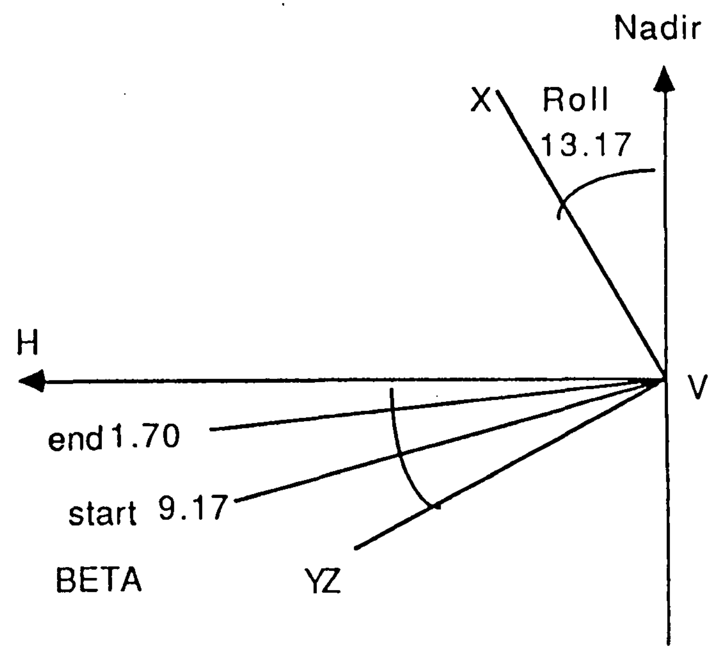
At poles, $\vec{S} = \vec{Snh}$, Beta = 9.63

Sun vector in Nadir,Velocity,Momentum Frame

1st Eclipse



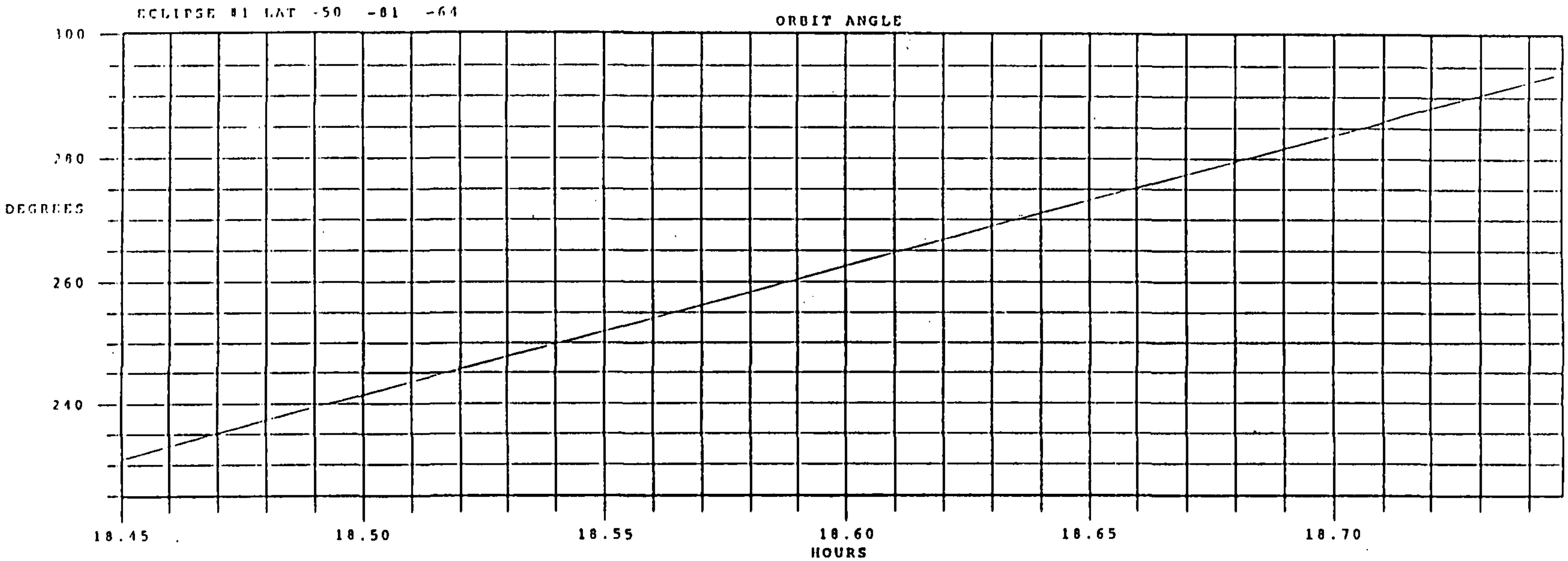
2nd Eclipse

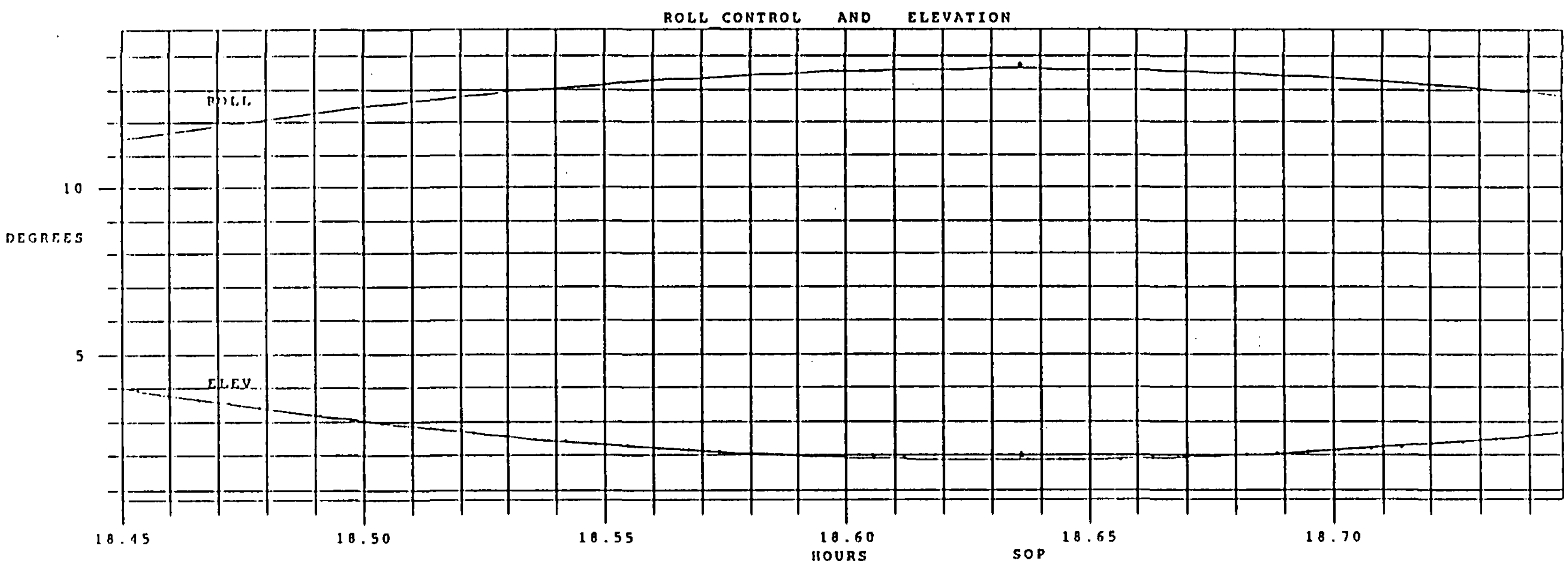


Sun Motion varies Elevation

ACS-50

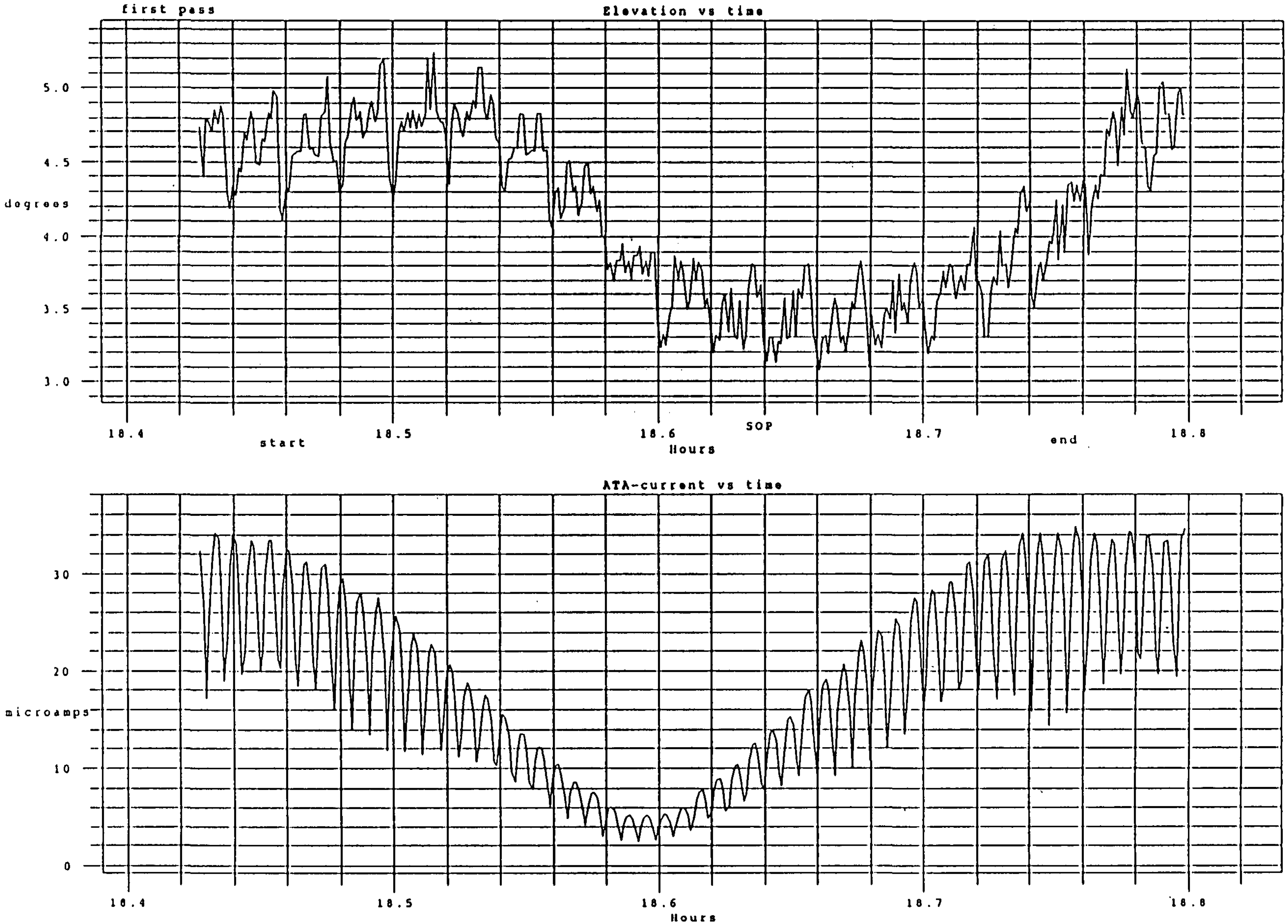
The time is 11:03:49.27 9-FEB-1990



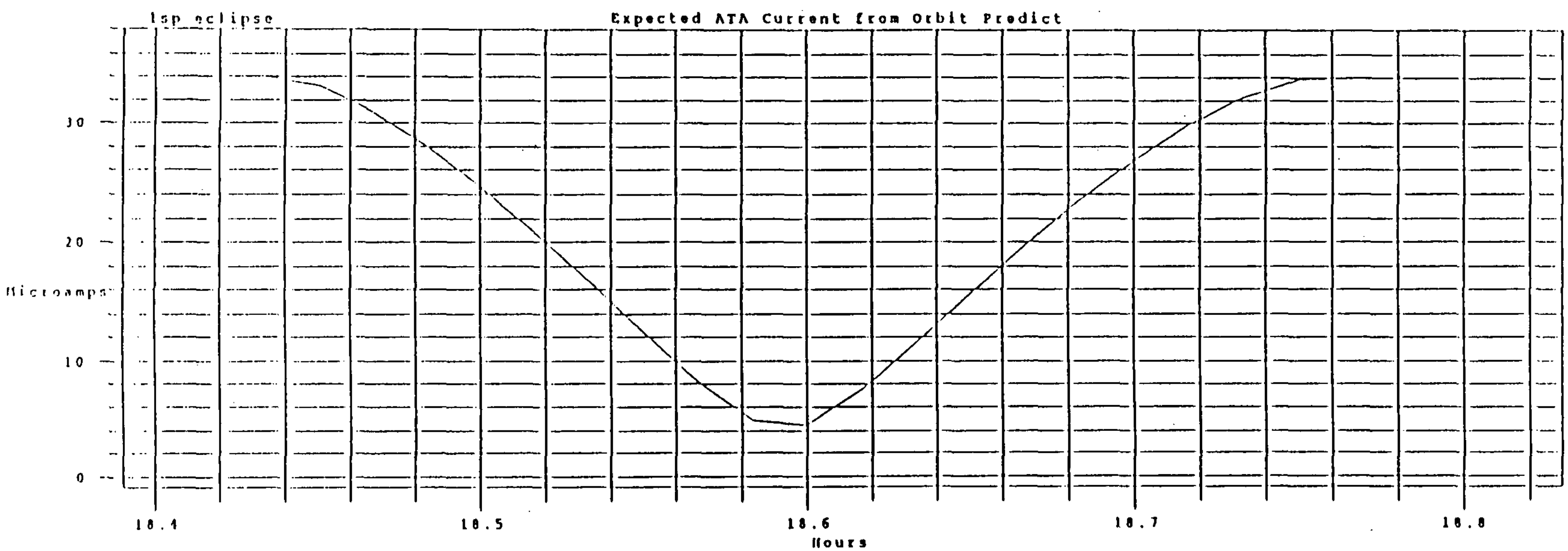


ACS-51

The time is 15:06:35.82 8-FEB-1990



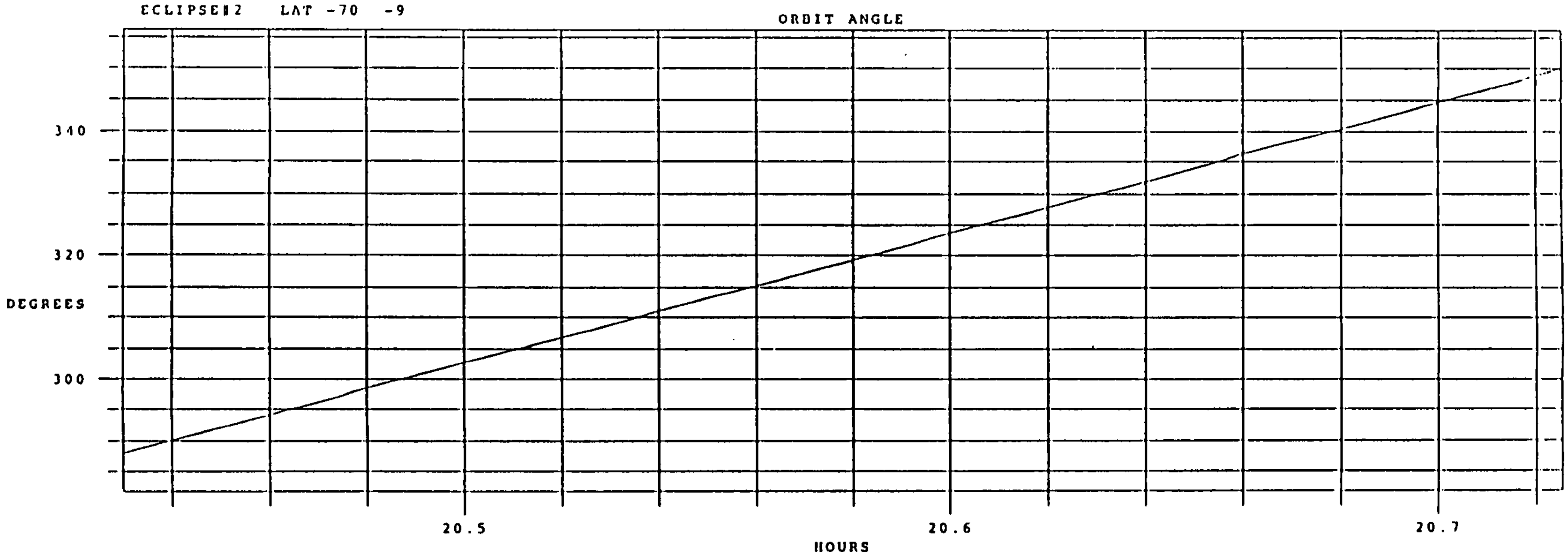
ACS-52

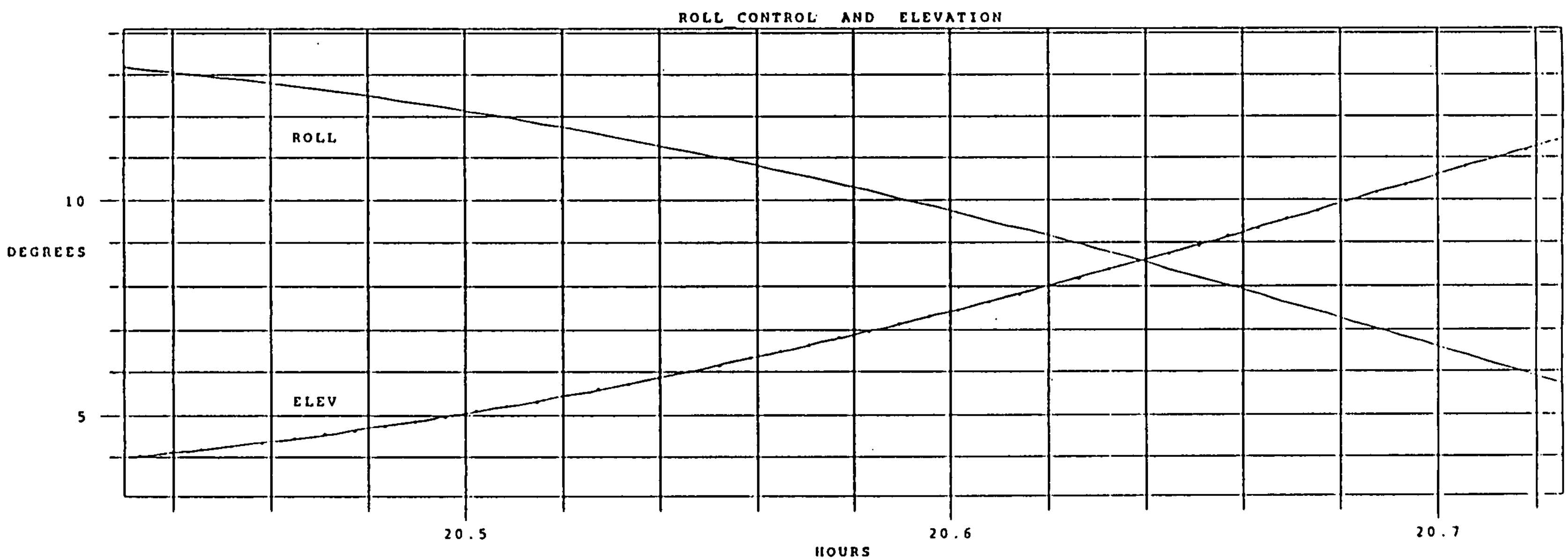


lsp eclipse
Expected ATA Current from Orbit Predict
30
20
10
0
Microamps
18.4
18.5
18.6
18.7
18.8
Hours

ACS-53

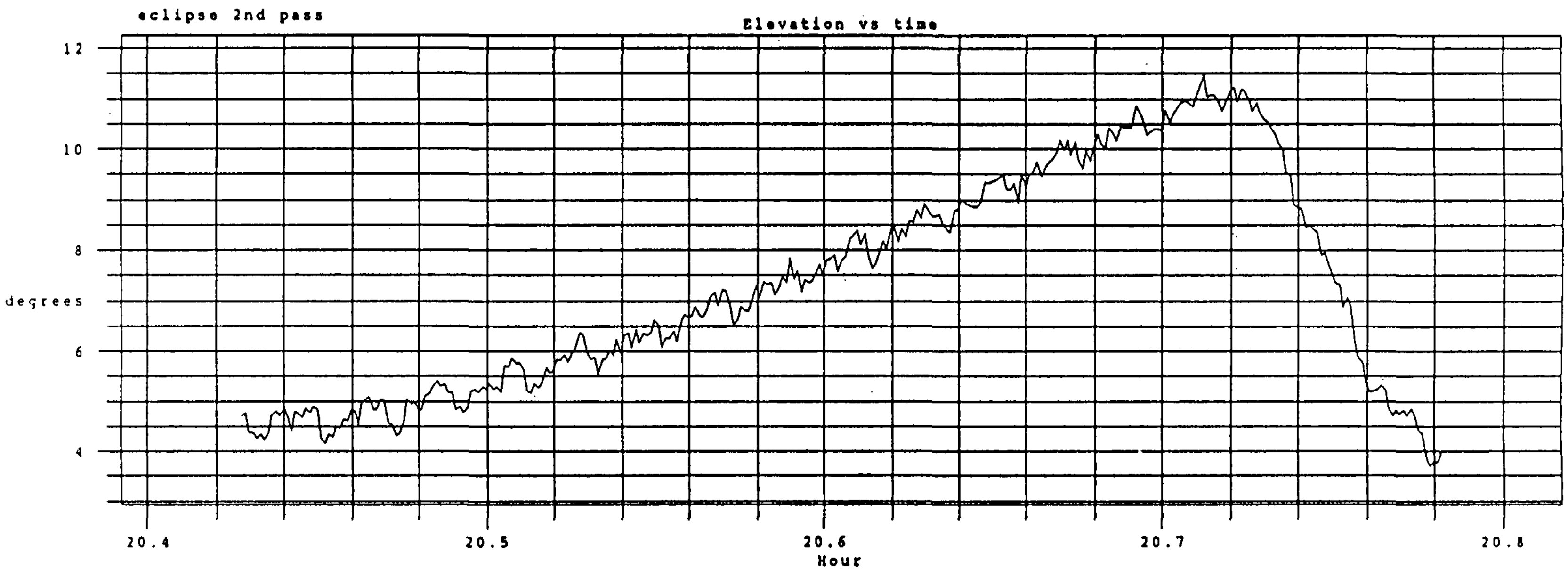
The time is 11:07:31.07 9-FEB-1990

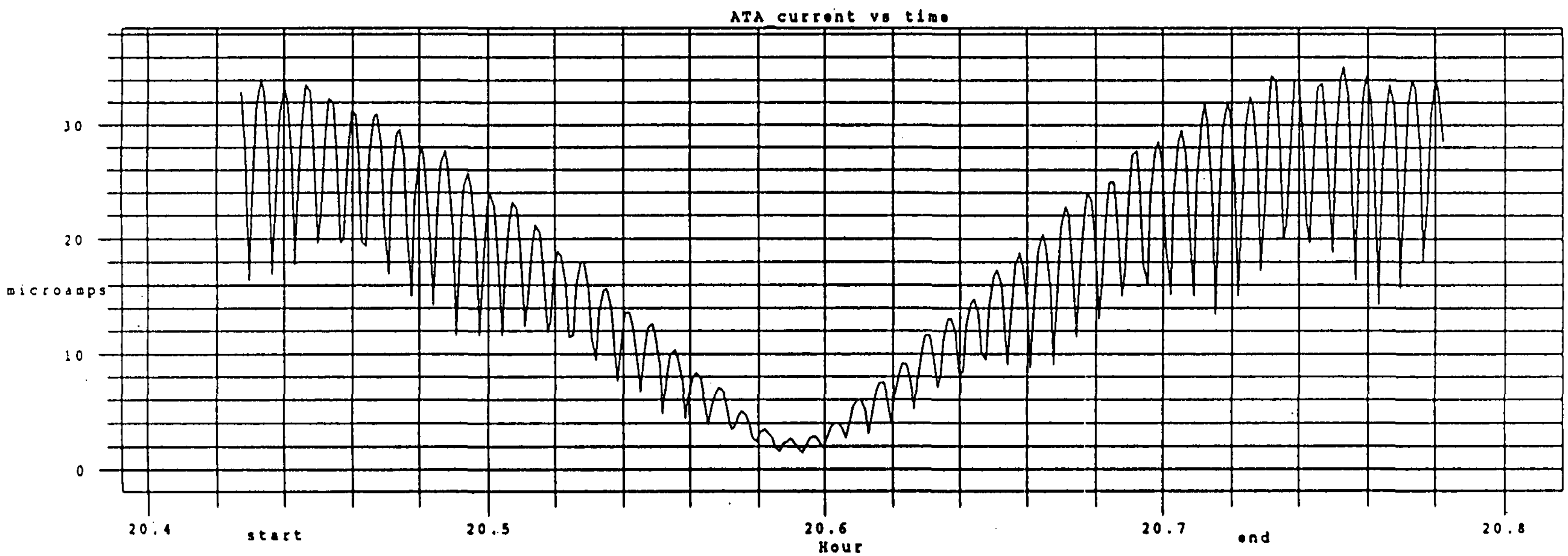




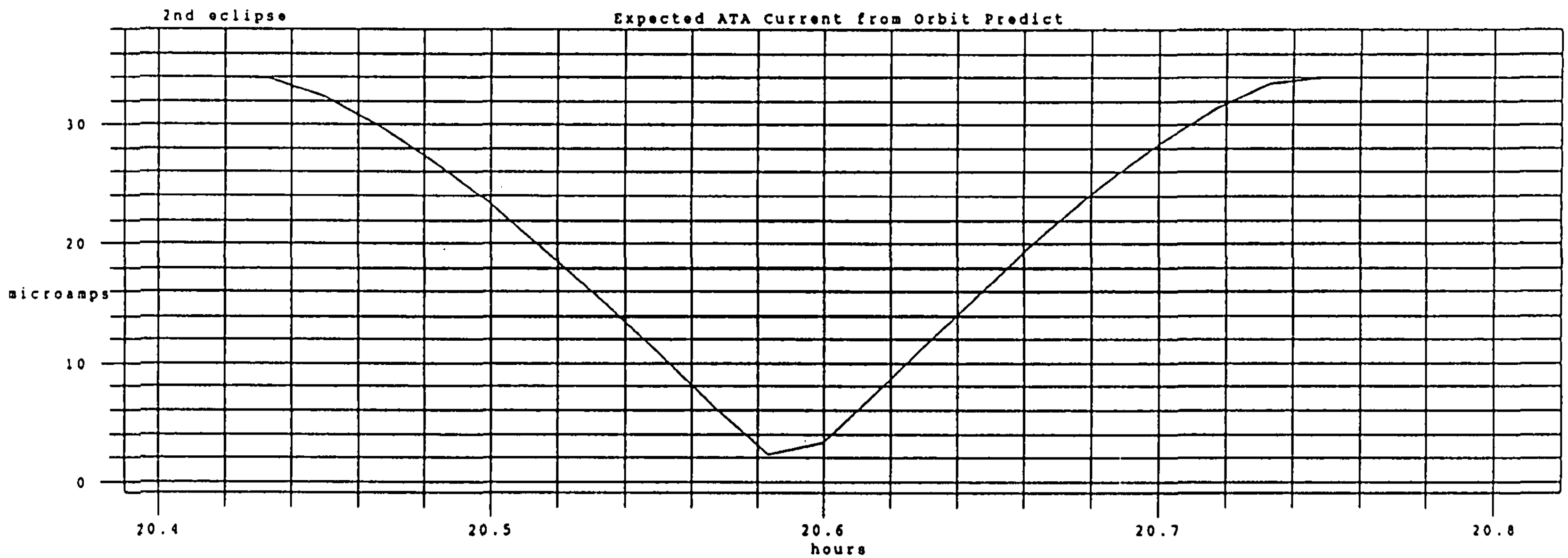
ACS-54

The time is 15:12:43.78 8-FEB-1990





ACS-55


2nd eclipse
Expected ATA Current from Orbit Predict
30
20
microamps
10
0
20.4
20.5
20.6
20.7
20.8
hours

ACS-56

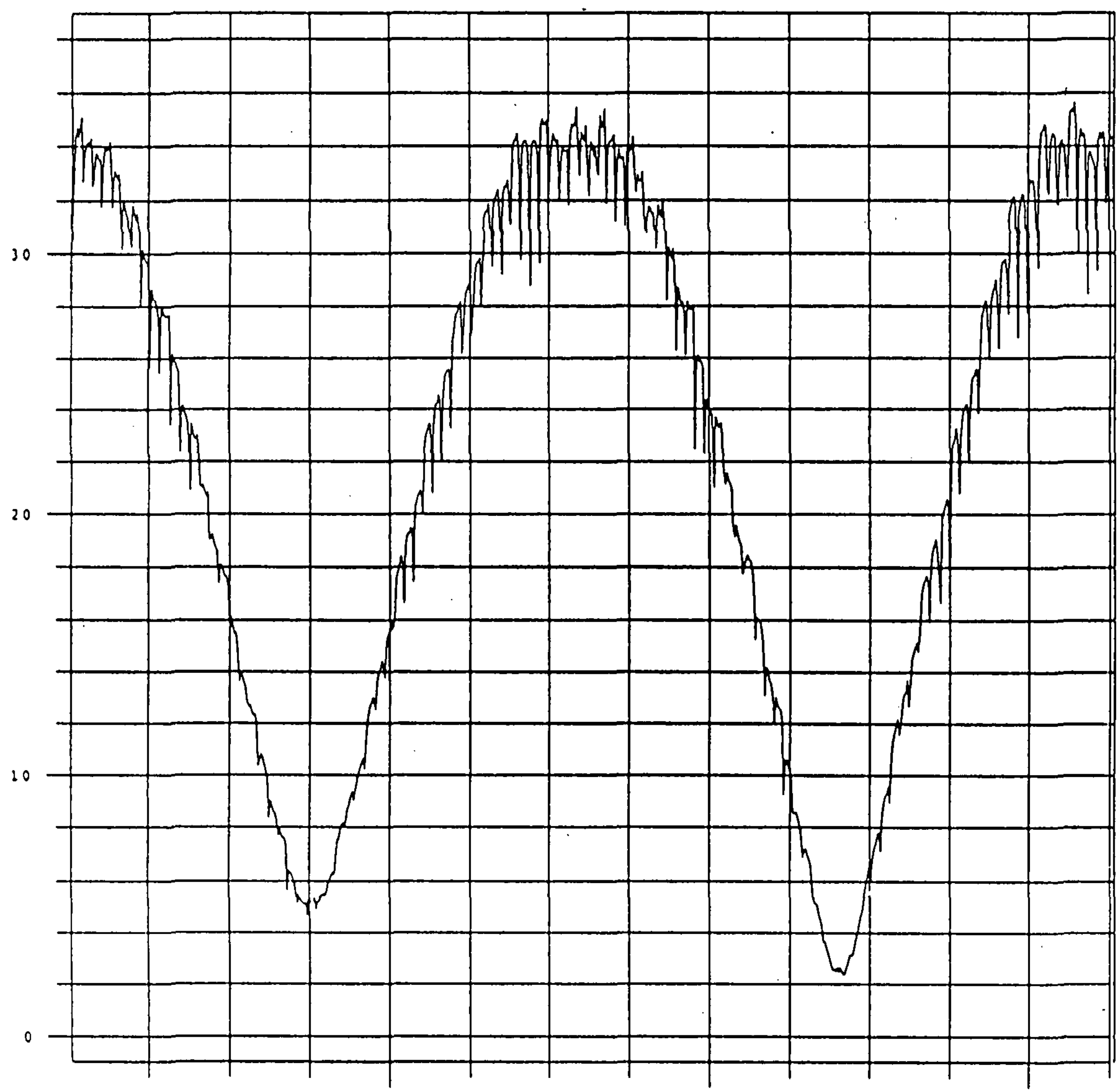
SUN SENSOR PERFORMANCE

T. FLATLEY

ACS-57

SUN SENSOR PERFORMANCE

* ANNULAR SOLAR ECLIPSE OCCURRED ON JANUARY 26, 1990 IN ANTARCTIC REGION
* COBE FLEW THROUGH THE MOON'S SHADOW TWICE
* SUN 87 PERCENT COVERED AT DARKEST POINT OF FIRST PASS (1827-1845 GMT)
* SUN 93 PERCENT COVERED AT DARKEST POINT OF SECOND PASS (2026-2044 GMT)
* CONCERNS WERE RAISED ABOUT TWO-AXIS DIGITAL SUN SENSOR PERFORMANCE WITH GREATLY DIMINISHED SUN AND EXCEPTIONALLY BRIGHT EARTH BELOW
* PRECAUTIONS WERE TAKEN TO AVOID POSSIBLY SERIOUS CONSEQUENCES OF ERRONEOUS SUN SENSOR OUTPUT
* TELEMETRY DATA INDICATES THAT PRECAUTIONS WERE UNNECESSARY
* SYSTEM PROVIDED NOMINAL INDICATION OF SUN VECTOR POSITION THROUGHOUT BOTH ECLIPSE PASSAGES
* DELAYED HEAD TRANSITIONS OCCURRED (DUE TO BY-DESIGN HYSTERESIS IN ELECTRONICS) BUT DATA QUALITY WAS UNEFFECTED

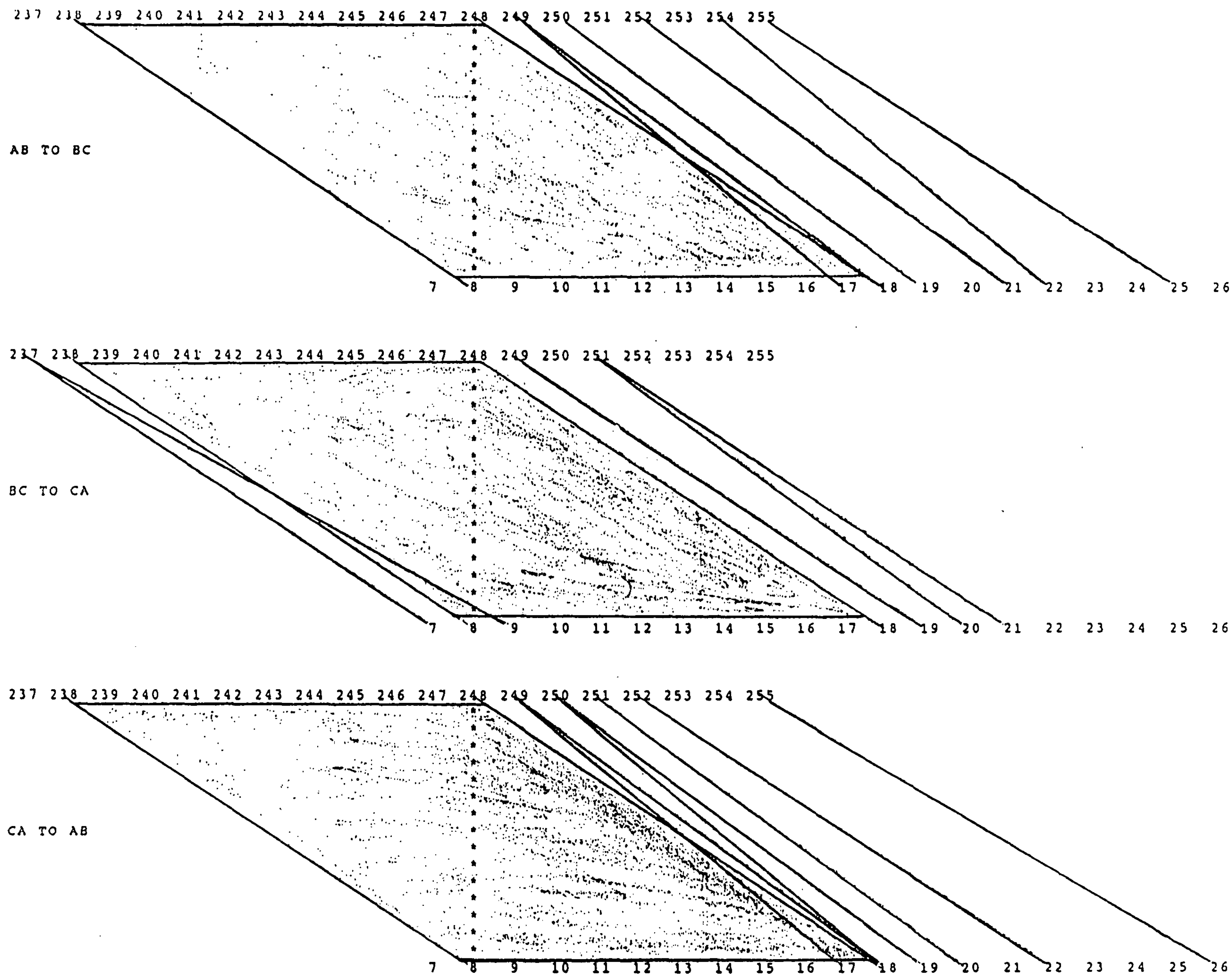


COBE ECLIPSES

$$\frac{ATA\ \mu A}{\cos(\text{SUN AZIMUTH})}$$

$\frac{34-5}{34} = 0.85$ $\frac{34-2.5}{34} = 0.93$

ACS-59



ACS-6

SUN SENSOR HEAD TRANSITIONS (NB COUNTS)

ACS-61

COBE ATTITUDE CONTROL ELECTRONICS

by Walter Squillari

The Attitude Control Electronics (ACE) is the heart of the COBE Attitude Control System. This unit contains all of the electronics to properly maintain the spacecraft's attitude and in addition processes and formats certain attitude sensor data for telemetry.

The ACE is an analog/digital system that does not employ microprocessors. As originally designed for a COBE shuttle launch, the ACE consisted of 26 printed circuit boards. After it was decided to launch the COBE spacecraft on a Delta vehicle, several circuits within the ACE required redesign. Along with the redesign, seven Orbit Transfer printed circuit boards were eliminated. As launched, the ACE consisted of 19 printed circuit boards and approximately 1500 interconnecting wires used to carry signals between the boards and interface to the spacecraft sensors, actuators and Command & Data Handling (C&DH) system. The ACE was approximately 24in x 13in x 11in and weighed 53 pounds. Fully powered, the unit consumed a low 8 watts of power.

The ACE is a fully redundant unit. The control system was designed as a triaxial system such that no single point failure existed and certain multiple failures could be sustained without jeopardizing the COBE mission.

ACS-62

Figure 1 shows a block diagram of the electronics within the ACE. Power distribution within the unit is accomplished via relays located on two of the printed circuit boards. These boards interface with two dc-dc converters and the C&DH system. Upon command from the spacecraft's C&DH, they apply or remove power to the remaining ACE boards.

The Common Electronics (CE) cards are fully redundant and process much of the sensor data prior to interfacing with the A,B,C mission electronics cards. These cards also format the Digital Sun Sensor and Gyro data for telemetry. These data are used by the Flight Dynamics Facility in determining the spacecraft's attitude.

The following is a brief explanation of the functions performed by the CE in conjunction with the on-board sensor inputs.

a. <u>Digital Sun Sensors:</u> The CE receives gray coded elevation, azimuth, head ID and sun presence information from these sensors at 10 millisecond intervals. This information is used to calculate the elevation error, spacecraft azimuth angle and when to energize the eclipse mode.

b. <u>Spin Gyros (Xa,Xb,Xc):</u> Upon command, one of the three gyros is selected for use by the CE. The incremental angle pulses from the gyro are accumulated by a counter and telemetered. The selected X gyro incremental angle pulses

ACS-63

are used by the electronics to develop the spacecraft's azimuth angle information required by the mission electronics in resolving the error signals during eclipse operations. This gyro information, along with a spin rate command, is also used by the Spin Rate Control card to develop the spin error signal required by the Magnetic Management Assy. The addition of the Spin Rate Control card was one of the modifications made to the ACE following the Challenger disaster.

c. Transverse Gyros (A,B,C): Incremental angle pulses from these gyros are used by the CE rate processing electronics to derive the spacecraft's A,B,C body rates. The electronics strips out orbit rate, removes gyro drift and the resulting body rate signal is applied to the mission electronics cards to control the reaction wheels.

d. Coarse Sun Sensors: This sensor information is processed for telemetry and used to determine the spacecraft's attitude if the sun is out of the field-of-view of the Digital Sun Sensors. The -X coarse sun sensors are further processed and a signal is sent to the DIRBE instrument to close its shutter when the sun angle to the spacecraft's -X axis exceeds a predetermined value.

e. Command & Data Handling (C&DH): The CE receives and stores numerous commands for the mode selections and bias adjustments. The unit also interfaces with the data system and telemeters the large amounts of formatted sensor data as well as analog housekeeping data.

ACS-64

Within the ACE are redundant 48 channel analog multiplexers. These cards were added in order to decrease the number of analog telemetry channels required by the ACE. The multiplexers telemeter the analog housekeeping data and control error signals.

The A,B,C electronics (Mission) cards further process the Common Electronics data. Figure 2 shows a more detailed block diagram of the electronics that resides within these three cards. The Mission cards perform the control laws required to maintain proper spacecraft attitude through all spacecraft operating modes. Their outputs are applied to the Reaction Wheel and Momentum Management Assemblies. These Mission cards receive their signals from either of the two Common Electronics. The Common Electronics selection occurs by either ground command or automatically from an autonomy circuit housed within Common Electronics #2.

The Mission cards incorporate many sensor and actuator cross-strapping modes that may be initiated through ground commands. This allows the spacecraft's control system to withstand numerous sensor and actuator failures without seriously affecting the mission objectives. In case of an Earth Sensor failure, the failed sensor may be switched out of the control loop and the negative sum of the other two earth sensors can be used for pitch control. If a Reaction

ACS-65

Wheel or mission electronics fails, the control law may be modified to compensate for this failure. In case of a gyro failure (as it happened four days after launch), the signal from the remaining "good" axes can be cross-strapped to drive the wheel in the "failed" axis without any loss of performance.

Within Common Electronics #2 resides an autonomy circuit that switches ACE control to Common Electronics #2 (if CE#1 is in use). This circuit monitors certain Common Electronics #1 parameters and if their predetermined values are exceeded and the autonomy is enabled, the Mission card inputs are switched to Common Electronics #2. These parameters are as follows:

a. Elevation error exceeds +/-3.75 degrees.

b. Digital Sun Sensor is turned OFF or the sensor ceases to transmit data for at least 70 milliseconds.

c. The CE#1 internal oscillator ceases to function.

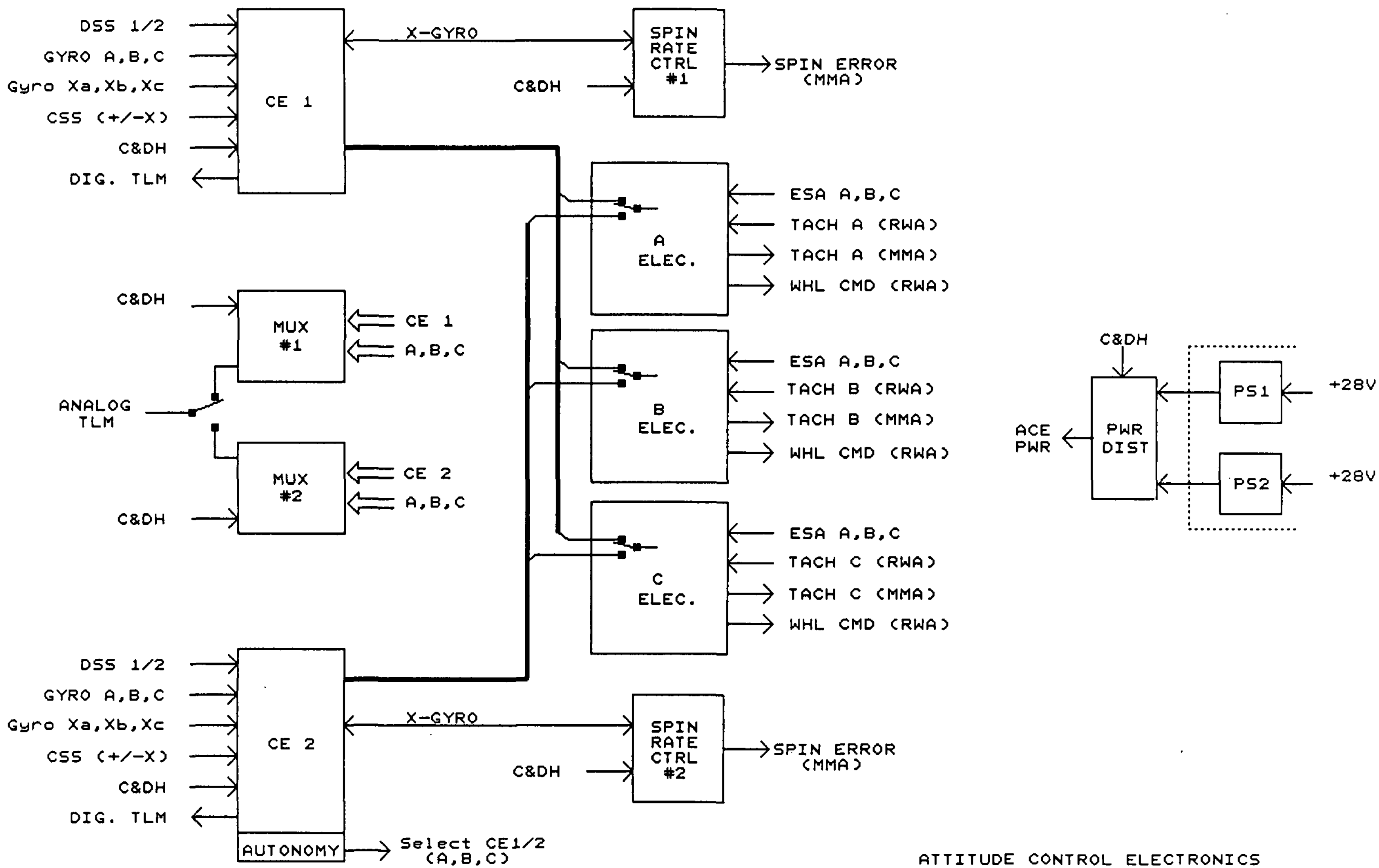
Only routine fabrication problems were encountered during the development and assembly of the ACE. No design problems were discovered during board testing, closed loop testing and finally, environmental testing.

The Attitude Control Electronics was successfully integrated without any problems to the spacecraft's power, command & data handling system and to the control system sensors and

ACS-98

actuators. The unit completed all required spacecraft tests and performed well within the system specifications.

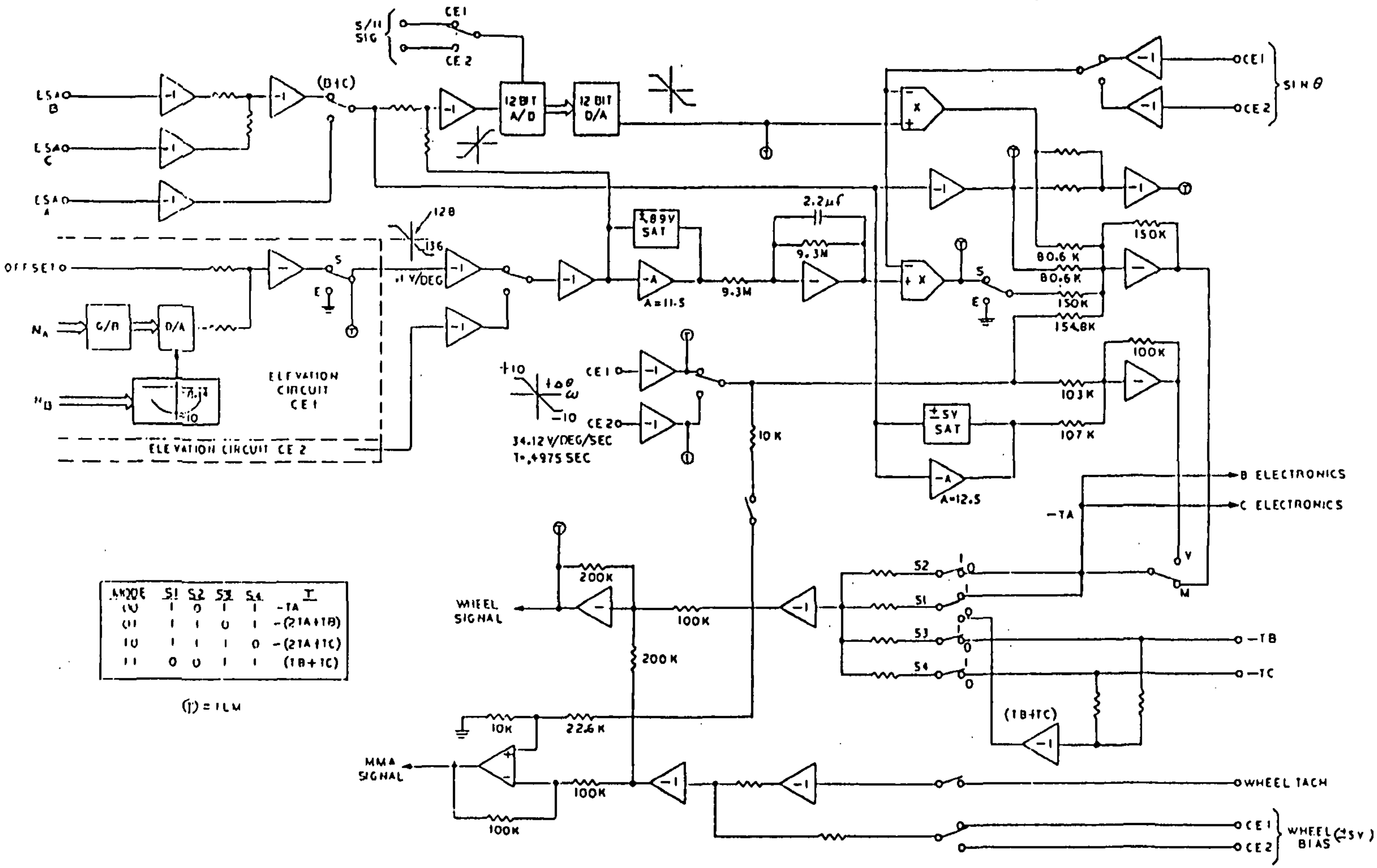
Since launch, the unit has performed flawlessly. The total system (prime and redundant) has been checked out.
One of the cross-strapping and gyro bias commands was used four days after launch when the Attitude Control System experienced a gyro failure.



ATTITUDE CONTROL ELECTRONICS
BLOCK DIAGRAM

Figure 1

ACS-67



MODE	S1	S2	S3	S4	T
00	1	0	1	1	-TA
01	1	1	0	1	-(2TA+TB)
10	1	1	1	0	-(2TA+TC)
11	0	0	1	1	(TB+TC)

COBE

MISSION ELECTRONICS

FIGURE 2

ACS-68

ACS-69

COBE MOMENTUM WHEEL ELECTRONICS ASSEMBLY

by Walter Squillari

The Momentum Wheel Electronics Assemblies (MWEA) are fully redundant drivers that power the two COBE Momentum wheels. These wheels are used for the control of the spacecraft's spin rate.

Figure 1 shows a block diagram of the MWEA. The MWEA consists of two sub-assemblies, the Power Control Electronics (PCE) and Speed Control Electronics (SCE). The PCE is the power stage that converts the low level logic of the wheel speed error and phase signals into the drive signals for the wheels. The PCE, an existing wheel driver design from a military contract, was purchased from TRW. Within the PCE are also a dc-dc converter and the electronics to develop the motor current signal used in controlling the motor voltage during run-up.

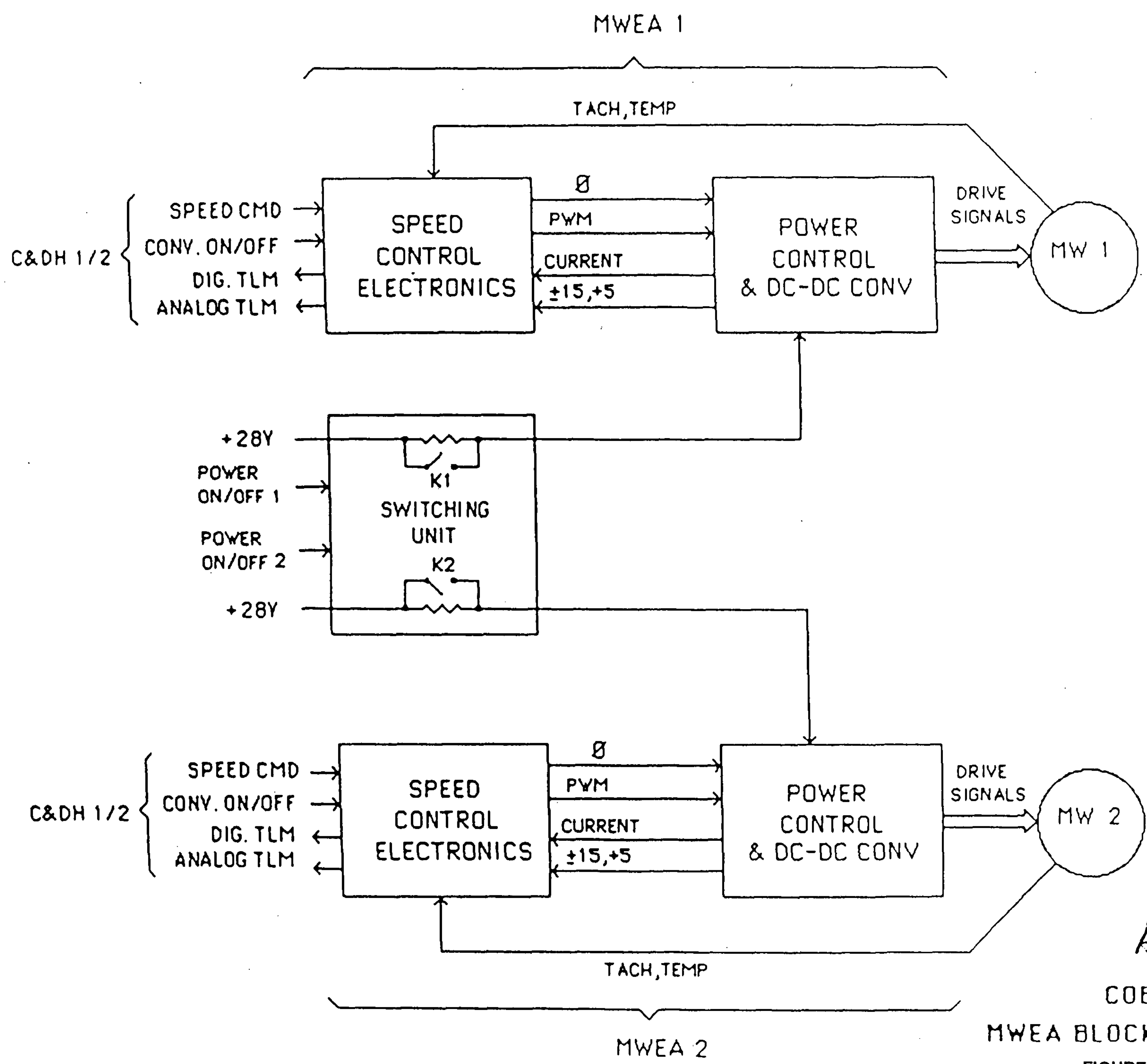
The second sub-assembly, the SCE, was designed and fabricated by Code 712. Figure 2 shows a block diagram of the SCE. The SCE interfaces with the magnetic wheel tach and provides the PCE with a pulse width modulated signal that represents wheel speed error. The SCE contains a linear motor frequency drive of 125hz to 250hz. This variable frequency was provided in order to maintain a 20% motor slip and reduce the wheel power by approximately 5 watts. A current limiter is

ACS-70

incorporated to limit surge currents to 7 amps and motor run-up current to 3 amps. If motor current were not limited, the motor could overheat. Through speed commands from the Command & Data Handling System, the wheels may be controlled from 440 rpm to 4500 rpm with a controlling accuracy of +/-0.02%.

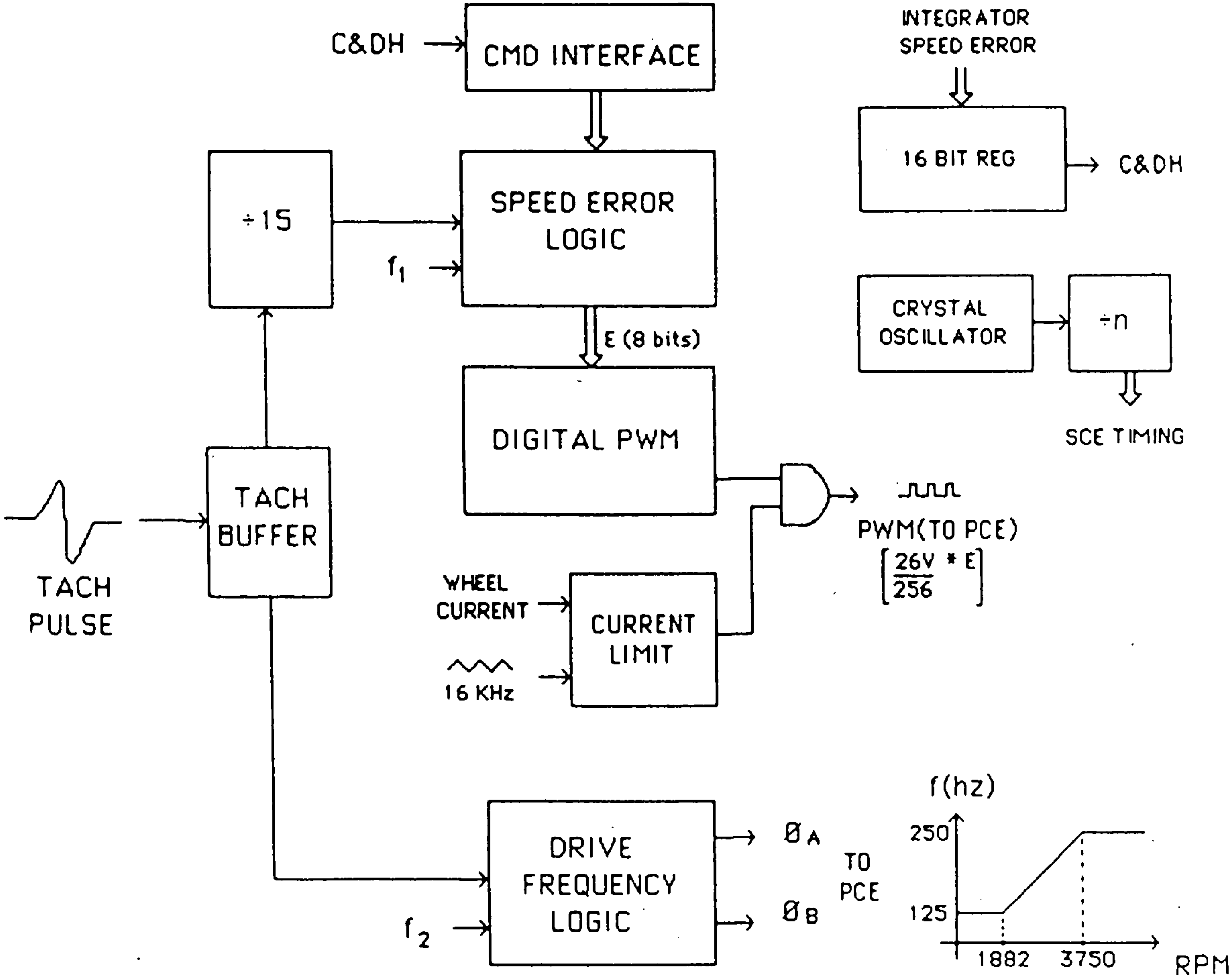
The only problem encountered prior to delivery of the units to the spacecraft was an integrated circuit part failure that occurred during the second hot soak of the MWEA#2 thermal-vacuum tests. This part was replaced and both MWEA's successfully completed all environmental tests.

The two MWEA's have operated flawlessly since launch and the wheel speeds have been trimmed to 1600.2 rpm, giving the spacecraft a spin rate of approximately 0.815 rpm. As measured from telemetry, speed control is better than +/-0.02% and the wheels are consuming a low 2.5 watts each.



ACS-71

COBE
MWEA BLOCK DIAGRAM
FIGURE 1



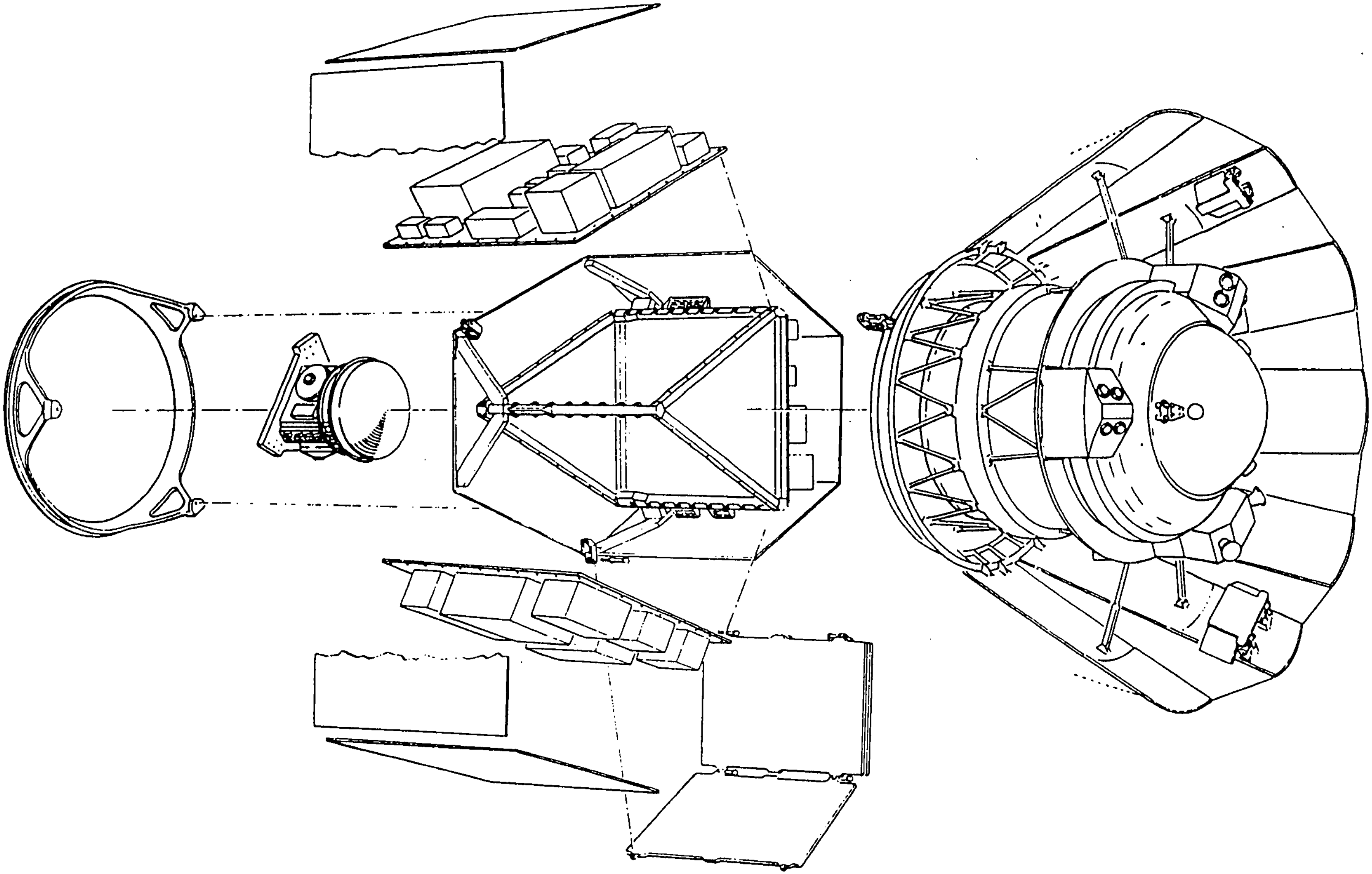
ACS-72

COBE MWEA
SPEED CONTROL ELECTRONICS
FIGURE 2

COBE ON-ORBIT ENGINEERING PERFORMANCE

STRUCTURES SUBSYSTEM

OREN R. SHEINMAN
MARCH 7-8, 1990



STRUCTURAL SUBSYSTEM REQUIREMENTS

- DELTA 5920 INTERFACE

- MASS PROPERTIES
  - STRUCTURE
    - WEIGHT $\leq$ 980 LBS.
  - OBSERVATORY

	PREDICTED		ACTUAL
$(Ixy^2 + Ixz^2)^{1/2} \leq$ 50 SLUG-FT	21		31
Ixx KNOWN TO $\pm$ 2%	766	(1.2%)	775
$\mid Iyy - Izz \mid \leq$ 300 SLUG-FT	20		--
C.G. OPPOSITE POROUS PLUG			

COBE/DELTA STRUCTURE WEIGHT SUMMARY

COMPONENT	CONCEPT (LBS)	CDR (LBS)	FINAL (LBS)
THERMAL SHIELD	115	108	107.3
SOLAR ARRAY	150	210	223.2 *
DMR SUPPORT STRUCTURE	110	43	47.1
COWLING	65	110	62.0
PRIMARY ** STRUCTURE	305	278	286.5
CONE	---	25	26.8
ACS MODULE	40	53.5	54.0
OMNI BOOM	10	10.5	12.5
BRACKETS	75	62	59.8
HARDWARE	80	80	19.0
MISCELLANEOUS	50	50	0.0
STRUCTURE TOTAL	1000	980.0	898.2

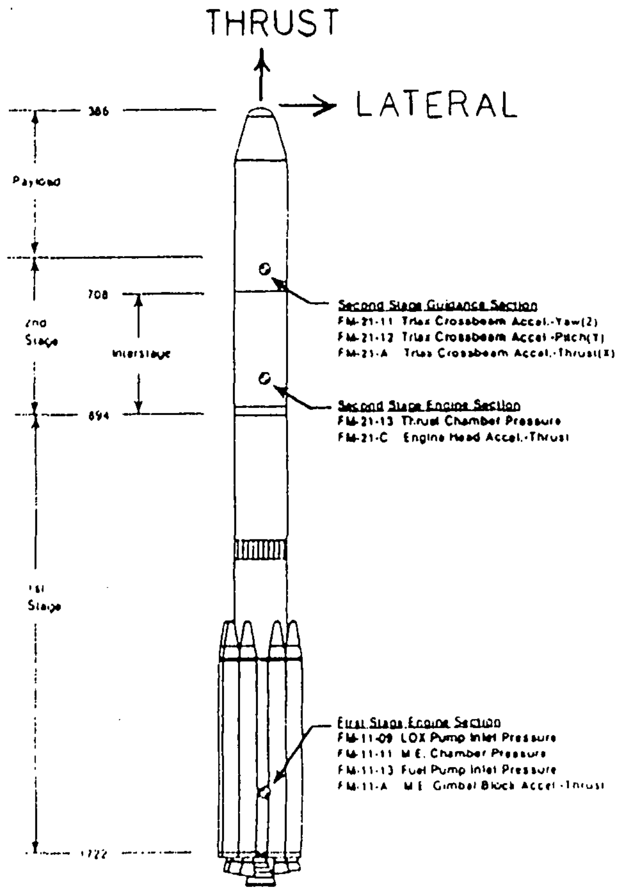
* PIN PULLER/PYRO INCLUDED

** PRIMARY STRUCTURE CONSISTS OF THE MAIN FRAME, TOP DECK, BOTTOM DECKS, A FRAMES, EQUIPMENT PANELS, PANEL TO PANEL DISCONNECT BRACKETS, AND HARDWARE.

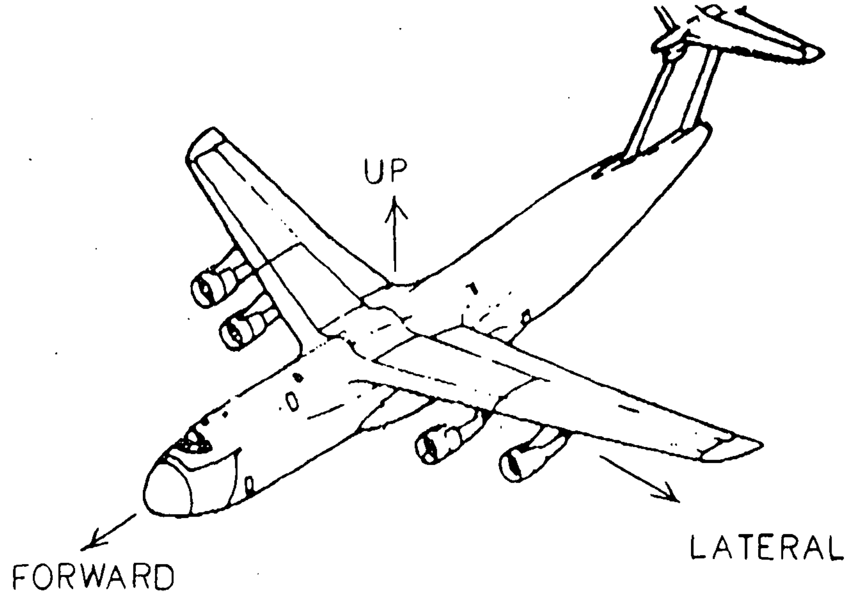
STRUCTURAL SUBSYSTEM REQUIREMENTS (CONT.)

- INTERFACES
  - 6019 PAF
  - DEWAR
  - SOLAR ARRAYS
  - THERMAL SHIELD
  - OMNI ANTENNA
  - ELECTRONIC BOXES
  - ACS
  - ELECTRICAL
  - THERMAL
  - HANDLING

- MAINTAIN STRUCTURAL INTEGRITY WHEN SUBJECTED TO DELTA LAUNCH AND C-5A/I-95 SHIPPING ENVIRONMENTS.
  - DESIGN TO THE FOLLOWING DESIGN LIMIT LOAD FACTORS
    - DELTA +10.7/-0.71g's (THRUST); +/- 2.1g (LATERAL)
    - C-5A 3.0g FORWARD; 1.5g AFT,LATERAL; 2g UP; 3.5g DOWN
  - SHOW +VE MARGINS FOR 1.4 AND 1.25 FACTORS ON ULTIMATE AND YIELD/TEST RESPECTIVELY TIMES THE DESIGN LIMIT LOAD.
  - DESIGN LIMIT LOAD = CONFIDENCE FACTOR x FLIGHT LOAD.
  - CONFIDENCE FACTOR = 1.08 (THRUST); 1.31 (LATERAL); DELTA



DELTA



C-5A

VEHICLE DIRECTION DEFINITIONS

ON-ORBIT PERFORMANCE VS. SPECIFIC REQUIREMENTS

- TELEMETRY BEING RECEIVED - SPACECRAFT IS ALIVE!
  - NO DIRECT INDICATORS
  - SPACECRAFT SEPARATION ACHIEVED
  - DEPLOYABLES FUNCTIONED
  - ALIGNMENT MAINTAINED

- LAUNCH PERFORMANCE CHARACTERISTICS

EVENT	LOCATION	FREQUENCY (HZ)		ACCELERATION (G'S)	
		PREDICTED	ACTUAL	PREDICTED *	ACTUAL **
MECO POGO	SECOND STAGE GUIDANCE SECTION	19.027	17.7	1.76 (T)	1.85
				0.64 (L)	0.51
MINI POGO 1	SECOND STAGE GUIDANCE SECTION	26 - 27	27.3	0.12 (T)	0.16
				0.40 (L)	0.23
MINI POGO 2	SECOND STAGE GUIDANCE SECTION	32 - 38	32.0	0.12 (T)	0.02
				0.22 (L)	0.14

* DOES NOT INCLUDE STEADY STATE ACCELERATIONS

** VALUES REFERENCED FROM R. COLADONATO

TRADE-OFFS

- MACHINED VS. HONEYCOMB STRUCTURE
  - CANDIDATES
    - TOP DECK
    - EQUIPMENT PANELS
    - BOTTOM DECKS
  - MACHINED STRUCTURE WOULD BE SIGNIFICANTLY HEAVIER THAN HONEYCOMB.
  - HONEYCOMB PROVIDES GREATER MOUNTING FLEXIBILITY.
  - HONEYCOMB IS MORE EXPENSIVE.

- BASE CONFIGURATION
  - SIX-SIDED STRUCTURE
  - MMS STRUCTURE
  - THREE-SIDED STRUCTURE

NEXT TIME AROUND

- ATTACH EQUIPMENT PANELS TO CORNERPOSTS USING THROUGH BOLTS OR SHEAR PINS INSTEAD OF HELICOILS.
- INCREASE SHEAR/TRANSITION AREA OF EQUIPMENT PANEL TIE DOWNS.
  - SHOW POSITIVE MARGINS ANALYTICALLY WITHOUT REQUIRING ADDITIONAL TEST DATA.
- LIFT SPACECRAFT ABOVE THE C.G.
  - PROVIDE LIFT CAPABILITY FOR THE S/C THROUGH THE DEWAR.
  - WE HAD THE LOAD CARRYING CAPABILITY, BUT THE NUMBER OF LIFTPOINTS WERE EITHER INSUFFICIENT OR COVERED UP BY THE DMR'S.
- USE SINGLE INSTEAD OF DOUBLE CASTORS AND PROVIDE A STEERABLE LINK ARM FOR THE FLIGHT DOLLY.

LESSONS LEARNED

- HONEYCOMB PANELS PROVIDED EXTREME FLEXIBILITY IN DESIGN WITH OPTIONS FOR CHANGE.
  - FINAL BLACK BOX AND HARNESS TIE DOWNS NOT REQUIRED UP FRONT DURING DESIGN PHASE OF THE PANELS.
  - BRACKETS, TIE DOWNS, SMALL PACKAGES CAN BE MOUNTED FAR DOWNSTREAM IN THE PROGRAM USING RIV-NUTS, DELRON INSERTS, OR THROUGH BOLTS.

- ETU INVALUABLE FOR THIS TYPE OF PROGRAM.
  - PROVIDES A TOOL THROUGH WHICH PROOF OF CONCEPT AND DRESS REHEARSAL CAPABILITY FOR TESTS AND FIT CHECKS CAN BE PERFORMED.
  - DECOUPLES FLIGHT STRUCTURE FROM SUBSYSTEM TESTING FOR EARLY DELIVERY TO INTEGRATION.
  - PROVIDES REPRESENTATIVE STRUCTURE FOR FABRICATION OF HARNESS AND THERMAL BLANKETS.
  - PROVIDES PERSONNEL EXPERIENCE IN HANDLING BEFORE FLIGHT STRUCTURE IS PROCESSED.

- CAMLOC FASTENERS TYPICALLY USED IN THE AIRCRAFT INDUSTRY HAVE GOOD APPLICATIONS FOR NON-STRUCTURAL, REMOVABLE PANELS.

LESSONS LEARNED (CONT.)

- GOOD PRACTICE TO RUN ASSEMBLY DRAWINGS THROUGH THE TECHNICIANS PRIOR TO SUBMITTING PARTS TO FABRICATION.
- DOCUMENTATION IS EXTREMELY IMPORTANT.
- DO MAJOR TESTS ON SATURDAYS AND SUNDAYS AT MIDNIGHT!
- DESIGN AS MUCH AS POSSIBLE UP FRONT FOR CLOSEOUTS AND GSE.
- FOUND USE OF ASAP (AUTOMATED STRESS ANALYSIS PROGRAM) USEFUL.
  - ONCE EQUATIONS ARE CODED, INPUT FROM FINITE ELEMENT ANALYSIS IS THE ONLY REQUIREMENT.
    - DELTA LIFTOFF AND MECO POGO
    - C-5A
    - TRANSPORTATION (HIGHWAY)
    - STATIC LOAD TEST CASES
    - HANDLING

LESSONS LEARNED (CONT.)

- THE SHIPPING CONTAINER CAN BE AS BIG A DESIGN JOB AS THE STRUCTURE IN MANY RESPECTS --- START IT EARLY!
- PROCEDURES FOR ASSEMBLY, INSTALLATION, TESTS, ETC. SHOULD BE INTO THE REVIEW CYCLE EARLY, PARTICULARLY WHEN DEALING WITH AN OUT-OF-HOUSE PARTY.
- WHEN MOUNTING HONEYCOMB IN A SIMILAR CONFIGURATION AS THE EQUIPMENT PANELS TO THE FRAMES, ASSUME 75% OF THE LOAD WILL GO DOWN THE REAR FACESHEET FOR STRESS CALCULATIONS.

COBE ON-ORBIT ENGINEERING PERFORMANCE

SOLAR ARRAY SUBSYSTEM

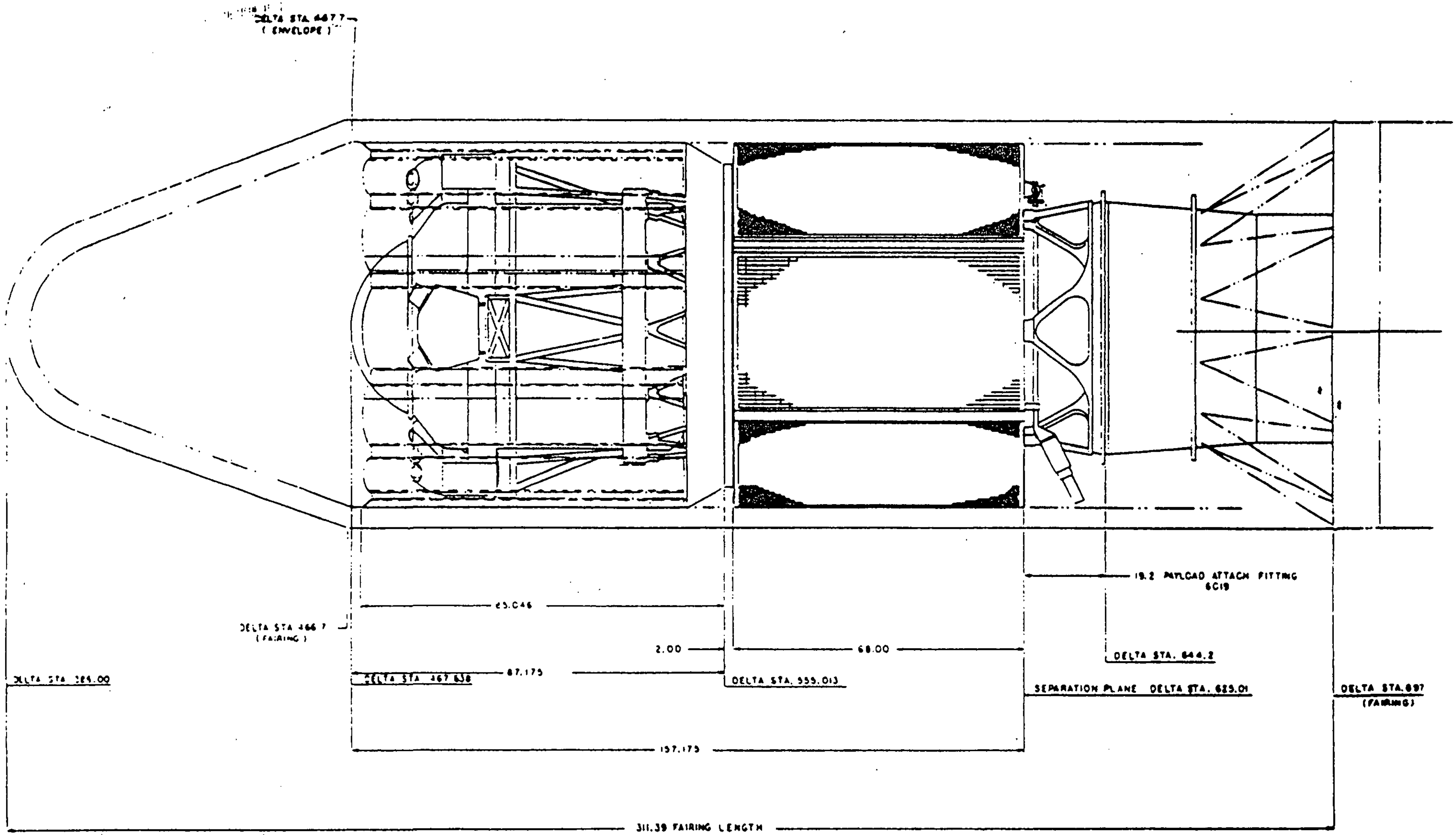
MARCH 1990
CODE 731
GODDARD SPACE FLIGHT CENTER

SOLAR ARRAY SUBSYSTEM

AGENDA

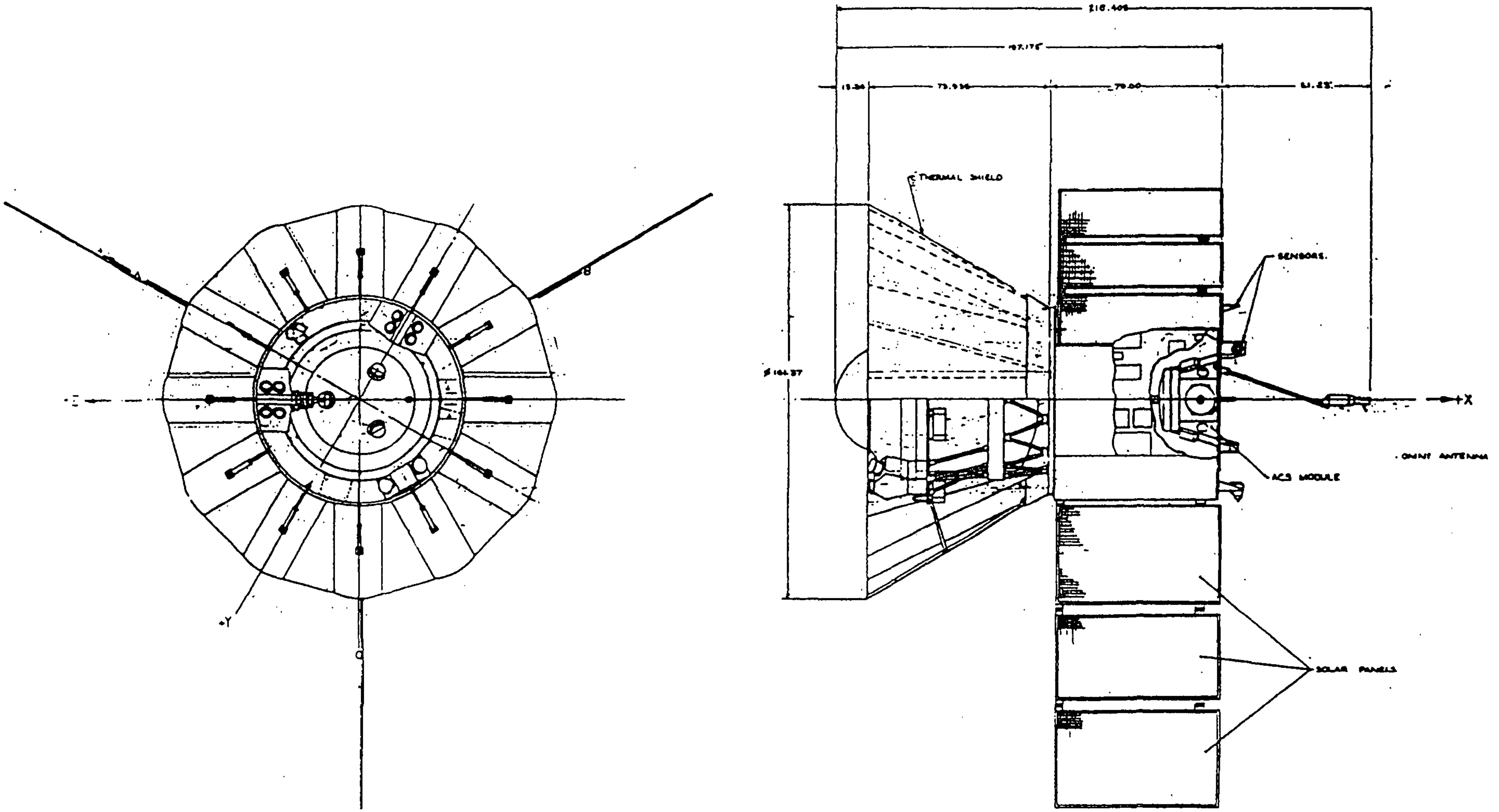
- INTRODUCTION
- DESIGN REQUIREMENTS
- TRADE-OFFS DURING DEVELOPMENT
- SIGNIFICANT PROBLEMS DURING I & T
- ON-ORBIT PERFORMANCE VS. SPECIFIC REQUIREMENTS
- FUTURE CONSIDERATION
- LESSONS LEARNED

SNN-01
3/90



LAUNCH CONFIGURATION - SIDE VIEW

SNN-02
3/90



COBE DELTA ORBIT CONFIGURATION

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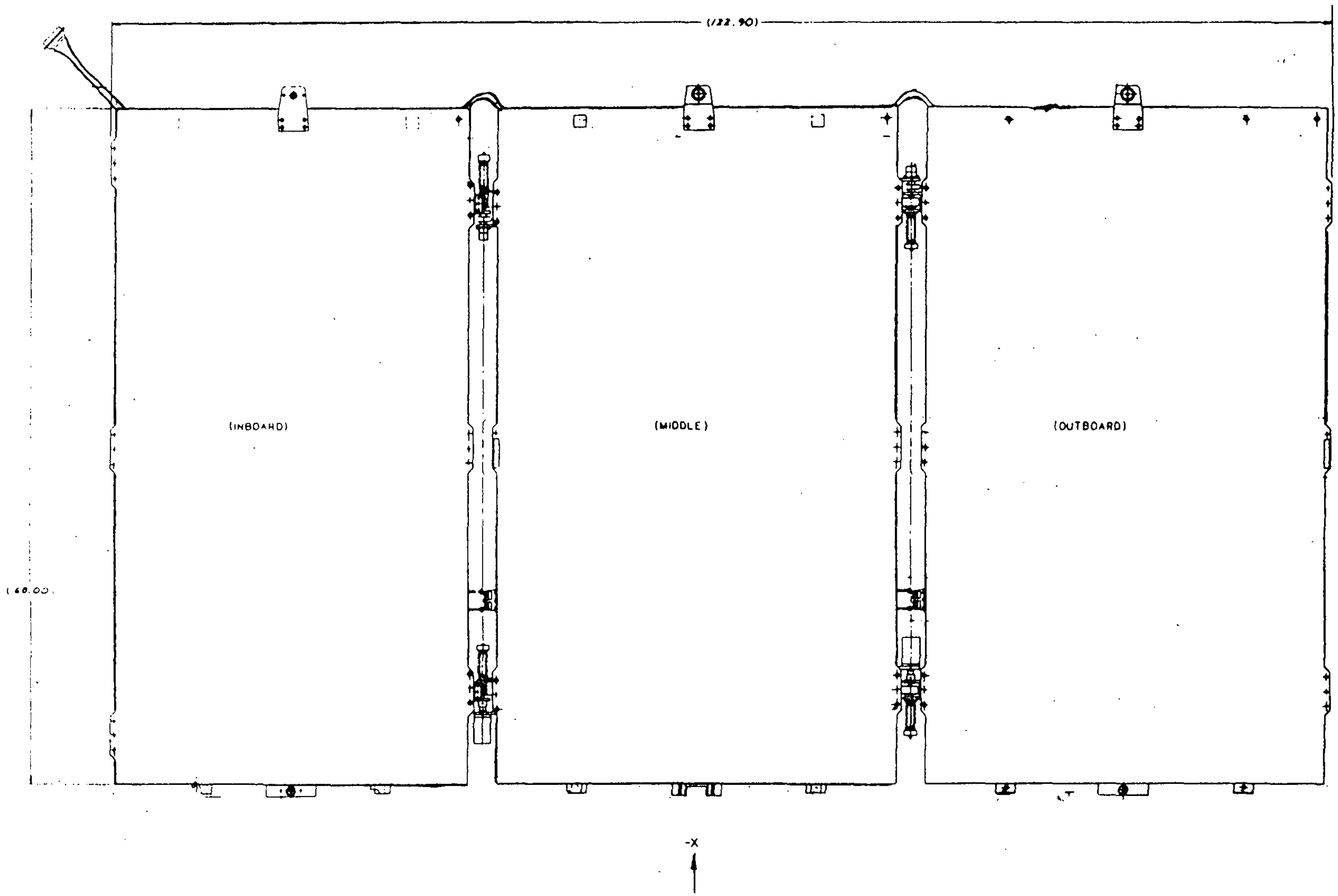
3/90



SOLAR ARRAY WING

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3/90

REQUIREMENTS FOR THE COBE SOLAR ARRAY

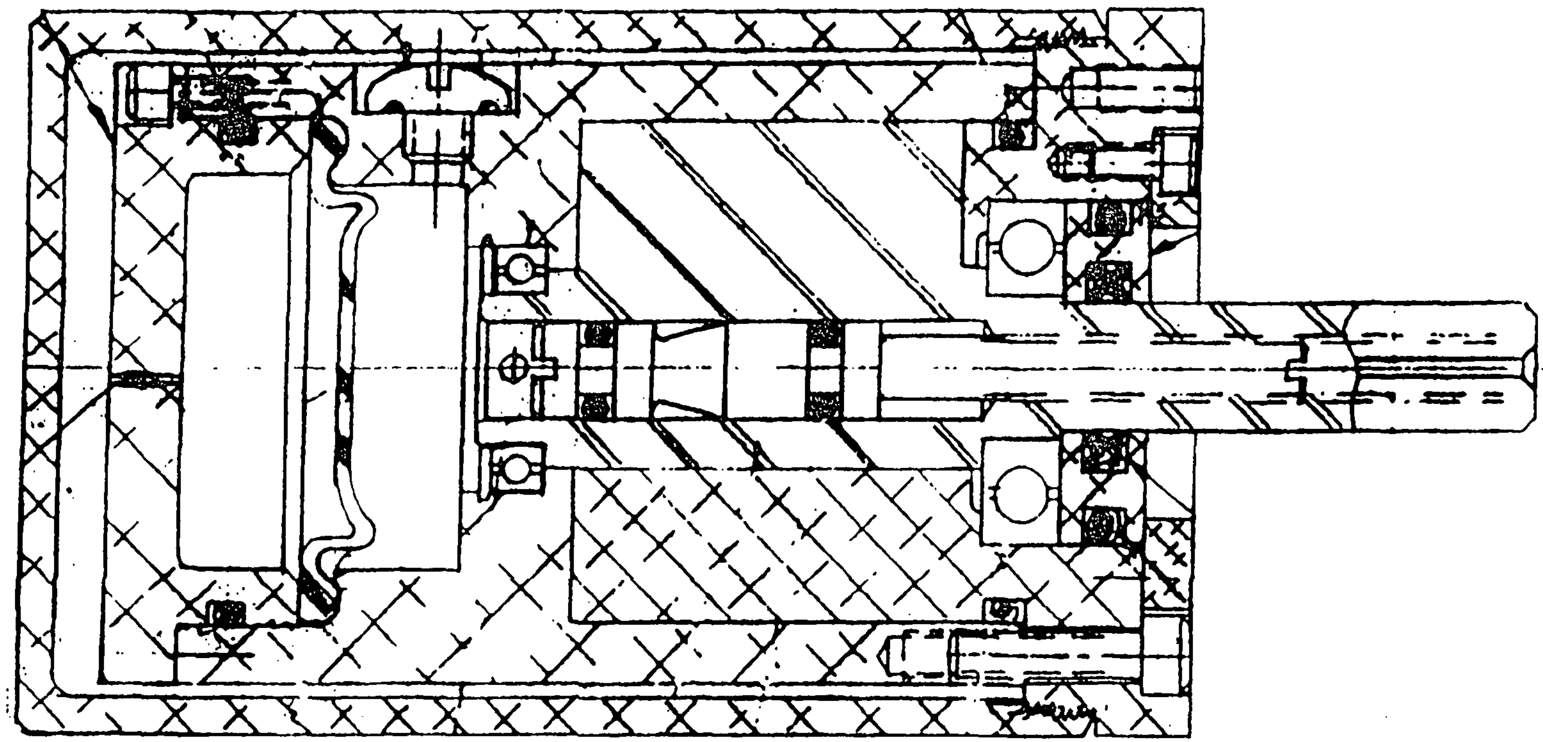
REQUIREMENTS PARAMETER	VALUE	BASIS	VERIFICATION PLAN
CELL AREA	328 FT^2	-	-
ELV LAUNCH • STOWED SPACE • LOADS	 - 18G, 3G	 DESIGN ANALYSIS	 - STRUCTURAL TEST
DEPLOYED TORQUE RATIO	2 TO 1	DESIGN	COMPONENT TEST DEPLOYED TEST
DEPLOYED STIFFNESS	>1HZ	ANALYSIS	STRUCTURAL TEST
TEST TEMPERATURE (°C)	-20°C HOT +50°C COLD	ANALYSIS ANALYSIS	DEPLOYED TEST DEPLOYED TEST
FULLY REDUNDANT PYROTECHNIC IN RELEASE MECHANISM	-	DESIGN	COMPONENT TEST DEPLOYED TEST
ELECTRICAL GROUNDING	-	DESIGN	COMPONENT TEST ASSEMBLY TEST
CONTAMINATION CONTROL	-	-	CONT. MONITOR CLEANING PROC.

TRADE-OFFS DURING DEVELOPMENT

- PIN PULLER VS. BOLT CUTTER
- THERMAL VAC VS. THERMAL DEPLOYMENT TEST
- WRAP & STOWAGE OF PANELS
- DEPLOYMENT SEQUENCE (DRIVEN VS. INDEPENDANT)

PROBLEMS DURING I&T

- SLOT HOLES ON SHEAR BLOCKS DURING ASSEMBLY
- G-NEGATION SYSTEM (AIR PADS AND MOBILE SYSTEM)
- DAMPER REPLACEMENT @ LAUNCH SITE
- PIN PULLER REBOUND



DAMPER

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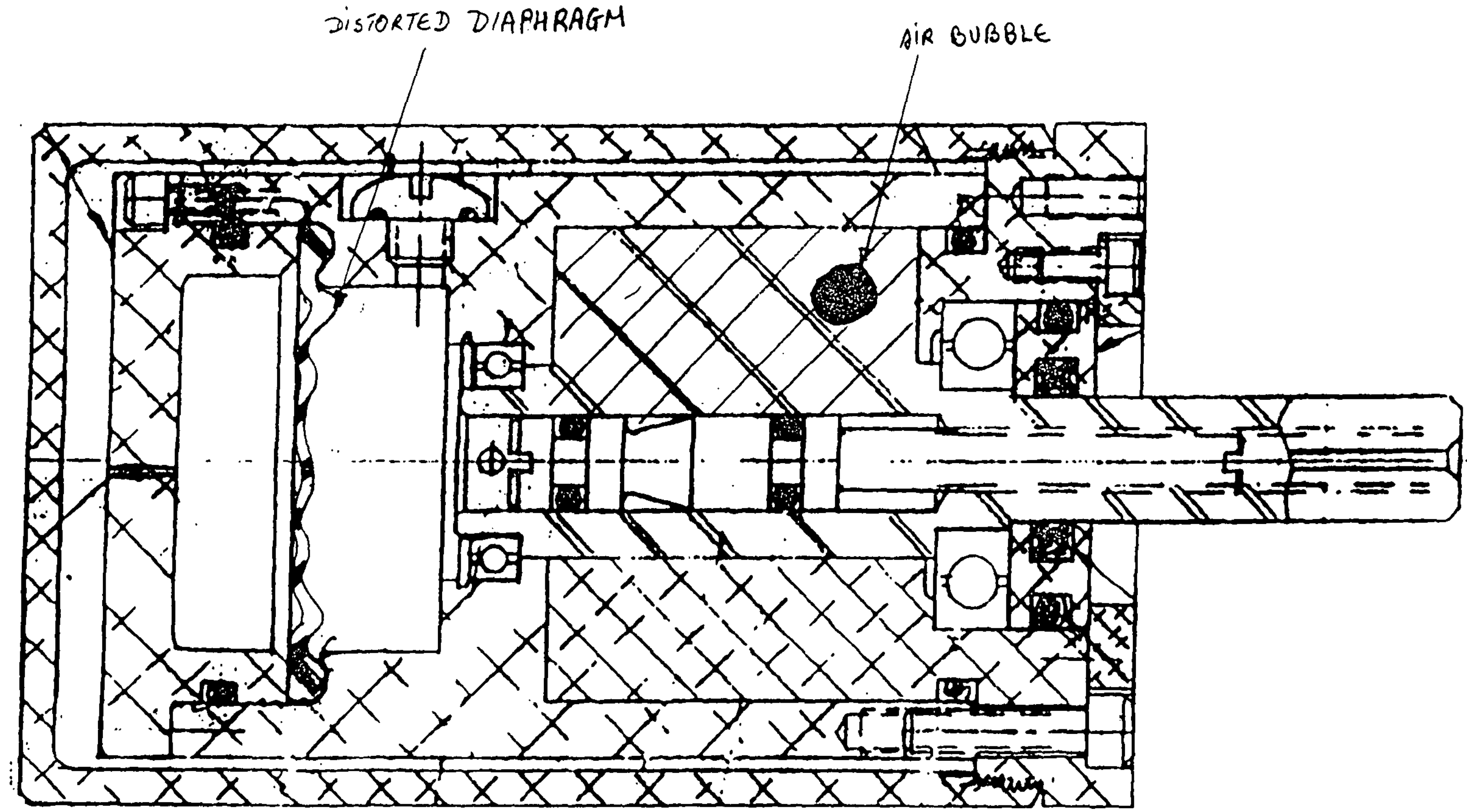

BAD DAMPER.

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ON-ORBIT PERFORMANCE

- OUTBOARD PRIMARY MICROSWITCH B DID NOT WORK
- PANEL DEPLOYMENT SEQUENCE AS EXPECTED

DEPLOYMENT TIMES COMPARISON

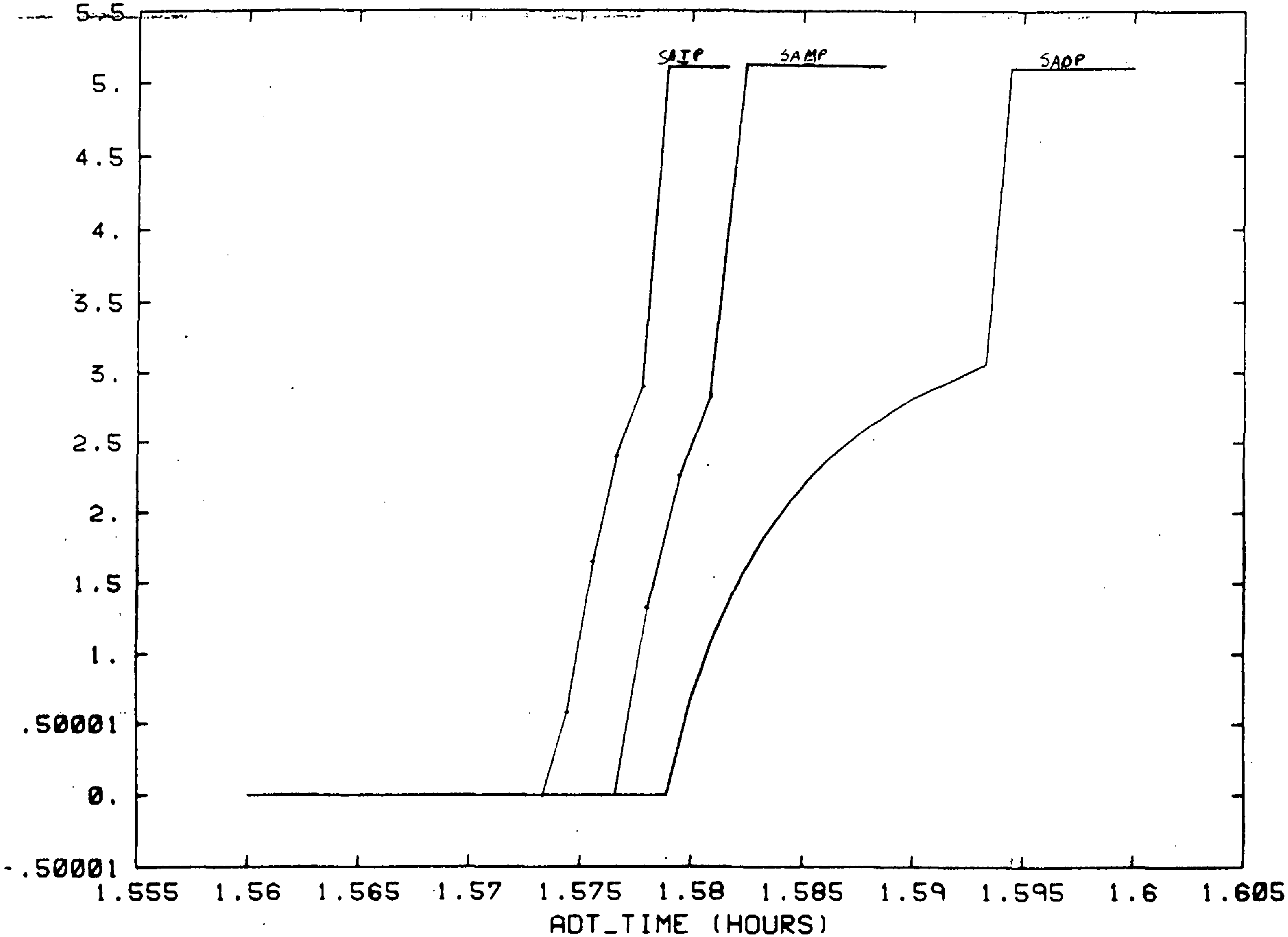
	TEST	ON-ORBIT	
WING A	22 sec	22 sec	Root hinge @ 50 C
	13 sec	16 sec	60° hinge @ 25 C
	52 sec	56 sec	180° hinge @ 25 C
WING B	25 sec	21 sec	Root hinge @ 50 C
	11 sec	17 sec	60° hinge @ 25 C
	55 sec	50 sec	180° hinge @ 25 C
WING C	32 sec	25 sec	Root hinge @ 50 C
	12 sec	25 sec	60° hinge @ 25 C
	56 sec	68 sec	180° hinge @ 25 C

* ROOT HINGE HAS HEATERS TO MAINTAIN DAMPER'S TEMPERATURE AT 45 TO 50 DEGREE C.

* ON-ORBIT DEPLOYMENT TIMES ARE ± 16 SEC DUE TO DATA CONVERSION

* ROOT HINGE DAMPER B WAS REPLACED @ LAUNCH SITE

LPLOT NSB_MNQ_GC_001:ADT_TIME/UNITS=HOURS/WINDOW=(1.56,1.6) PSAAOP/TAL



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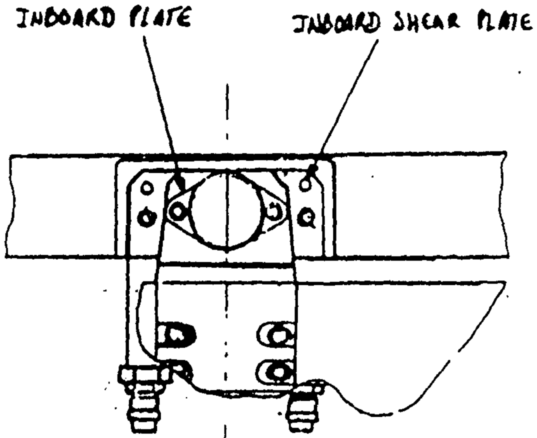
By:SHIV

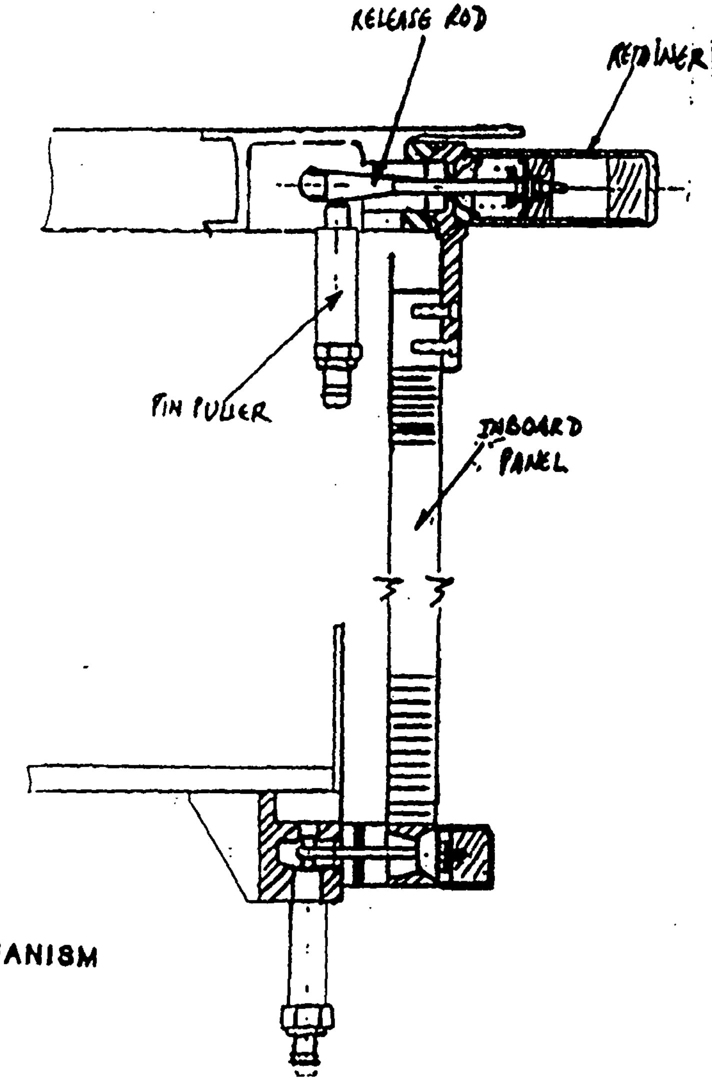
18-NOV-1989 14:01:08.30

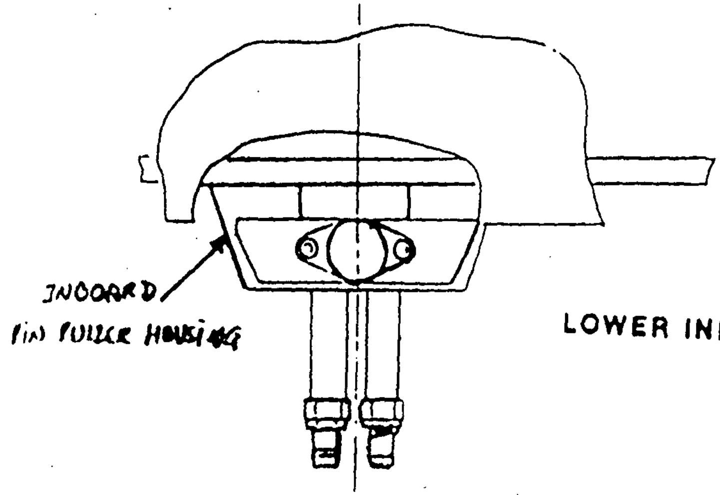
FUTURE CONSIDERATION

- DESIGN THE BOTTOM RELEASE MECHANISM SIMILAR TO THE UPPER RELEASE MECHANISM
- PROVIDE BETTER ADJUSTMENT & ACCESS FOR THE ASSEMBLY OF THE WING
- PROVIDE SUSPENSION SYSTEM ON THE AIR PAD SYSTEM
- CHECK THE DAMPER AFTER EACH DEPLOYMENT TO INSURE NO DAMAGE. AS PART OF THE DAMPER QUAL TESTS, ADD AN OVER-TORQUE TEST TO ELIMINATE THE DAMPERS WITH WEAK SEALS
- DOCUMENT MORE EFFICIENTLY

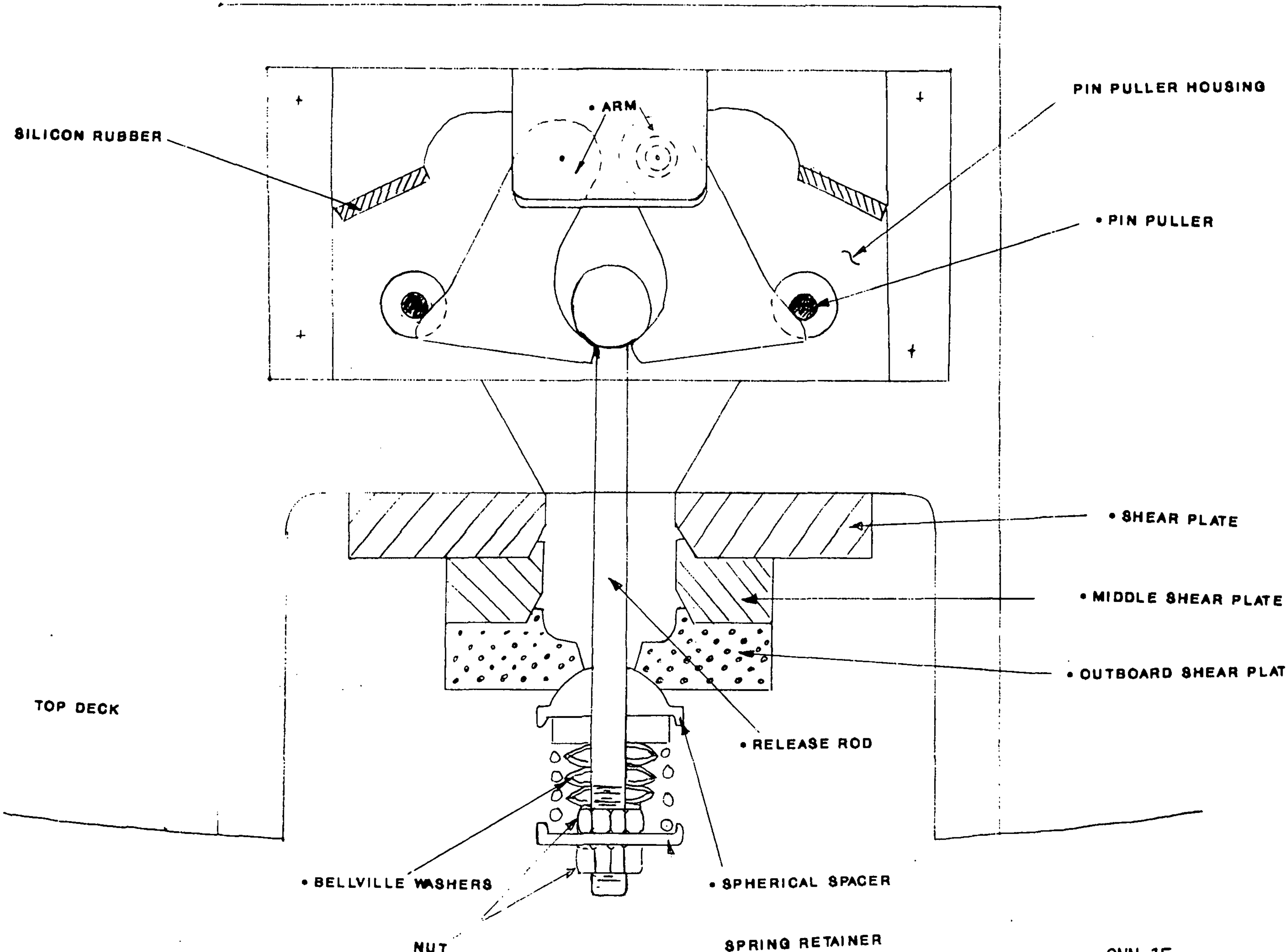
UPPER INBOARD RELEASE MECHANISM







LOWER INBOARD RELEASE MECHANISM



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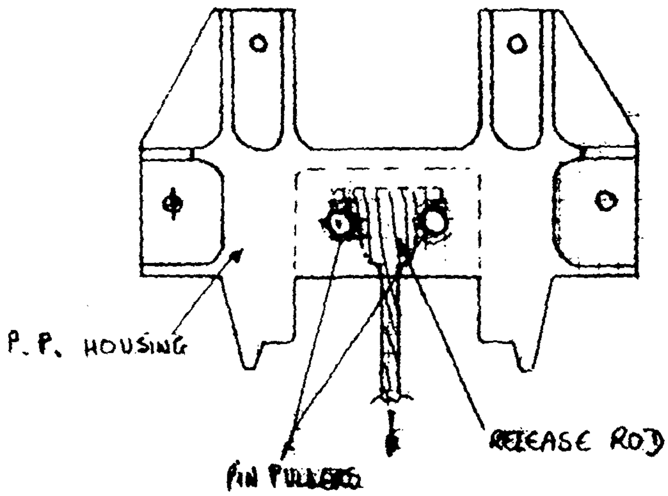
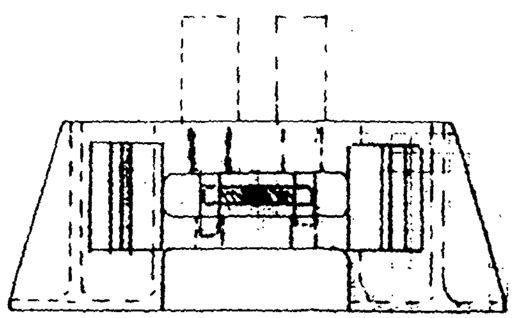
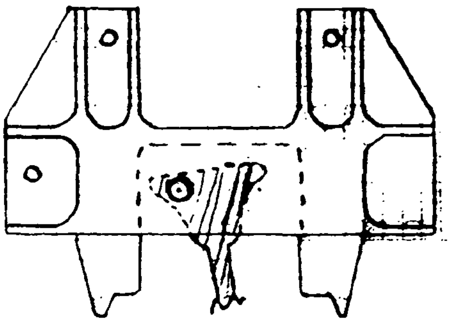
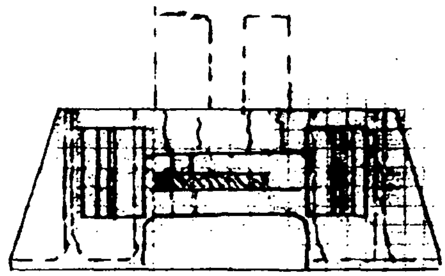
• UPPER MIDDLE/OUTBOARD RELEASE MECHANISM





DURING STOWED POSITION





PIN PULLER FIRED & REBOUNDED

SNN-17
3/90

LESSONS LEARNED

- USE DAMPER IN THE SPRING DEPLOYABLE SYSTEM
- HARNESS TORQUE TEST AS SOON AS POSSIBLE
- STUDY THE INTERFACE AROUND THE DEPLOYABLE
- ALWAYS BE CONSCIOUS ON THE FAILURE MODES OF THE DEPLOYABLE

COBE/DELTA

THERMAL/RF SHIELD

ON-ORBIT ENGINEERING

PERFORMANCE

ALPHONSO C. STEWART

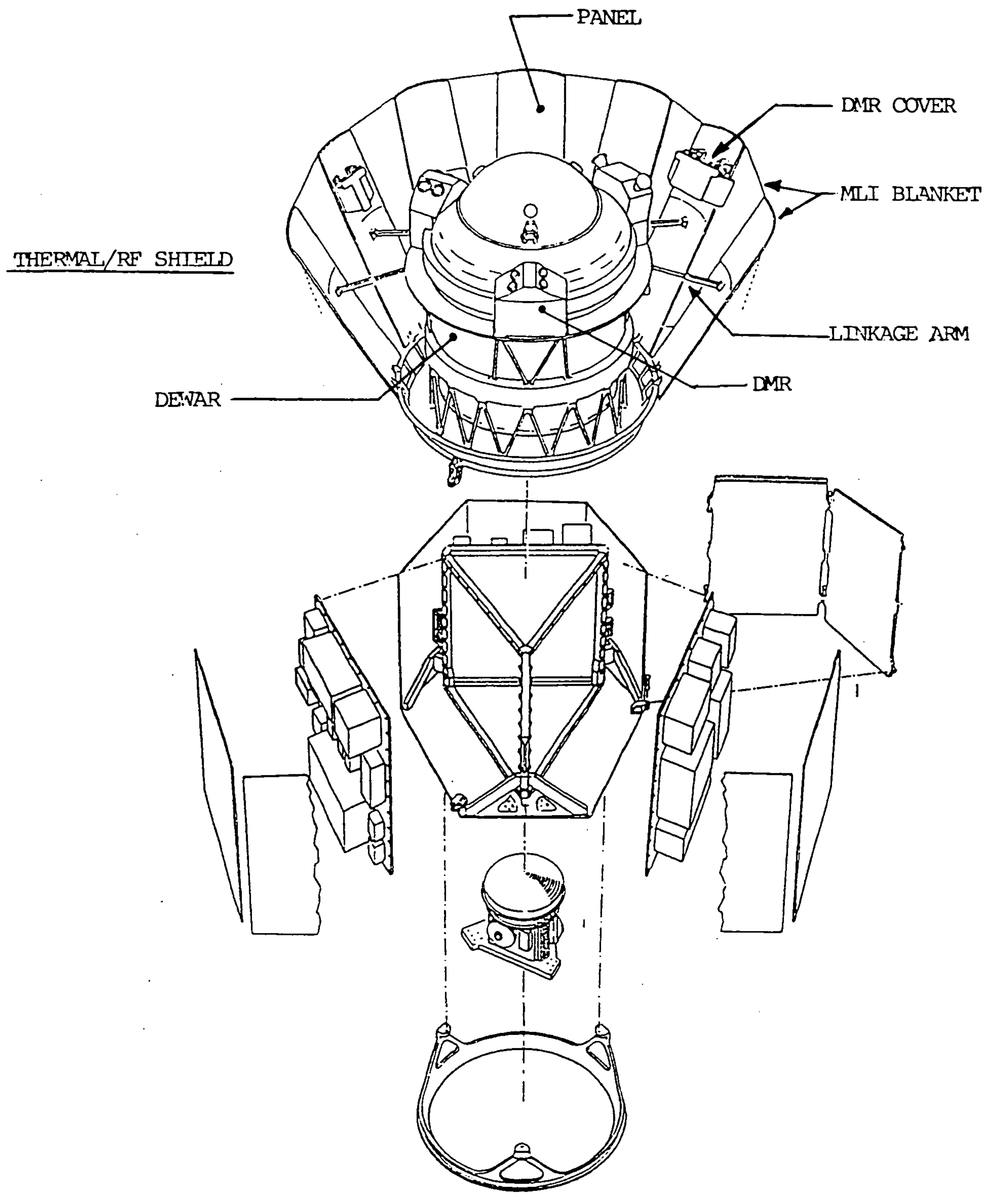
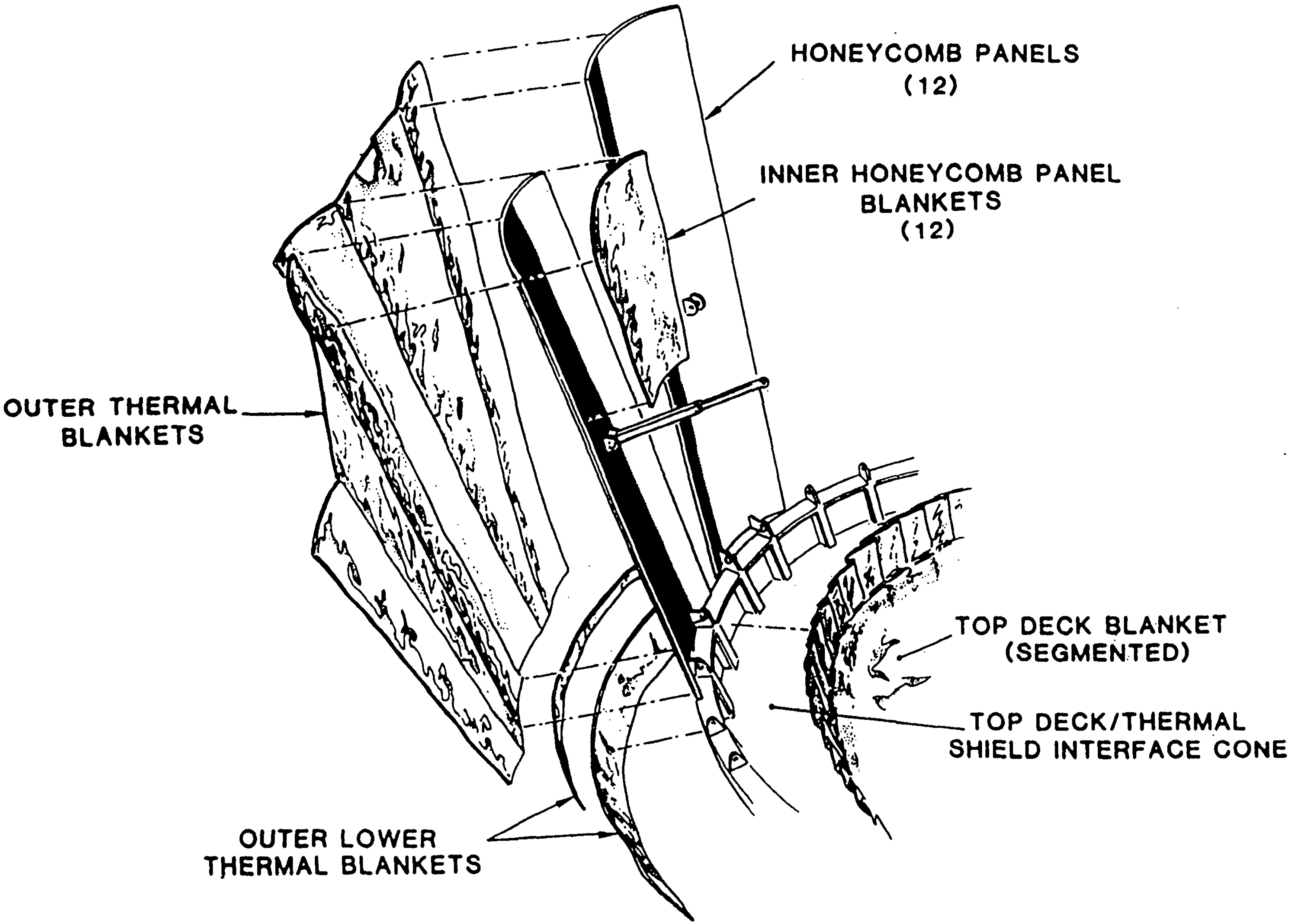


FIGURE 1: EXPLODED VIEW OF COBE SPACECRAFT



EXPLODED VIEW OF THERMAL/RF SHIELD

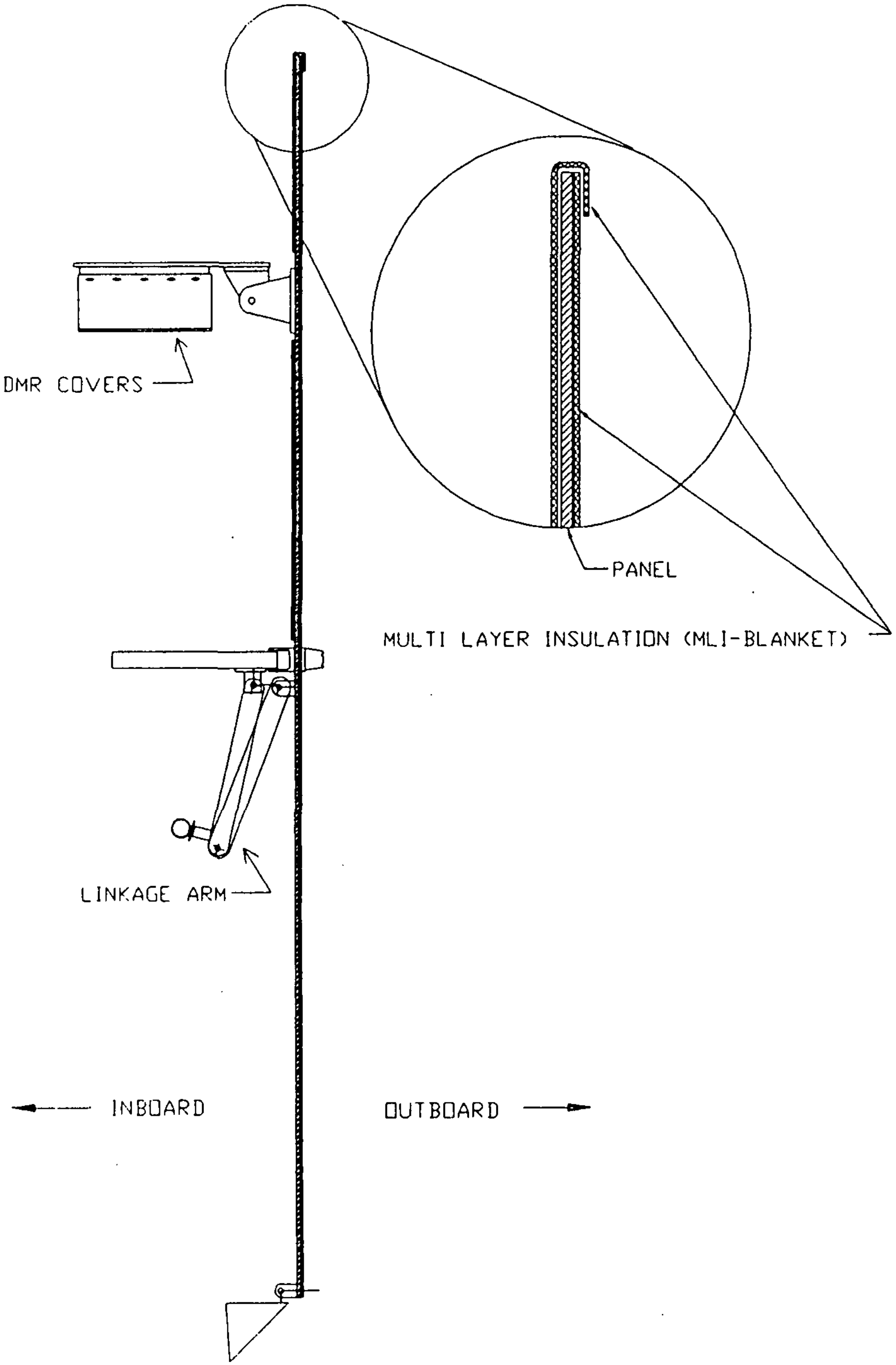


FIGURE 3: STOWED THERMAL/RF SHIELD WITH MLI AND DMR COVER

T/RF ACS
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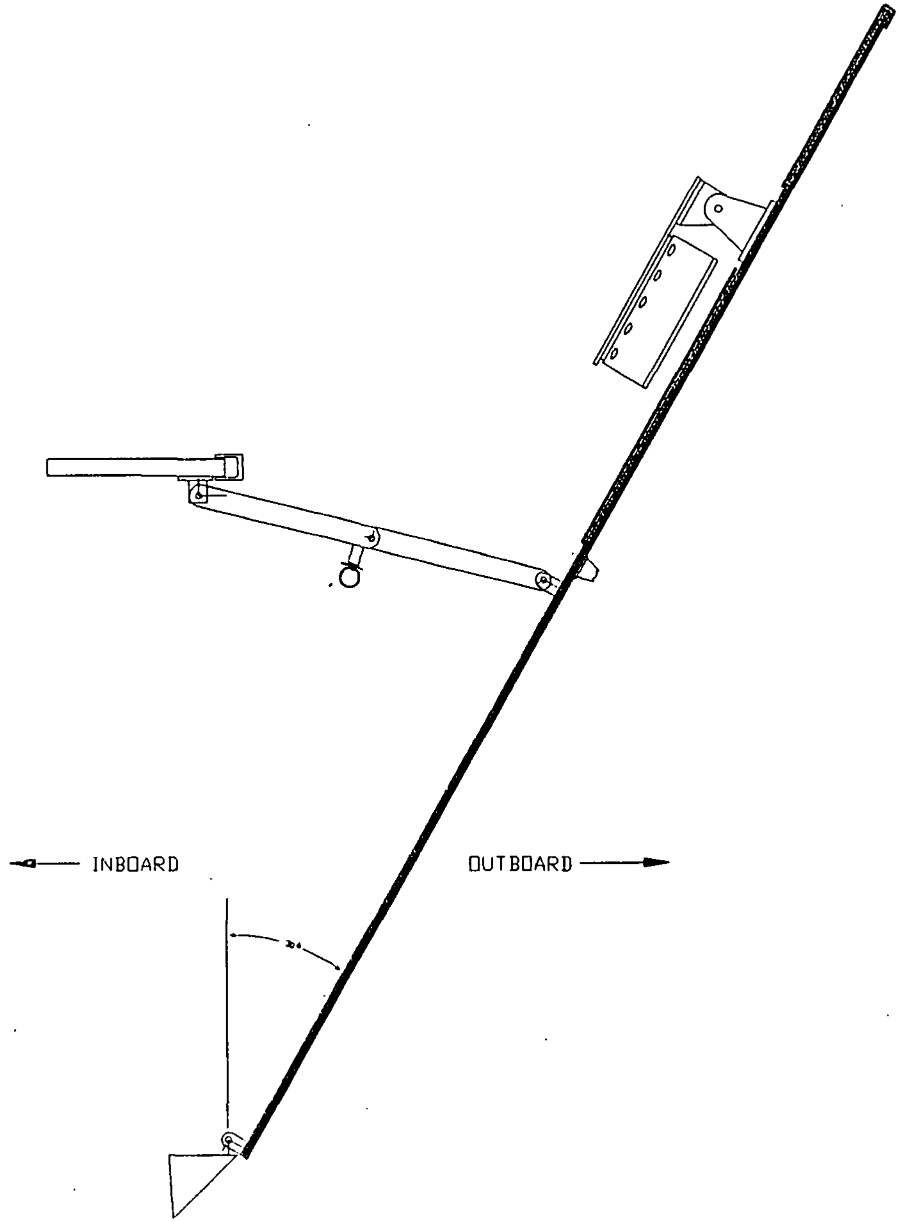

FIGURE 4: DEPLOYED THERMAL/RF SHIELD WITH MLI AND DMR COVER

T/RF ACS
3/7/90

TRADE-OFFS

- PANELS

  SHEET AND STIFFENER VS HONEYCOMB

  - SHEET AND STIFFENER UNABLE TO MEET THE WEIGHT BUDGET AND GEOMETRIC ENVELOPE REQUIREMENTS

  - HONEYCOMB LIGHTER, MORE RIGID, AND FIT WITH SIZE REQUIREMENTS

- LINKAGE ARM ASSEMBLY

  ALUMINUM LINKAGE ARM VS STAINLESS STEEL

  - ALUMINUM IS LIGHTER AND HAS A HIGH THERMAL CONDUCTIVITY

  - STAINLESS STEEL IS HEAVIER AND HAS A LOWER THERMAL CONDUCTIVITY

- FLEXIBLE SHIELD PARTS (BLANKET)

  RF FABRICS VS MLI

  - RF FABRICS ATTENUATE SIGNALS BUT FAIL TO MEET THERMAL REQUIREMENTS

  - MLI (ALUMINIZE KAPTON) MET BOTH THERMAL AND RF REQUIREMENTS

THERMAL/RF SHIELD DESIGN REQUIREMENTS

- THERMAL

  TO PROVIDE THERMAL ISOLATION FOR THE DEWAR AND DMR INSTRUMENTS ON THE INSIDE FROM THE EARTH AND SUN ON THE OUTSIDE. (INNER SHIELD TEMP. LESS THAN 220 K)

- EMI

  PREVENT RADIATION FROM EARTH, SUN, AND S/C COMMUNICATIONS FROM REACHING THE INSTRUMENTS.
  (ATTENUATE 60 dB AT 2.2 GHz)

- PHYSICAL DIMENSION

  STOWED CONFIGURATION MUST FIT WITHIN PAYLOAD ENVELOPE

PROBLEMS DURING I & T

- LINK ARM

  o PANEL(#8) UNABLE TO STOW PROPERLY DURING FLIGHT INTEGRATION

  o LINKAGE ARM FLANGE MADE CONTACT WITH PANEL BRACKET RADIUS DURING S/C INTEGRATION.

  SOLN.: RESHAPED PANEL BRACKET RADIUS

- GLINT

  o LIGHT (LASER) BEAM REFLECTED OF THE EDGE OF THE T/RF BLANKET DOWN TOWARDS THE DEWAR DURING GLINT TEST.

  SOLN.: INSTALL LIGHT BLOCKER ON T/RF PANEL CORNERS (5 MIL ALUMINIZE KAPTON)

- CABLE RESTRAINT

  o DEVELOP A SYSTEM THAT CAPTURES THE THERMAL/RF SHIELD RETAINER CABLE DURING DEPLOYMENT TEST.

  SOLN.: ATTACH AN ADDITION GSE CABLE TO THE ENDS OF THE RETAINER TO CAPTURE IT DURING DEPLOYMENT TEST.

- G-NEGATION SYSTEM

  o DEVELOP A SYSTEM THAT WILL MINIMIZE THE EFFECTS OF GRAVITY ON THE THERMAL SHIELD DURING DEPLOYMENT TESTING

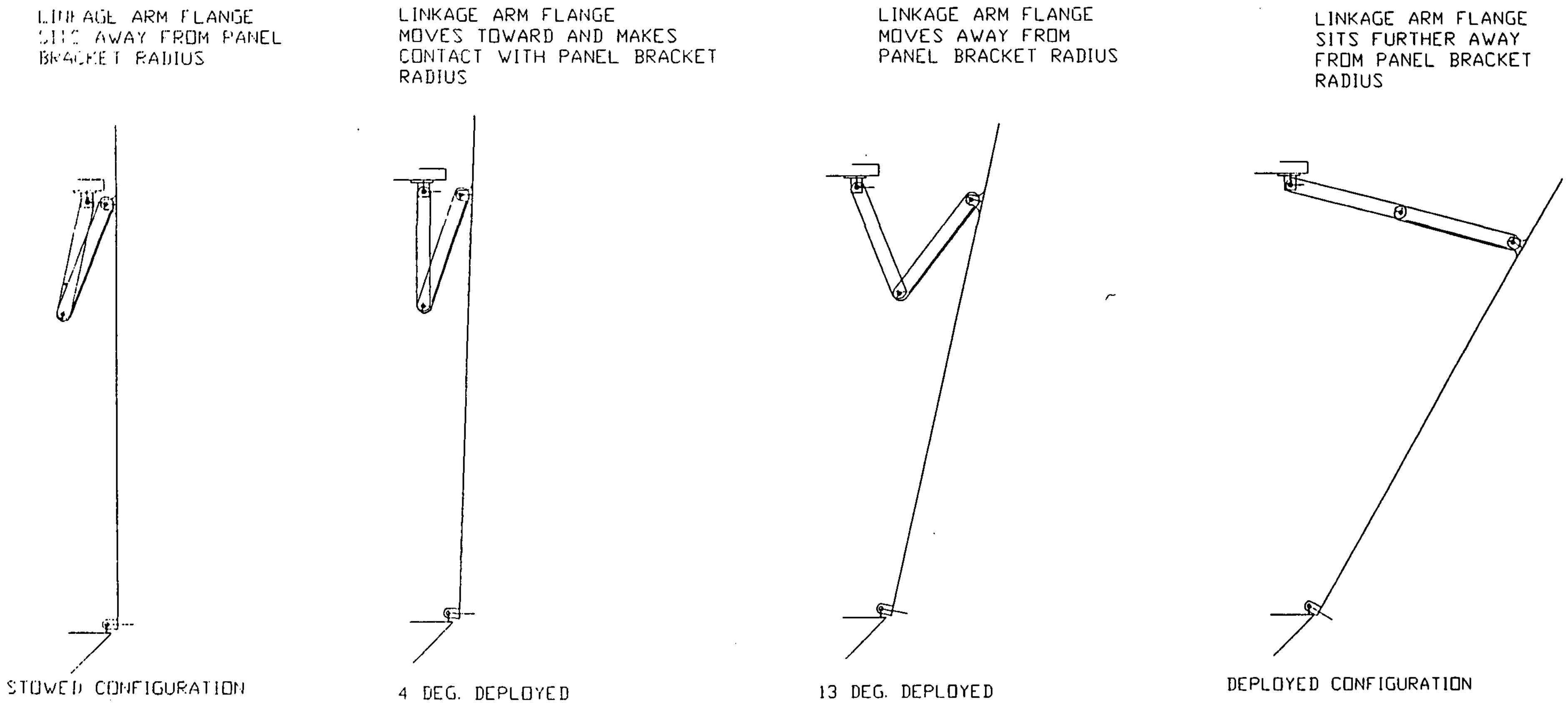

FIGURE 4: THERMAL/RF SHIELD DEPLOYMENT SEQUENCE

T/RF ACS
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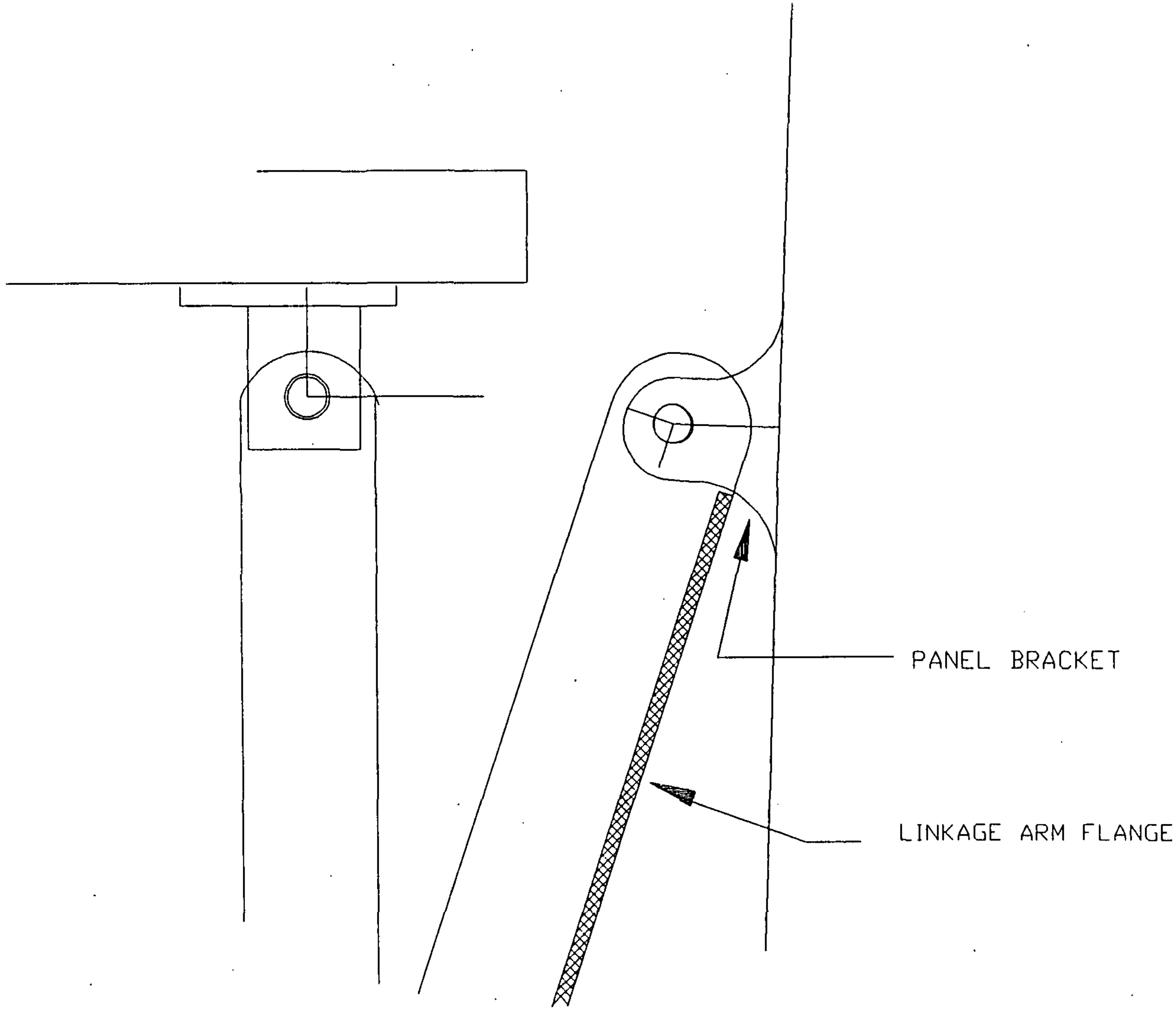
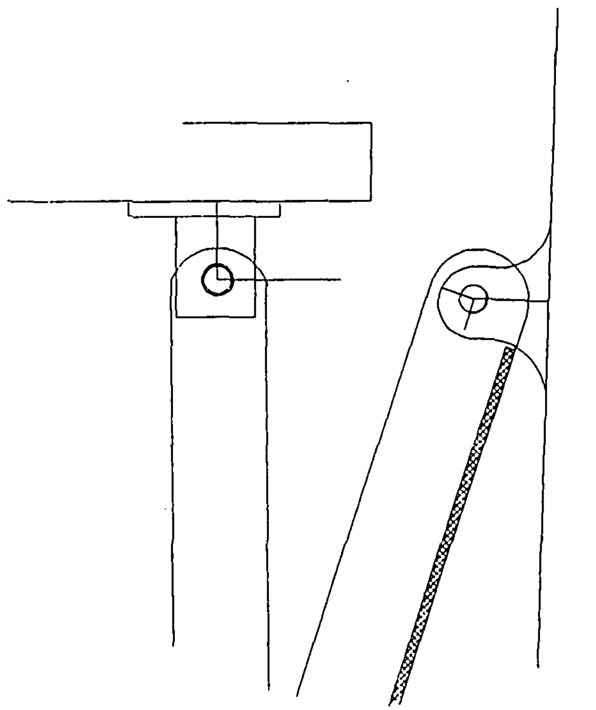
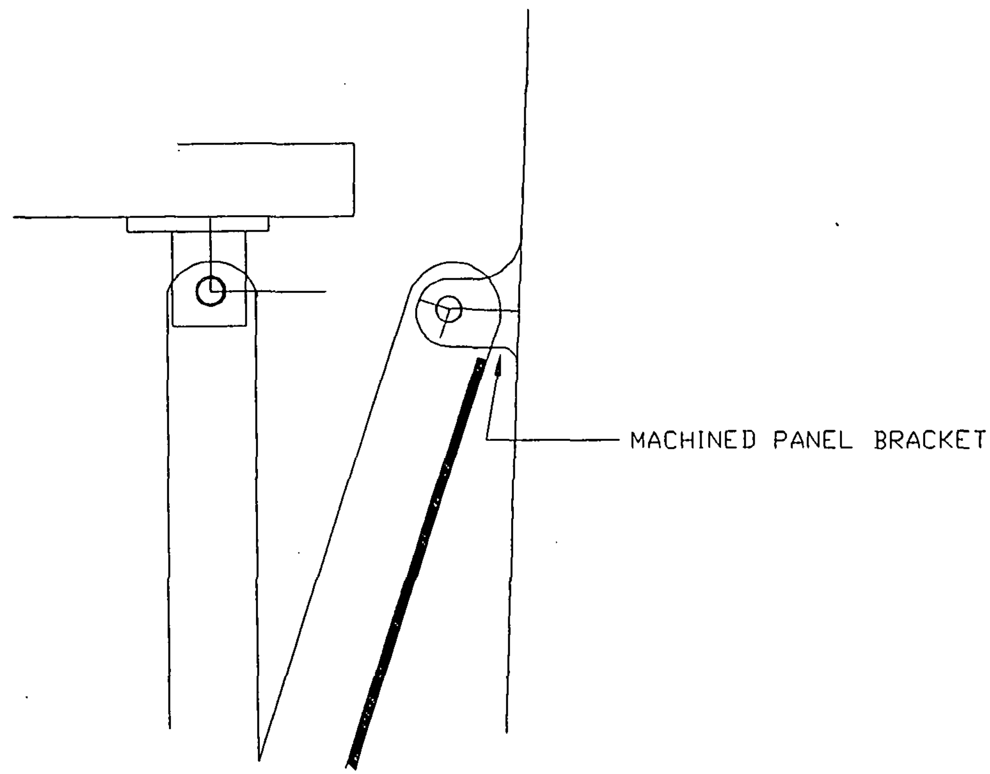


LINKAGE ARM FLANGE AGAINST PANEL BRACKET RADIUS

T/RF ACS
3/7/90



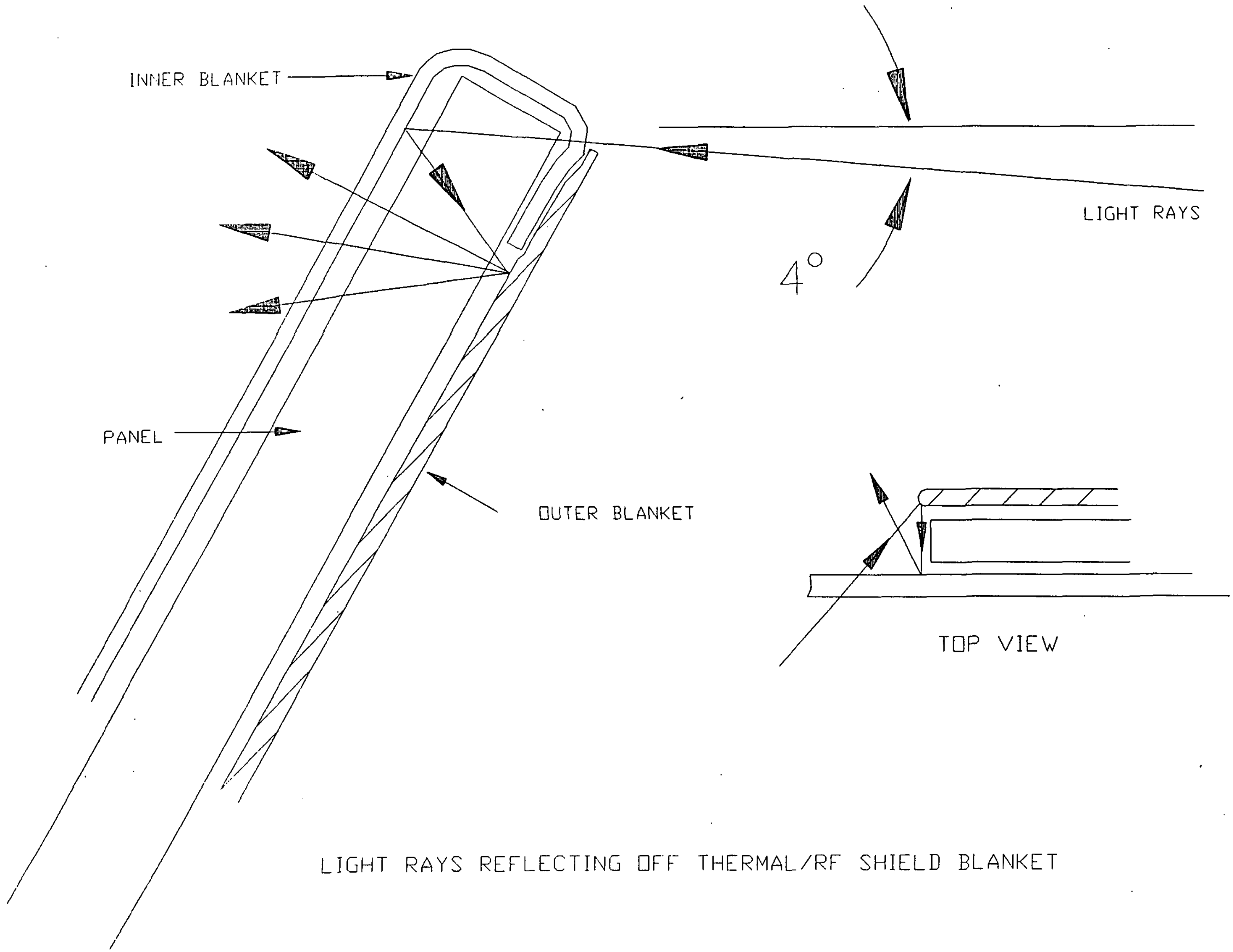
BEFORE



AFTER

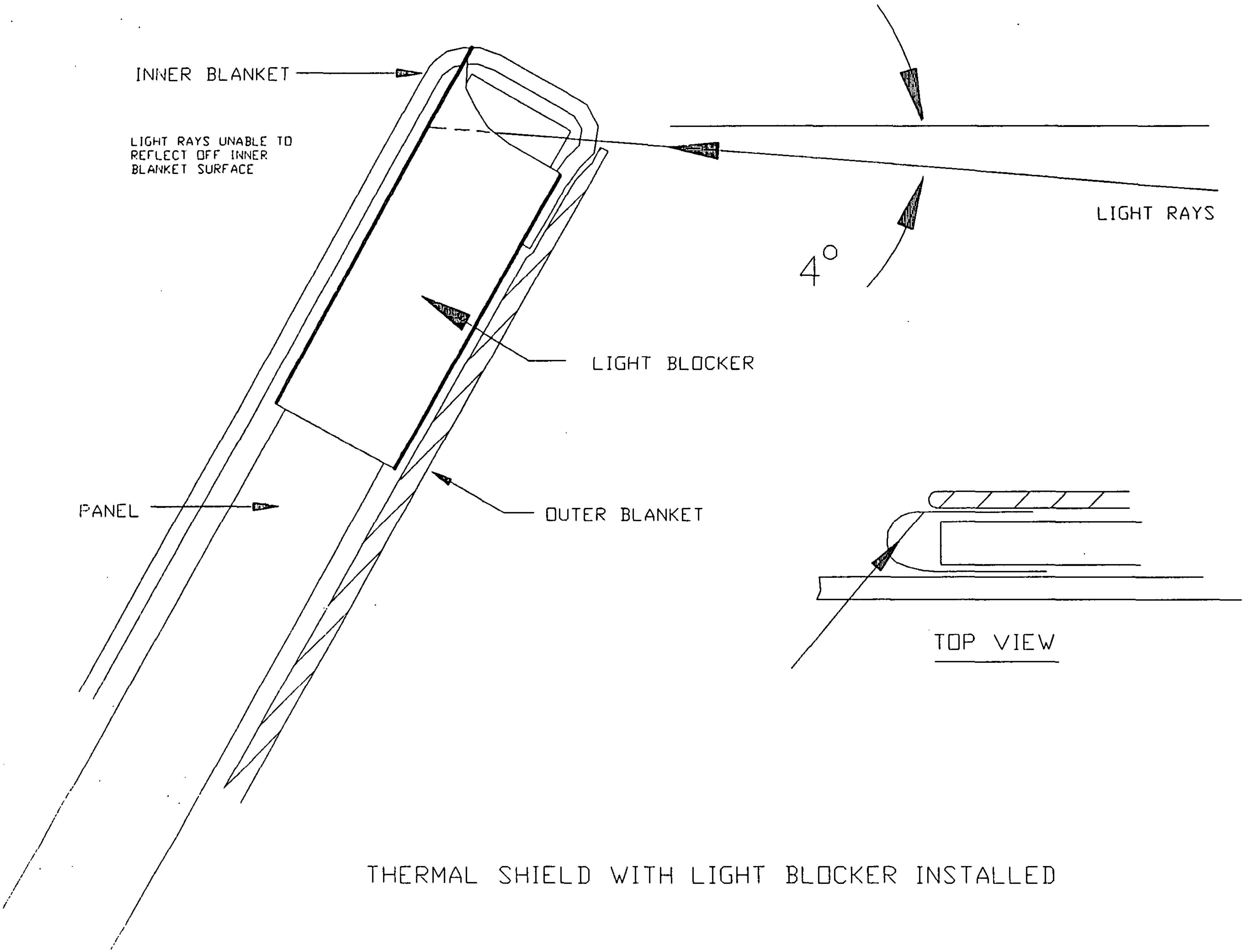
THERMAL/RF SHIELD WITH MACHINED PANEL BRACKET

T/RF ACS
3/7/90



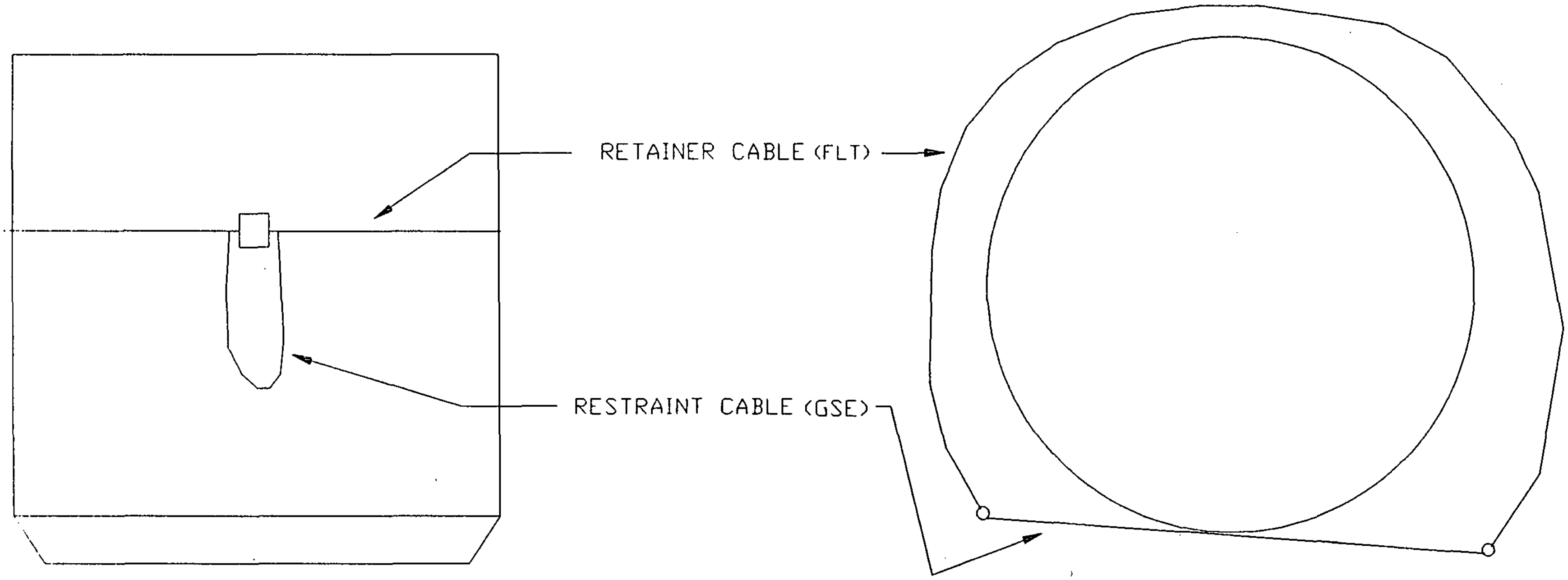
LIGHT RAYS REFLECTING OFF THERMAL/RF SHIELD BLANKET

T/RF ACS
3/7/90



THERMAL SHIELD WITH LIGHT BLOCKER INSTALLED

T/RF ACS
3/7/90



THERMAL/RF SHIELD CABLE WITH
RESTRAINT SYSTEM (STOW CONFIGURATION)

TOP VIEW OF THERMAL/RF SHIELD CABLE
BEING RELEASE IN A DEPLOYMENT TEST

THERMAL/RF SHIELD WITH RETAINER CABLE RESTRAINT SYSTEM

ON-ORBIT PERFORMANCE

- DEPLOYMENT OCCURRED AS EXPECTED (I.E. DEPLOYMENT INDICATED BY MICROSWITCH READINGS)
- ALL TEMPERATURE REQUIREMENTS FOR THERMAL/RF SHIELD HAVE BEEN MET. (ALL LESS THAN 220 K)
- AT PRESENT, NO EVIDENCE OF LIGHT GLINT OVER THE TOP EDGE OF THE T/RF SHIELD.
- AT PRESENT, NO EFFECTS OF EMI ON INSTRUMENTS.

WHAT WOULD I DO DIFFERENTLY

DESIGN AND DEVELOP A TESTING PROGRAM FOR THE THERMAL/RF SHIELD THAT INVOLVES AS FEW PERSONNEL AS POSSIBLE AND USES THE LEAST AMOUNT OF TIME.

- DEVELOP GSE FOR TENSIONING THE T/RF SHIELD RETAINER CABLE.

- DEVELOP A STOWING SYSTEM WHICH IS EASIER TO EXECUTE.

LESSONS LEARNED

- FOR A DEPLOYABLE SYSTEM, BE AWARE OF THE PATH IN WHICH SYSTEM TRAVELS FROM ONE POSITION TO ANOTHER. (I.E. STOWED TO DEPLOYED)
- CONCENTRATE MORE ON GSE SYSTEMS EARLY IN THE DESIGN STAGE AND DO NOT ASSUME IT IT CAN BE TAKEN CARE OF LATER.
- BE CRITICAL OF THE INTERFACES IN WHICH YOUR SUBSYSTEM IS ATTACHED TO. (I.E. ARE THE INTERFACE TOLERANCES SUITABLE FOR YOUR SUBSYSTEM?)

COBE/DELTA OMNI DEPLOYMENT MECHANISM

ON-ORBIT ENGINEERING PERFORMANCE

NASA/GODDARD SPACE FLIGHT CENTER
MECHANICAL ENGINEERING BRANCH
STRUCTURAL LOADS & ANALYSIS SECTION

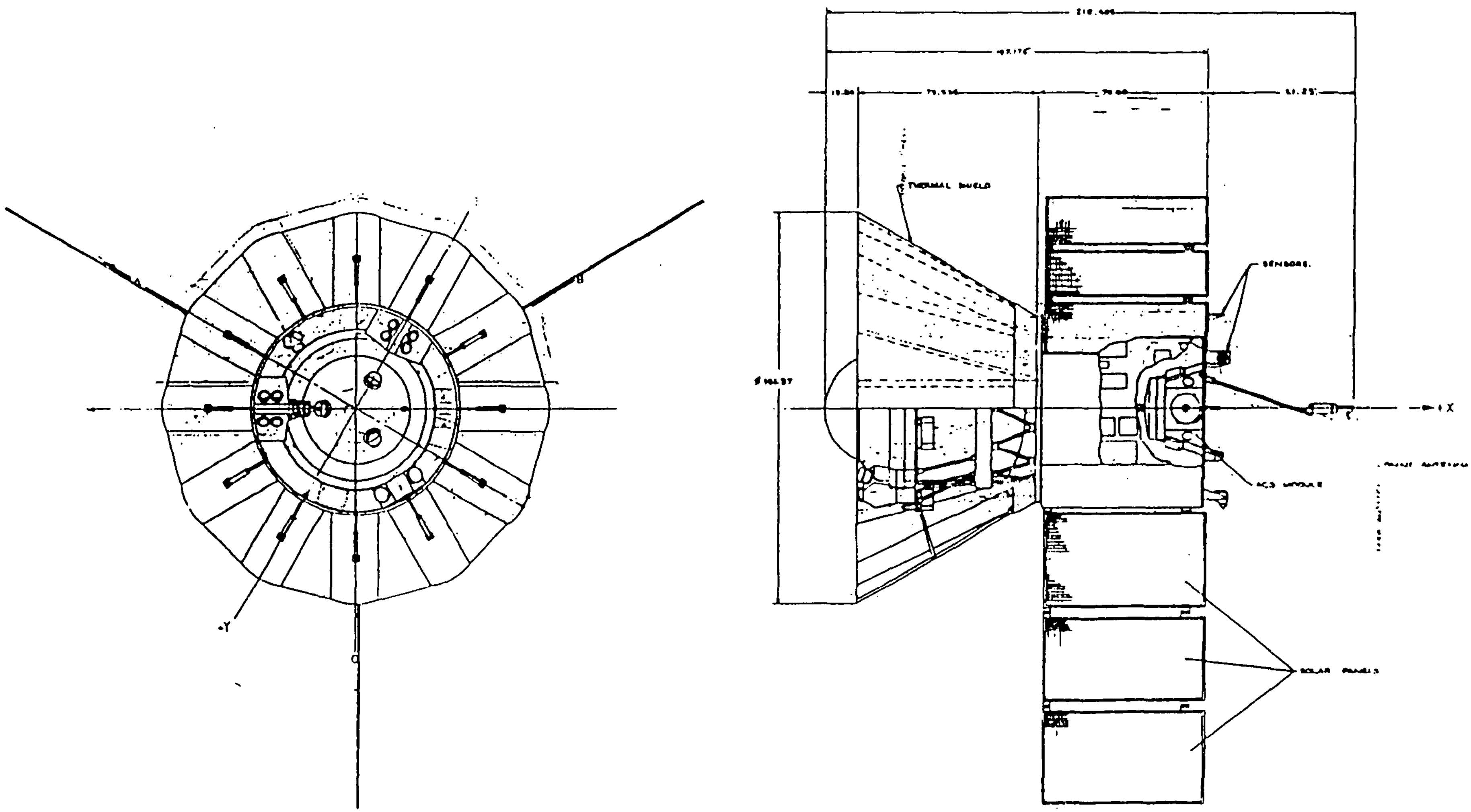
MINH C. PHAN

MARCH 7,8 1990

CONTENTS

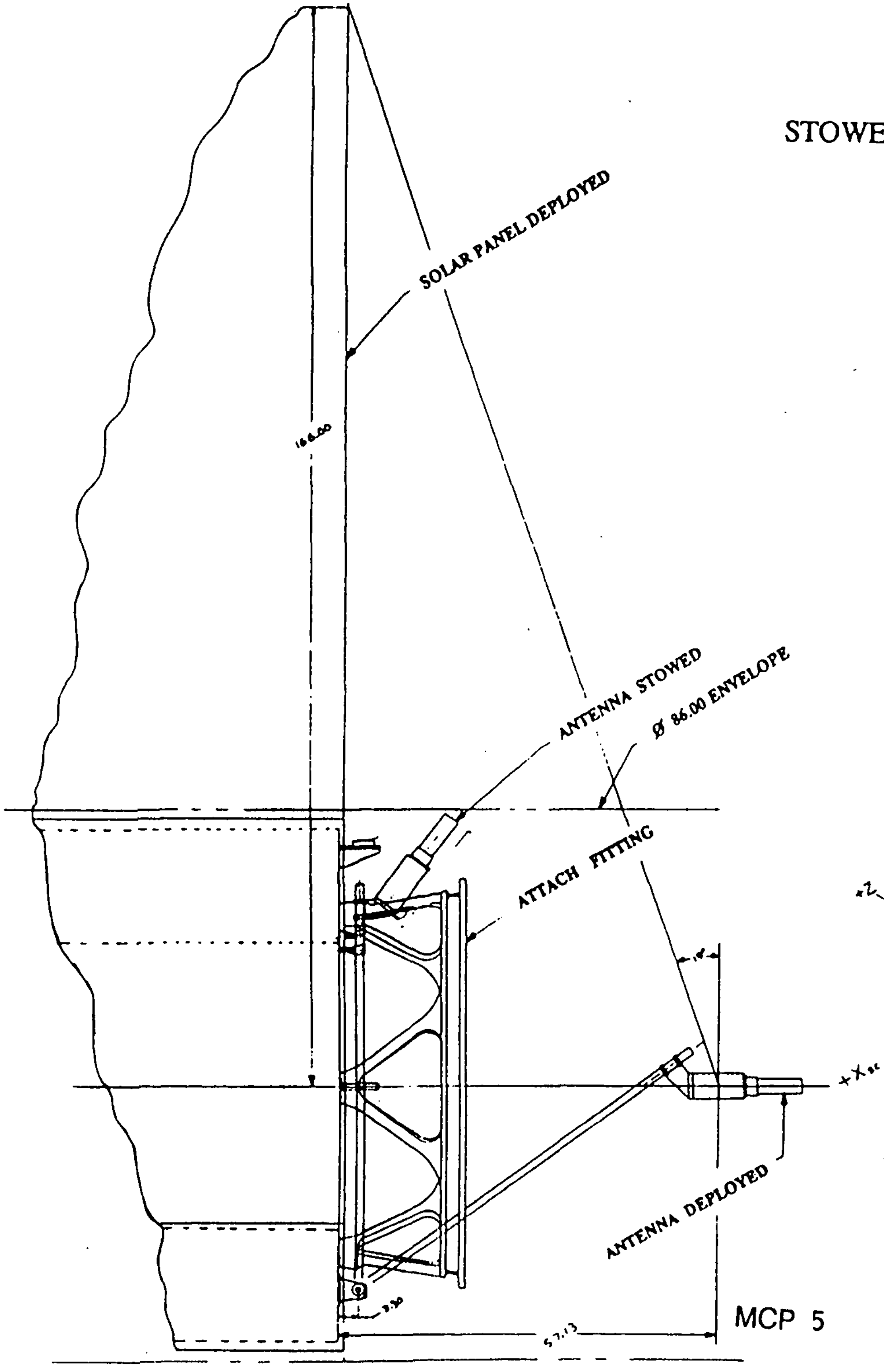
- SUBSYSTEM OVERVIEW
- SPECIFICATIONS AND REQUIREMENTS
- TRADE-OFFS REQUIRED DURING DEVELOPMENT
- SIGNIFICANT PROBLEMS DURING I&T
- WAIVERS/DEVIATIONS
- ON-ORBIT PERFORMANCE VS. SPECIFIC REQUIREMENTS
- DO DIFFERENTLY
- LESSON LEARNED

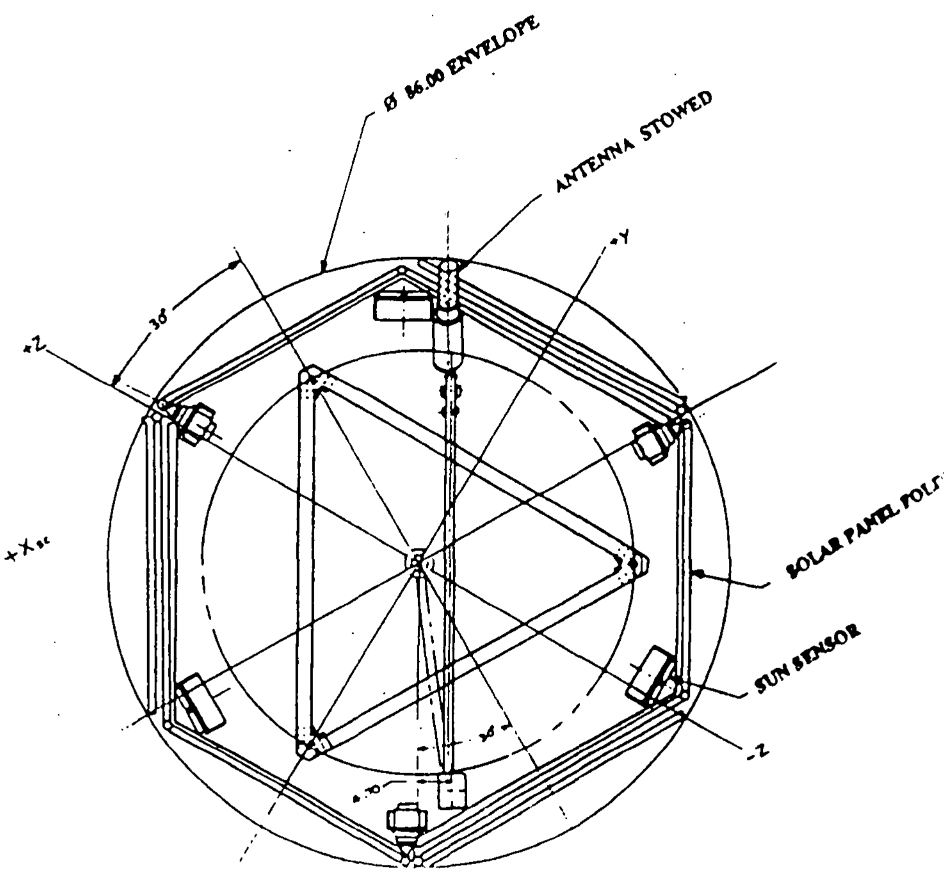
SUBSYSTEM OVERVIEW



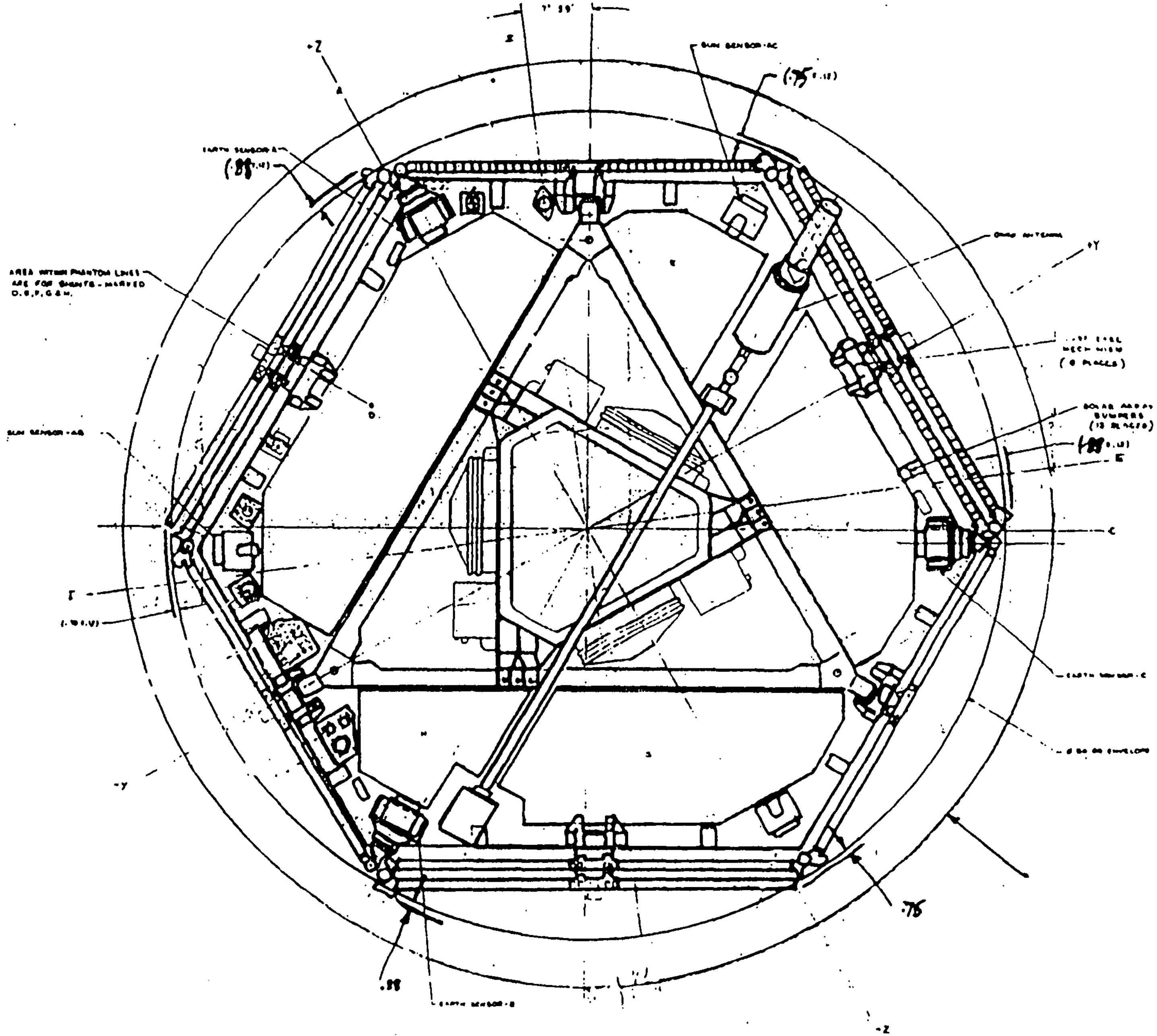
COBE SPACECRAFT ON-ORBIT CONFIGURATION

STOWED/DEPLOYED CONFIGURATION





MCP 5



VIEW LOOKING IN -X DIRECTION
(BOTTOM DECK)

MCP 6

REQUIREMENTS

STOWED CONFIGURATION

- o PREFERRED ORIENTATION WITHIN ALLOWABLE ENVELOPE
- o FAILURE OF OMNI DEPLOYMENT MUST NOT CAUSE COBE MISSION FAILURE

DEPLOYED CONFIGURATION

- o ALIGN ON COBE GEOMETRIC CENTER WITH +- 0.5 DEGREE TILT FROM THE X-AXIS
- o CLEARANCE OF 19 DEGREES FROM THE SOLAR ARRAY

DESIGN LOADS

LAUNCH LOAD [18-G THRUST, 3-G LATERAL] FOR ALL COMPONENTS

REQUIREMENTS (CONT.)

STRUCTURAL DYNAMICS

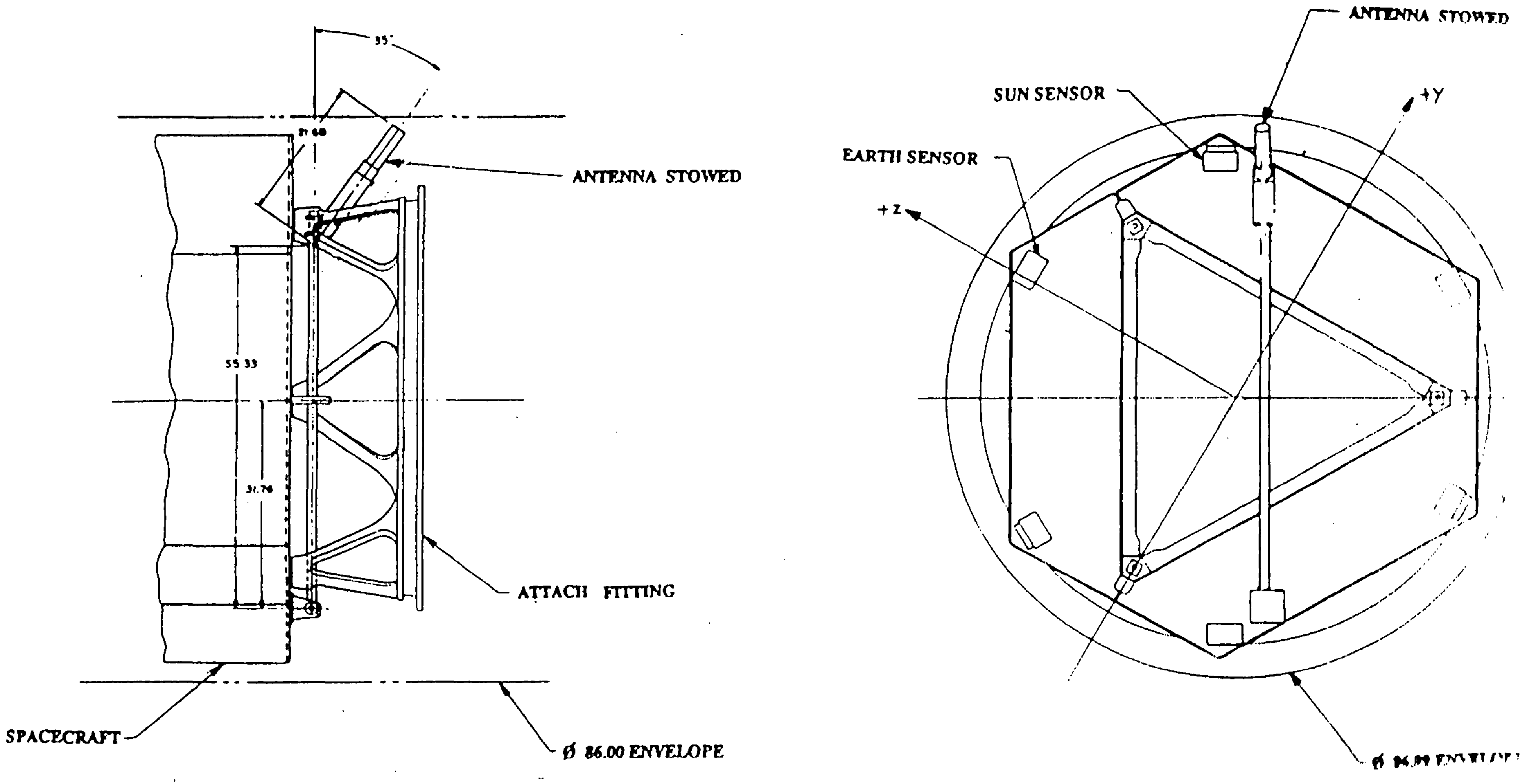
- o IN STOWED CONFIGURATION, FIRST STRUCTURAL NATURAL FREQUENCY MUST BE ABOVE 40 HZ
- o IN THE DEPLOYED CONFIGURATION, FIRST STRUCTURAL NATURAL FREQUENCY MUST BE ABOVE 1 HZ

THERMAL

- o OPERATING TEMPERATURE RANGE -20 C <---> +60 C
- o SURVIVAL TEMPERATURE RANGE -40 C <---> +75 C

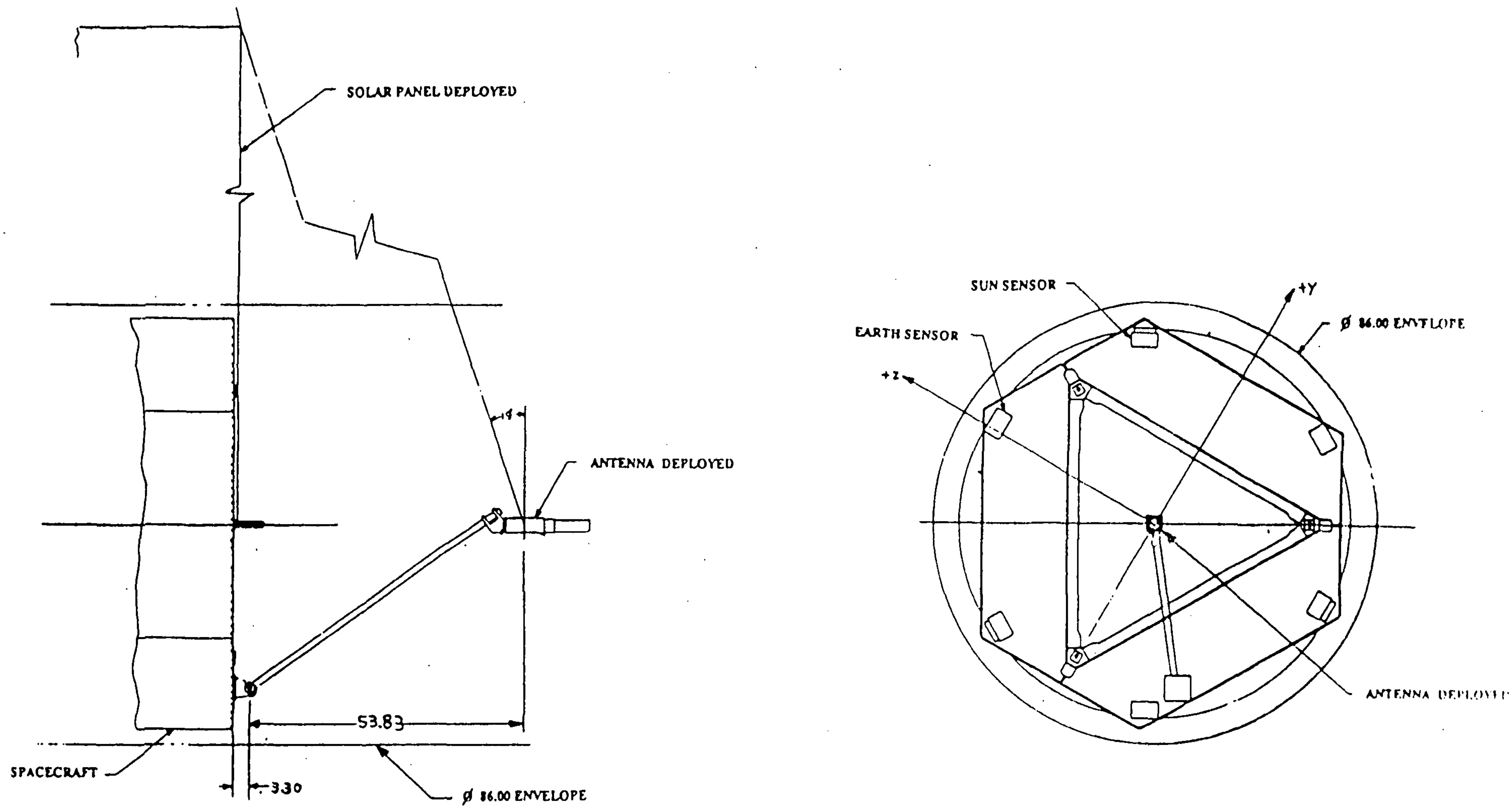
WEIGHT

- o OMNI ANTENNA WEIGHS 5 LBS
- o ALLOTTED WEIGHT FOR DEPLOYMENT MECHANISM IS 10 LBS



STOWED CONFIGURATION

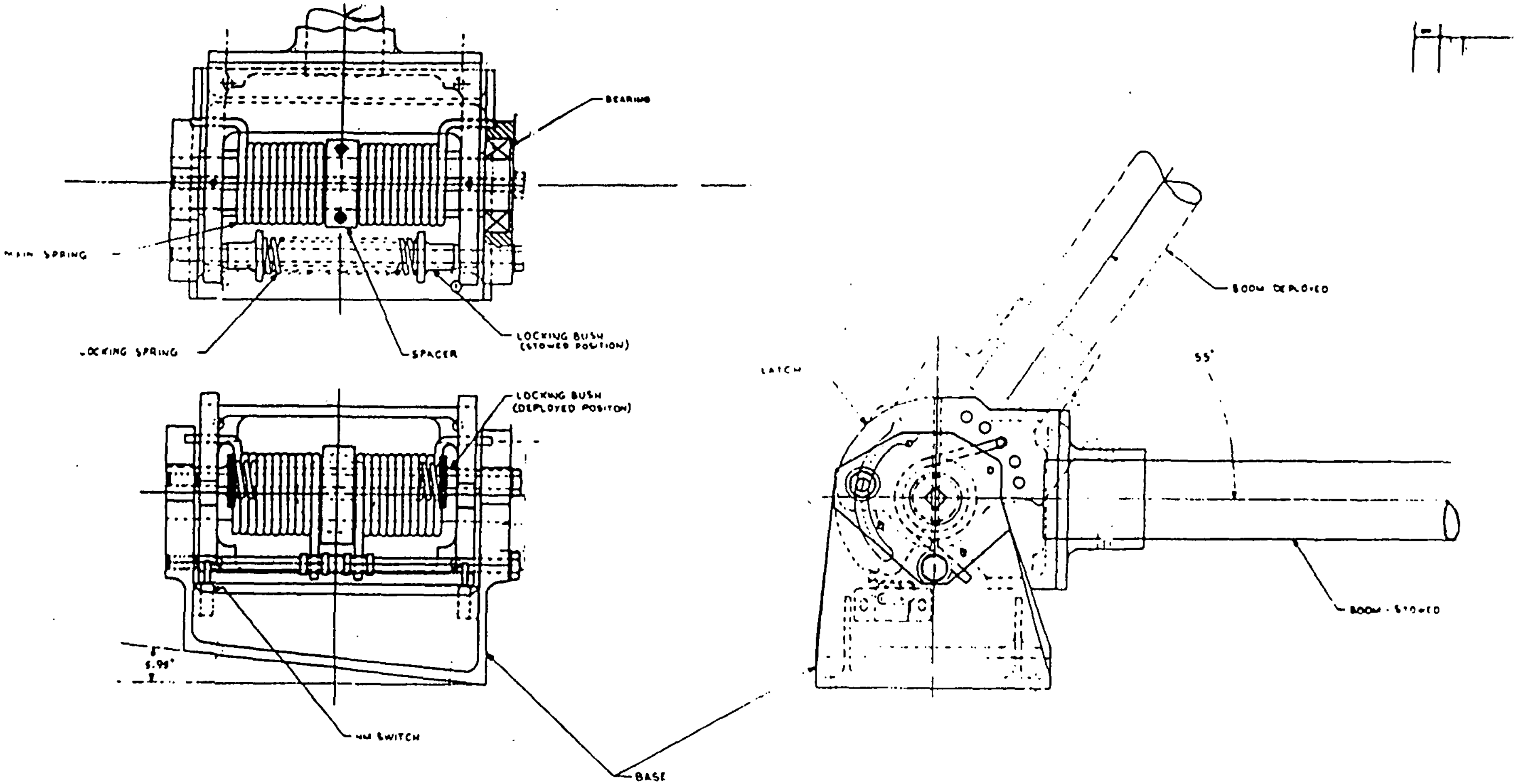
MCP 9



DEPLOYED CONFIGURATION

TRADE-OFFS DURING DEVELOPMENT

- DESIGN WITHOUT DAMPER VS. DESIGN WITH DAMPER
- PIN PULLER VS. CABLE CUTTER
- ANTENNA SAFETY BUMPER VS. ANTENNA SIGNAL GAIN
- BASE HINGE DESIGNS FOR VARIOUS COAXIAL CABLES
- STRAIGHT DEPLOYMENT VS TILT ANGLE DEPLOYMENT



LIST OF MATERIAL

ITEM NO	PART NO	DESCRIPTION	MAT'L

	NAME	INIT	DATE
DESIGNER	KANNAN	KS	3/3/88
DRAWN	KANNAN	KS	3/3/88
CHECKED			
APPROVED			
APPROVED			

UNLESS OTHERWISE SPECIFIED-DIMENSIONS ARE IN INCHES

HINGE, BASE
DEPLOY MECH
OMNI ANTENNA
Δ COBE

SCALE 1/1

GODDARD SPA[CE] FLIGHT CENT[ER]

GE138701

MCP 12

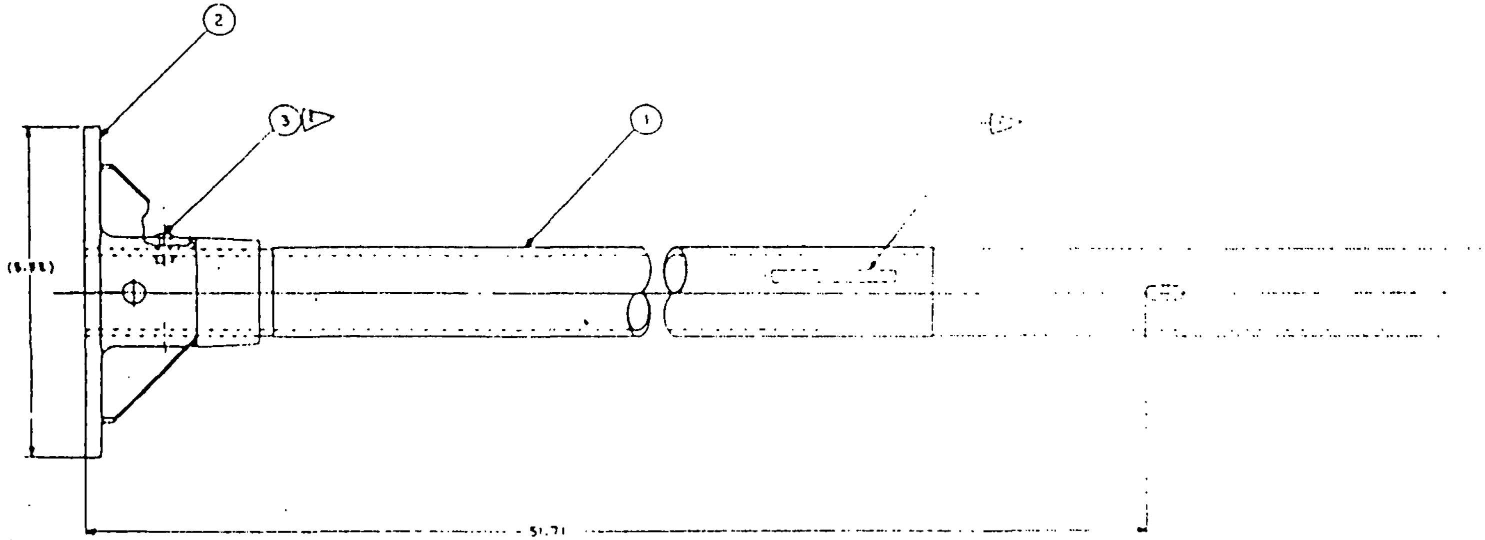
SIGNIFICANT PROBLEMS DURING I&T

- o RIVETING THE BOOM TO THE BASE HINGE BRACKET
- o CRACKING OF THE ALIGNMENT BLOCK
- o INTEGRATING TWO VERY SENSITIVE CO-AXIAL CABLES
- o DEINTEGRATION OF THE BOOM/ANTENNA TO FACILIATE HANDLING SPACECRAFT, GROUND SUPPORT EQUIPMENT INSTALLATION TO & REMOVAL FROM SPACECRAFT
- o ROUTING OF THE THERMISTER WIRES ACROSS BASE HINGE

NOTES: UNLESS OTHERWISE SPECIFIED

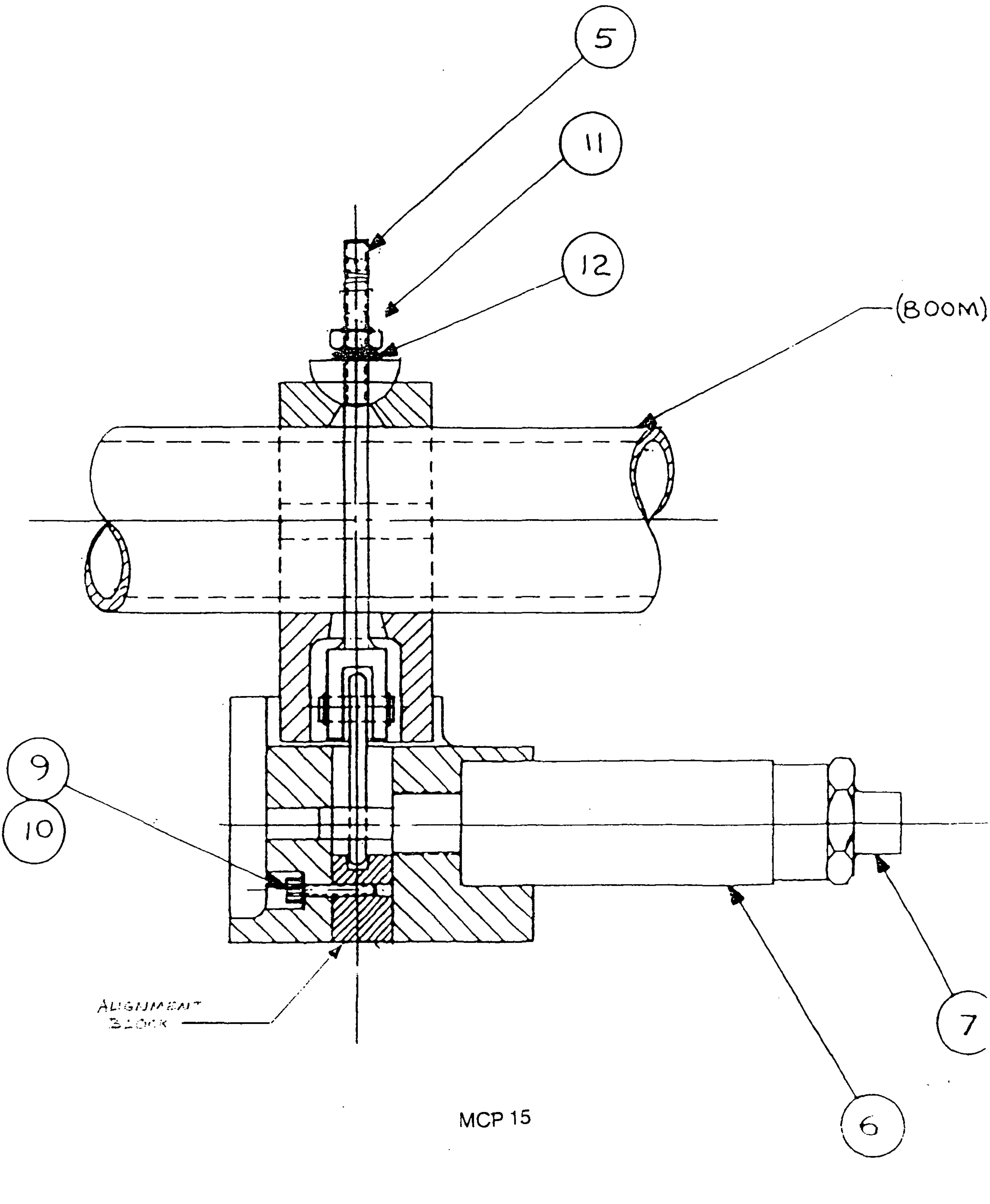
1. [illegible] STAMP PART NO. IN THE APPROX. POSITION SHOWN WITH [illegible] HIGH CHARACTER USING [illegible] SERIES [illegible], COLOR BLACK. SOURCE OF SUPPLY: [illegible] PRODUCTS DEPT [illegible] CA [illegible].

2. INSTALL RIVETS PER MIL-STD-403.



BOOM/BASE HINGE BRACKET ASSEMBLY

OMNI ANTENNA RELEASE MECHANISM



WAIVERS & DEVIATIONS

- NO WAIVER
- NO DEVIATION

ON-ORBIT PERFORMANCE VS. SPECIFIC REQUIREMENTS

- o MISSION OBJECTIVE

  DEPLOYABLE SYSTEM SURVIVED THROUGH LAUNCH ENVIRONMENT

  SUCCESSFULLY DEPLOYED THE OMNI ANTENNA

- o STRUCTURAL DYNAMICS

  STIFF BOOM (>1HZ) TO SATISFY THE ALTITUDE CONTROL SYSTEM

- o THERMAL

  SURVIVE AND OPERATE WITHIN THE TEMPERATURE RANGE

- o DEPLOYMENT TIME AND TELEMETRY

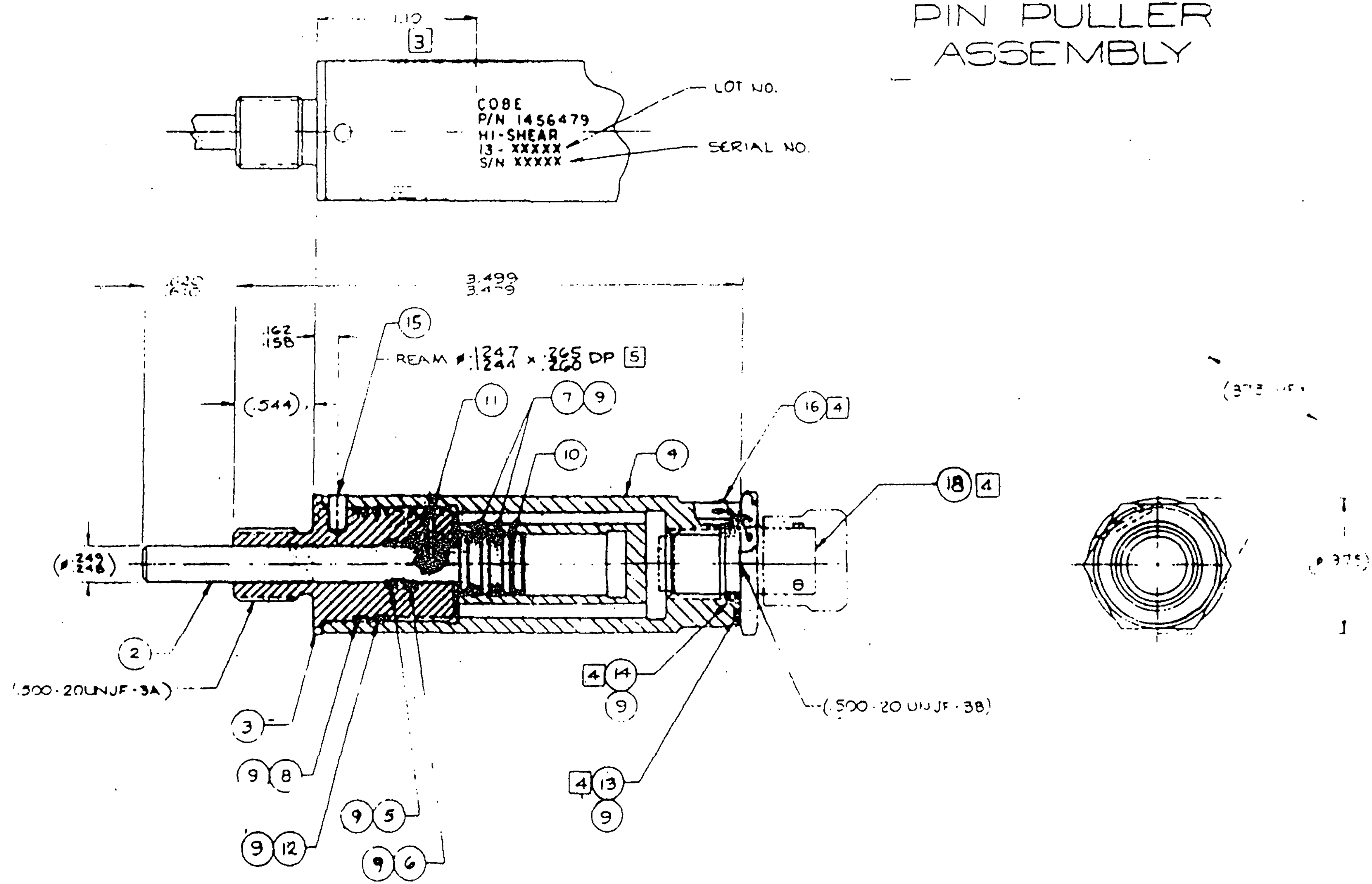
  ACTUAL DEPLOYMENT TIME WAS WITHIN PREDICTED TIME

  ALL TELEMETRY INDICATED AS PREDICTED

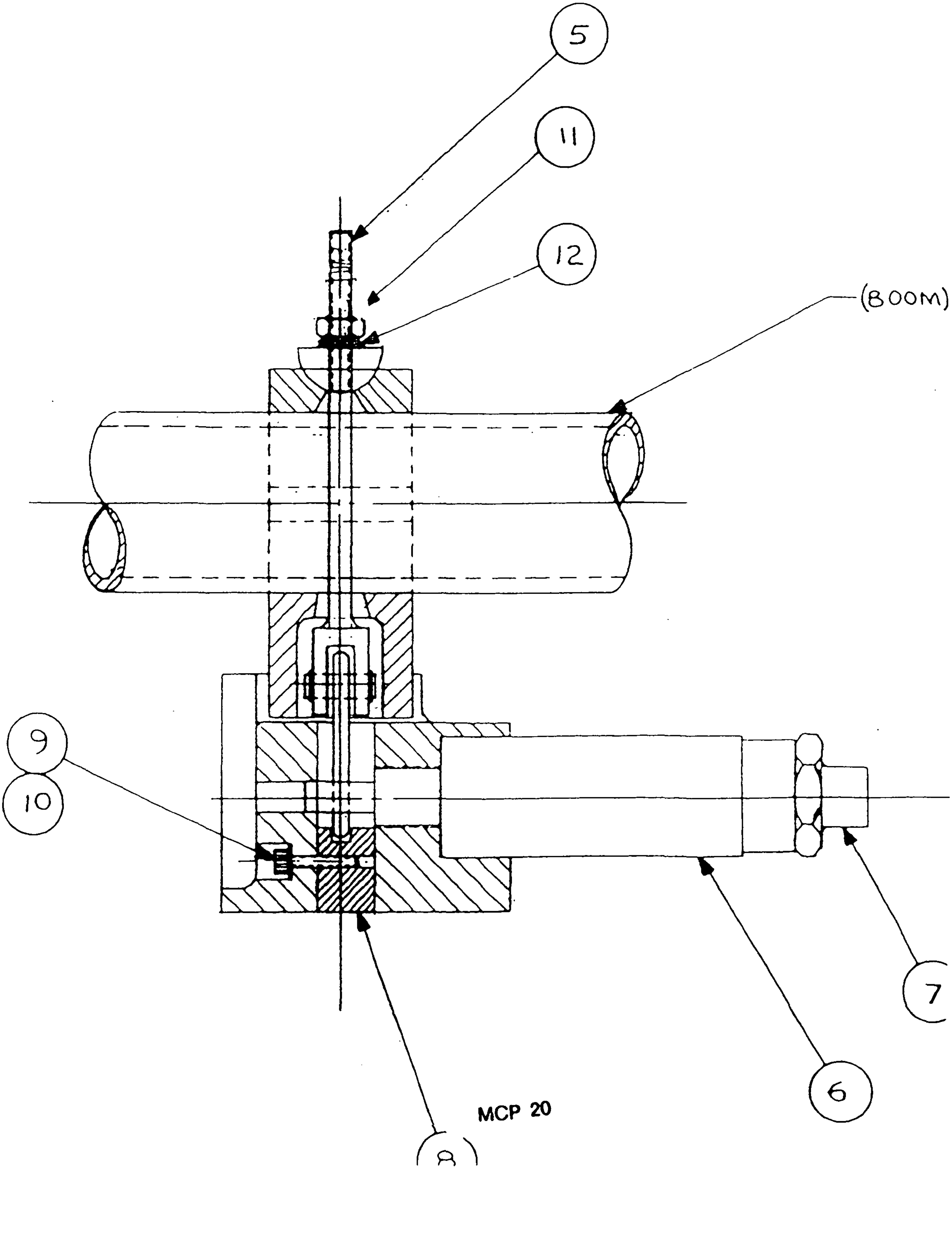
DO DIFFERENTLY

- o REDUCE WEIGHT OF THE OVERALL DEPLOYABLE SYSTEM
- o CHANGE HOUSING & PISTON MATERIAL OF COBE PYROTECHNIC PIN PULLER
- o DESIGN THE RELEASE MECHANISM DIFFERENTLY

PIN PULLER ASSEMBLY

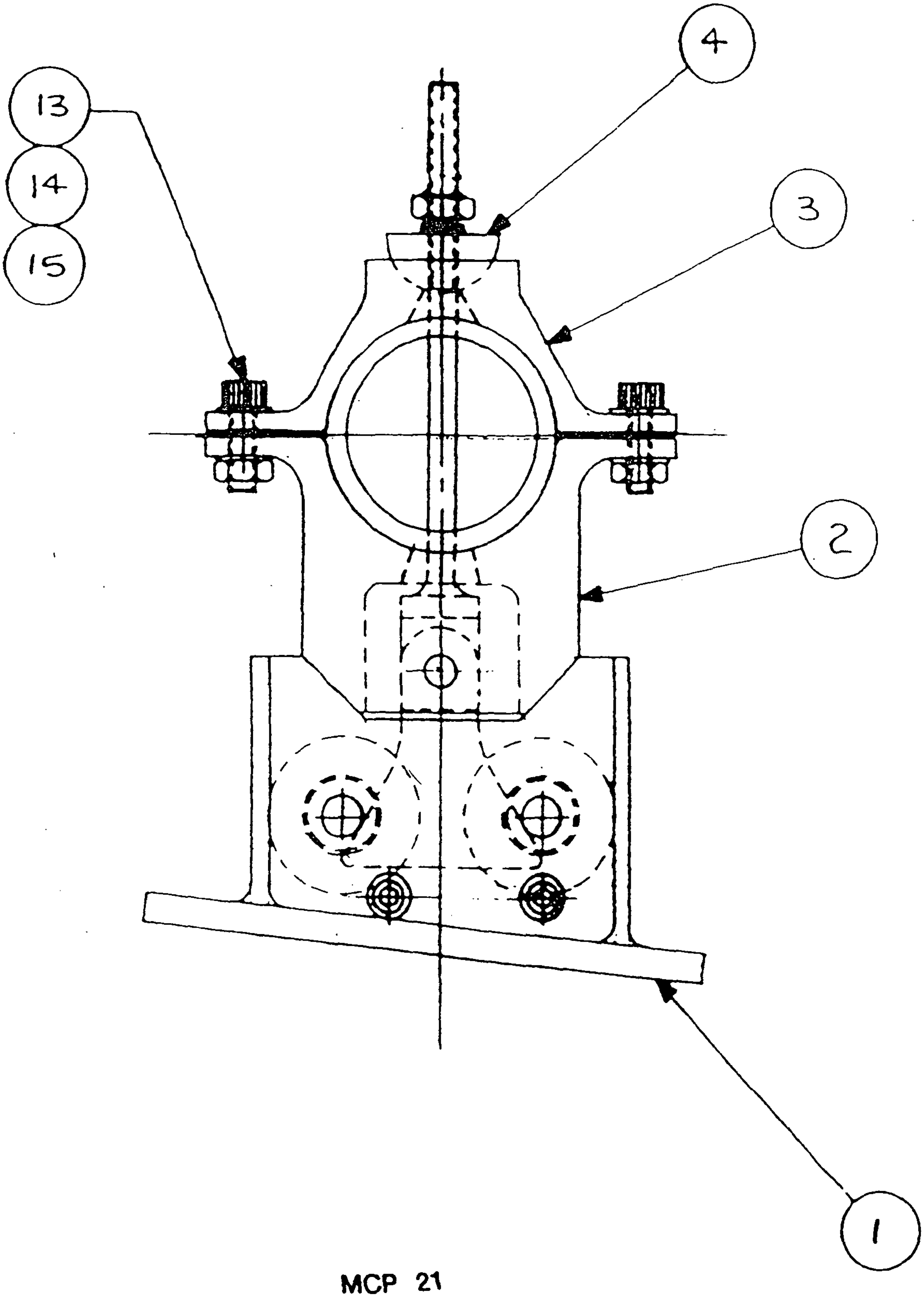


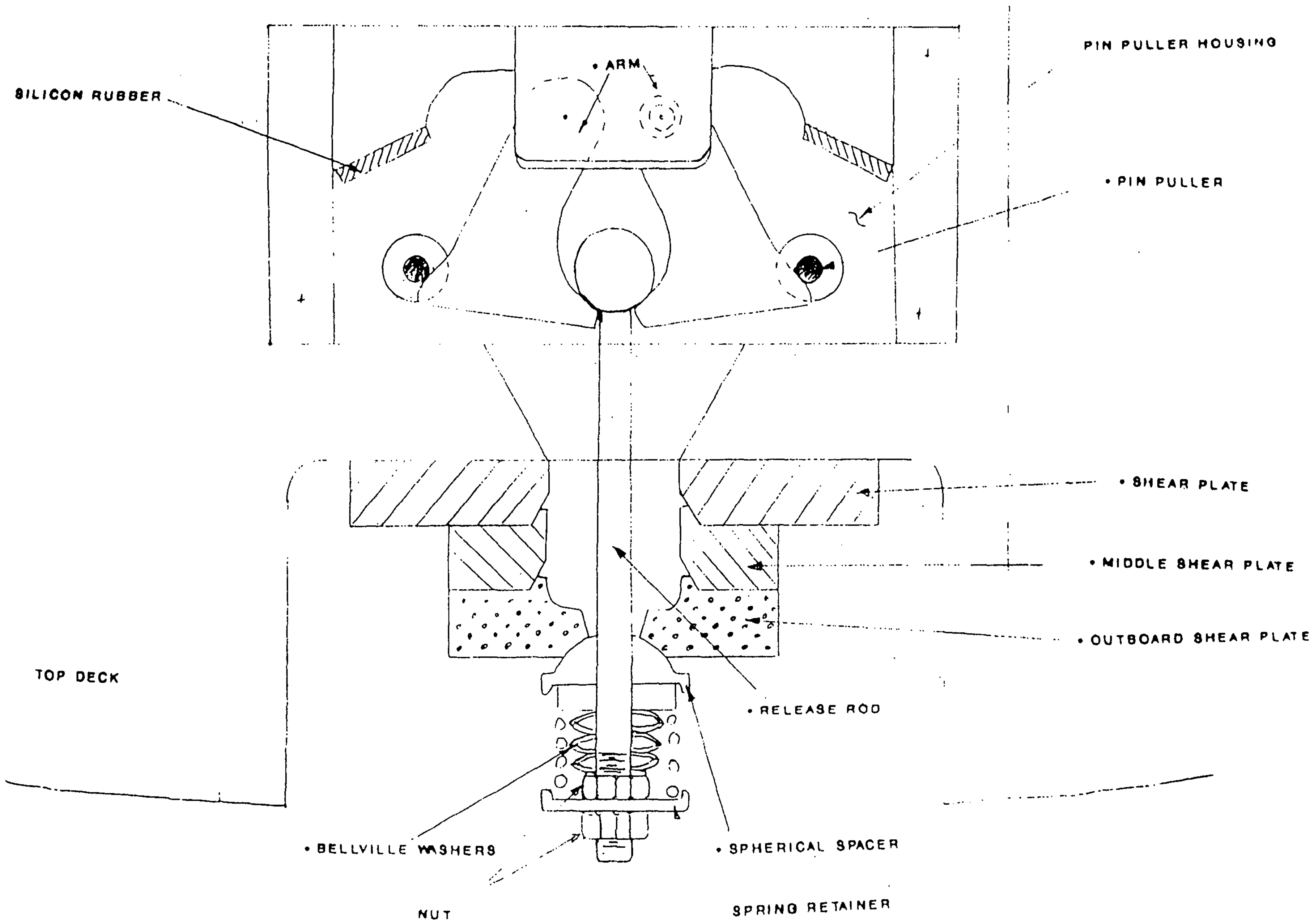
OMN. ANTENNA RELEASE MECHANISM



MCP 20

OMNI ANTENNA RELEASE MECHANISM





• UPPER MIDDLE/OUTBOARD RELEASE MECHANISM

LESSONS LEARNED

- o ALWAYS DESIGN DEPLOYABLE SYSTEM WITH REDUNDANCY PHILOSOPHY
- o BEWARE OF FAILURE MODES OF DEPLOYABLE SYSTEM
- o ALWAYS DESIGN SYSTEM/COMPONENT WITH THE CAPABILITY OF FUTURE ADJUSTMENT
- o CAREFULLY INSPECT ALL COMPONENTS OF DEPLOYABLE SYSTEM BEFORE AND AFTER A TEST
- o X-RAY OF PYROTECHNIC PIN PULLER SHALL BE TAKEN FOR 2 ORTHOGONAL VIEWS SHALL BE CAREFULLY INSPECTED
- o ALWAYS PERFORM A CAREFUL INSPECTION OF THE FINAL FLIGHT CONFIGURATION OF THE DEPLOYABLE SYSTEM AND THE SURROUNDING AREA FOR THE POSSIBILITY OF SNAGGING UP DURING DEPLOYMENT
- o LEARN THE DIFFERENT PHASES, REQUIREMENTS, AND INVOLVEMENTS OF BUILDING A FLIGHT DEPLOYABLE SYSTEM AND A SPACECRAFT

COSMIC BACKGROUND EXPLORER

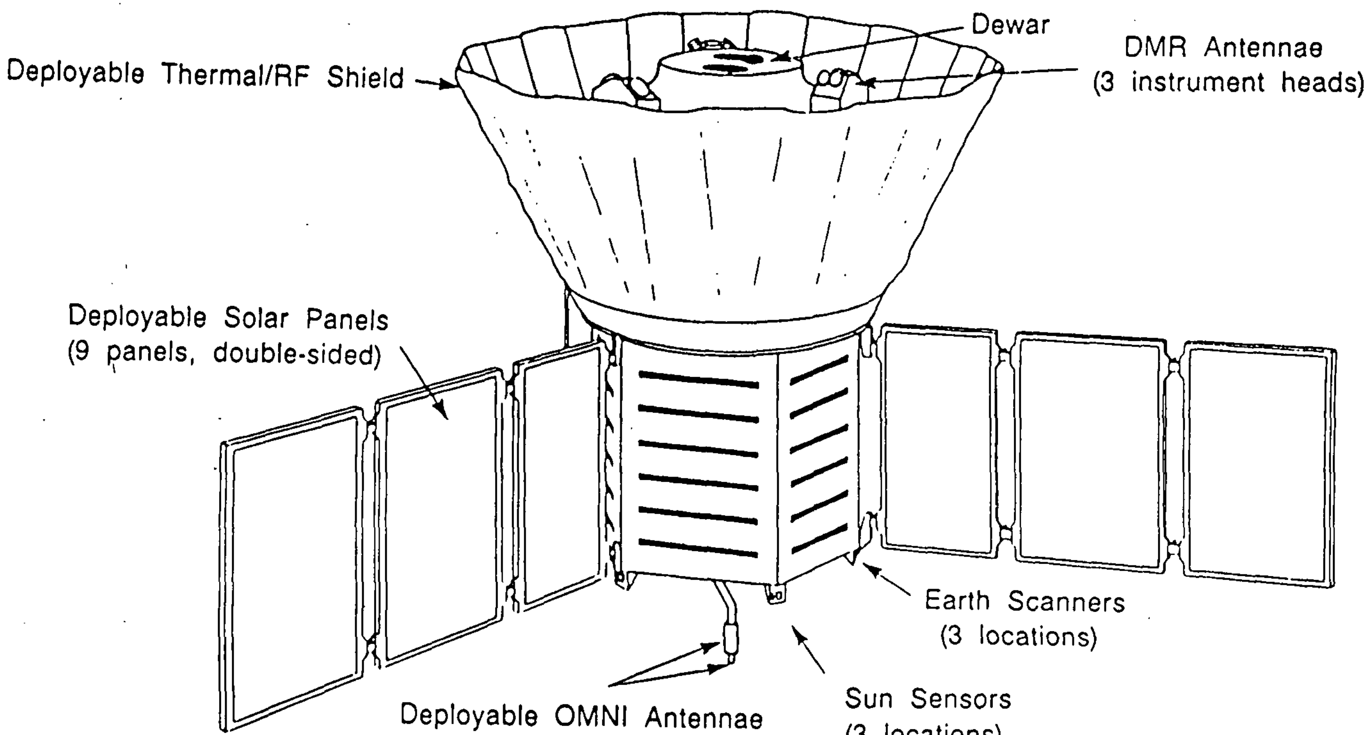
THERMAL SUBSYSTEM ON-ORBIT PERFORMANCE REPORT

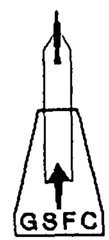
R. A. Chalmers

March 7-8, 1990

COSMIC BACKGROUND EXPLORER

MISSION ORBIT CONFIGURATION





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THERMAL SUBSYSTEM
TOP-LEVEL REQUIREMENTS

- maintain dewar mainshell (exterior) temperature below 150 K during mission operations
- meet emittance and temperature requirements on interior of thermal shield (e < 0.07, t < 240 K)
- maintain spacecraft components within their survival range (typically -25 to 50 C) at all times
- maintain spacecraft components within their operating temperature range (typically 0 to 40 C) whenever performance is required
- minimize heater power requirements, especially during the shadow season
- provide capability to safely dissipate excess power while spacecraft is attached to launch vehicle

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THERMAL SUBSYSTEM
MISSION ENVIRONMENT

spacecraft spin and pointing requirements ensure a near-constant thermal environment:

- .815 rpm body spin rate
- angle between spin axis and sun maintained at 94°
- 16 minute maximum shadow period

THERMAL SUBSYSTEM
DESIGN FEATURES

- thermal subsystem is designed to operate autonomously during mission mode; monitoring and intervention through the POCC are required during initial checkout period

- passive elements:
    - optimized exterior coating patterns
    - mli blankets
    - thermal isolators
    - aluminum heat sink plates
    - thermally-conductive interface materials

- active elements:
    - shadow season heaters
    - instrument make-up heaters
    - special function heaters

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THERMAL SUBSYSTEM
DESIGN QUALIFICATION

ANALYSES

- component-level thermal analyses and/or design reviews were conducted for all flight electronics
- assembly-level thermal analyses were performed for the OMNI antenna, shunt dissipator panels, thermal shield and retention cable, solar arrays, and dewar ejectable cover
- observatory-level thermal analyses were performed for hot/cold mission orbit, prelaunch, and ascent conditions
- observatory thermal math models were correlated with thermal balance test results

THERMAL SUBSYSTEM
DESIGN QUALIFICATION

TESTING

- all electronics underwent at least four thermal cycles at temperature levels 10 C in excess of predicted operating extremes
- the solar array panels, shunt dissipator assemblies, antennae, thermal shield, and deployment mechanisms underwent additional thermal vacuum testing
- the flight observatory was subjected to eight thermal vacuum soaks (4 hot/4 cold)

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THERMAL SUBSYSTEM
ON-ORBIT PERFORMANCE

- no violations of qual temperature limits have been seen to date and none are expected during a nominal mission
- one non-critical temperature sensor was lost on the thermal shield during deployment, possibly due to pyro shock
- temperature readings for most spacecraft components are within 2-3 C of BOL predicts. due to the degradation of thermal control coatings, temperatures next winter will be somewhat higher.
- initial cooldown of dewar mainshell was more rapid than expected due to specularity of thermal shield
- thermal shield is performing better than expected with very stable inner surface temperatures of 180 K or lower (spec is 220 K)

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THERMAL SUBSYSTEM
ON-ORBIT PERFORMANCE

FLIGHT TEMPERATURES VS. QUALIFICATION LIMITS

SUBSYSTEM	COMPONENT	TEMPERATURE (C) 1ST HOT SEASON	QUAL LIMITS
ACS	ACS Power Supplies	20	-10 to 50
ACS	Attitude Control Electronics	17	-10 to 50
ACS	Digital Sun Sensors	32	-30 to 60
ACS	Earth Scanner Electronics	21	-25 to 50
ACS	Earth Scanners (avg of 2 thermistors)	36	-25 to 50
ACS	Magnetometer Electronics	19	-25 to 65
ACS	MMA Switching Unit	17	-10 to 50
ACS	Momentum Management Assembly	21	-10 to 50
ACS	Momentum Wheel Assembly 1	21	-10 to 43
ACS	Momentum Wheel Assembly 2	16	-15 to 43
ACS	MWEA 1	20	-10 to 50
ACS	MWEA 2	21	-10 to 50
ACS	MWEA Relay Unit	17	-10 to 50
ACS	Reaction Wheel Drive Electronics	23	-10 to 50
ACS	Reaction Wheels	23	-10 to 50
ACS	RMAs (gyros)	21	-10 to 50
ACS	Sun Presence Sensors (+X)	31	-50 to 50
ACS	Sun Presence Sensors (-X)	(-20)	-50 to 50
ACS	Sun Sensor Electronics	18	-20 to 60
ACS	Three-Axis Magnetometers	22	-25 to 65
ACS	Torquer Bars	25	-10 to 50
C&DH	Central Command Unit 1	18	-15 to 50
C&DH	Central Command Unit 2	16	-15 to 50
C&DH	Central Telemetry Unit 1	19	-15 to 50
C&DH	Central Telemetry Unit 2	16	-15 to 50
C&DH	Instrument Command Unit 1	17	-15 to 50
C&DH	Instrument Command Unit 2	19	-15 to 50
C&DH	Instrument Telemetry Unit 1	18	-15 to 50
C&DH	Instrument Telemetry Unit 2	19	-15 to 50
C&DH	Spacecraft Command Unit 1	21	-15 to 50
C&DH	Spacecraft Command Unit 2	15	-15 to 50
C&DH	Spacecraft Telemetry Unit 1	18	-15 to 50
C&DH	Spacecraft Telemetry Unit 2	17	-15 to 50
C&DH	Stable Oscillator 1	18	-15 to 50
C&DH	Stable Oscillator 2	18	-15 to 50
C&DH	Tape Recorder 1	16	-10 to 45
C&DH	Tape Recorder 2	18	-10 to 45
COMM	Band Reject Filter	(20)	-10 to 55
COMM	Diplexers	20	-20 to 60
COMM	Navigation Oscillator (USO)	19	-20 to 55
COMM	OMNI Antenna (TDRSS/GSTDN)	35	-50 to 80
COMM	RF Transfer Switches	(20)	-20 to 60
COMM	Transponder 1	21	-10 to 55
COMM	Transponder 2	23	-10 to 55
DEWAR	Dewar Mainshell	-138	-138 to 25
DEWAR	Valve Drive Electronics	8	-25 to 50

() temperature inferred using correlated thermal math model

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THERMAL SUBSYSTEM
ON-ORBIT PERFORMANCE

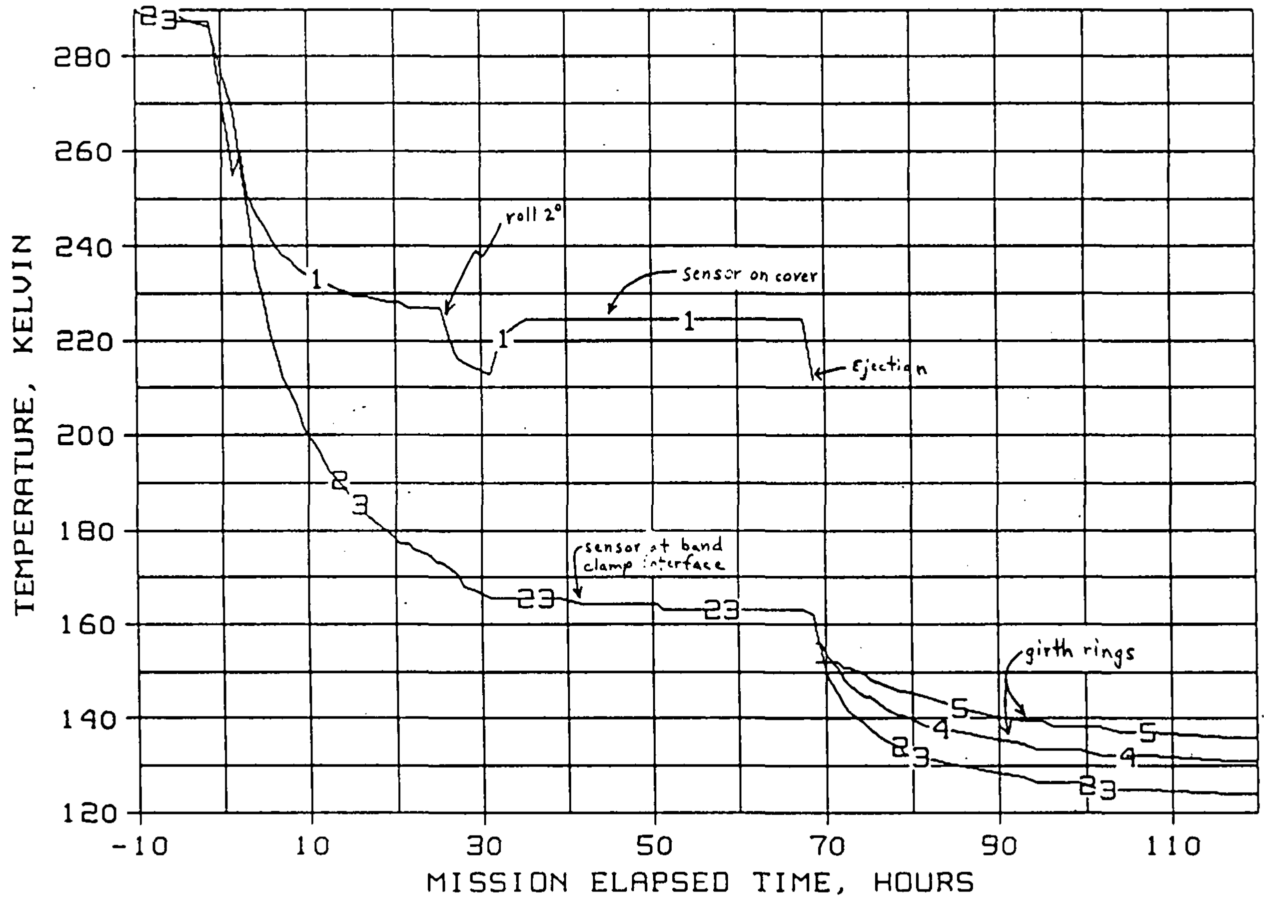
FLIGHT TEMPERATURES VS. QUALIFICATION LIMITS
(continued)

		TEMPERATURE (C)	
SUBSYSTEM	COMPONENT	1ST HOT SEASON	QUAL LIMITS
DIRBE	DIRBE Analog Unit	24	-10 to 50
DIRBE	DIRBE Digital Unit	17	-10 to 50
DIRBE	DIRBE IPDU	20	-10 to 50
DIRBE	DIRBE Preamp	23	-20 to 50
DMR	Data Electronics Units	25	0 to 50
DMR	DMR IPDU 1	20	-10 to 50
DMR	DMR IPDU 2	19	-10 to 50
ELECT	Signal Conditioning Unit	17	-10 to 50
FIRAS	FIRAS Fiberoptics Preamp	12	-20 to 30
FIRAS	FIRAS IPDU	26	-10 to 50
FIRAS	FIRAS Main Electronics Unit	21	-10 to 50
FIRAS	FIRAS Preamplifier	21	-10 to 50
FIRAS	MTM Electronics Unit	18	-10 to 40
POWER	20 AH Batteries	15	-10 to 25
POWER	Power Supply Electronics	21	-25 to 50
POWER	Shunt Dissipator Panels	135 (hottest)	-60 to 160
POWER	Solar Array Panels	30 (hottest)	-65 to 50
THERMAL	Thermal/RF Shield Interior Blankets	-100	<-50 (220 K)

() temperature inferred using correlated thermal math model

THERMAL SUBSYSTEM
ON-ORBIT PERFORMANCE

COOLDOWN OF COBE DEWAR MAINSHELL AND EJECTABLE COVER
(FROM FLIGHT DATA)



1 - WACMBCT
2 - TCLAMPS3
3 - TCLAMPYG
4 - WUGR90GT
5 - WLGR90GT

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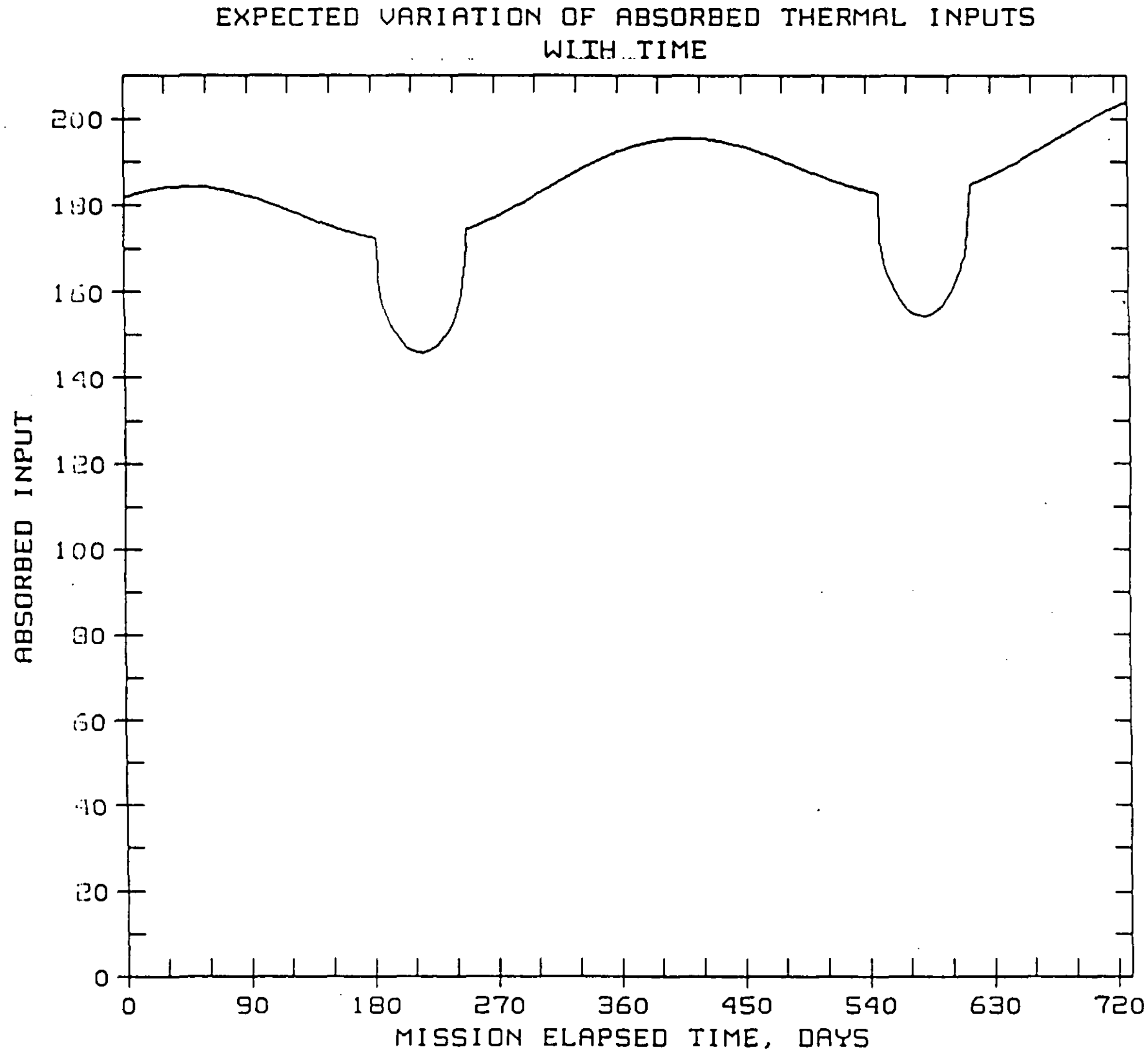
THERMAL SUBSYSTEM

ON-ORBIT PERFORMANCE



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THERMAL SUBSYSTEM
ON-ORBIT PERFORMANCE

MAXIMUM/MINIMUM EXPECTED FLIGHT TEMPERATURES
(not correlated with flight data)

SUBSYSTEM	COMPONENT	TEMPERATURES (C) FULL SUN	SHADOW	QUAL LIMITS
ACS	ACS Power Supplies	26	2	-10 to 50
ACS	Attitude Control Electronics	22	2(10)	-10 to 50
ACS	Digital Sun Sensors	18	-18	-30 to 60
ACS	Earth Scanner Electronics	31	9	-25 to 50
ACS	Earth Scanners	25	-5(5)	-25 to 50
ACS	Magnetometer Electronics	23	2	-25 to 65
ACS	MMA Switching Unit	21	3	-10 to 50
ACS	Momentum Management Assembly	28	6	-10 to 50
ACS	Momentum Wheel Assembly 1	25	5	-10 to 43
ACS	Momentum Wheel Assembly 2	20	-2	-15 to 43
ACS	MWEA 1	23	2	-10 to 50
ACS	MWEA 2	27	7	-10 to 50
ACS	MWEA Relay Unit	21	3	-10 to 50
ACS	Reaction Wheel Drive Electronics	29	7	-10 to 50
ACS	Reaction Wheels	27	5	-10 to 50
ACS	RMAs (gyros)	27	5	-10 to 50
ACS	Sun Presence Sensors (+X)	17	-4	-50 to 50
ACS	Sun Presence Sensors (-X)	-13(4)	-40(4)	-50 to 50
ACS	Sun Sensor Electronics	21	0	-20 to 60
ACS	Three-Axis Magnetometers	35(8)	8(8)	-25 to 65
ACS	Torquer Bars	24	1	-10 to 50
C&DH	Central Command Unit 1	26	5	-15 to 50
C&DH	Central Command Unit 2	21	3	-15 to 50
C&DH	Central Telemetry Unit 1	27	6	-15 to 50
C&DH	Central Telemetry Unit 2	21	1	-15 to 50
C&DH	Instrument Command Unit 1	22	1(10)	-15 to 50
C&DH	Instrument Command Unit 2	22	1	-15 to 50
C&DH	Instrument Telemetry Unit 1	26	5	-15 to 50
C&DH	Instrument Telemetry Unit 2	24	4	-15 to 50
C&DH	Spacecraft Command Unit 1	24	4	-15 to 50
C&DH	Spacecraft Command Unit 2	24	3	-15 to 50
C&DH	Spacecraft Telemetry Unit 1	28	9	-15 to 50
C&DH	Spacecraft Telemetry Unit 2	22	2	-15 to 50
C&DH	Stable Oscillator 1	30	11	-15 to 50
C&DH	Stable Oscillator 2	25	2	-15 to 50
C&DH	Tape Recorder 1	20	1	-10 to 45
C&DH	Tape Recorder 2	26	4	-10 to 45
COMM	Band Reject Filter	22	-1	-10 to 55
COMM	Diplexers	25	-1	-20 to 60
COMM	Navigation Oscillator (USO)	25	4	-20 to 55
COMM	OMNI Antenna (TDRSS/GSTDN)	46	-22	-35 to 65
COMM	RF Transfer Switches	21	-2	-20 to 60
COMM	Transponder 1	28	6	-10 to 55
COMM	Transponder 2	34	5	-10 to 55
DEWAR	Dewar Mainshell	-132	-120	-138 to 25
DEWAR	Valve Drive Electronics	6	-9(5)	-25 to 50

() assumed heater power in Watts

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THERMAL SUBSYSTEM
ON-ORBIT PERFORMANCE

MAXIMUM/MINIMUM EXPECTED FLIGHT TEMPERATURES
(continued)

SUBSYSTEM	COMPONENT	TEMPERATURES (C) FULL SUN	SHADOW	QUAL LIMITS
DIRBE	DIRBE Analog Unit	23	3(4)	-10 to 50
DIRBE	DIRBE Digital Unit	20	1(10)	-10 to 50
DIRBE	DIRBE IPDU	27	7	-10 to 50
DIRBE	DIRBE Preamp	16	4	-20 to 50
DMR	Data Electronics Units	25	25	0 to 50
DMR	DMR IPDU 1	27	5	-10 to 50
DMR	DMR IPDU 2	23	2(4)	-10 to 50
ELECT	Signal Conditioning Unit	24	4	-10 to 50
FIRAS	FIRAS Fiberoptics Preamp	22(2)	17(4)	-20 to 30
FIRAS	FIRAS IPDU	27	5	-10 to 50
FIRAS	FIRAS Main Electronics Unit	28	8	-10 to 50
FIRAS	FIRAS Preamplifier	22	7	-10 to 50
FIRAS	MTM Electronics Unit	25	20(52)	-10 to 40
POWER	20 AH Batteries	18	-2	-10 to 25
POWER	Power Supply Electronics	25	5	-25 to 50
POWER	Shunt Dissipator Panels	143	-45	-60 to 160
POWER	Solar Array Panels	35	-50	-65 to 50
THERMAL	Thermal/RF Shield Interior Blankets	-50	-70	<-50 (220 K)

() assumed heater power in Watts

THERMAL SUBSYSTEM
TRADE-OFFS

- weight constraints (20 mil honeycomb facesheets, placement of components primarily determined by harness weight considerations)
- power limitations would not permit a design that required a large amount of power during the shadow season; this constraint forced a great deal of analytical 'fine-tuning' to optimize the exterior coating design.
- active temperature control for magnetometers was ruled out due to incompatibility with thermostats (steel); this constraint, aggravated by schedule pressures, resulted in a thermal design that is not fully redundant.

THERMAL SUBSYSTEM
TRADE-OFFS

- concerns over electrostatic discharge problems originally impacted choice of exterior thermal control materials; this constraint was partially removed as a result of overriding concerns (thermal inputs to FIRAS external calibrator). it would be interesting to determine whether or not any on-orbit discharge events can be detected during the shadow season.

- high cost and complexity led to a decision not to fully simulate environmental heat inputs on the thermal shield during observatory TV/TB testing. test results were adjusted using analytical models to compensate for the missing heat inputs.

THERMAL SUBSYSTEM
TRADE-OFFS

- performing thermal analysis of the instrument module was accomplished using programs that do not account for specular reflections. the actual couplings to space are higher than these programs predicted, but a program to correctly model specular surfaces was not available. as a result, the dewar mainshell and thermal shield temperatures are colder than predicted.

- choice of thermal control coatings greatly limited by non-thermal issues such as contamination (shedding), adhesion problems, and electrostatic charging concerns.

- long procurement lead-times for flight quality heaters and thermostats made it necessary to use COBE/STS hardware. in order to achieve the necessary power levels, several heater circuits were constructed by combining the elements into series and parallel arrangements.

THERMAL SUBSYSTEM
TECHNICAL CHALLENGES

- determination of incident and absorbed environmental fluxes was a difficult and unwieldy process due to COBE's unusual mission orbit attitude
- thermal analysis of cryogenic instruments never before performed by Code 732
- specular analyses performed to determine reflections of solar array panels onto cowling
- large temperature differences between spacecraft and instrument module components required careful modeling of cabling and other frequently-ignored heat paths

THERMAL SUBSYSTEM
PROBLEMS ENCOUNTERED DURING I&T

- presence of thermostats made it very difficult to check proper operation of heater circuits
- accurate calibration of PRTs not possible due to inadequate characterization of SCU sensor conditioning circuitry
- late discovery of heat inputs to FIRAS external calibrator from thermal shield and DMR heads led to major rework; these inputs should have been considered much earlier
- last-minute design and installation of earth scanner RFI shields precluded thorough analysis. in addition, the flight RFI shields were irridited instead of being left bare. combined, these two factors are largely responsible for warmer-than-expected earth scanner temperatures.

THERMAL SUBSYSTEM
LESSONS LEARNED

- confusion appeared to exist for a prolonged period regarding the definition of design, qual, and operating temperatures. as a consequence, some contractor-furnished components were designed and tested to the operating limits.
- present in-house capabilities to analyze specular surfaces are extremely limited
- thermal design of cryogenic instruments requires careful attention to every detail
- thermal analyses of contractor-supplied boxes were frequently inadequate. in retrospect, it would have been preferable to make the thermal analysis deliverable to Goddard at the time of each component's CDR and to make review and acceptance of the analysis by Code 732 one of the CDR action items.

THERMAL SUBSYSTEM
LESSONS LEARNED

- the power dissipations of several components were not measured until observatory-level thermal vacuum testing. it is recommended that a requirement to measure power dissipation be included in all box-level acceptance test procedures.
- the dewar ejectable cover and clamp band should have been more heavily instrumented with thermocouples during the observatory TV/TB test. this information would have been helpful in assessing the flight data prior to cover deployment.
- a method for easily verifing the operation of heater circuits should have been developed prior to TV/TB testing.

COBE POWER SUBSYSTEM

POWER SUBSYSTEM TEAM

Joel Jermakian	-	Subsystem Engineer, Solar Array Engineer
Dominic Manzer	-	Electronics Engineer & much more
Sid Tiller	-	Battery Manager
Dave Sullivan	-	Battery Test Engineer
Nick Mejia	-	Solar Array Technician

COBE POWER SUBSYSTEM

SYSTEM OVERVIEW:

Specification Requirements:

Direct Energy Transfer (DET) System:

- 3 Deployable Solar Array wings, each consisting of 3 panels having solar cell on both sides.
- 2 NiCd Modified NASA Standard 20 Amp-Hour Batteries (18 instead of 22 cells each)
- Power Supply Electronics which provides power management, bus regulation, and primary power distribution functions.
- Shunt Dissipator Panels.

System Load Capability:

100% Sun	920 W (EOL)
Max Eclipse	712 W (EOL)
Pre-Array Deployment	400 W (BOL)

COBE POWER SUBSYSTEM

SHUNT DISSIPATORS:

Description:

- 6 Aluminum panels, with Kapton film heaters bonded on exterior surface.
- Shunt elements use cancellation to approach zero net dipole magnetic moment.
- Large weight savings versus STS shunts.

Performance:

- No shunt failures detected at present.
- Shunt temperatures at or below thermal predictions (max. recorded shunt temperature 130°C).

COBE POWER SUBSYSTEM

SOLAR ARRAY:

Specification Requirements:

Performance:

- Support 1-year mission life.
- 1000 W, End Of Life, Summer Solstice, spin average array output at 28.5 volts.
- Meet magnetic cleanliness requirements of ACS and instruments.

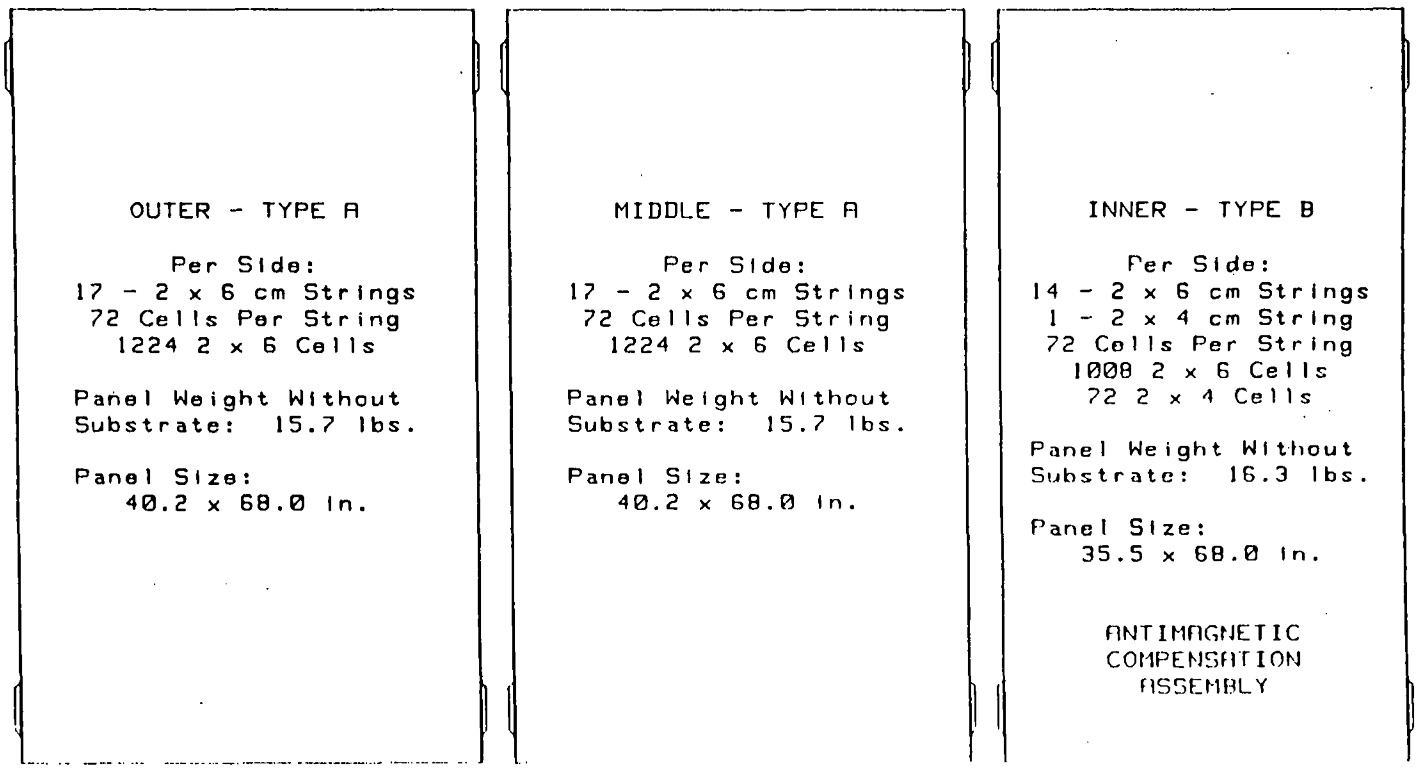
Derived and Other:

- Utilize remaining STS COBE solar cells (~11000 cells or 53% of total cells in array).
- Move array blocking diodes from box internal to spacecraft onto the array.
- No single point failure to impact mission.
- Compatible with impact of changing to deployable system.

COBE POWER SUBSYSTEM

SOLAR ARRAY cont'd:

COBE SOLAR ARRAY WING



COBE POWER SUBSYSTEM

SOLAR ARRAY cont'd:

Trade-offs required during development:

Magnetic Compensation

- STS solar array incorporated anti-magnetic returns for all strings directly underneath cells. Analysis showed that in new array this heavy, costly and time consuming compensation was only needed on inboard panels. Other panels' magnetic characteristics were improved by optimizing layout.

Bypass Diodes:

- New array experiences extensive shadowing of strings. A shadow study of entire array showed that with proper layout most strings could avoid shadow situations which require bypass diodes. This saved weight, cost, and schedule and increased array output.

Position Telemetry

- In order to determine possible corrective action a lockout indicator was required on the array hingelines. Historically, a microswitch has been used. A simple circuit was designed which using only a passive analog channel reads out position of the array as well as giving a positive lockout indication.

COBE POWER SUBSYSTEM

SOLAR ARRAY cont'd:

Significant Problems During I & T:

Handling:

- Due to time, area, weight, (inexperience?) little attention was given to handling of panels and array system during design phase. Even though spacecraft technicians (mechanical and electrical) did a super job some damage to array occurred and had to be repaired.
- Array inter-panel harness was essentially built in-place and installed with great difficulty while deployed array was on spacecraft.
- Facility at VAFB should be carefully examined prior to next Goddard launch from there. Significant hardware risks could be removed if minor measures are taken, eg. all cranes work, access to S/C labs are improved (other than outside or through computer/operations room.

Waivers/Deviations:

- Laydown of one portion of one panel was reversed by Solarex due to operator error. No impact to mission.

COBE POWER SUBSYSTEM

SOLAR ARRAY cont'd:

On-orbit Performance:

Electrical:

- Array exceeds original predicts by 7%, latest prelaunch estimates by 2%.
- Current variation due to spacecraft rotation and roll angle is identical to analytical model.
- No anomalous degradation in performance.
- South Pole has a high albedo.

Other:

- One microswitch did not trip to indicate a locked out hinge. Fortunately, we know the position of the panel to be 100% deployed.

COBE POWER SUBSYSTEM

SOLAR ARRAY cont'd:

Things To Do Differently Next Time:

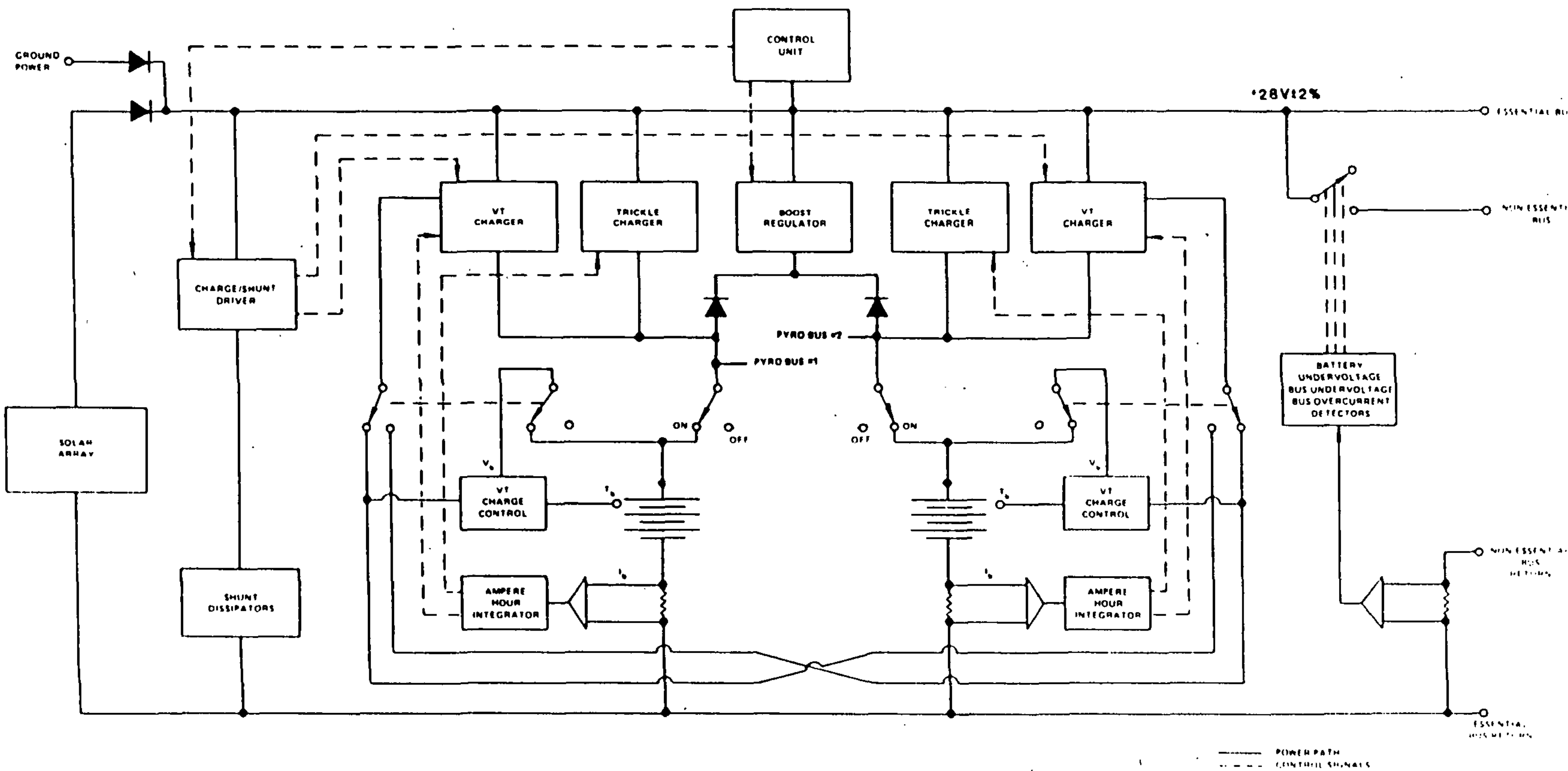
- Use a full scale mockup for harness build.
- Give more attention to handling and integration of panels.

COBE Power System Electronics (PSE)

Requirements:

Bus Voltage	+28 Volts DC +-2%
Output Power	700 Watts, Orbital Average @ 500 Km
Output Ripple	200 Millivolts P-P, 1 KHz to 100 KHz 500 Millivolts P-P in Dead Band
Output Impedance	0.01 Ohm 1 Hz to 1 KHz, increasing to 0.1 Ohm at 10 KHz, 0.1 to 100 KHz
Bus Voltage Transient	26.88 Volts for less than 0.5 mS
Fault Protection	Nonessential load removal should fault be detected Detectors disabled via Ground Command
Bus UV	Trip point; 26.3 Volts, Time Constant 20 mS
NEBus OC	Trip point; 40 Amps, Time Delay 100 mS
Battery UV	Trip Point; Half Bat. 9.0 Volts, Time Delay 100 mS
Pyro Bus	+22 +-5 volts, 40 Amp
Instrumentation	Bus and Battery voltages, PSE and Bat. temp. Bus, S.A., Bat., and Shunt Currents Ampere-Hour Integrator Output

COBE POWER SUBSYSTEM BLOCK DIAGRAM



DM 2

COBE Power System Electronics (PSE)

Tradeoffs:

(1) Shunt Capacity

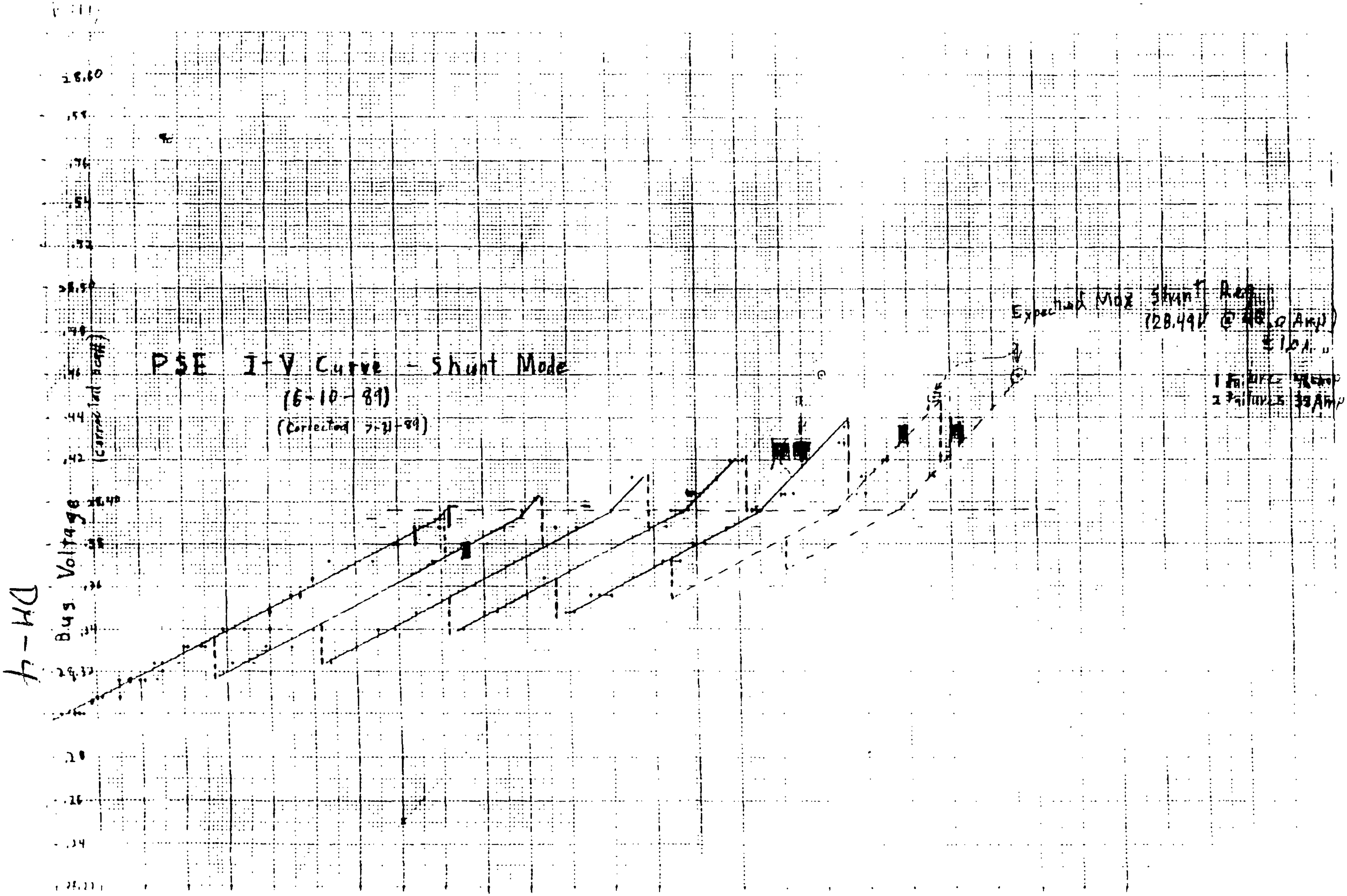
Requirements change:	Out put from S.A. increased from 36 Amps to 45 Amps Peak..
Constraints:	2% Bus regulation, stability, number of shunt driver stages, thermal dissipation limits of shunt transistors
Solution:	Change operating mode of 6 drivers from linear to non-linear

(2) Power Distribution Fusing

Power distribution PC Board connected redundant fusses to all parallel wires causing conflict between wire derating and fuse derating.

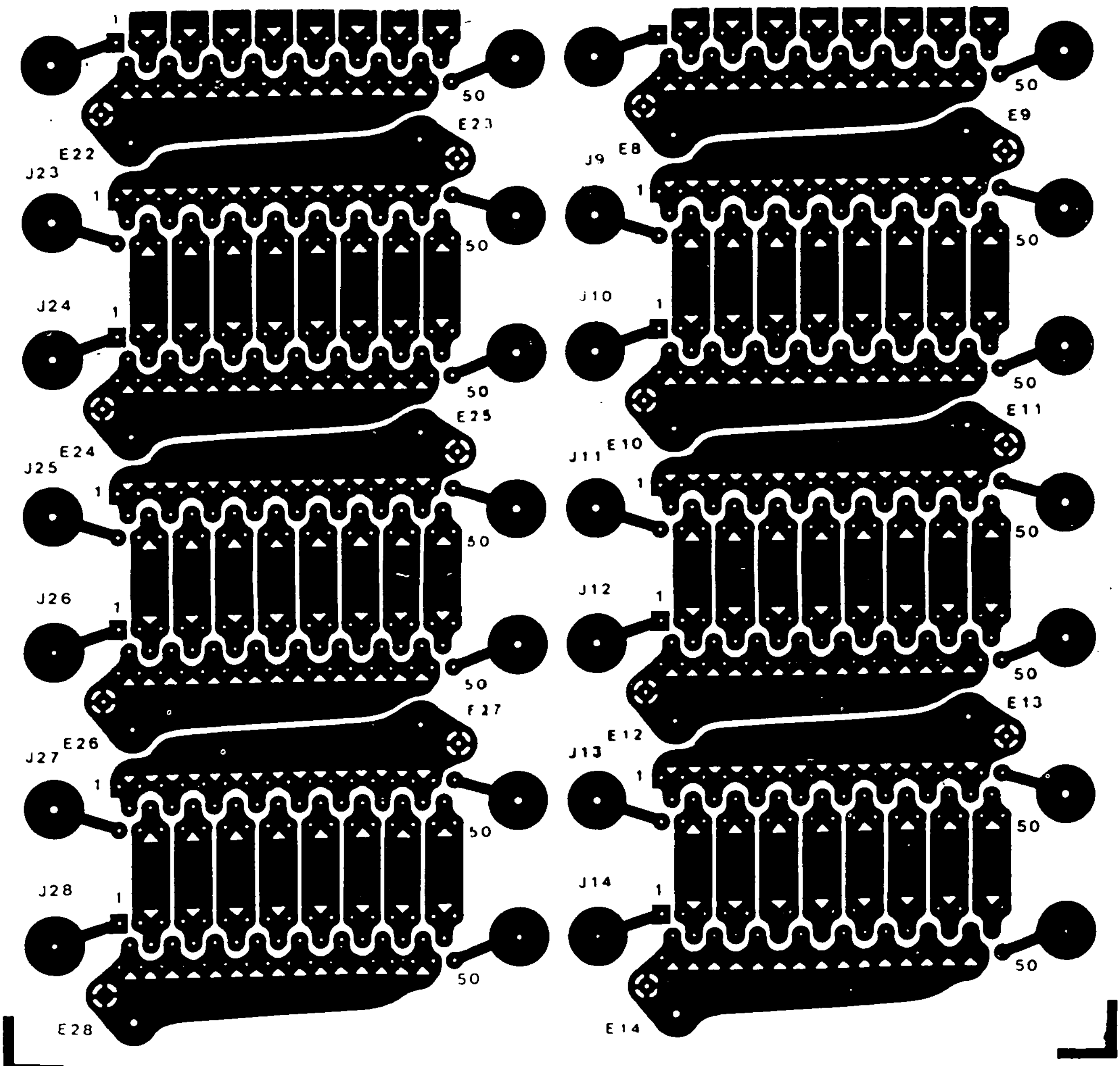
(3) Battery Safety

Used additional Fusing to control electrical hazards caused by requiring test batteries to be on spacecraft for Integration testing.



PSE I-V Curve - Shunt Mode
(6-10-89)
(Corrected 7-31-89)
Bus Voltage
Expected Max Shunt Reg
(28.49V @ 48.0 Amp)
7-H4

DM 5


J23
E22
E23
J24
J25
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J26
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F27
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J9
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J11
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J12
J13
E12
E13
J14
E14
1
50

DM-5

COBE Power System Electronics (PSE)

Problems during I &T

(1) Finding Equipment - Solution would be to have a full time person to control equipment, review equipment required in work orders and kit it prior to need.

(2) Battery maintenance, GSE and cabling.

(3) Integration schedule required components be integrated before the components were completed. Simulators for the PSE and the Shunt assembly allowed good work arounds.

(4) Calibration of spacecraft EGSE voltage and temperature monitors.

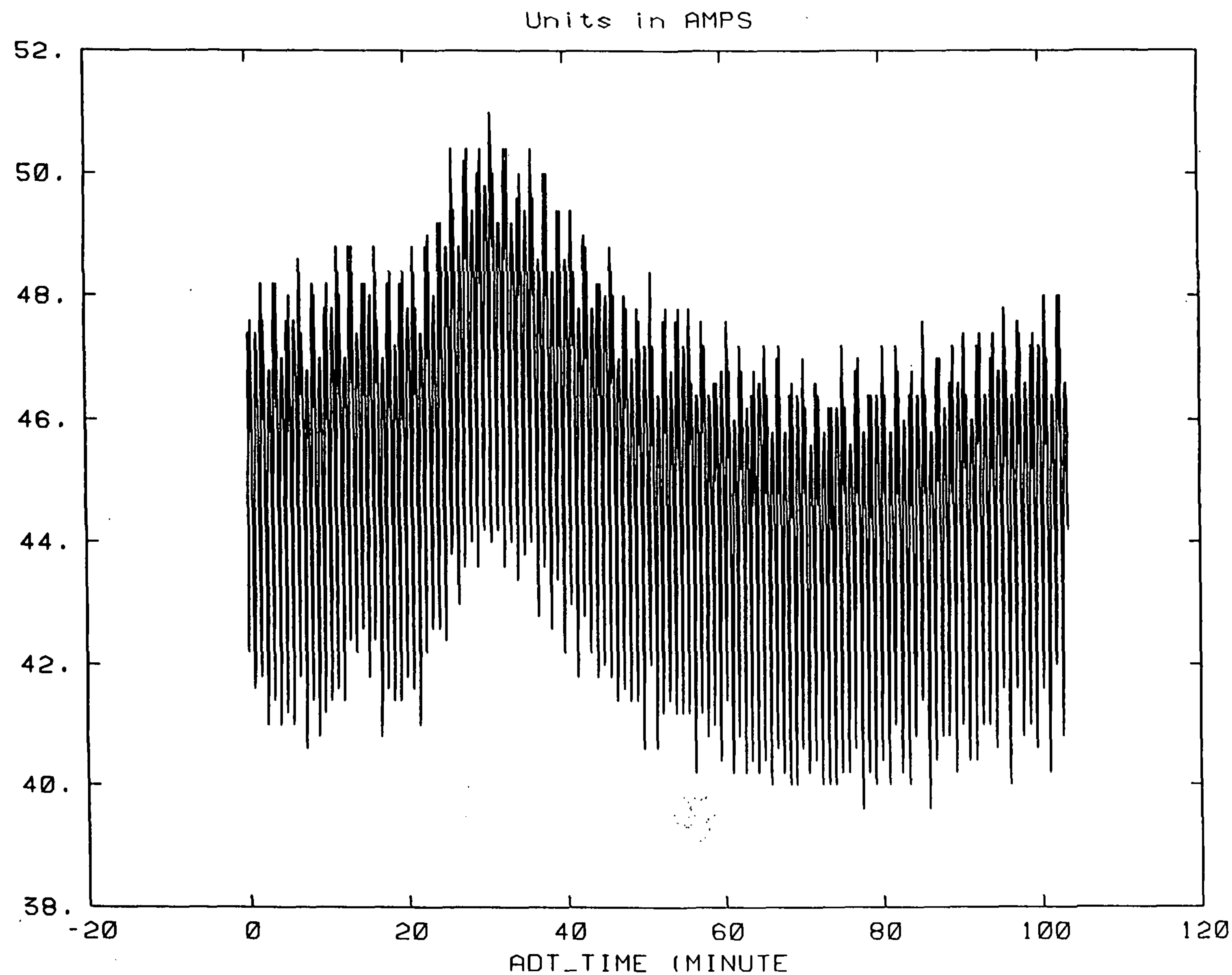
Waivers/Deviations

Battery A Ampere Hour Integrator state of charge readout failed during I &T.

On Orbit performance:

Performance within specification and expectations for all PSE parameters. The following features have been verified; Bus voltage regulation, +-2%, Boost mode, charge mode shunt mode, AHI and VT battery charge control, 500 mV ripple, Thermal predicts for 100% Sun.

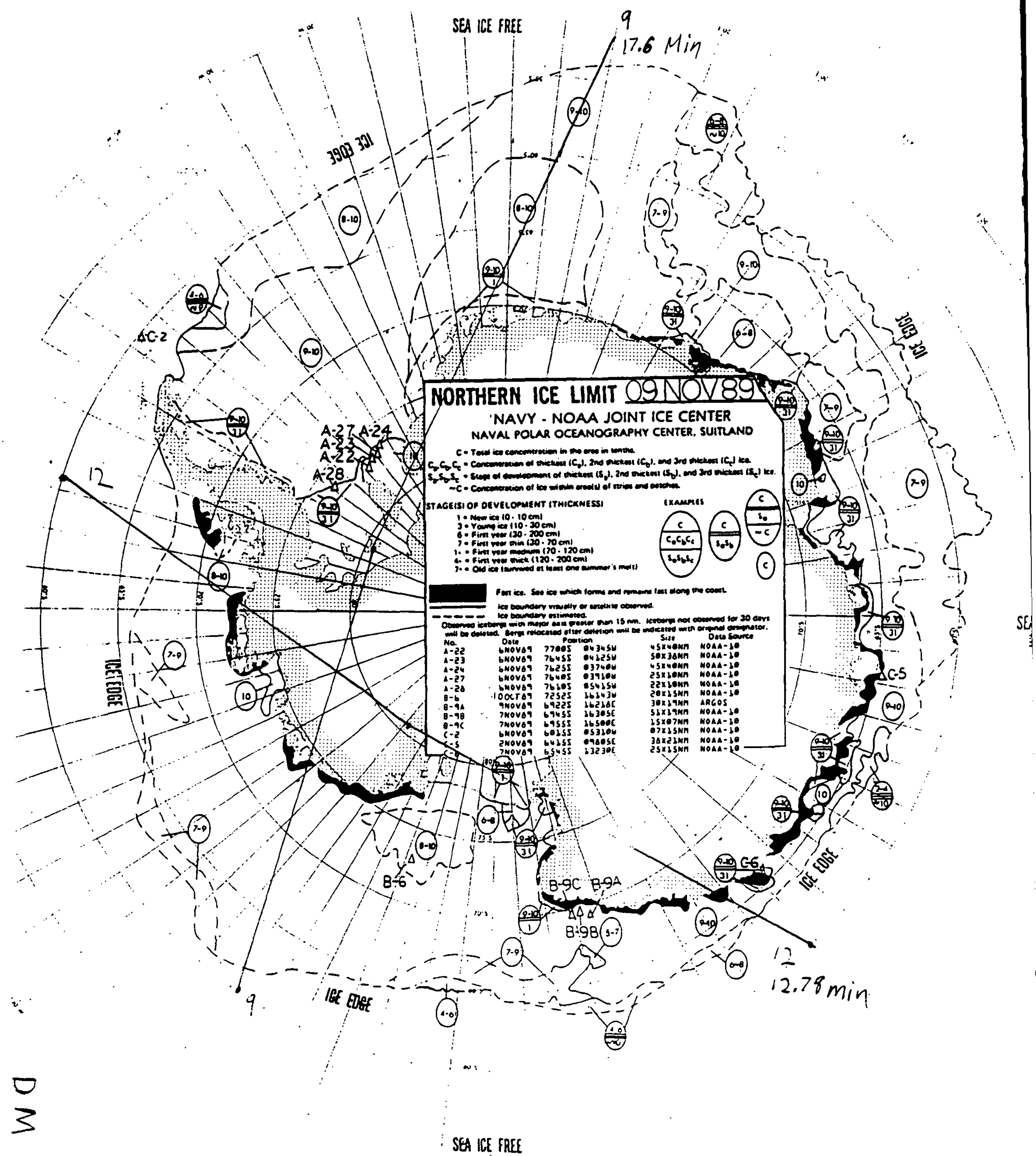
10 to 15% Power increase at South Pole.



PSAI

Drawn:30-NOV-1989 21:02
By:SHIV
19-NOV-1989 09:26:28.30

DM 7



SEA ICE FREE
9
17.6 Min
ICE EDGE
ICE EDGE
NORTHERN ICE LIMIT 09 NOV 89
NAVY - NOAA JOINT ICE CENTER
NAVAL POLAR OCEANOGRAPHY CENTER, SUITLAND
C = Total ice concentration in the area in tenths.
C_a, C_b, C_c = Concentration of thickest (C_a), 2nd thickest (C_b), and 3rd thickest (C_c) ice.
S_a, S_b, S_c = Stage of development of thickest (S_a), 2nd thickest (S_b), and 3rd thickest (S_c) ice.
~C = Concentration of ice within area(s) of strips and patches.
STAGE(S) OF DEVELOPMENT (THICKNESS)
1 = New ice (0 - 10 cm)
3 = Young ice (10 - 30 cm)
6 = First year (30 - 200 cm)
7 = First year thin (30 - 70 cm)
1• = First year medium (70 - 120 cm)
4• = First year thick (120 - 200 cm)
7• = Old ice (survived at least one summer's melt)
EXAMPLES
Fast ice. Sea ice which forms and remains fast along the coast.
Ice boundary visually or satellite observed.
Ice boundary estimated.
Observed icebergs with major axis greater than 15 nm. Icebergs not observed for 30 days will be deleted. Bergs relocated after deletion will be indicated with original designator.

No.	Date	Position		Size	Data Source
A-22	6NOV89	7700S	08434W	45X40NM	NOAA-10
A-23	6NOV89	7645S	04125W	50X38NM	NOAA-10
A-24	6NOV89	7625S	03740W	45X40NM	NOAA-10
A-27	6NOV89	7640S	03910W	25X10NM	NOAA-10
A-28	6NOV89	7610S	05415W	22X10NM	NOAA-10
B-6	10OCT89	7252S	16143W	20X15NM	NOAA-10
B-9A	9NOV89	6922S	16216E	30X19NM	ARGOS
B-9B	7NOV89	6945S	16305E	51X19NM	NOAA-10
B-9C	7NOV89	6955S	16500E	15X07NM	NOAA-10
C-2	6NOV89	6015S	05310W	07X15NM	NOAA-10
C-5	2NOV89	6415S	09805E	38X21NM	NOAA-10
C-6	7NOV89	6545S	13230E	25X15NM	NOAA-10

A-27 A-24
A-23
A-22
A-28
C-2
12
B-6
B-9C B-9A
B-9B
C-5
C-6
ICE EDGE
12
12.78 min
9
ICE EDGE
SEA ICE FREE

DM 8

DM-8

COBE Power System Electronics (PSE)

What would I do Differently?

(1) Battery Safety — Safe the battery power leads.
Have safing of signal leads integral to battery design.
Reduce the need for cables to Battery EGSE.

(2) Battery mechanical and thermal design.

(3) Shunt Blankets

(4) Additional Shunting Capacity

(5) Eliminate current measuring shunts as a Single Point Failure

(6) Improve power distribution assembly to include:
- One fuse per wire
- Switching of loads
- Final filter for each load
- 37 pin connectors

COBE Power System Electronics (PSE)

Lessons Learned:

(1) EGSE must be reliable and include strip-chart recorders.

(2) Close out of MR's.

(3) Improve ability to do battery maintenance without exclusive use of Spacecraft.

COBE BATTERY

ON-ORBIT ENGINEERING PERFORMANCE

MARCH 1990

SMITH TILLER
Code 711.2

- SPECIFICATION REQUIREMENTS

  - Temperature Range: -5 To +25 Degrees Centigrade
  - Output Voltage Range: 18 To 27 Volts
  - Output Current (Eclipse): 20 Amperes/Battery
  - Peak Current Capability: 60 Amps For 5 Minutes

  - Mission Life (2 Eclipse Seasons/Year): 1 Year Design Requirement

    2 Year Design Goal

- SUBSYSTEM OVERVIEW

  - Battery Photo - Figure 1
  - Battery Mechanical Structural Design - Figure 2
  - Battery Signal Fuse Assembly Photo - Figure 3
  - Cell Voltage Fuse Assembly Photo - Figure 4

- TRADEOFFS REQUIRED DURING DEVELOPMENT

  - Standard 20 Ampere-Hour Battery Redesigned For Smaller Capacity And Fewer Cells to Reduce Weight For Delta Configuration

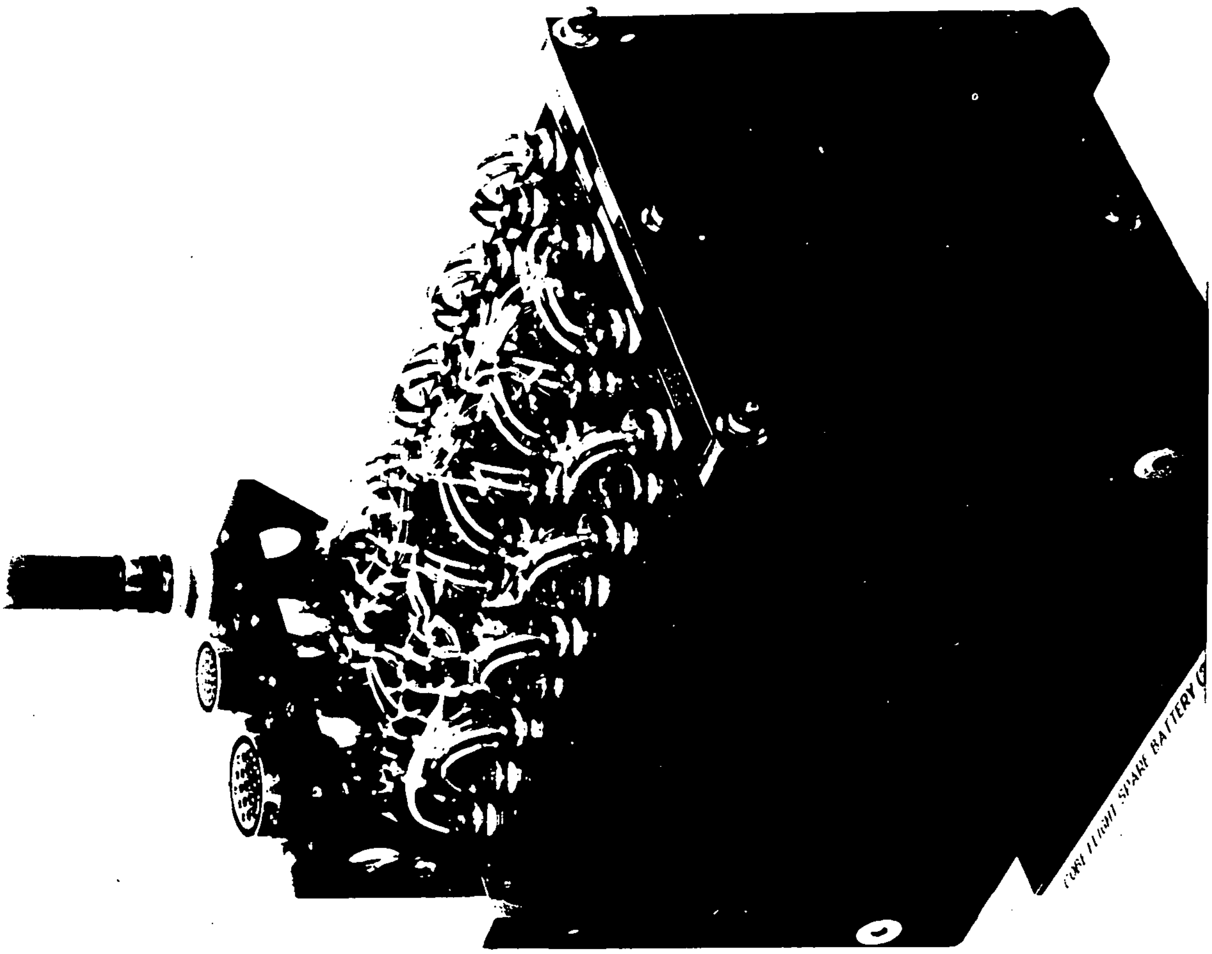


FIGURE 1

BATTERY MECHANICAL/STRUCTURAL DESIGN

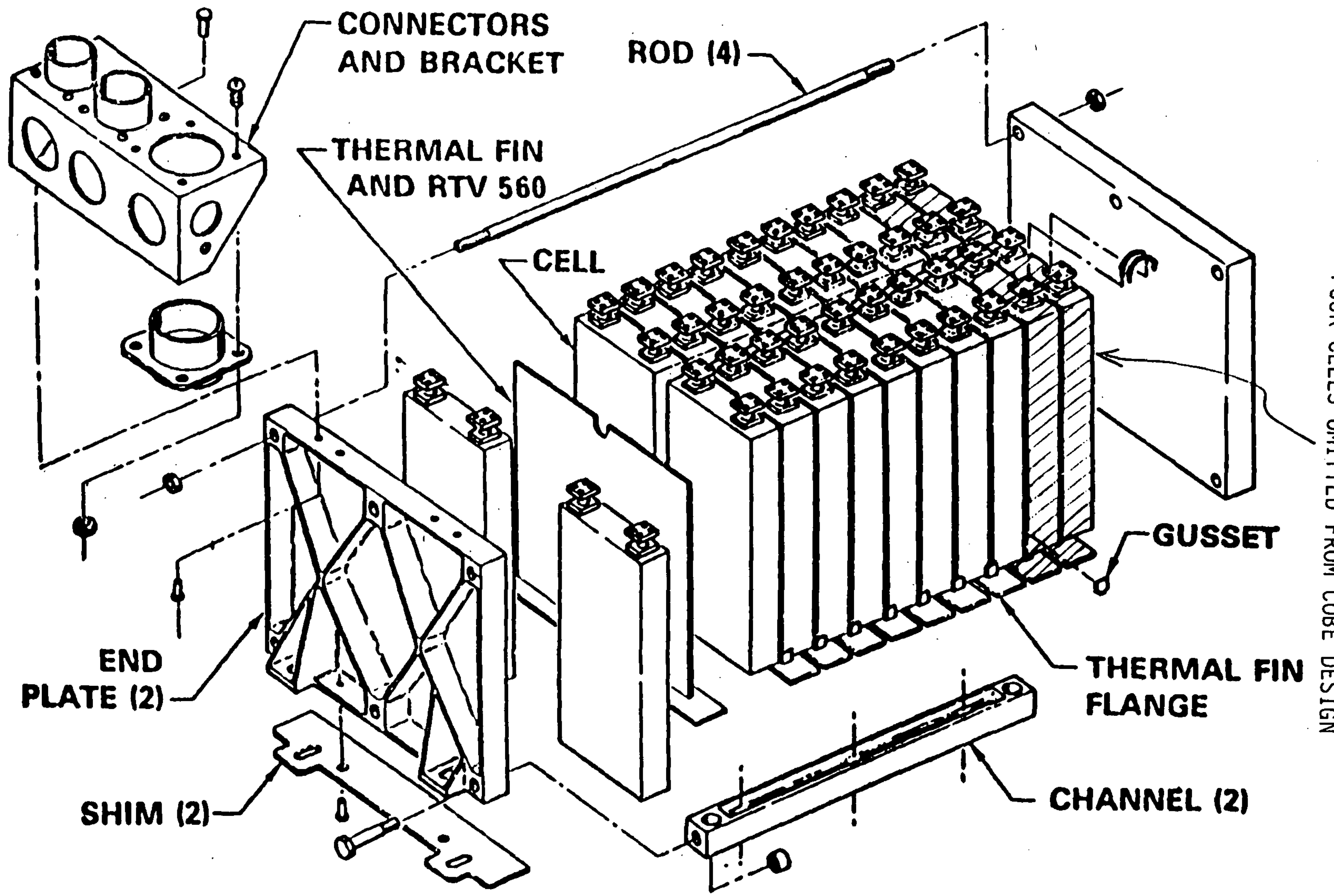
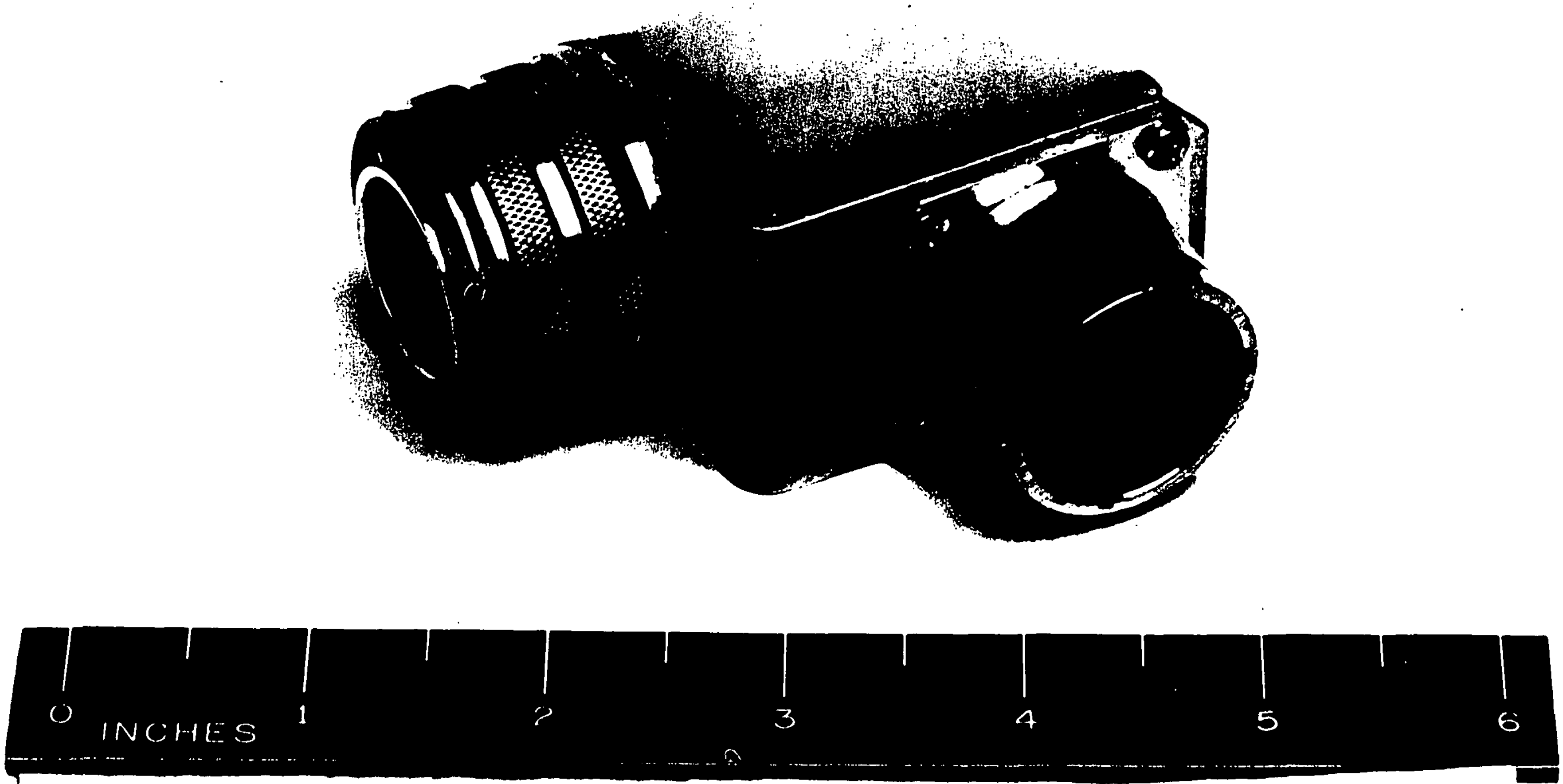


FIGURE 2



COBE BATTERY FUSE ASSEMBLY (J2)

FIGURE 3

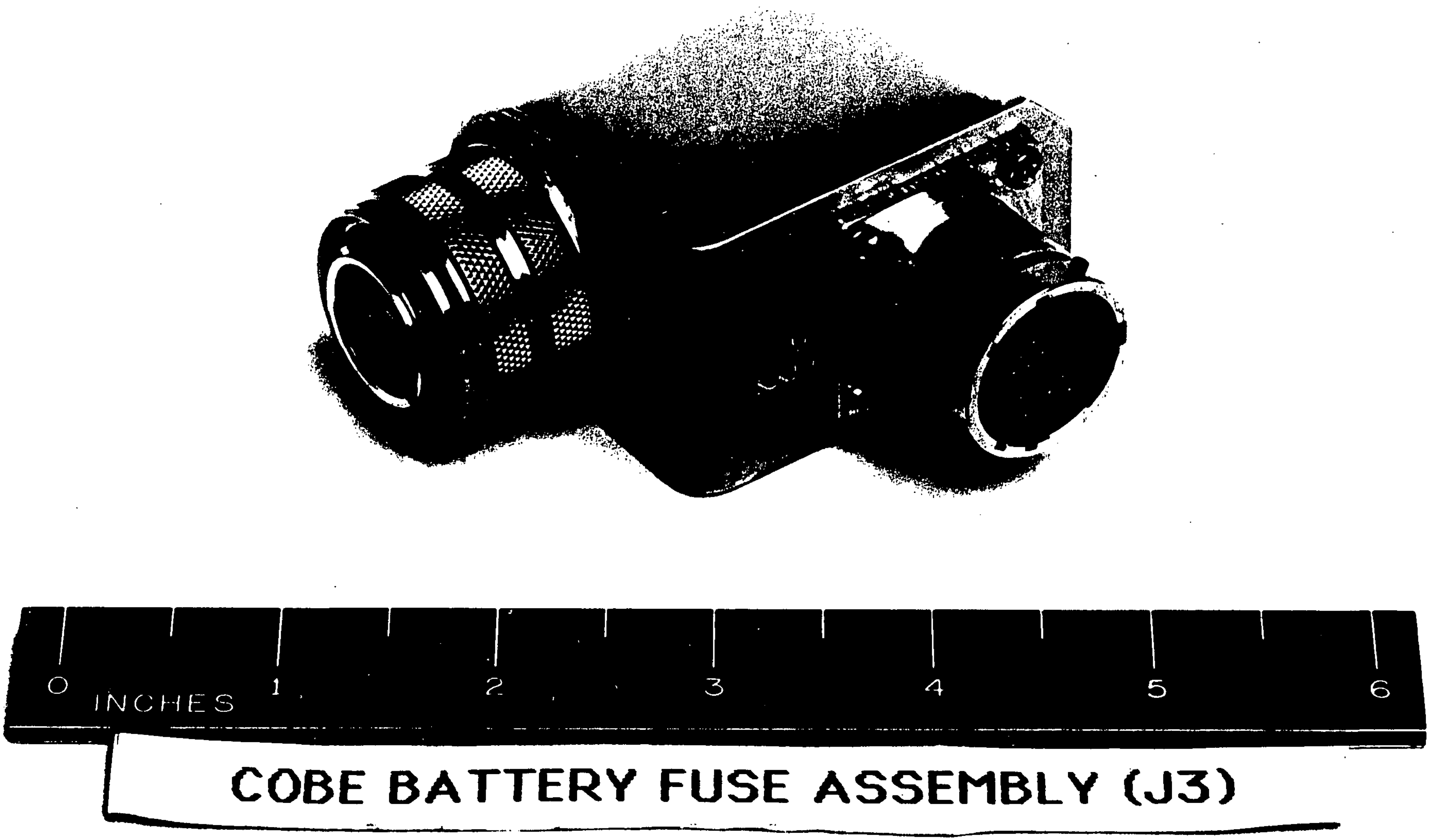


FIGURE 4

- **SIGNIFICANT PROBLEMS/ANOMALIES DURING I&T**
  - **GSE Test Connector Connected To Spacecraft Bottom Deck Damaged While Battery Charging During Other I&T Activities in SES Chamber**
  - **Battery Cells Manufacturered with 2536 Separator Material Have Slightly Higher Terminal Voltages Than Previously Experienced**
  - **Many WTR Facilitie's AC Power Wired Incorrectly Caused Damage To GSE And Test Delays**

- WAIVERS/DEVIATIONS

  - Replaced Suspect 2505 Cell Separator Material In Battery Cells With 2536 Separator Material During Manufacture.

  - Added 3 Wires To Battery Positive and Negative Terminals To Support Termination Points on COBE Power System Electronics

  - Implemented V-Notch To Maintain Battery Vibration Level at ± 32db Flight Specification During Acceptance at Manufacturer's Facilities

  - Accepted Batteries With Cracked (Cosmetic) Thermal Fins Following Manufacturer's Vibration Test

  - Added In-Line Fuse Assemblies Between Battery and Spacecraft Harness To Prevent Possible Spacecraft Damage Due To Harness Shorts.

  - Normal Manufacturing Procedures Had To Be Modified To Avoid To Avoid Contamination Of Spacecraft By Use Of Thermal Grease During Test And Evaluation

- **ON-ORBIT PERFORMANCE VS SPECIFIC REQUIREMENTS**

  - **Battery Trickle Charge Performance Excellent:**

DATA	ON-ORBIT	SPECIFIC REQUIREMENTS
Cell Voltage	1.43V ±10%	1.43V ±10%
Half Batt. V_Delta	0.000V To 0.007V	-0.5V To 0.5V
Batt. Temp.	12 To 15 Deg. C.	0 To 20 Deg. C.

  - **Battery Life Is Currently Expected To Be 3 Years Or More Due To The excellent Performance Observed Since Launch And Lower Mission Loads Than Initially Predicted**

  - **Spacecraft Thermal-Vac Battery Data Shows Slightly Higher Cell Voltage Than Former Aerospace Batteries**
  - **Anticipate Use Of Slightly HighVoltage/Temperature (VT) Levels During Battery Recharge In Orbit**

  - **Critical Battery Performance Data Will Be Available After First Spacecraft Eclipse Season**

- THINGS I WOULD DO DIFFERENTLY

  - Suggest That Battery Turn-On Connectors On Spacecraft Bottom Deck Be Mounted Perpendicular To Spacecraft Thrust Axis To Facilitate Easy Access

  - Suggest That Battery Turn-On Connectors On Spacecraft Bottom Deck Be Adequately Spaced To Allow Easy Access

  - Suggest That All Battery Turn-On Connectors On Spacecraft Bottom Deck Be Scoop Proof

  - Suggest Access Port To Battery Fuse Assemblies For Replacement If Necessary To Prevent Launch Delays

  - Add Additional Wires To Battery Positive And Negative Terminals In the Cell Voltage Connector To Facilitate Battery Conditioning With The Battery Power And Signal Turn-On Connectors Installed In Flight Configuration

LESSONS LEARNED

- Newly Designed Battery GSE Failsafe Systems Proven Effective During Several I&T Mishaps

- Use Of Non-Flight Batteries During Portions Of I&T Provided Experience Necessary For Launch Support Activities

- Fully Trained Battery Personnel Must Be Present For All Flight Battery Operations

- Always Coordinate SubSystem Activities With The Spacecraft Structural Engineering Group, Quality Assurance, Contamination Control Etc. To Prevent Operational Impacts

- Personally Check AC Power In Test Facilities With A Meter For Correct Voltages And Phasing

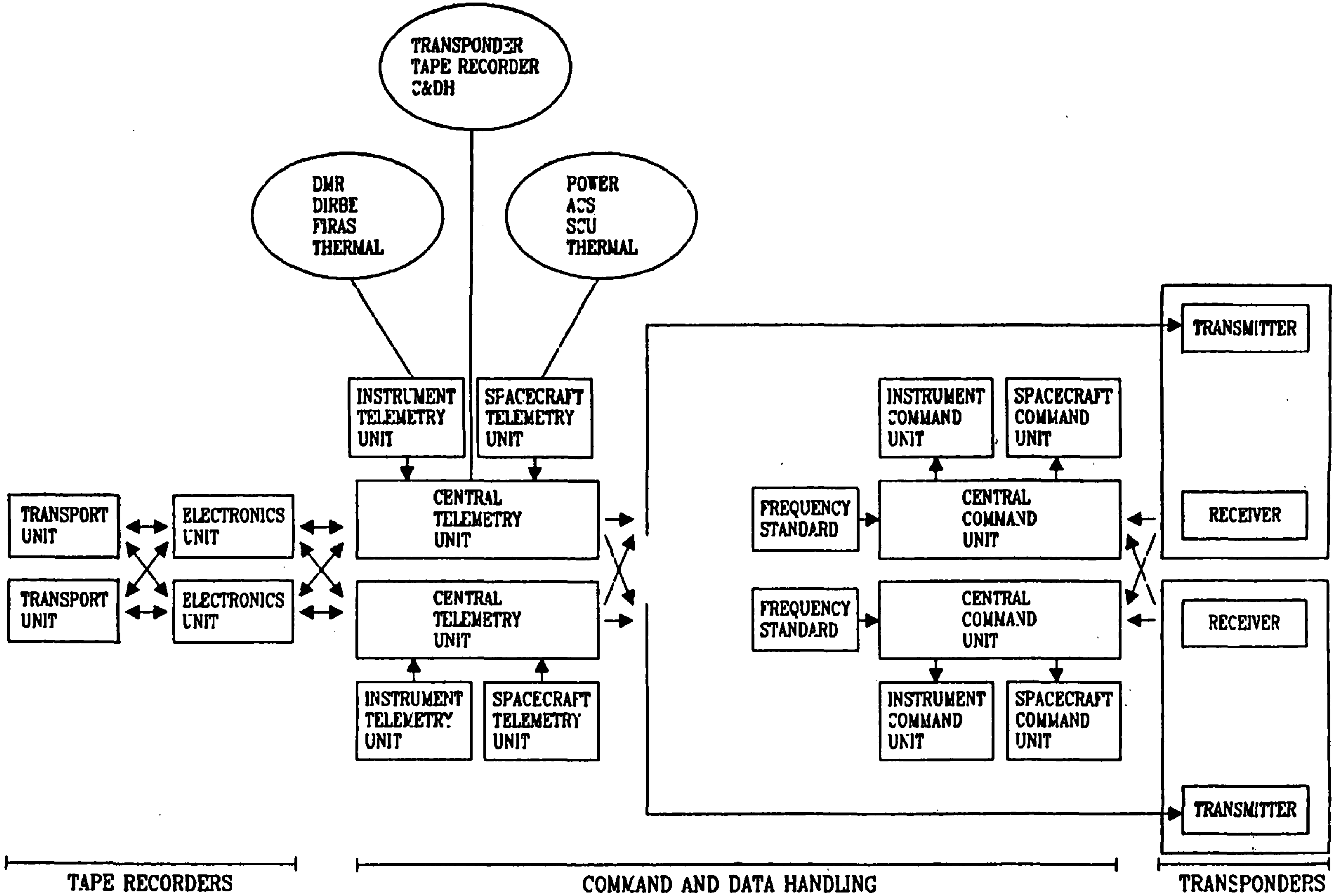
- Ascertain That AC Power Provided For GSE Is Independent and Not Shared By Other Users

COMMAND AND DATA HANDLING, TAPE RECORDERS AND SIGNAL CONDITIONING UNIT

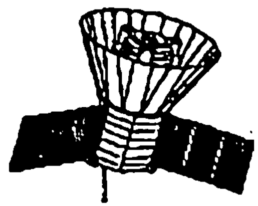
COBE LESSONS LEARNED

MARK FLANEGAN/CODE 735
MARCH 1990

COBE C&DH AND TAPE RECORDERS



MCF-2



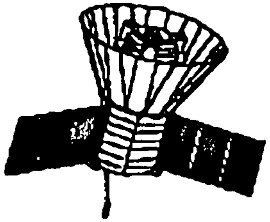
SIGNAL CONDITIONING UNIT (SCU) FUNCTIONS

AUTOMATIC DEWAR VENT VALVE OPERATION DURING ASCENT.

SEQUENCER FUNCTION INITIATED BY DELTA COMMAND:

- POWER TRANSMITTER #1
- DEPLOY THERMAL SHIELD
- POWER MOMENTUM WHEELS
- DEPLOY SOLAR ARRAYS
- DEPLOY ANTENNA BOOM
- POWER REACTION WHEELS

PRT AND GRT TELEMETRY CONDITIONED AND SENT TO C&DH.



C&DH
COMMAND UNITS RELAY
CHANGEOUT

BACKGROUND:

C&DH SYSTEM SWITCHES POWER TO SUBSYSTEMS THROUGH THE USE OF RELAYS WITHIN THE C&DH BOXES.

PROBLEM:

ISOLATION FAILURE DURING ACCEPTANCE TESTING AT CONTRACTOR. TRACED TO SHORT WITHIN POWER SWITCHING RELAYS.

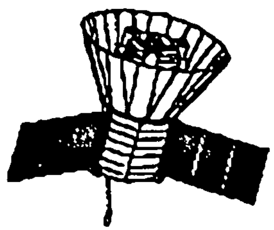
CAUSE:

USE OF SHARP TOOLS TO INSERT WIRES IN RELAYS PINCHED WIRES WITHIN RELAYS CAUSING SHORT TO CASE.

SOLUTION/LESSON:

NASA PERSONNEL SUPERVISED PRODUCTION OF NEW RELAYS. ALL POWER SWITCHING RELAYS IN C&DH WERE REPLACED.

SIMPLE WORKMANSHIP PROBLEMS CAN HAVE MAJOR CONSEQUENCES.



C&DH

COMMAND FAILURE IN

SPACECRAFT COMMAND UNIT

BACKGROUND:

FIRING A COMMAND ON COBE CAUSES A ROW AND A COLUMN DRIVER TO FIRE IN THE C&DH. WHERE THEY MEET IS THE COMMAND THAT IS SENT OUT.

PROBLEM:

ONE COMMAND WAS SENT, TWO RECEIVED IN REDUNDANT SPACECRAFT COMMAND UNIT.

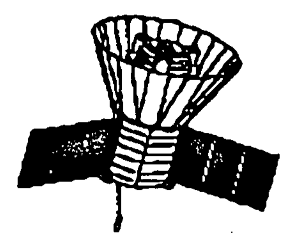
CAUSE:

STRAND OF WIRE ON BOARD UNDER CONFORMAL COATING CAUSED HIGH IMPEDANCE SHORT WHICH BRIDGED A COLUMN AND CAUSED TWO COMMANDS TO BE ACTIVATED. THIS WIRE WAS THERE FOR ABOUT TWO YEARS (8 THERMAL VACUUM CYCLES) BEFORE CAUSING A PROBLEM.

SOLUTION/LESSON:

REMOVE THE WIRE.

CONTAMINATION CONTROL MUST BE CAREFULLY MAINTAINED THROUGHOUT PRODUCTION/TEST. CONTAMINANTS CAN MOVE IN CONFORMAL COATING CAUSING A PROBLEM LATER.



C&DH
COMMAND UNIT
PULL UP RESISTOR CHANGE

BACKGROUND:

SERIAL DIGITAL AND LOW LEVEL LOGIC COMMANDS FROM THE C&DH USE OPEN COLLECTOR DEVICES WHICH ARE PULLED UP AT THE USER END OF THE WIRE.

PROBLEM:

SERIAL DIGITAL COMMAND TO THE C&DH DID NOT OPERATE RELIABLY. SIGNAL RISE TIMES WERE TOO SLOW.

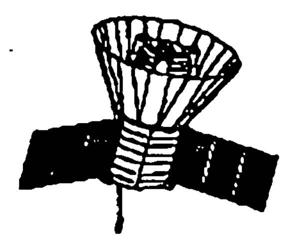
CAUSE:

MILLER EFFECT IN DRIVING TRANSISTORS.
RESISTORS TO MINIMIZE THIS EFFECT HAD BEEN REMOVED TO PREVENT A TRANSISTOR FAILURE FROM PROPAGATING TO THE OTHER TRANSISTORS. PULL UP RESISTOR (10 K) WAS TOO LARGE WITH THIS EFFECT TAKEN INTO ACCOUNT.

SOLUTION/LESSON:

REPLACE 10 K RESISTORS USED IN C&DH WITH 1 K OHM. REVIEWED OTHER USERS INTERFACES AND DECIDED THAT THEY HAD ENOUGH MARGIN TO OPERATE RELIABLY.

GIVE CAREFUL THOUGHT TO THE TOTAL CAPACITANCE OF SYSTEM BEFORE CHOOSING PULL-UPS.



C&DH
TELEMETRY UNIT PULL-UP RESISTOR CHANGE

BACKGROUND:

CAPACITANCE IN THE HARNESS HAD BEEN A CONCERN VERY EARLY IN THE DESIGN OF THE C&DH.

PROBLEM:

ANALYSIS INDICATED THAT SOME C&DH TELEMETRY POINTS WOULD NOT BE SAMPLED CORRECTLY.

CAUSE:

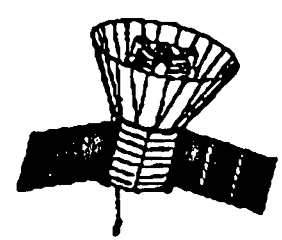
CAPACITANCE MEASUREMENTS OF THE HARNESS SHOWED MUCH HIGHER VALUES THAN HAD BEEN ESTIMATED BEFORE HARNESS CONSTRUCTION (50 PF/FOOT).

SOLUTION/LESSON:

REDUCE VALUE OF PULL-UP RESISTORS IN TELEMETRY UNITS TO GIVE MORE MARGIN.

GIVE CAREFUL CONSIDERATION TO THE HARNESS CAPACITANCE BEFORE PICKING PULL-UP RESISTORS. THE ESTIMATES MADE BEFORE HARNESS CONSTRUCTION WERE ABOUT HALF OF WHAT WAS EVENTUALLY MEASURED.

MCF-7



C&DH
TELEMETRY SYSTEM
CROSS-STRAPPING PROBLEM

BACKGROUND:

CROSS-STRAPPING OF USERS SIGNALS WAS ADDED TO THE C&DH AFTER ITS INITIAL DESIGN. SIGNALS ARE WIRE-ORED IN THE C&DH.

PROBLEM:

BILEVEL TELEMETRY BITS STARTING DROPPING OUT AFTER REDUNDANT C&DH I&T BEGAN. THE UNPOWERED C&DH SIDE WAS LOADING DOWN THE POWERED SIDE.

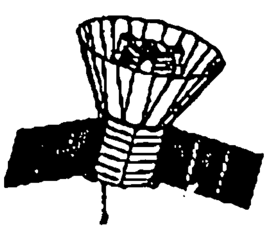
CAUSE:

THE MULTIPLEXERS USED TO SWITCH THE SIGNALS IN THE C&DH EXHIBITED A LOW INPUT IMPEDANCE (ABOUT 100 K OHMS) INTERMITTANTLY WHEN UNPOWERED. THE MANUFACTURER DID NOT SPEC THE UNPOWERED INPUT IMPEDANCE OF THE DEVICE. A MISTAKE WAS MADE IN USING THE SPEC FOR A POWERED DEVICE WHEN DESIGNING THIS CIRCUIT.

SOLUTION/LESSON:

CROSS STRAP THE MULTIPLEXER SUPPLIES BETWEEN TELEMETRY UNITS SO THAT WHEN ONE UNIT IS ON, THE MULTIPLEXERS IN BOTH UNITS ARE ON. THIS PUTS US IN THE SPEC CONDITIONS FOR THE MULTIPLEXERS.

PAY ATTENTION TO THE CONDITIONS UNDER WHICH A DEVICE IS SPECIFIED. BE CAREFUL OF UNPOWERED CONDITIONS.



C&DH
INSTRUMENT REMOTE
TELEMETRY UNIT FAILURE

BACKGROUND:

THE INSTRUMENT REMOTE IS THE PATH FOR ALL INSTRUMENT TELEMETRY ON COBE.

PROBLEM:

LOADING DOWN OF PRIME INSTRUMENT TELEMETRY UNIT SIGNALS DURING FUNCTIONAL TEST AT LAUNCH SITE.

CAUSE:

METALLIC CONTAMINATION ON BOARD BRIDGED TWO LINES CAUSING A MULTIPLEXER IN THE UNIT TO SELECT MORE THAN ONE LINE AT A TIME. CONTAMINATION APPEARED TO BE MACHINE SHOP SHAVING.

SOLUTION/LESSON:

REMOVE THE CONTAMINATION.

CONTAMINATION MUST BE ADDRESSED THROUGHOUT THE PRODUCTION/TEST PROCESS. CONFORMAL COATING WILL NOT HOLD CONTAMINANTS IN PLACE.

TAPE RECORDER
RECORD MODE PROBLEM

BACKGROUND:

THERE ARE TWO TAPE RECORDERS ON COBE. EACH IS COMPOSED OF A TRANSPORT AND AN ELECTRONICS UNIT.

PROBLEM:

TAPE RECORDER #1 WOULD NOT GO INTO RECORD MODE FROM STANDBY MODE DURING SPACECRAFT THERMAL VACUUM TESTING. IT WOULD GO INTO PLAYBACK, FAST FORWARD AND FAST REVERSE.

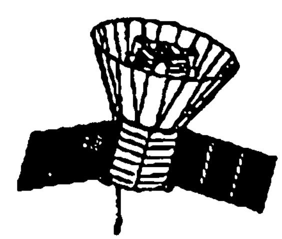
CAUSE:

VOLTAGE DIP APPEARS ON FIVE VOLT LINE FROM CONVERTER WHEN MODE CHANGE IS COMMANDED. DIP IS LARGER FOR SMALLER CURRENT DRAWS. OVER TIME, AS THE UNIT IS WORKED IN, IT DRAWS LESS CURRENT. EVENTUALLY IT REACHED A LEVEL WHERE THE DIP ON THE FIVE VOLT LINE WAS ENOUGH TO CAUSE THE RESET CIRCUIT TO ACTIVATE. RESET (ACTIVE LOW) PUTS THE TAPE RECORDER IN STANDBY. OTHER MODES DRAW MORE CURRENT.

SOLUTION/LESSON:

CAPACITOR WAS ADDED TO RESET CIRCUIT TO FILTER OUT VOLTAGE DIP. REWORK WAS PERFORMED ON BOTH UNITS.

TEST DEVICES EXTENSIVELY IN MISSION MODES IN MISSION CONFIGURATION.



C&DH
CLOCK NOISE

BACKGROUND:

THE C&DH PROVIDES TELEMETRY CLOCKS TO THE OTHER SUBSYSTEMS ON COBE.

PROBLEM:

NOISE ON CLOCK LINES WAS SEVERE.

CAUSE:

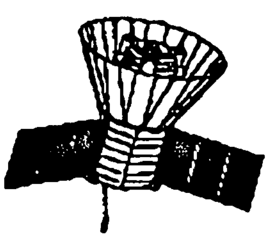
IMPEDANCE MISMATCHING OF DRIVER TO LINE.

SOLUTION/LESSON:

ADD 27 OHM RESISTORS TO DRIVER (C&DH) END TO MATCH LINES BETTER. WORKED VERY WELL.

PAY ATTENTION TO TRANMISSION LINE CONSIDERATIONS DURING DESIGN.

MCF-11



C&DH
FREQUENCY STANDARD
OUTPUT AMPLITUDE

BACKGROUND:

THE FREQUENCY STANDARDS ON COBE PROVIDE THE 4 MHz SIGNAL WHICH IS USED FOR ALL SPACECRAFT TELEMETRY TIME (INCLUDING THE PB5 CLOCK).

PROBLEM:

FS *2 OUPUT AMPLITUDE OF THE 4 MHz SIGNAL WAS LOW. THE TELEMETRY SYSTEM STILL WORKED BUT WITH LITTLE MARGIN.

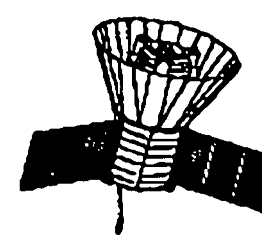
CAUSE:

MICA CAPACITOR FAILED OPEN IN UNIT.

SOLUTION/LESSON:

REPLACED ALL MICA CAPACITORS IN BOTH FREQUENCY STANDARDS WITH CERAMIC CAPACITORS. REPLACED MICA CAPACITORS IN CENTRAL COMMAND UNITS WHEN OPPURTUNITY PRESENTED ITSELF.

PARTS BRANCH PREFERS CERAMIC CAPACITORS TO MICA'S.



C&DH
OPEN ITEMS
AT LAUNCH

1. FREQUENCY STANDARD *2 TELEMETRY BIT FOR FREQUENCY ADJUST RELAY *2 STUCK AT ZERO ON FIRST TRY DURING DELTA I&T.
   (5/12/88; WOA D-046-03; MR 296)

   TESTED MANY TIMES SINCE AND PROBLEM NEVER APPEARED AGAIN.

   FUNCTION IS NOT CRITICAL IF LOST.

   HAVE NOT SEEN PROBLEM IN FLIGHT.

2. TEN HIGH LEVEL PULSE COMMANDS FROM SPACECRAFT COMMAND UNIT *2 DID NOT WORK WHEN INITIALLY CHECKED.
   (6/1/88; WOA D-064-01; MR 297)

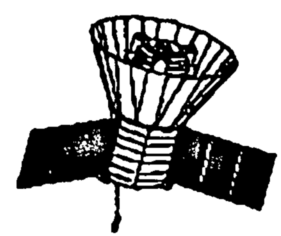
   COMMANDS:

   8 SPARES
   BATTERY *2 UNDERVOLTAGE DETECTOR DISABLE
   BATTERY *1 VT CONTROL NORMAL

   I&T CREW CHECKED OUT EQUIPMENT AND THEN RETESTED COMMANDS. THEY WORKED AND HAVE WORKED SINCE.

   COMMANDS ARE REDUNDANT.

   HAVE NOT SEEN PROBLEM IN FLIGHT.



C&DH
OPEN ITEMS
AT LAUNCH

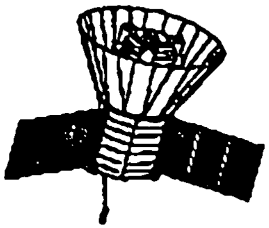
3. CENTRAL TELEMETRY UNIT #2 PASSIVE ANALOG CHANNELS 28 THROUGH 31 READ LOW IN COUNTS INTERMITTANTLY.
(6/11/89; WOA-246-137; MR 3530)

CHANNEL	FUNCTION
28	USO FLASK TEMPERATURE
29	SOLAR ARRAY WING B INNER POSITION INIDCATOR
30	STABLE CLOCK #1 FOOTPRINT
31	SOLAR ARRAY WING A INNER POSITION INDICATOR
32	STABLE CLOCK #2 FOOTPRINT

TESTING INDICATES FAILURE IS MOST LIKELY INPUT MULTIPLEXER.

THESE CHANNELS ARE NOT MISSION CRITICAL. FAILURE OF INPUT MULTIPLEXER WOULD NOT EFFECT OTHER TELEMETRY CHANNELS. UNIT WILL BE USED AS IS.

SAME PROBLEM IS OCCURRING IN FLIGHT. IT IS NOT GETTING ANY WORSE.



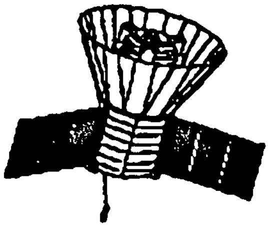
SCU
OPEN ITEMS
AT LAUNCH

1. SCU HIGH LEVEL COMMAND OUTPUT REGISTERS 11.67 V WHEN IT IT SHOULD REGISTER 0 V. HIGH LEVEL COMMAND IS FOR MOMENTUM WHEEL #1 DRIVER OFF FROM THE B SIDE.
(12/20/88; WOA D-492-15; MR 3256)

LEAKAGE CURRENT IN OPTO-ISOLATOR OF AROUND 10 MICROAMPS IS CAUSE OF PROBLEM. THIS IS WITHIN SPECIFICATION FOR PARTS (UP TO 250 MICROAMPS).

10 MICROAMP CURRENT MUST INCREASE TO 40 MILLAMPS TO CAUSE MOMENTUM WHEEL TO TURN OFF. TURNING OFF SEQUENCER #2 BYPASS POWER WOULD ELIMINATE PROBLEM IF LEAKAGE GOT TO 40 MILLIAMPS. THIS LEAKAGE DOES NOT EFFECT THE OPERATION OF MOMENTUM WHEEL #1.

HAVE NOT SEEN PROBLEM IN FLIGHT.



C&DH I&T
FREQUENCY STANDARD
FREQUENCY DROP

BACKGROUND:

FREQUENCY STANDARD FREQUENCY IS 4 MHz. IT IS CHECKED USING A CESIUM STANDARD TO MILLIHERTZ. THERE ARE ADJUSTMENT RELAYS TO ADJUST THE FREQUENCY BY ABOUT 1.3 Hz AROUND 4,096,000.000 Hz.

PROBLEM:

FREQUENCY OF FS *1 DROPPED TOO LOW FOR THE ADJUSTMENT TO BRING IT BACK TO 4 MHz.

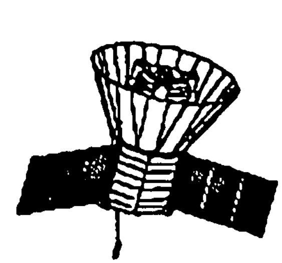
CAUSE:

CAUSE WAS TRACED TO ONE BOARD ON THE UNIT. FAILURE ANALYSIS DETERMINED THAT SOME OF THE CAPACITORS ON THE BOARD HAD SHIFTED IN VALUE. THIS APPEARS TO HAVE BEEN MISLEADING. PROBLEM IS OCCURRING IN FLIGHT.

SOLUTION/LESSON:

ALL PARTS ON THE BOARD EXCEPT THE CRYSTAL WERE REPLACED. THE CRYSTAL HAD TOO LONG A LEAD TIME TO REPLACE.

TREND IMPORTANT TREND PARAMETERS CLOSELY.



C&DH FLIGHT
FREQUENCY STANDARD
FREQUENCY DROP

BACKGROUND:

DROP IN FREQUENCY DURING I&T.

PROBLEM:

FREQUENCY STANDARD #1 IS DROPPING IN FREQUENCY AGAIN.
IF DROP CONTINUES FS #1 WILL FALL OUT OF ITS ADJUSTMENT RANGE IN FEW MONTHS. THIS MEANS PB5 TIME WILL STEADILY DRIFT AWAY FROM GROUND TIME.

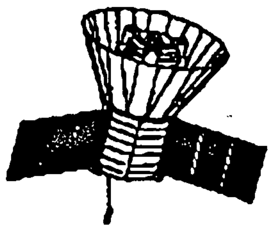
CAUSE:

UNKNOWN.

SOLUTION:

THREE OPTIONS:

1. LIVE WITH THE DRIFT.
2. ADJUST THE TIME (FAST COUNT) EVERY HOUR OR SO USING COMMAND MEMORY.
3. SWITCH TO C&DH SIDE #2.



TAPE RECORDER #1
PRESSURE DROP

BACKGROUND:

THE TAPE RECORDER TRANSPORT UNIT IS PRESSURIZED.

PROBLEM:

TR #1'S PRESSURE IS DROPPING AT ABOUT 0.6 PSI A MONTH. THIS IS MORE THAN EXPECTED. TR#2 IS NOT DROPPING SIGNIFICANTLY.

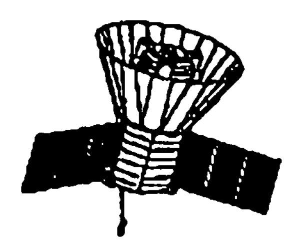
CAUSE:

UNKNOWN.

SOLUTION:

NONE. THIS IS FASTER THAN EXPECTED BUT WILL NOT CAUSE COBE A PROBLEM.

MCF-18



LESSONS LEARNED SUMMARY

1. CONTAMINATION CONTROL IS IMPORTANT DURING PRODUCTION/TEST.

2. CONFORMAL COATING WILL NOT HOLD CONTAMINANTS STATIONARY.

3. BE CAREFUL NOT TO UNDERESTIMATE CAPACITANCE IN SYSTEM WHEN SIZING PULL-UPS.

4. BE AWARE OF THE CONDITIONS SPECIFICATIONS ARE MADE UNDER.

5. BE AWARE OF TRANSMISSION LINE CONSIDERATIONS WHEN DESIGNING.

6. CERAMIC CAPACITORS ARE PREFERRED OVER MICA'S.

7. TEST THINGS IN MISSION MODE.

8. DESIGN SYSTEM FOR EASY MATE/DEMATE OF CONNECTORS; ESPECIALLY THOSE CONNECTORS THAT NEED TO BE ACCESSED OFTEN.

9. DESIGN SEPARATE ARMING CONNECTORS FOR SEPARATE GROUPS OF PYROS.

10. WRITE PROCEDURES TO AVOID USING LOTS OF BREAK-OUT-BOXES AT ONCE, IF POSSIBLE.

11. BUILD BTE TO PERFORM TIME INTENSIVE TASKS AUTOMATICALLY.

12. USE CROSS-STRAPPING TO PROVIDE FLEXIBILITY.