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(NASA-TM-109699) ARTEMIS COMMON LUNAR LANDER. PHASE 2: STUDY RESULTS FOR EXTERNAL REVIEW (NASA) 131 p

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# Artemis

## Common Lunar Lander

## Phase 2 Study Results for External Review

NASA Johnson Space Center

March 10, 1992

# Common Lunar Lander
# Phase 2
## Design Review

10-Mar-92

| Speaker | Topic |
|---|---|
| Steve Bailey | (Project Manager) |
| Jonette Stecklein | (Lead Engineer)<br>Engineering Overview |
| Lynn Wagner | Trajectory/Performance |
| Tim Pelischek | Structures/Thermal |
| Nancy Smith | GN&C |
| Dave Pruett | Data System |
| Henry Chen | Communications |
| Bill Culpepper | Tracking |
| Steve Rickman | Thermal |
| Don Hyatt | Propulsion |
| Betsy Kluksdahl | Power/Pyrotechnics |

Presentation to Mike Griffin; Nova Bldg, Rm 122

Artemis

# *Artemis* Common Lunar Lander

## Phase 2 Study Results for External Review

March 19, 1992

Stephen Bailey/IE3/283-5411

# Mission Statement

**"The purpose of the Artemis Program is to gather vital reconnaissance data by conducting robotic exploration missions to the lunar surface both prior to and concurrent with human exploration missions. The Artemis Program includes rapid, near-term development of a variety of small experimental and operational payloads, provides a low-cost capability to deliver these payloads to any location on the lunar surface, and supports the analysis of the data returned. The Artemis Program will improve the understanding of lunar geosciences, demonstrate the Moon's unique capability as an astronomical platform to study the universe, and to conduct scientific and technology development experiments, and will prepare for, enhance, and compliment human missions."**

Artemis

# The Artemis Lander Fits the JSC Mission

*"The mission of the Johnson Space Center is the expansion of human presence in space through exploration and utilization of space for the benefit of all"*

- The Artemis lander is the only near term exploration-oriented project suitable for JSC participation
- It will allow JSC to reestablish its expertise in the design, development, and operations of extra-LEO spacecraft
- This is vital to the health and future of the Center

# Artemis Fits the JSC Strategic Plan

- JSC is the lead center for exploration
- Builds the talent, knowledge and capability of our people
  - *"Perform selected in-house projects..."*
  - *"Perform selected precursor activities..."*
  - Strengthens requirements and project management capabilities
  - Create experience wedge for future exploration missions
  - Emphasize the role of civil servants early in the project
- Helps to establish requirements for human exploration
- Will help prepare us for living and working on the moon

Artemis

*Artemis*

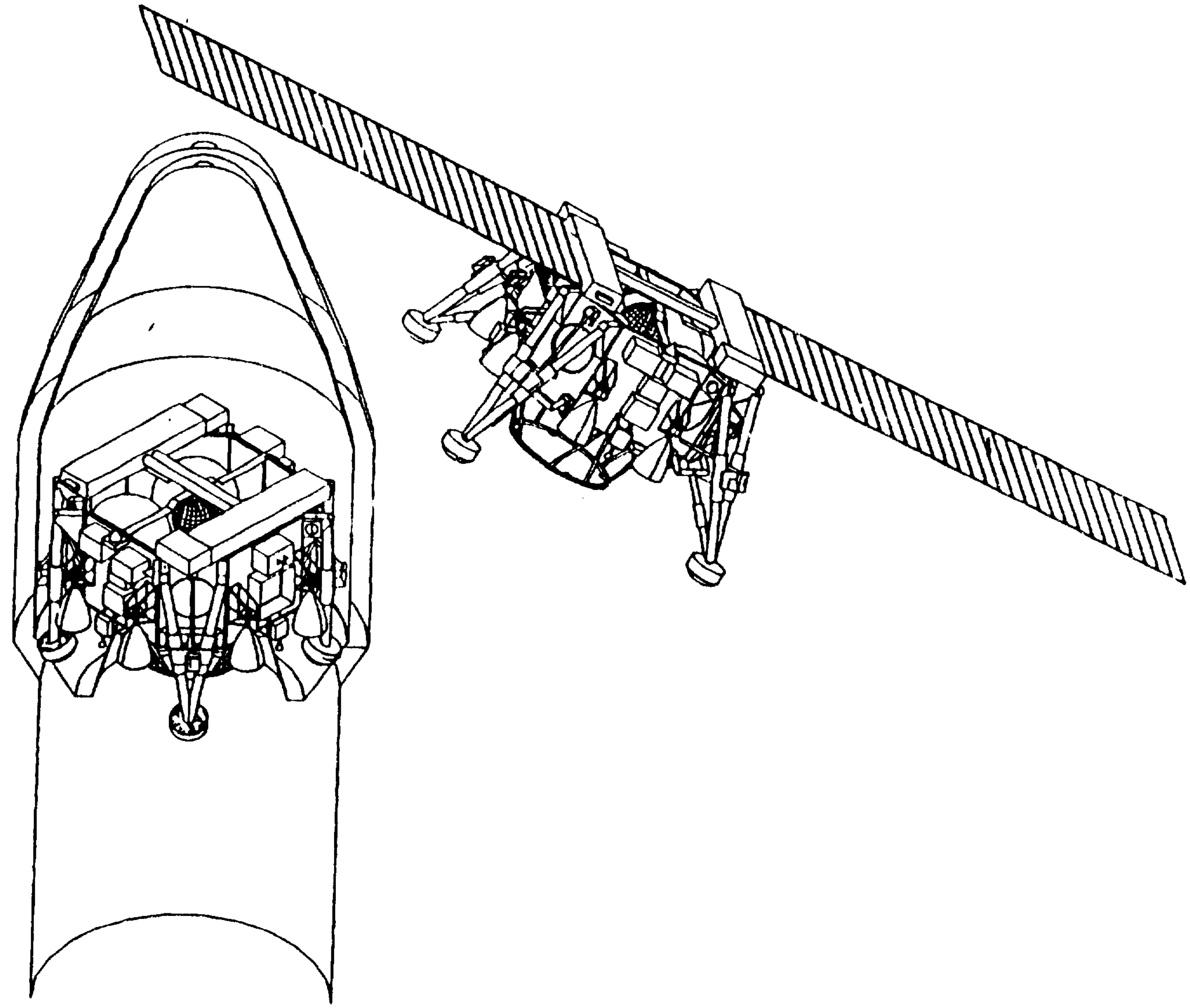
# Common Lunar Lander, Phase 2

# Design Review

March 10, 1992

Jonette Stecklein, Lead Engineer

Artemis



Jonette Stecklein/ET2/x36624

# CLL Phase 2 Study: Results

## I. CLL TASK

- Products
- Schedule
- CLL Team

## II. CLL DESIGN

- Mission and Requirements
- Spacecraft Design
- Reliability

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## III. WRAP-UP

- Conclusions
- Performance Options

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# CLL Phase 2: Engineering Products

- **Lander Design**
  A single design to be mated with the Delta II 7925 launch vehicle
- **Estimate of payload mass delivered to lunar surface**
- **Manpower and Schedules for CLL in-house development effort**
  including Facility and Equipment needs
- **Cost of CLL development**

J Stecher
2/28/92

# CLL II Study Schedule

| | Mon | Tues | Wed | Thurs | Fri |
|---|---|---|---|---|---|
| December 1991 | | | | KICKOFF MTG 12 | 13 |
| | 16 | 17 | TEAM MEETING B.17; Rm 2037 3-5pm 18 | 19 | 20 |
| | 23 | 24 | Holiday 25 | 26 | 27 |
| | 30 | 31 | Holiday 1 | 2 | 3 |
| January 1992 | 6 | TEAM MEETING B.15; Rm 131 2-4 pm 7 | 8 | 9 | 10 |
| | System Integration Week 13 | Subsystem Input 14 | 15 | Subsystem Input 16 | Lander Design Baselined<br>Pohl Status 17 |
| | Holiday 20 | SCHEDULING INTERVIEWS 21 | 22 | 23 | 24 |
| | 27 | Re-integrate System 28 | 29 | Wt,power reduc data dump 30 | 31 |
| February 1992 | 3 | EMS draft to team<br>TEAM MEETING 4 | Team Review of EMS 5 | EMS Redlines due<br>Wt,power reduc study completed 6 | 7 |
| | Update ES Dwgs 10 | Procurement<br>TEAM MEETING<br>Revised EMS 11 | Team Review of EMS<br>EMS Redlines due 12 | Reliability Study 13 | Drawings, Schedules Completed 14 |
| | Holiday 17 | TEAM MEETING<br>Reliability Study 18 | Dry Run 19 | Presentation to Pohl 20 | 21 |
| | Presentation to Cohen 24 | 25 | Integ Testing Mtg 2-5pm 26 | 10 weeks 27 | 28 |

March 10
CLL Design Review
1-5pm

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# Common Lunar Lander Team

## Subsystem Team Leads

| | | |
|---|---|---|
| Trajectory/Perf. | ET3/ET4 | Lynn Wagner/Lee Bryant |
| GN&C | EG2 | Nancy Smith |
| Space Data System | EK | Dave Pruett |
| Communications | EE7 | Henry Chen |
| Tracking | EE6 | Bill Culpepper |
| Propulsion | EP4 | Don Hyatt |
| Power/Pyro. | EP5 | Betsy Kluksdahl |
| Structure | ES | Tim Pelischek |
| Thermal | ES3 | Steve Rickman |
| EMS Development | ET3 | Nancy Wilks |

## Supporting Organizations

| | |
|---|---|
| SR&QA | NB/ND |
| Procurement | BE2 |
| Personnel | AH7 |
| Technical Services | JH2 |
| Operations | DA15 |
| White Sands Facility | RD |
| Facilities Planning | JD2 |

## Systems Engineering Team

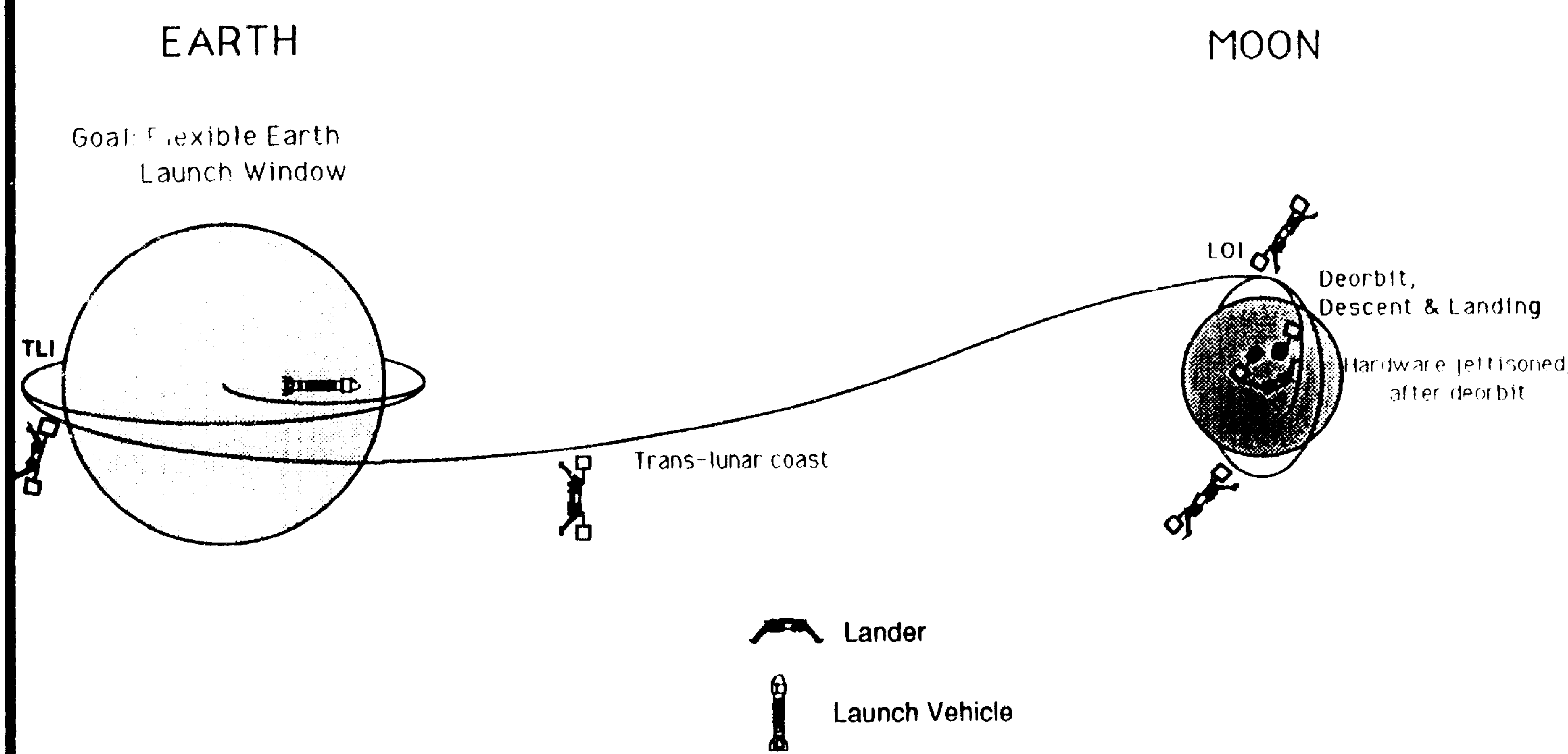
| | |
|---|---|
| Wayne Peterson | ET2 |
| Ed Robertson | ET2 |
| Mark Valentine | ET2 |
| Andy Petro | ET2 |
| David Rodriguez | ET3 |

# CLL Phase 2 Study: Results

## II. CLL DESIGN

- Mission and Requirements
- Spacecraft Design
- Reliability

# CLL Mission


EARTH
MOON
Goal: Flexible Earth Launch Window
TLI
LOI
Deorbit, Descent & Landing
Hardware jettisoned, after deorbit
Trans-lunar coast
Lander
Launch Vehicle

Artemis

# CLL Reference Mission

- Payload set mated to pallet. CLL spacecraft built in parallel. Payload pallet and CLL spacecraft integrated (structural interface only).
- CLL one-stage spacecraft is launched by Delta II 7925 from Cape Canaveral.
- Launch vehicle initiates Trans-Lunar Injection burn. CLL completes TLI burn.
- 5 day trip to moon.
- CLL performs Lunar Orbit Insertion burn, into circular orbit about Moon.
- Nominal 1 day wait in lunar orbit. (Able to wait in lunar orbit up to 7 days, if sufficient altitude management propellant exists.)
- CLL performs deorbit burn.
- Jettison hardware (solar arrays, controller) separates from CLL.
- CLL performs descent and landing burns, targeting for a given lunar lat/long, and landing at given lighting condition (typically lunar dawn).
- CLL transmits final system performance and landing location information to Earth. Power requirements sized for 1 hour lifetime on lunar surface.

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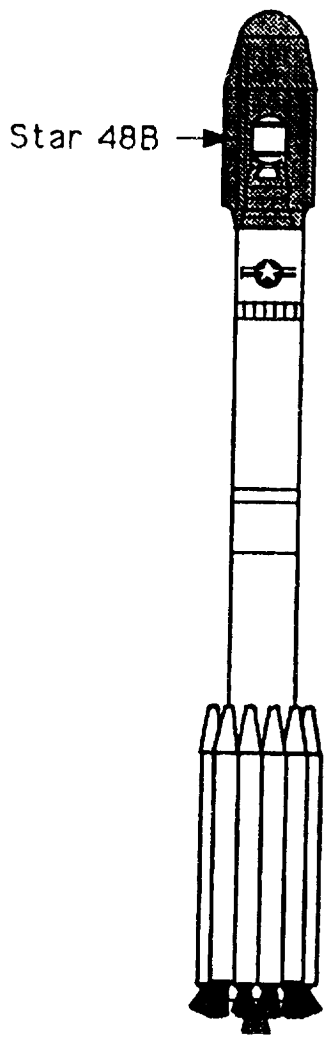
# DRIVING Requirements Levied on Engineering

| GUIDELINES | | SYSTEM DRIVERS |
|---|---|---|
| - Small, simple, cheap, & quick | → | ① Schedule<br>② Cost<br>③ Risk<br>④ Performance |

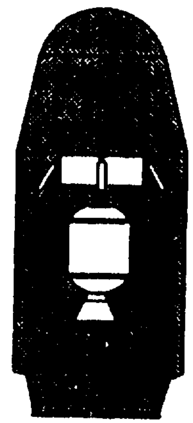
- Land a payload at any Lunar latitude/longitude
- Capable of capture into lunar orbit prior to descent, and loiter
- **First Launch by June 1, 1996**
- **Architecture Imposed**
  - **- Launched by Delta II 7925**
  - **- One-Stage Spacecraft**
  - **- Pulse Modulated Engines used for lunar landing**

- **Payload weight: best available, i.e. unconstrained**




McDonnell Douglas
Delta II 7925

One-Stage
Reference



3-Axis RCS control
of Star 48 during TLI

Growth Potential

Add LOI solid motor,
Launch on higher performance launch vehicle

Jonette Stecklein/ET2/x36624

Artemis

# Requirements Levied on Engineering

Lander

- Design loads and limits are constrained by launch vehicle, ~6 g peak for Delta LV
- Capable of communications when line of sight exists between spacecraft and ground station
- Targeting Accuracy: location: within 5 km (3 sigma); time: within 10 hours
- Lunar surface characteristics: 15 degree slope, 0.2m obstruction
- Provide structural interface to payload(s)
- Assuming use of higher performance launch vehicle:
  - Do not preclude attachment of propulsive stage [goal - not addressed]
  - Do not preclude landing of heavier payloads [goal - not addressed]

Payload Imposed Requirements

- On lunar surface, provide unobstructed hemispherical view of the sky
- Provide payload(s) with mounting locations having direct access to surface [goal]
- Keep deck height low, preferably within 1.5 m of surface [goal]

Lander Subsystems

- Emphasis on choosing existing system, rather than new design
- Avoid block redundancy when a single string system can provide adequate reliability

NOTE: ALL MASS IS IN KILOGRAMS.

# DESIGN MASS SUMMARY

## Common Lunar Lander (CLL)

REFERENCE: Baseline C
**February 26,1992 MASS STATEMENT**
**"BASELINE C Rev 1"**

| FUNCTIONAL SUBSYSTEM CODE | Lander | Jettison Hardware | | CLL Launch Adapter | |
|---|---|---|---|---|---|
| 1.0 STRUCTURE | 72 | 0 | | 11 | |
| 2.0 PROTECTION | 15 | 0 | | | |
| 3.0 PROPULSION | 89 | 0 | | | |
| 4.0 POWER | 51 | 74 | | | |
| 5.0 CONTROL | 0 | 0 | | | |
| 6.0 AVIONICS | 98 | 0 | | | |
| 7.0 ENVIRONMENT | 0 | 0 | | | |
| 8.0 OTHER | 31 | 8 | | 0 | |
| 9.0 GROWTH (15%) | 44 | 12 | | | |
| **DRY MASS** | 399 | 94 | | 11 | |
| 10.0 NON-CARGO | 20 | 0 | | | |
| 11.0 PAYLOAD | 64.1 | 0 | | | |
| **INERT MASS** | 484 | 94 | | 11 | |
| 12.0 NON-PROPELLANT | 0 | 0 | | | |
| 13.0 PROPELLANT | 861 | 0 | | | |
| **GROSS MASS** | 1,345 | 94 | | 11 | |

NOTE: ALL DIMENSIONS ARE IN METERS.

**NOTE:** Single string systems.
Lander: 3.36 m dia legs deployed, 2.54 m dia lander stowed
Payload: as available; limited by launch vehicle lift mass
Jettison Hardware: jettisoned after deorbit burn

Total Spacecraft Launch Mass (kg) = 1450

| **CLL Lander** | Tl. Mass KG | | Tl. Mass KG |
|---|---|---|---|
| **1.0 Structure** | **72** | **8.0 Other** | **31** |
| - Primary Structure | 57 | - Landing System | 29 |
| - Secondary Structure | 14 | - Pyrotechnics | 1 |
| - Subsystem Installation (info) | (30) | | |
| **2.0 Protection** | **15** | **9.0 Growth** | **44** |
| - Insulation | 10 | | |
| - Misc. Thermal Control | 6 | | |
| | | **CLL Lander Dry Mass** | **399** |
| **3.0 Propulsion** | **89** | | |
| - Integrated Propulsion System | 89 | **10.0 Non-Cargo** | **20** |
| | | - Reserve and Residual Fluids | 20 |
| **4.0 Power** | **51** | | |
| - Generation | 12 | **11.0 Cargo** | **64** |
| - Electrical Pwr Dist. & Control (EPDC) | 22 | - Payload | 64 |
| - Wiring | 18 | | |
| | | **CLL Lander Inert Mass** | **484** |
| **6.0 Avionics** | **98** | | |
| - Guidance, Navigation & Control (GNC) | 31 | 12.0 Non-Propellant (Consumables) | 0 |
| - Data Management System (DMS) | 22 | | |
| - Instrumentation | 0 | **13.0 Propellant** | **861** |
| - Communications & Tracking (C&T) | 46 | | |
| | | **CLL Lander Gross Mass** | **1,345** |

| CLL Jettison Hardware | | Tl. Mass KG |
|---|---|---|
| **1.0 Structure** | | 0 |
| - Subsystem Installation (info) | (11.6) | |
| **2.0 Protection** | | 0 |
| **3.0 Propulsion** | | 0 |
| **4.0 Power** | | 74 |
| - Generation | | 49 |
| - Electrical Pwr Dist. & Control (EPDC) | | 21 |
| - Wiring | | 5 |
| **5.0 Avionics** | | 0 |
| 8.0 Other | | 8 |
| - Mechanisms | | 4 |
| - Pyrotechnics | | 4 |
| 9.0 Growth | | 12 |
| CLL Jettison Hardware Dry Mass | | 94 |
| CLL Jettison Hardware Inert Mass | | 94 |
| CLL Jettison Hardware Gross Mass | | 94 |

| CLL Launch Adapter | Tl. Mass KG |
|---|---|
| **Total Mass** | 11 |

| CLL Total Launch Mass | Tl. Mass KG |
|---|---|
| **Total Launch Mass (Lander + Jettison Hardware + Launch Adapter)** | 1,450 |

| CLL: LANDER SUBSYSTEM: | Tl. Mass (KG) | No | Comments |
|---|---|---|---|
| **1.0 Structure:** | **71.6** | | |
| Primary Structure | 57.3 | | 4% of gross spacecraft mass .04(1431.5). Ratio of primary and secondary structure dependent on design. |
| Secondary Structure | 14.3 | | 1% of gross spacecraft mass .01(1431.5). |
| Subsystem Installation (info only) | 29.7 | | Subsystem Installation average percentage = 11.07% |

| CLL: LANDER SUBSYSTEM: | Tl. Mass (KG) | No | Comments |
|---|---|---|---|
| **2.0 Protection:** | **15.0** | | Contact Steve Rickman, x38867. Estimated. |
| Insulation | 9.5 | | |
| Tank insulation | 8.0 | | MLI tank wrap; assuming 4 70.9 cm diam. prop. tanks, 0.8 g/sq. in |
| Line Insulation | 1.5 | | MLI line wrap |
| Misc. Thermal Control Hardware | 5.5 | | |

| CLL: LANDER SUBSYSTEM: | Tl. Mass (KG) | No | Comments |
|---|---|---|---|
| **3.0 Propulsion:** | **88.8** | | Contact Don Hyatt, x39019. Performs partial TEI, LOI, MCC, att control, DD & L |
| Integrated Propulsion System | 88.8 | | Bipropellant RCS and Primary Engine System, Delta V=2872.7 m/s |
| Fuel Tanks | 13.6 | 2 | Spherical, 70.9 cm dia. |
| Oxidizer Tanks | 13.6 | 2 | Spherical, 70.9 cm dia. |
| Pressurant Tank | 9.1 | 1 | Spherical, 48.77 cm. (19.2") dia. |
| RCS Engines | 5.4 | 8 | Marquardt R-6C, 5 lbf thrust, ISP = 289 sec. @ AR = 100 |
| Descent Lander Engines | 30.1 | 8 | Marquardt R-4D, Isp=310 sec, 110 lbf thrust |
| Lines, Valves & Instrumentation | 9.8 | | Based on existing hardware, includes RCS isolation valve pyros |
| Installation - Mounting & Circuitry | 7.3 | | Structure factor of 9.0% supplied by George Sanger, 333-7254. |

| CLL: LANDER SUBSYSTEM: | Tl. Mass (KG) | No | Comments |
|---|---|---|---|
| **4.0 Power:** | **50.7** | | Contact Betsy Kluksdahl, x36484. |
| <u>Generation</u> | <u>11.6</u> | | |
| Secondary Batteries | 10.0 | 2 | No redundancy, divided into 2 boxes; 30.5x22.9x12.7 cm3 ea. |
| Installation - Mounting & Circuitry | 1.6 | | Structure factor of 15.6% supplied by George Sanger, 333-7254. |
| | | | |
| <u>Electrical Pwr Dist. & Control (EPDC)</u> | <u>21.5</u> | | Contact Betsy Kluksdahl, x36484. |
| Bus Controller | 18.6 | 1 | 38.1x38.1x15.2 cm |
| Installation - Mounting & Circuitry | 2.9 | | Structure factor of 15.6% supplied by George Sanger, 333-7254. |
| | | | |
| <u>Wiring</u> | <u>17.7</u> | | Contact Betsy Kluksdahl, x36484. |
| Wiring Harness | 15.3 | | Includes connectors, 7,000 cm3 |
| Installation - Mounting & Circuitry | 2.4 | | Structure factor of 15.6% supplied by George Sanger, 333-7254 |

| CLL: LANDER SUBSYSTEM: | Tl. Mass (KG) | No | Comments |
|---|---|---|---|
| **6.0 Avionics:** | **98.4** | | |
| <u>Guidance, Navigation and Control (GNC)</u> | <u>30.7</u> | | Contact Nancy Smith, x38275 |
| Inertial Navigation System | 9.1 | 1 | Honeywell 764 C3, Incl.2 CPU,3 Accel,3 RLGyro & GPS rcvr (+20W), 45W base [illegible] |
| Star Tracker | 1.0 | 1 | wide field of view, 12x12x15 cm, 2W/8Wpeak |
| Nav Base | 0.0 | 1 | ES will include in (primary) structure |
| RCS Jet Driver Electronics | 17.0 | 1 | GD estimate; drives 8 R4Ds and 8 R4Cs, 25x25x12.5 cm, 20W quiescent |
| 1553B Data Bus | 0.6 | 1 | 3048 cm. Mass based on vendor quote |
| Installation - Mounting & Circuitry | 3.0 | | factor of 10.8% on GNC equipment; see George Sanger, 333-7254 |
| | | | |
| <u>Data Management System (DMS)</u> | <u>22.0</u> | | Contact Dave Pruett, x35269. |
| General Avionics Processor | 10.0 | 1 | Vendor bid. 15.2 x 17.8 x 15.2 cm, 20.0 W |
| Time Generation Unit | 1.8 | 1 | 11.4x20.3x5.1 cm; 10 W |
| 1553B Data Bus | 0.6 | 1 | 3048 cm. Mass based on vendor quote |
| Timing Bus | 0.6 | 1 | 3048 cm. Mass comparable to 1553B bus. |
| Operational Instrumentation Bus | 6.2 | 1 | Est. Assumes 139 sensors, .5 vehicle circumference, 33.3k cm length |
| Installation - Mounting & Circuitry | 2.8 | | Structure factor of 14.5% supplied by George Sanger, 333-7254. |

| CLL: LANDER SUBSYSTEM: | Tl. Mass (KG) | No | Comments |
|---|---|---|---|
| **6.0 Avionics: (Continued)** | | | |
| Instrumentation | 0.0 | | Dist. among subsystems. |
| Sensors | 0.0 | | |
| Installation - Mounting & Circuitry | 0.0 | | Structure factor of 16.6% supplied by George Sanger, 333-7254. |
| | | | |
| Communications & Tracking (C&T) | 45.7 | | Contact Henry Chen, x30128. |
| • S-Band System | 23.2 | | Uses DSN 26m subnet, for telemetry, ranging and command. |
| Transponder | 4.1 | 1 | Motorolla Near Earth Xpndr,inc. cmd detector. 16x20x11cm, 3500 cm3, 4.5W recv., 40.5W trnsmt. |
| RF Assembly | 7.9 | 1 | 16x20x24 cm, 7800 cm3, 12.7W. |
| Processing Module | 6.0 | 1 | Process, signal condition, control and monitors. 18x18x26 cm, 8400 cm3, 27W |
| Antenna | 1.4 | 6 | Antenna, W-J, Log conical spiral.10 cm dia x 9 cm h,8640 cm3,0W |
| Coaxial Cable | 0.7 | 1 | Gore G2 cable, 900 cm3, (dependent on communication equip placement) |
| Installation - Mounting & Circuitry | 3.1 | | Structure factor of 15.6% supplied by George Sanger, 333-7254. |
| | | | |
| • Tracking | 22.6 | | Contact Bill Culpepper, x31479. Viking Heritage, Teledyne Ryan Co |
| Landing Radar | 16.6 | 3 | antenna is integral part of surface,3 38.1boxes, 8.26 cm ht, arranged as "T",51W |
| Altimeter | 4.1 | 1 | 23.4x14.7x20.1 cm, 27W |
| Altimeter Antenna | 0.7 | 1 | Conical horn,15.25 cm dia, 15.25 cm length, 1721 cm3 |
| Coax cable | 0.1 | 1 | Connection between altimeter and antenna. Estimate |
| GPS Antenna System | 0.0 | 0 | 1.8 kg, 12.7 x 12.7 x 5.1cm; 1W [OPTION] |
| GPS Antenna Switch | 0.0 | 0 | 0.3 kg, 5.1 x 5.1 x 2.5 cm, 1 W [OPTION] |
| GPS Antenna System Cables | 0.0 | 0 | ? kg; 1 control, 2 power, 3 RF [OPTION] |
| Installation - Mounting & Circuitry | 1.1 | | Structure factor of 5.0% supplied by George Sanger, 333-7254 |

| CLL: LANDER SUBSYSTEM: | Tl. Mass (KG) | | No | Comments |
|---|---|---|---|---|
| **8.0 Other:** | **30.6** | | | |
| Landing System | 29.2 | | | Contact George Sanger, LESC, 333-7254. |
| Lander Legs | | 24.0 | 4 | Alum. honeycomb. 6 kg each. |
| Installation - Mounting & Circuitry | | 5.2 | | Structure factor of 21.6% supplied by George Sanger, 333-7254. |
| Pyrotechnics System | 1.4 | | | Contact Betsy Kluksdahl, x36484. |
| RCS Isolation | | 0.0 | 0 | Mass carried in propulsion system |
| Landing Strut Deployment | | 1.0 | 4 | Uplock cutters,Apollo,16x12x5 cm,960 cm3,5A @ 30 mSec peak pwr. |
| Installation - Mounting & Circuitry | | 0.4 | | Structure factor of 38.5% supplied by George Sanger, 333-7254. |

| CLL: LANDER SUBSYSTEM: | Tl. Mass (KG) | | No | Comments |
|---|---|---|---|---|
| **9.0 Growth:** | **44.3** | | | 15% of all subsystems (2.5% on existing hardware) |
| 1.0 Structure | | 10.7 | | |
| 2.0 Protection | | 2.3 | | |
| 3.0 Propulsion | | 6.5 | | 2.5% growth calculated on engine, pressurant tank, & LV&I mass |
| 4.0 Power | | 7.6 | | |
| 6.0 Avionics (GN&C, DMS, Instr., C&T) | | 12.7 | | 2.5% growth calculated on INS,star track,s band transp&ant, alt ant |
| 8.0 Other (Landing sys., Pyrotechnics) | | 4.5 | | 2.5% growth calculated on N/C pyrovalves & uplock cutters |

| | |
|---|---|
| CLL Lander DRY MASS | 399 |

| CLL: LANDER SUBSYSTEM: | Tl. Mass (KG) | | No | Comments |
|---|---|---|---|---|
| **10.0 Non-Cargo** | **20.1** | | | |
| Reserve and Residual Fluids | 20.1 | | | |
| IPS Fuel Reserves | | 0.0 | | 0% reserves, D. Hyatt. |
| IPS Fuel Residuals | | 6.6 | | 2% residuals, Monomethylhydrazine (MMH), Contact D. Hyatt. |
| IPS Oxidizer Reserves | | 0.0 | | 0% reserves, D. Hyatt. |
| IPS Oxidizer Residuals | | 10.9 | | 2% residuals, Nitrogen Tetroxide (NTO), Contact D. Hyatt. |
| IPS Pressurant | | 2.5 | | Helium, D. Hyatt. |

| CLL: LANDER SUBSYSTEM: | Tl. Mass (KG) | No | Comments |
|---|---|---|---|
| 11.0 Cargo | 64.1 | | {Delta 7925 Lift Capability (1450 kg)} - {lander+jettison+LV adapter} |
| Payload | 64.1 | | |

| | |
|---|---|
| CLL Lander INERT MASS | 484 |

| CLL: LANDER SUBSYSTEM: | Tl. Mass (KG) | No | Comments |
|---|---|---|---|
| 12.0 Non-Propellant (Consumables) | 0.0 | | |

| CLL: LANDER SUBSYSTEM: | Tl. Mass (KG) | No | Comments |
|---|---|---|---|
| 13.0 Propellant | 860.9 | | Delta V = 2872.7 m/s |
| Fuel | 324.9 | | Monomethylhydrazine (MMH) |
| Oxidizer | 536.0 | | Nitrogen Tetroxide (NTO) |

| | |
|---|---|
| CLL Lander GROSS MASS | 1,345 |

| CLL: JETTISON HARDWARE SUBSYSTEM: | Tl. Mass (KG) | | No | Comments |
|---|---|---|---|---|
| **4.0 Power:** | **74.3** | | | Contact Betsy Kluksdahl, x36484. |
| Generation | 48.9 | | | |
| Solar Arrays | | 35.0 | 2 | Silicon cells, accordian-style, 10.4 m2 total (non tracking), sized for 700 W total |
| Solar Array Container | | 7.3 | 2 | 1 box for each array, 152.4x30.48x15.2 cm |
| Installation - Mounting & Circuitry | | 6.6 | | Structure factor of 15.6% supplied by George Sanger, 333-7254 |
| Electrical Pwr Dist. & Control (EPDC) | 20.8 | | | Contact Betsy Kluksdahl, x36484. |
| Solar Array Controller | | 9.0 | 1 | 25.4x38.1x15.2 cm |
| Battery Charger | | 9.0 | 1 | 25.4x30.5x12.7 cm |
| Installation - Mounting & Circuitry | | 2.8 | | Structure factor of 15.6% supplied by George Sanger, 333-7254 |
| Wiring | 4.6 | | | Contact Betsy Kluksdahl, x36484. |
| Wiring Harness | | 4.0 | | Includes connectors, 2300 cm3 |
| Installation - Mounting & Circuitry | | 0.6 | | Structure factor of 15.6% supplied by George Sanger, 333-7254 |

| CLL: JETTISON HARDWARE SUBSYSTEM: | Tl. Mass (KG) | | No | Comments |
|---|---|---|---|---|
| 8.0 Other: | 7.8 | | | |
| Mechanisms | 4.2 | | | Contact Tim Pelischek, x38843 |
| Solar Array Deployment | | 3.6 | 2 | Telescoping booms for load support, No Tracking |
| Installation - Mounting & Circuitry | | 0.6 | | Structure factor of 15.6% supplied by George Sanger, 333-7254 |
| Pyrotechnic System | 3.6 | | | Contact Betsy Kluksdahl x36484. |
| Solar Array Deployment | | 0.2 | 2 | Pin Pullers; 7.6x2.5x3.0 cm each (2 arrays) |
| Solar Array Jettison | | 0.9 | 8 | Explosive bolts; 10 cm x 2.5 cm dia. ea. |
| Severing of Electric Wire Bundle | | 1.5 | 2 | Guillotine; 25x20x9 cm; STS Kuband |
| Installation - Mounting & Circuitry | | 1.0 | | Structure factor of 38.5% supplied by George Sanger, 333-7254 |

| CLL: JETTISON HARDWARE SUBSYSTEM: | Tl. Mass (KG) | No | Comments |
|---|---|---|---|
| **9.0 Growth:** | 12.0 | | 15% growth |
| 4.0 Power (incl. EPDC) | 11.1 | | |
| 8.0 Other (Mechanisms & Pyrotechnics) | 0.8 | | 2.5% growth on pyrotechnic components |

| | | |
|---|---|---|
| **CLL Jettison Hardware DRY MASS** | 94 | |

| | | |
|---|---|---|
| **CLL Jettison Hardware INERT MASS** | 94 | |

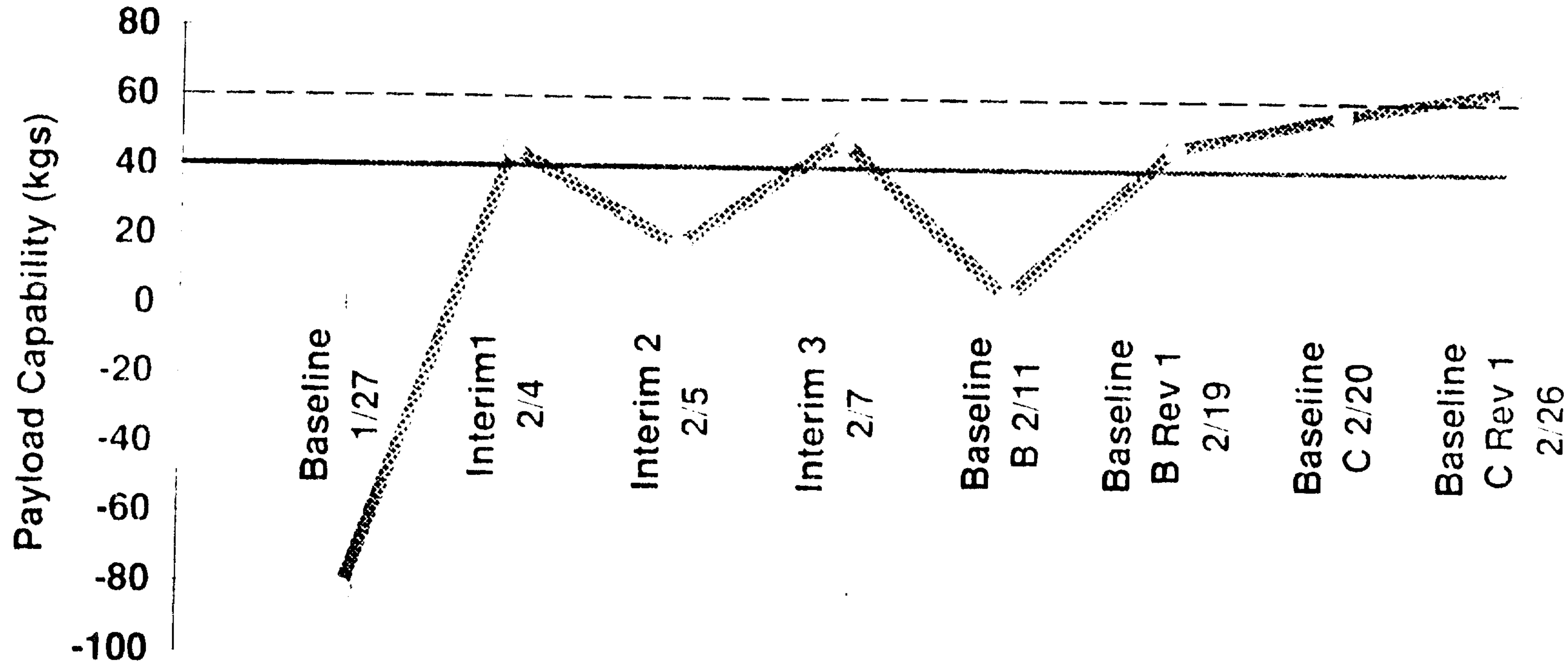
| | | |
|---|---|---|
| **CLL Jettison Hardware GROSS MASS** | 94 | |

CLL: Launch Adapter & Support Equipment

| SUBSYSTEM: | Tl. Mass (KG) | No | Comments<br>Adapter between CLL spacecraft and LV PAF |
|---|---|---|---|
| **1.0 Structure** | 10.9 | | Similar to PAF. |
| **8.0 Other** | 0.4 | 4 | Stage separation, explosive bolts; 10 cm x 2.5 cm dia each |
| **9.0 Growth** | 0.0 | | |

| | | |
|---|---|---|
| **Launch Adapter & Support** | 11 | |

# Common Lunar Lander Change in Payload Capability



# Reliability Study

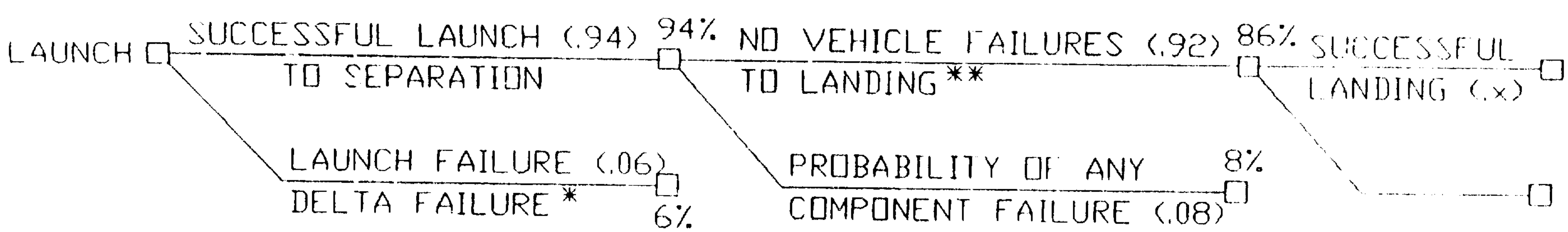
Artemis

- **Mean Time Between Failure Study**
  - Based on best available MTBF data
  - Duty cycles assumed for Tracking System and Star Tracker; all other hardware assumed continuously operating
  - Produces a conservative estimate of mission success
    - More aptly termed "probability of failure free operation"
      - Exponential failure distribution was assumed
      - Pass/fail study
  - **Result**
    - **6 day mission 92 %**
    - **9 day 87 %**
    - **12 day 82 %**
- **Surveyor (for comparison)**
  - Goal (by MTBF): 75 %
  - Measured (based on reliability and performance data): 86 %

Artemis

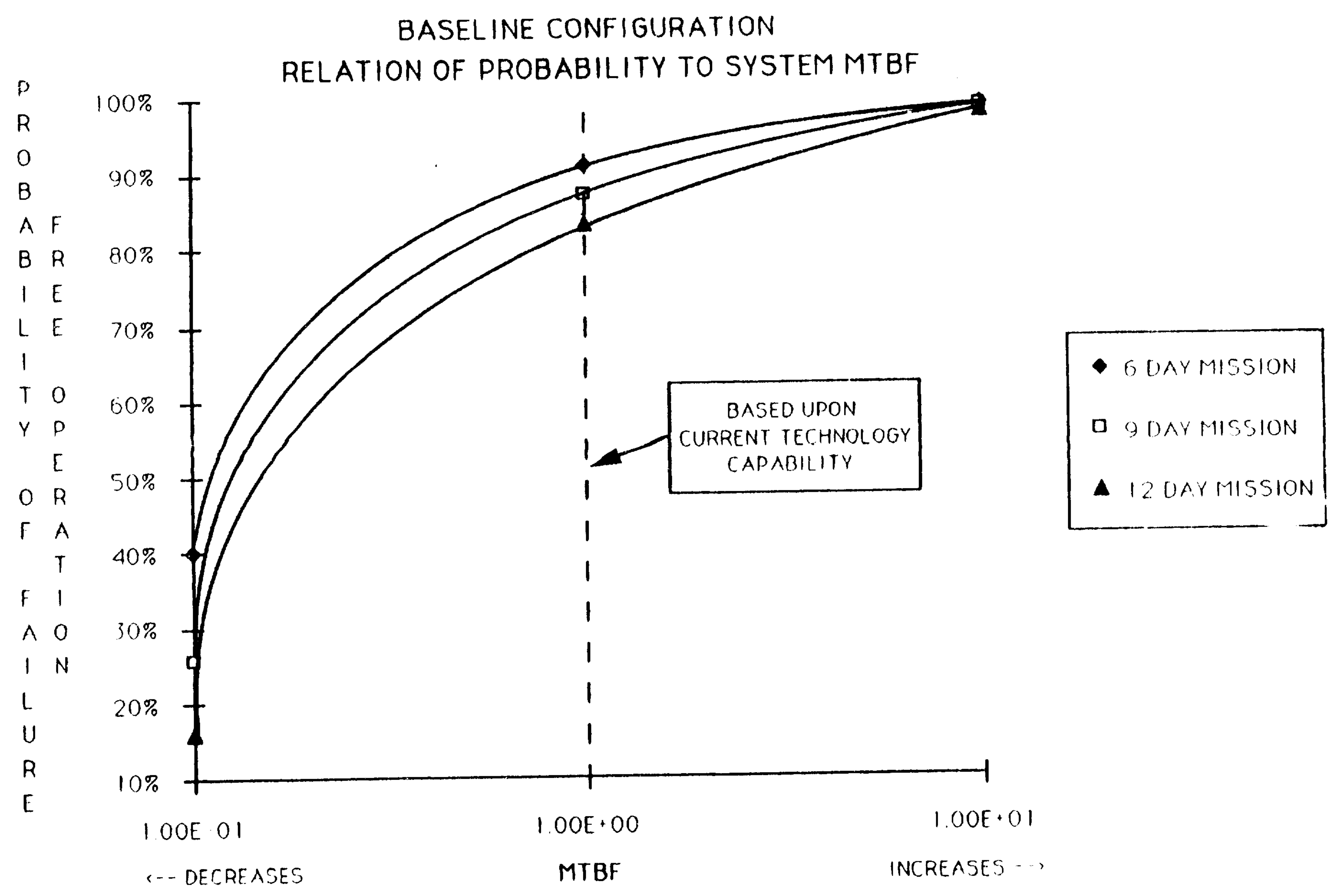
# Reliability Study

## Fault Tree: Launch to Landing




*Based on historical launch data, presented in: Air Force Report # AL-TR-89-013

**Assuming a six day mission, and perfect vehicle operation



BASELINE CONFIGURATION
RELATION OF PROBABILITY TO SYSTEM MTBF
PROBABILITY OF FAILURE FREE OPERATION
100%
90%
80%
70%
60%
50%
40%
30%
20%
10%
1.00E-01
1.00E+00
1.00E+01
<-- DECREASES
MTBF
INCREASES -->
BASED UPON CURRENT TECHNOLOGY CAPABILITY
6 DAY MISSION
9 DAY MISSION
12 DAY MISSION

# CLL Phase 2 Study: Results

## III. WRAP-UP

- Conclusions
- Performance Options

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# CONCLUSIONS

## SPACECRAFT DESIGN

### DESIGN MATURITY

- Avionics, Propulsion relatively mature, given design assump.
- Power Subsystem fluctuates with changing power rqmts.
- Structure, Thermal Subsystems in early definition stage

## ENGINEERING MASTER SCHEDULE

### TESTING STRATEGY

- Subsystem level
- Integrated avionics testing
- Integrated System Testing

### EMS MATURITY

- 3 iterations completed on details
- Need to stand back and review the whole
- Is there a better way to do business?

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# Performance Options

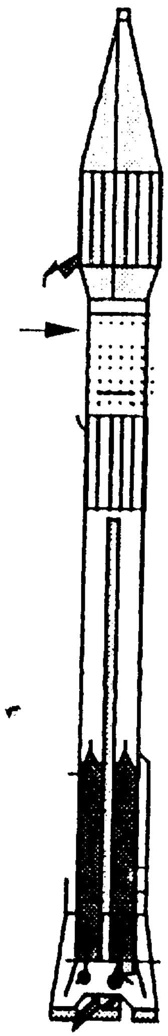
- ARCHITECTURAL OPTIONS
  - Multi-stage CLL Vehicle
    - Considered staging opportunities
      - add LOI Solid
      - add Descent Solid
      - add Drop Tanks
    - Added Complexity
    - No Performance Benefit
  - Adding a liquid first stage would increase performance
    - Also increases development and recurring costs
    - Potential schedule increase
    - See CLL Phase 1 Study for performance

# Performance Options

Artemis

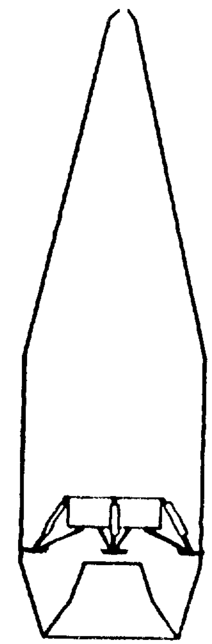
- **LAUNCH VEHICLE OPTIONS** (Preliminary)
  - **One-stage spacecraft, based on CLL Phase 2 Baseline B, Rev. 1 design**
  - **Designed specifically for each of the following launch vehicles**
    - **Delta** — **54 kg lunar payload**
    - **Atlas II** — **196 kg lunar payload**
    - **Atlas IIA** — **215 kg lunar payload**
    - **Atlas IIAS** — **375 kg lunar payload**

- **GROWTH OPTIONS** (Preliminary)
  - **Based on CLL Phase 2 Baseline C, Rev. 1 design**
  - **Design for Atlas II, with LOI Solid (Isp=260 sec, MF = 0.7), 4 kg scar wt.**
    - **Delta** — **60 kg lunar payload**
    - **Atlas II + LOI Solid** — **132 kg lunar payload**
  - **Design for Atlas II, 8 R-4Ds for landing**
    - **Atlas I, off-loaded propellant** — **54 kg lunar payload**
    - **Atlas II** — **230 kg lunar payload**




General Dynamics
Atlas II

One-Stage



High accuracy 3 axis TLI provided by Centaur

Lander performs all subsequent maneuvers

Growth possibilities up through Atlas IIAS

Jonette Stecklein/ET2/x36624

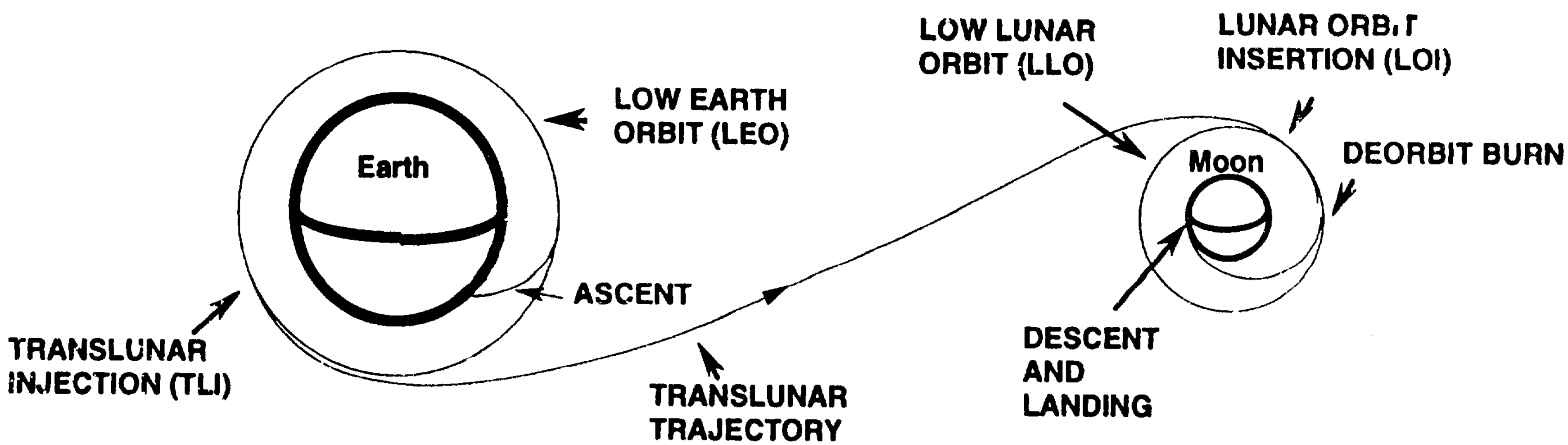
# COMMON LUNAR LANDER TRAJECTORY ANALYSIS

Lynn A. Wagner, Jr./ET3
Lee Bryant/ET4
March 10, 1992

# COMMON LUNAR LANDER ASSUMPTIONS

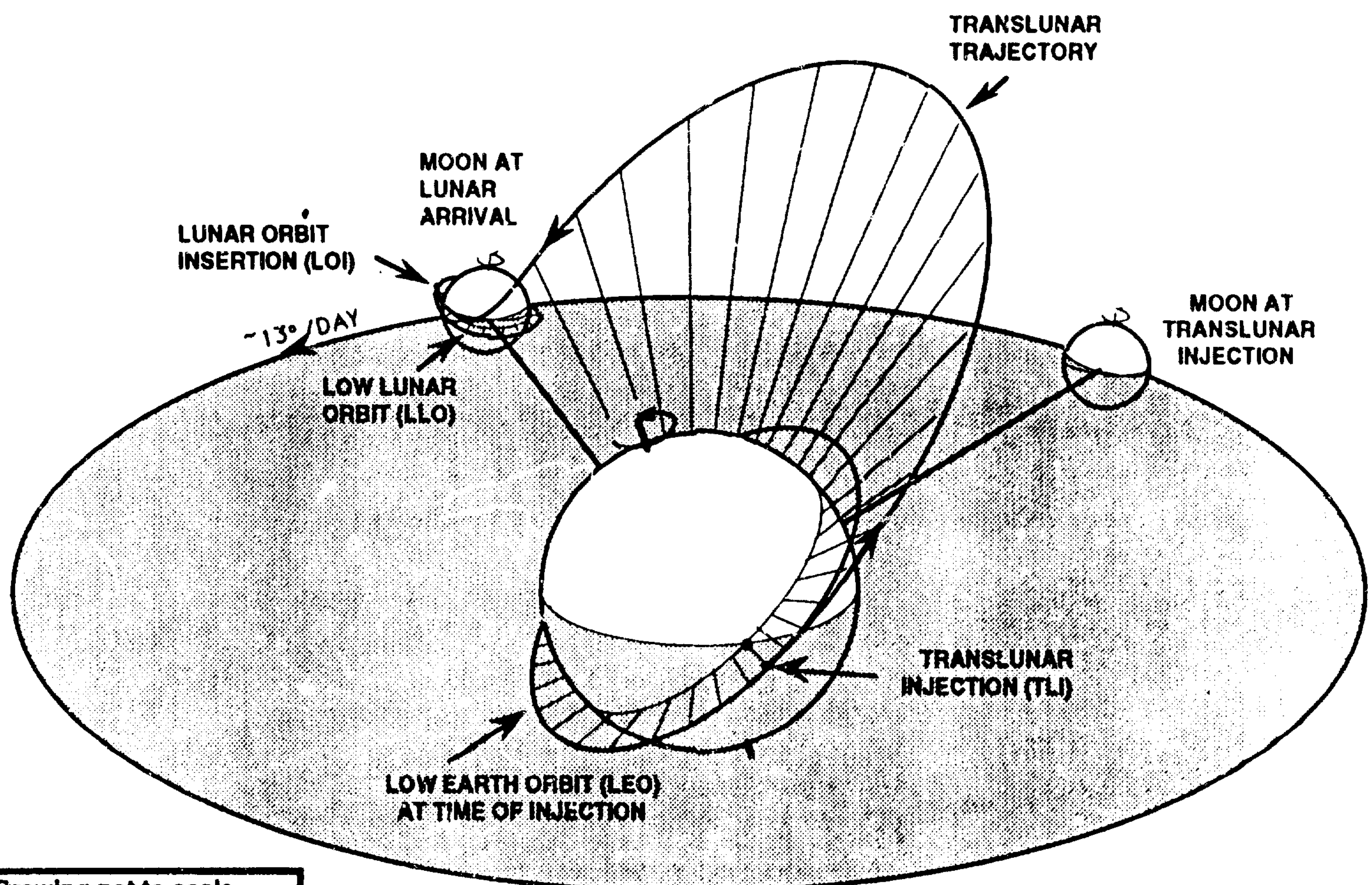
- Earth Parking Orbit
  - Due east launch from ETR into a 28.45 deg inclination and varying the inclination for a launch window
  - 185 km circular orbit
- Translunar Trajectories
  - Approximately 4-5 day transfer times (minimum energy)
- Lunar Parking Orbit
  - 100 km circular orbit
  - To help minimize deorbit, descent , and landing delta-V cost
- All lunar landing sites are accessible

# COMMON LUNAR LANDER TRAJECTORY




Drawing not to scale

# COMMON LUNAR LANDER TRAJECTORY




Drawing not to scale

# COMMON LUNAR LANDER TRAJECTORY TIMELINE

| TRAJECTORY EVENT | DURATION | DELTA-V |
|---|---|---|
| Launch | 20-30 min | - - - |
| Earth Parking Orbit Coast (185 km Circular Orbit) | 0-90 min | - - - |
| Translunar Injection (TLI) | - - - | $C3 = -4.4\ km^2/sec^2$ (Earth relative C3) |
| Separation Maneuver/ TLI Continuation | - - - | 102.1 m/sec (minimum) |
| Translunar Coast | 5 days | 30 m/s (Midcourse Corrections) |
| Lunar Orbit Insertion (LOI) | - - - | $C3 = 1.0\ km^2/sec^2$ (887 m/sec) (Moon relative C3) |
| Lunar Parking Orbit Coast (100 km Circular Orbit) | 1 day (nominal) | (Orbit Maintenance = 0 m/sec) |
| Deorbit | 60 min | 18.6 m/s |
| Descent and Landing | 6 min | 1835 m/s |

Note: C3 is the energy of the orbit per per unit mass of the spacecraft

# DELTA-V BUDGET CONSIDERATIONS

- For LOI, a lunar C3 of 1.0 $km^2/sec^2$ was assumed
  - Provides capability to transfer to the Moon at anytime during the 18.6 year lunar cycle
  - Provides a small triptime window for the worst case lunar conditions (larger triptime window when the Moon is in a better geometry)
  - In general when the Moon is at perigee, the LOI delta-V is highest while the TLI delta-V is lowest. (The reverse is also true for when the Moon is at apogee.)
- Total delta-V budget based on delta-V required when the Moon is at perigee and worst lunar conditions
- TLI delta-V split
  - The TLI burn is accomplished in three stages
    - The Delta-II second stage initiates the maneuver
    - The Star-48B (solid) continues the burn until the Earth C3 is -4.4 $km^2/sec^2$
    - The CLL finishes the burn to the proper Earth C3 for the translunar trajectory
  - TLI split accomplished by
    - Using part of the LOI delta-V budget to perform part of the TLI maneuver (when Moon is not at perigee)
    - By loading the propellant tanks full and using the extra fuel to augment the TLI maneuver

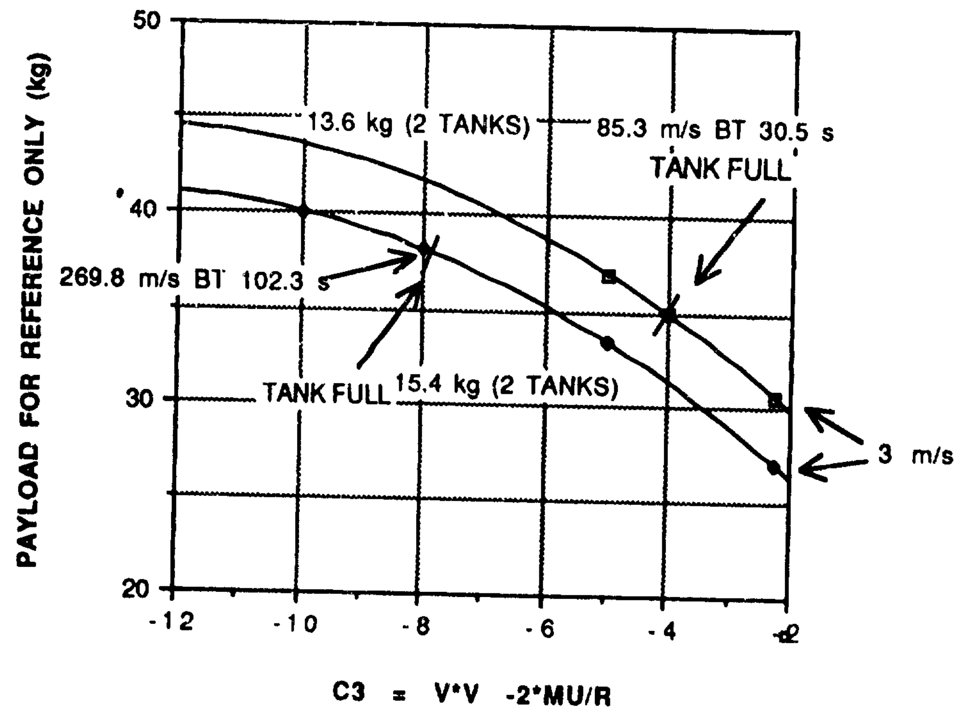
# PAYLOAD INCREASE (CURRENT PLAN)

- Delta-II quotes are given for a Earth C3 of -1.0 $km^2/sec^2$
- To increase the performance several changes were incorporated into the initial plan (mass increases to CLL include payload, propellant, and structural masses)
  - Strip the spin table of 22.7 kg unnecessary mass and move to CLL will reduce Earth C3 to -1.485 $km^2/sec^2$
  - Additional mass of 21.3 kg to CLL will reduce Earth C3 to -2.245 $km^2/sec^2$
  - Tanks were load full and FPR split (additional mass of 63 kg to CLL) to reduce Earth C3 to -4.4 $km^2/sec^2$
- Total CLL injected mass is now 1450 kg (5 kg of this will be used for launch window)
  - Launch window is 2.8 hours with a 1.4 hours hole in the middle of the window due to range safety reasons

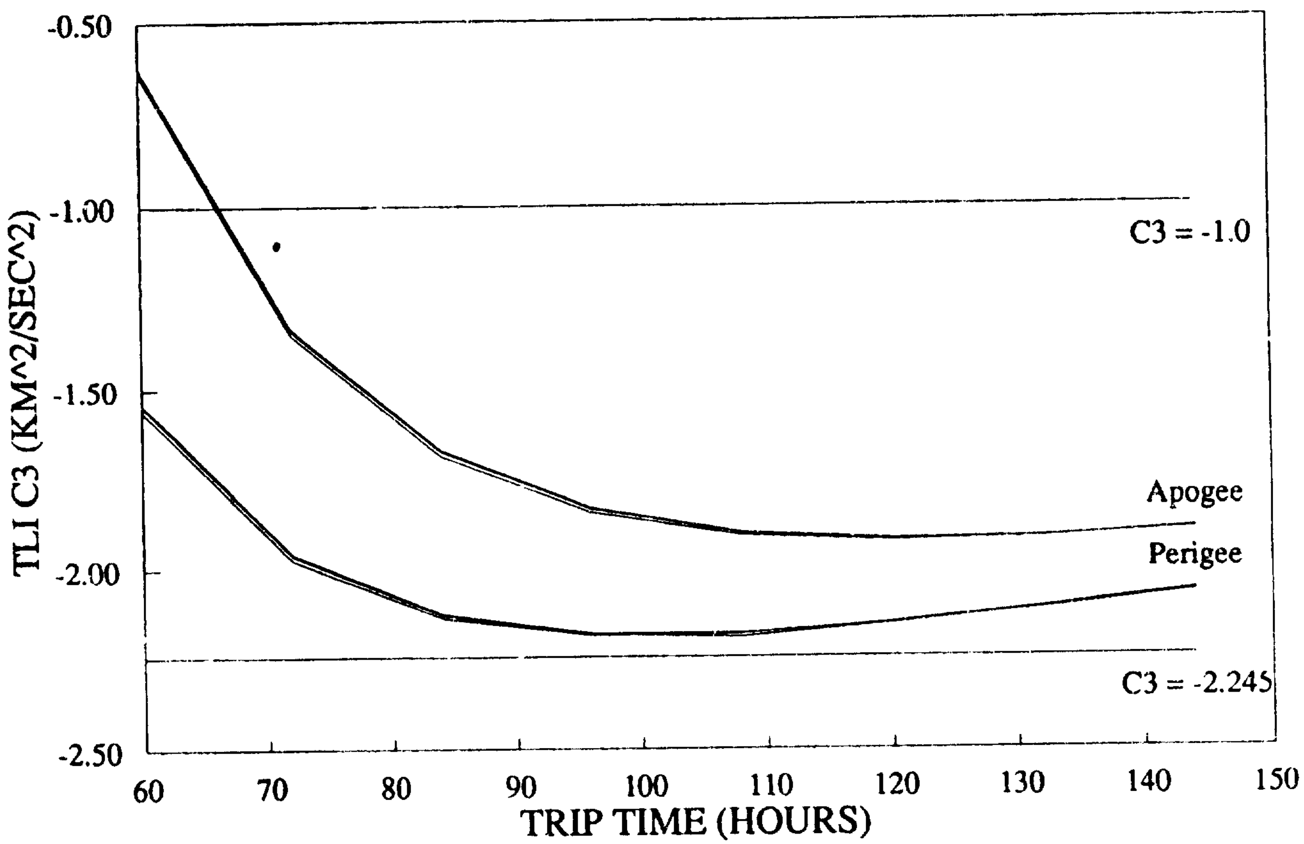
# PAYLOAD INCREASE CONSIDERATIONS

- Removing two of the eight engines (5.1 kg payload increase)
  - THIS WILL BE A PROBLEM FOR ATTITUDE CONTROL DURING TLI AND IS NOT RECOMMENDED FOR THIS CONFIGURATION
- Use drop tanks for separation, mid-course, LOI and deorbit (11.1 kg payload increase subject to additional hardware and residuals)
  - POTENTIAL LEAK PROBLEM WITH THE VALVES AND ADDED COMPLEXITY
- Use a solid for the large portion of descent then finish up with the liquid system
  - PERFORMANCE DATA DELIVERED TO PROPULSION (GAIN IS NOT EXPECTED)
- Propellant tanks can be enlarged by inserting a 2-inch cylindrical band. The new tank will hold about 980 kg of usable propellant (current tanks hold 860 kg of usable payload). (3.0 kg payload increase)
  - NOT INCLUDED IN THE BASELINE

# TLI SPLIT (2ND STAGE, 3RD STAGE, AND CLL)

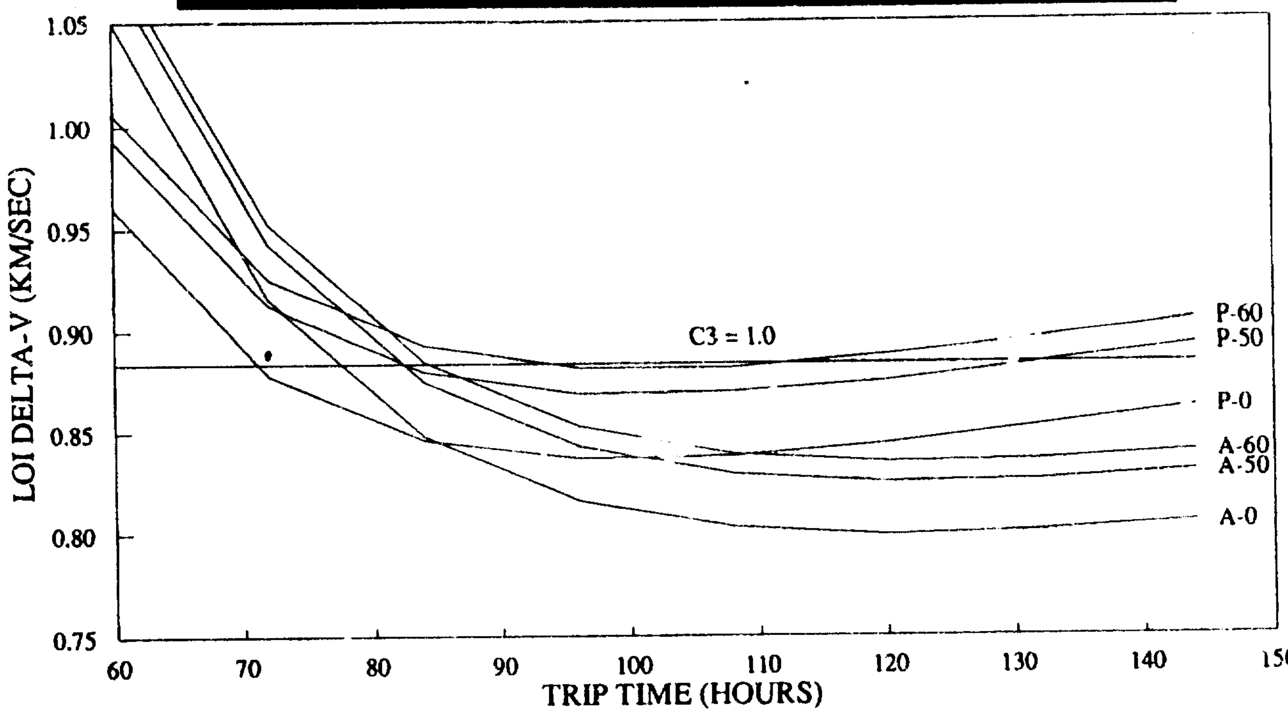


# TRANSLUNAR INJECTION (TLI)




Apogee/Perigee indicates the position of the Moon at lunar arrival time

# LUNAR ORBIT INSERTION (LOI)




Note 1: Gravity Loss was not included in the Delta-V

Note 2: The labels on the side of the LOI Delta-V curves indicates the geometry of the translunar trajectory

a. The letter indicates the Moon's geometry:

A - The Moon is at apogee at lunar arrival time

P - The Moon is at perigee at lunar arrival time

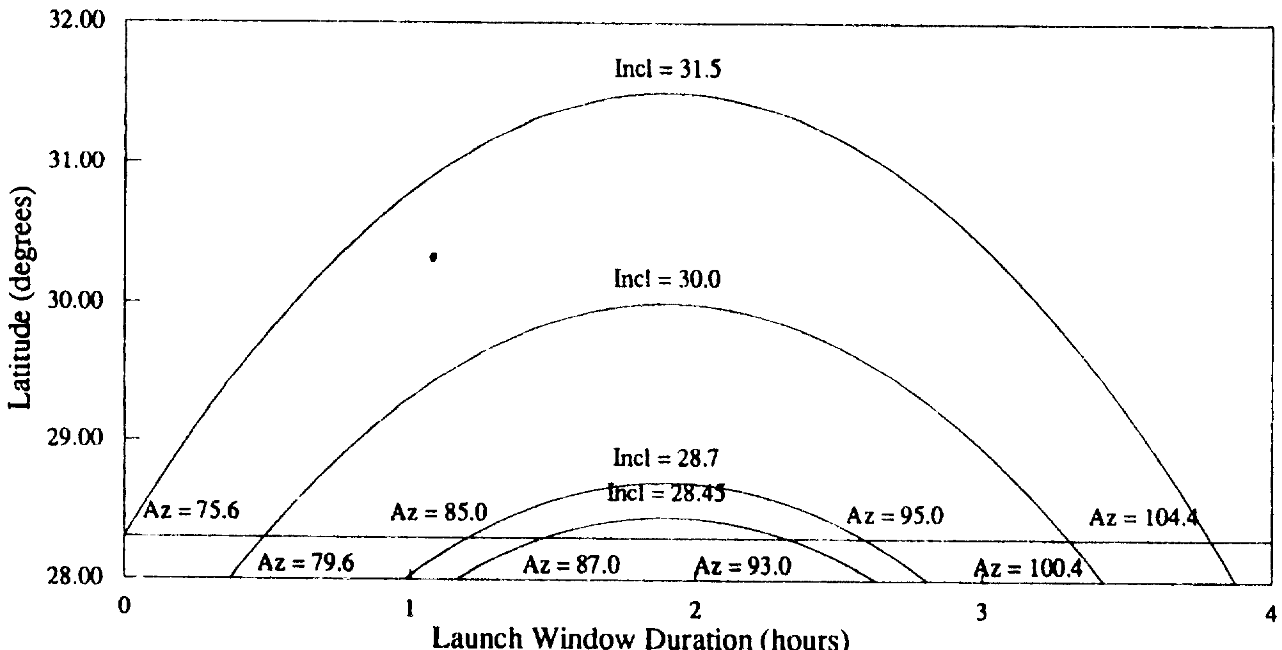
b. The number indicates the wedge angle between the Earth parking orbit (EPO) and the Earth Moon plane (EMP):

0 - The EPO lies in the Earth-Moon plane

50 - Maximum wedge angle assumming that the Moon is at minimum inclination (18.5 degrees) and the EPO is at 31.5 degrees inclination

60 - Maximum wedge angle assumming that the Moon is at maximum inclination (28.5 degrees) and the EPO is at 31.5 degrees inclination

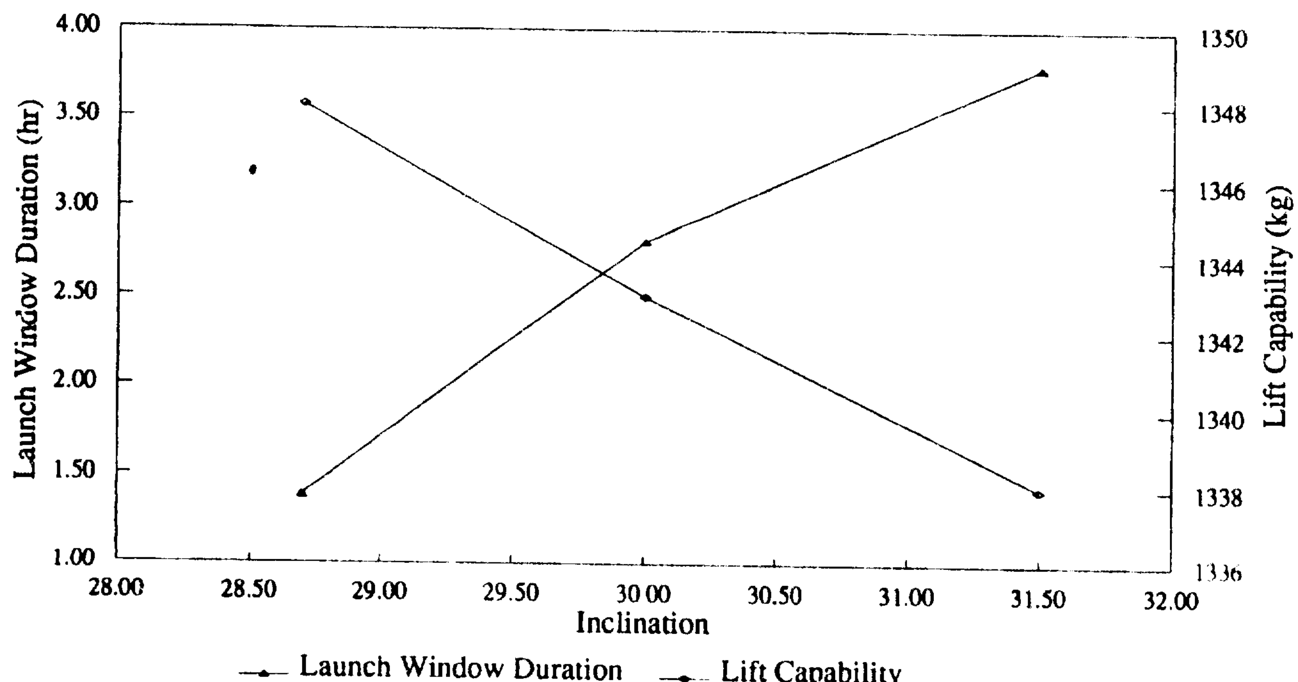
# CLL LAUNCH WINDOW




The given Launch Azimuths (Az) pertain to where the indicated orbit (curves labeled by their inclinations) intersect the horizontal Launch Site line (For the Delta II, the Launch Site is at 28.3 degrees Latitude)

| Inclination (degrees) | Launch Window Duration (hours) |
|---|---|
| 28.45 | 0.851643 |
| 28.70 | 1.386354 |
| 30.00 | 2.812842 |
| 31.57 | 3.792104 |

# LAUNCH WINDOW CONSIDERATIONS




Note : TLI C3 = -1.0 km^2/sec^2

STRUCTURES AND MECHANICS DIVISION

- TASK SUMMARY
- DESIGN-TO REQUIREMENTS
- VEHICLE DESCRIPTION
- STRUCTURAL DESIGN
- MECHANICAL DESIGN
- POTENTIAL WEIGHT SAVINGS OPTIONS
- ISSUES, IDEAS, CONCERNS

# SMD TEAM ACKNOWLEDGEMENT

JIM BRADLEY

TIM CAO

TONY DAO

PETER FANTASIA

BARBARA FOWLER

DAVE HAMILTON

GUY KING

ROBERT MCELYA

MIKE PEDLEY

TIM PELISCHEK

ALLEN REED

STEVE RICKMAN

## SMD TASKS

- STRUCTURE LAYOUT
- OVERALL LAYOUT
- DELTA II INTERFACE
- SUBSYSTEM INTEGRATION INTO LAYOUT
- PRELIMINARY LOADS ANALYSIS
- PRELIMINARY FINITE ELEMENT ANALYSIS
- HISTORICAL WEIGHT ESTIMATE
  - ENGINEERING MASTER SCHEDULE INPUTS
- REQUIREMENTS DEVELOPMENT/DEFINITION
- MANPOWER ESTIMATE
- TEST PROGRAM PHILOSOPHY
- PRELIMINARY LANDING STABILITY ANALYSIS
- IDENTIFIED HARDWARE TO DESIGN
  - GSE
  - TEST FIXTURES
  - FLIGHT HARDWARE
  - TEST HARDWARE

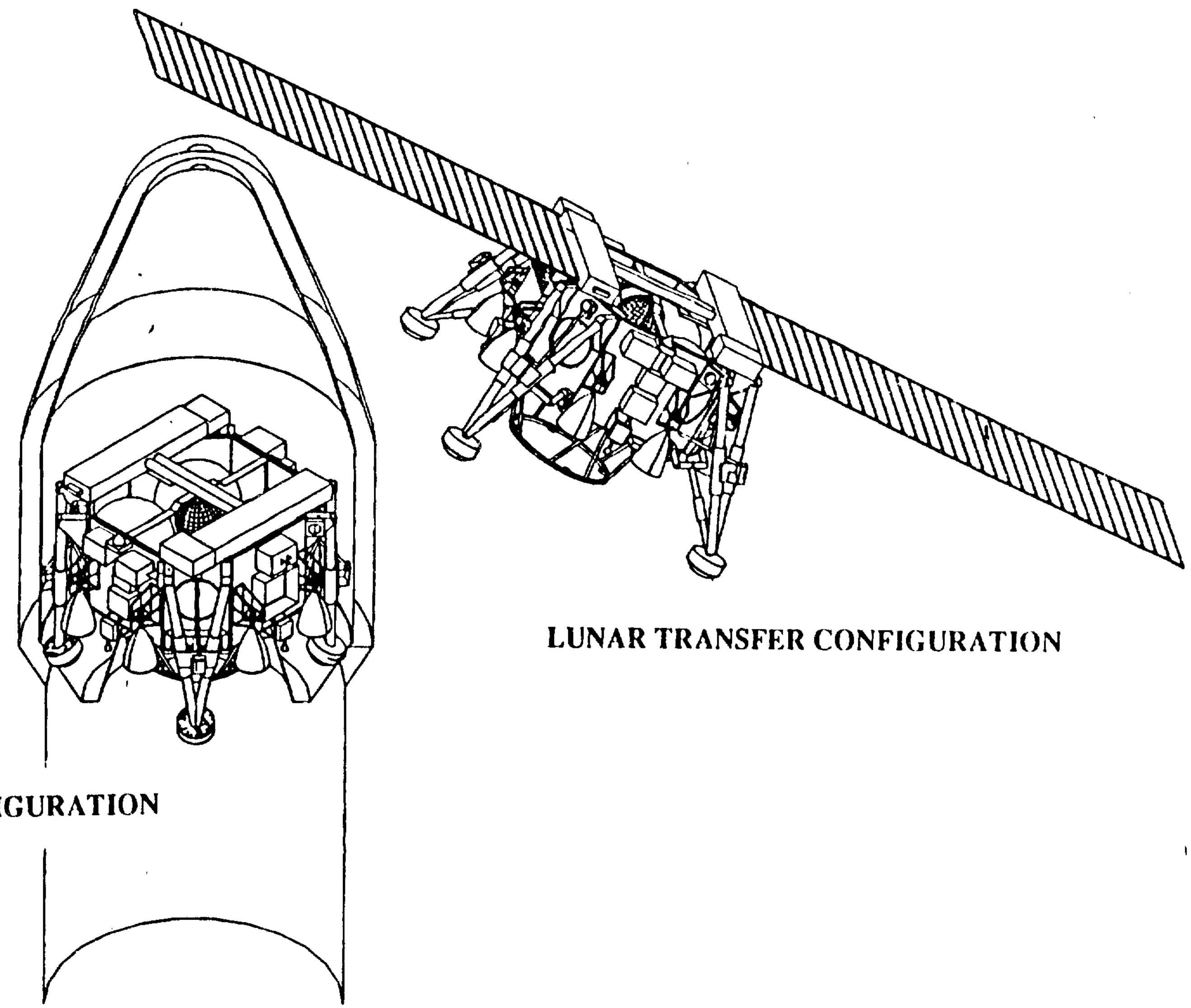
SMD TASKS (CONT'D)

- CONCEPTUAL DESIGN
  - LANDING GEAR
  - SOLAR PANEL DEPLOYMENT
  - TRANSITION STRUCTURE
  - ENGINE MOUNTS
- STRUCTURAL DESIGN OPTIMIZATION BRAINSTORMING
- THERMAL ASSESSMENT
- DEFINE THERMAL REQUIREMENTS
- TRADE SOLAR INERTIAL VS. PASSIVE THERMAL CONTROL FOR ENGINES
- HEATER POWER ESTIMATE FOR TANKS, ENGINES, LINES
- TCS WEIGHT ESTIMATE
- TCS CONCEPTUAL DESIGN

## STRUCTURES AND MECHANICS DESIGN-TO REQUIREMENTS

- 4 TANKS -- 4 LEGS FOR BI-SYMMETRY, AVAILABLE TANK
- 8 ENGINES , 8 RCS
- PAYLOAD ON TOP WITH 360 DEG. VIEW AND ON SIDES
- LANDING SINK RATE 15 FPS, 5 FPS HOR. 15 DEG. SLOPE
- SUPPORT ALL SUBSYSTEMS
- DELTA II SHROUD ENVELOPE
- DEPLOYABLE SOLAR ARRAYS
- SUBSYSTEM ACCESS ON PAD
- SOLAR INERTIAL
- STABILIZE, LOWER, LEVEL (DESIRABLE)
- TECH. SERVICES FAB
- OFF-THE-SHELF COMPONENTS AND TECHNOLOGY
- PASSIVE TCS
- SUBSYSTEM REQUIREMENTS

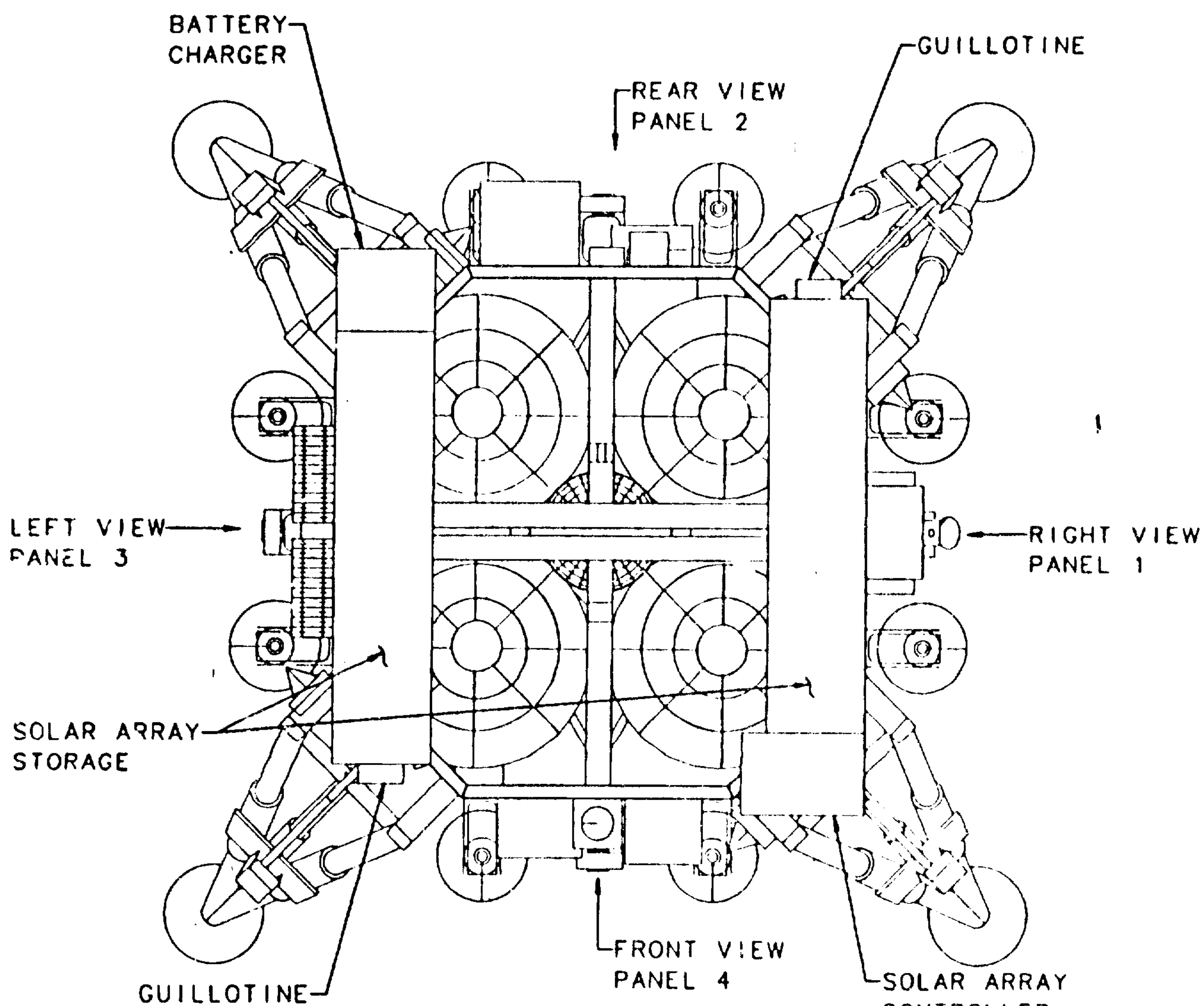
# COMMON LUNAR LANDER



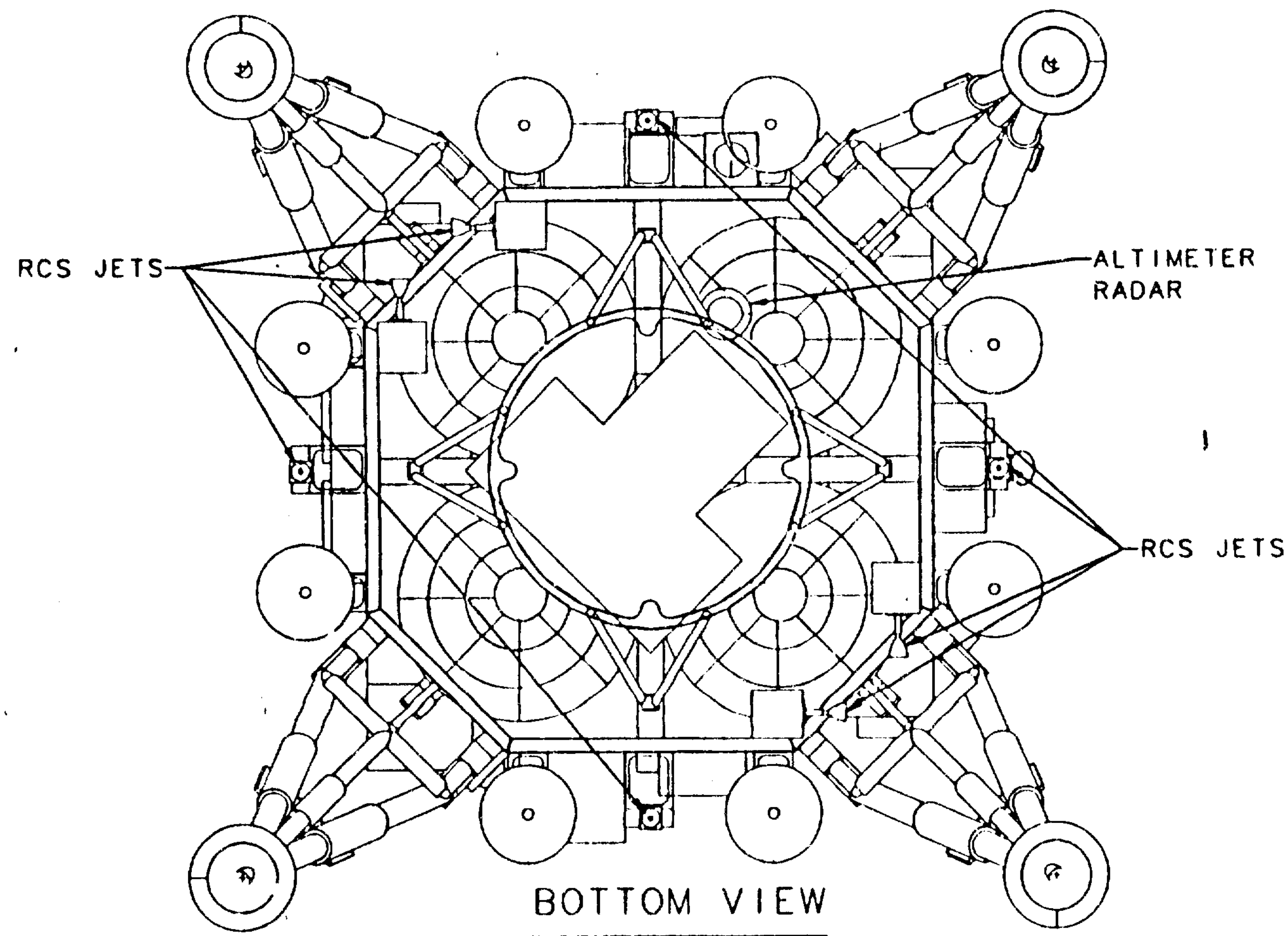
LAUNCH CONFIGURATION

LUNAR TRANSFER CONFIGURATION

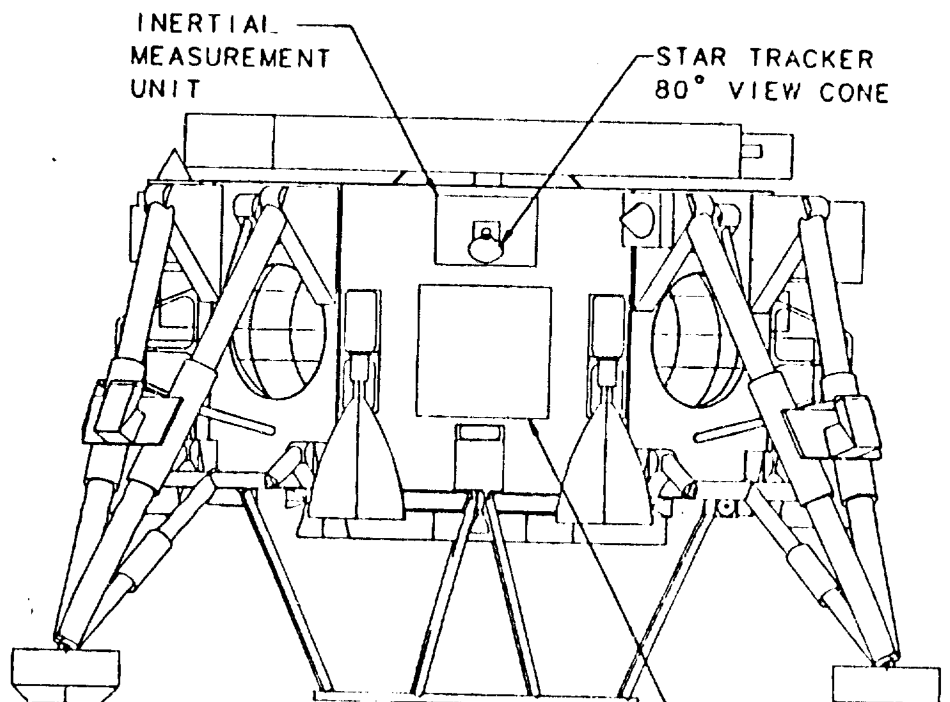
Tim Pelischek/ES2/x38843


BATTERY CHARGER
GUILLOTINE
REAR VIEW PANEL 2
LEFT VIEW PANEL 3
RIGHT VIEW PANEL 1
SOLAR ARRAY STORAGE
GUILLOTINE
FRONT VIEW PANEL 4
SOLAR ARRAY CONTROLLER

TOP VIEW

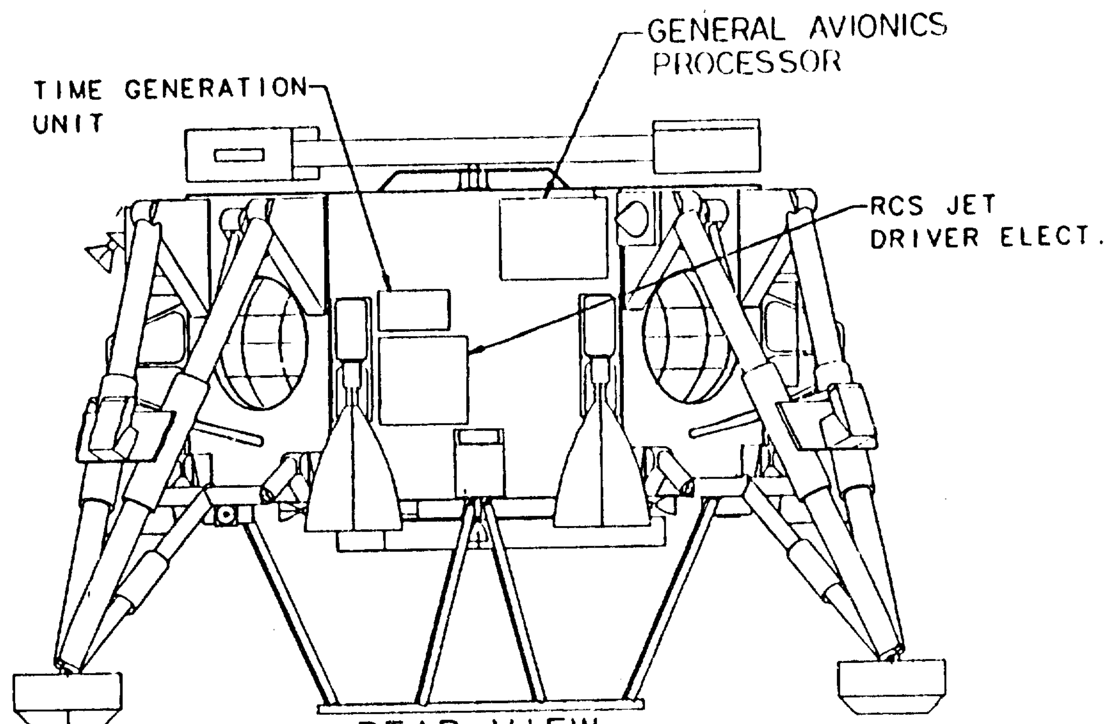



BOTTOM VIEW



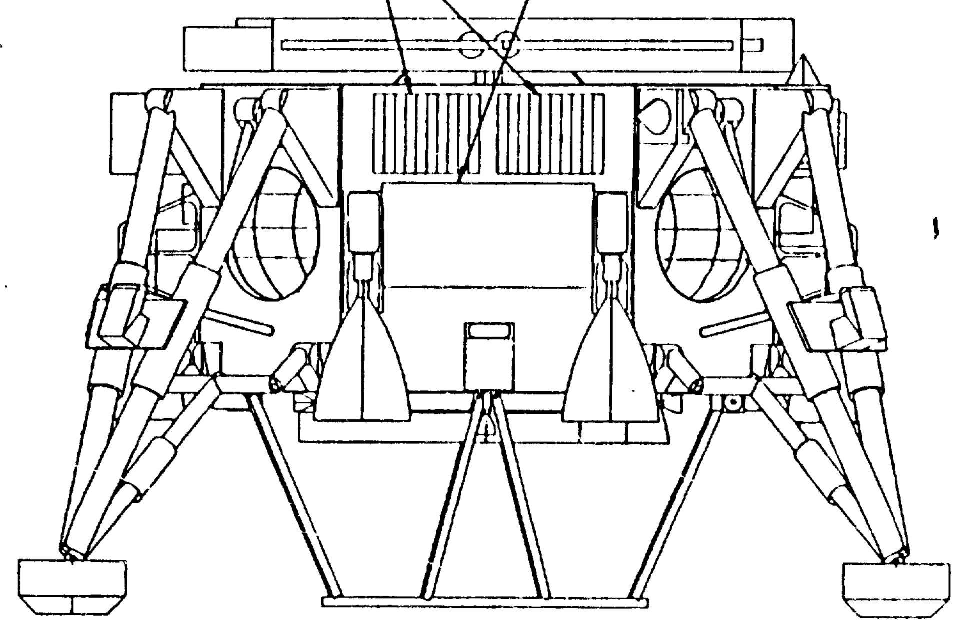

RIGHT VIEW

PANEL 1



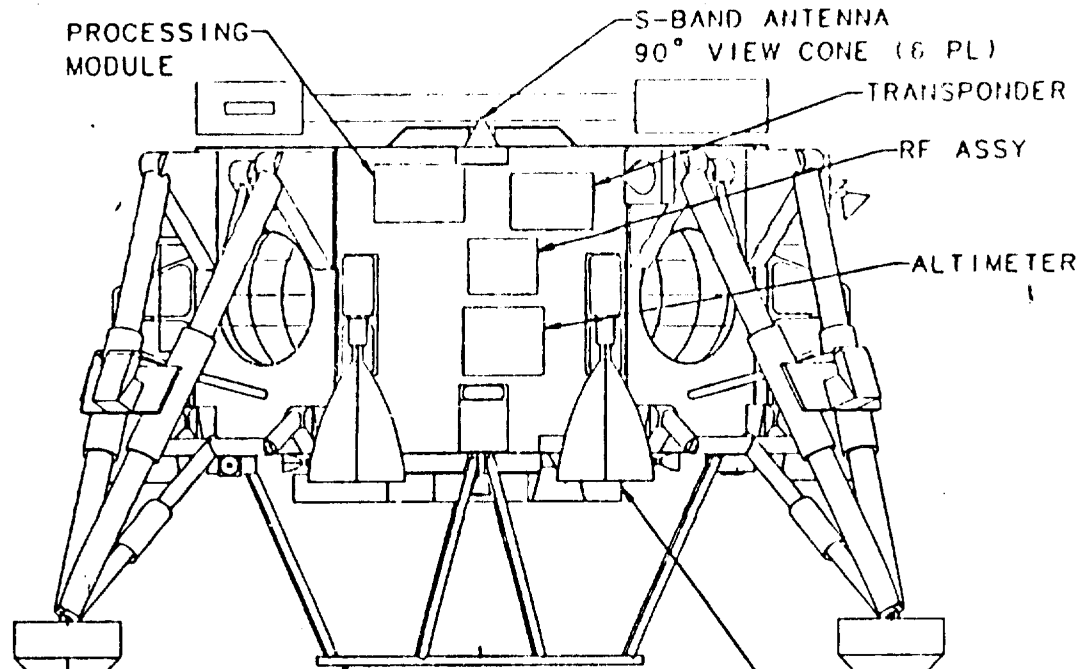

REAR VIEW

PANEL 2




LEFT VIEW

PANEL 3




FRONT VIEW

PANEL 4

STRUCTURAL DESIGN

- ISOGRID STRUCTURE
  - MULTIPLE ATTACH POINTS
  - PASSIVE RADIATORS
  - PRIMARY AND SECONDARY STRUCTURE
- LAUNCH VEHICLE ADAPTER
  - RING INTERFACE
  - THIN-WALL TUBE THRUST STRUCTURE
  - JETTISONED

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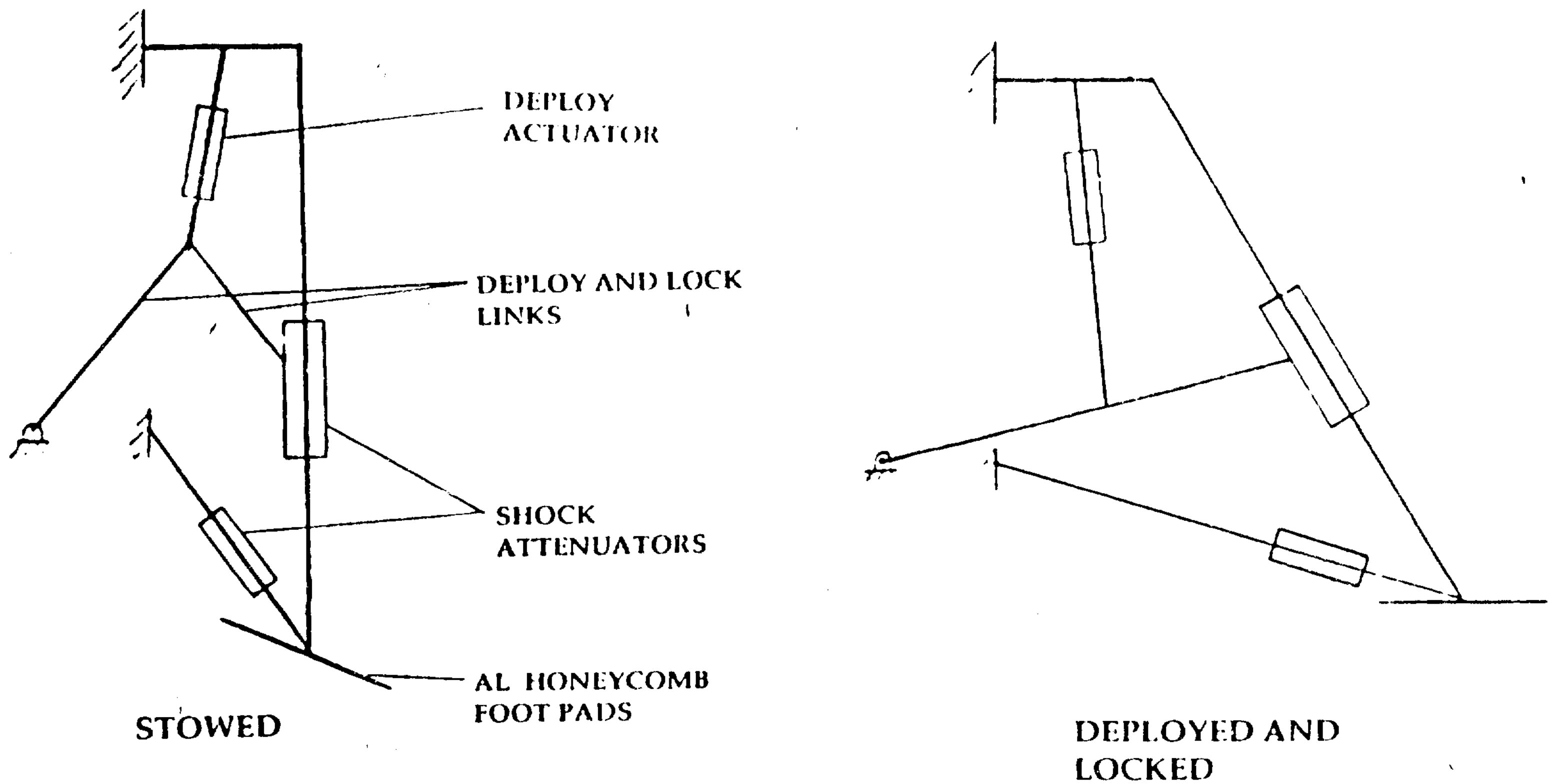
# LANDING GEAR




REQUIREMENTS

- CAPABLE OF WITHSTANDING 6G LANDING
- ATTENUATORS TO DISSIPATE LOAD
- CLEARANCE OF 8 IN UNDER LANDER AFTER STROKE
- FOUR LEG CONFIGURATION
- STOW WITHIN DELTA FAIRING (96 IN DIA)

FEATURES

- DEPLOYABLE MECHANISM (SIMPLE)
- THIN WALL AL LEG MEMBERS
- CRUSHABLE HONEYCOMB FOOT PADS
- POSITIVE LOCKING CAPABILITY
- SPRING OR PNEUMATIC ATTENUATION



S. Gov1

# Solar Array Deployment

- Two solar arrays are located on the top of vehicle to the avoid plume impingement of the main and RCS engines of the vehicle.
- The battery charger and array controller are located on top as well to reduce the number and complexity of the jettison systems.
- The arrays are stowed in an accordion fashion and are supported by telescoping poles for withstand the force of a 0.4G burn.
- The arrays will be deployed after Trans-Lunar Insertion burn.
- The arrays, the array controller, and battery charger will be jettisoned after Lunar Deorbit burn.

POTENTIAL WEIGHT SAVING OPTIONS

- NOT OFF-THE-SHELF
- REPACK AND INTEGRATE AVIONICS BOXES
- 3 LEGS LIGHTER STRUCTURE?
- DEVELOP MORE OPTIMUM VEHICLE CONCEPT

# GN&C Subsystem

NCAD team effort:

| EG2 | EG3 | EG4 |
|---|---|---|
| Gene McSwain | Steve Fitzgerald | Jim Blucker |
| John Ruppert | Mike Jansen | Ed Jiongo |
| Nancy Smith | Stan Johnson | Malcom Jones |
| Al Strahan | Rob Meyerson | Tony Pham |
| Tim Woeste | | Andy Saulietis |

NCAD Tasks:

- GN&C hardware architecture & trade studies
- GN&C algorithms & flight code
- Upgrades to JAEL to support integrated avionics testing
- Manpower & cost estimates
- Test program philosophy

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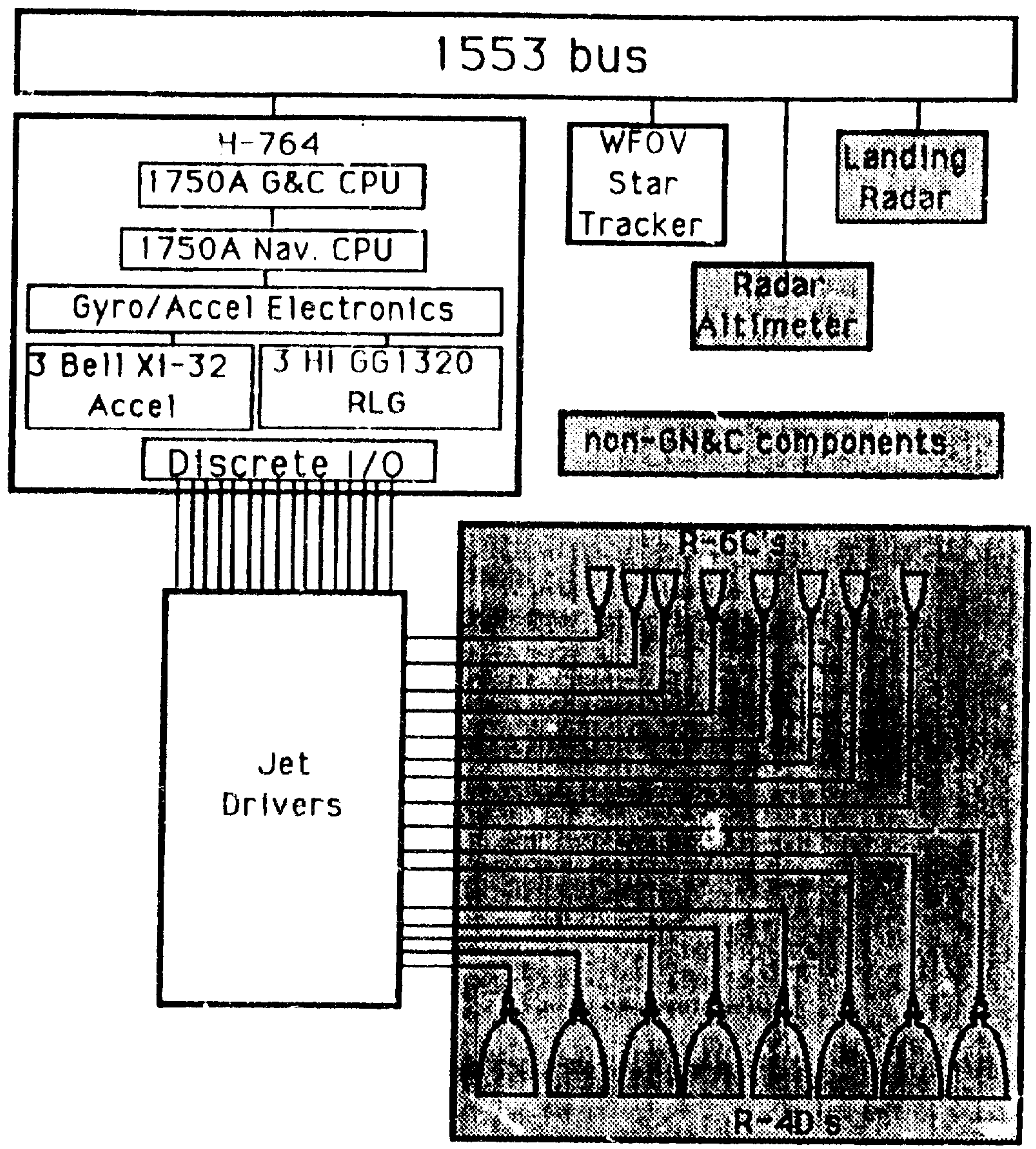
# GN&C Baseline *

| Unit | Weight (kg) | Size (cm/unit) | Power (W) | Cost (FY'92) | Description |
|---|---|---|---|---|---|
| INS H-764 | 9.1 | 17.8 x17.8 x 28 | 45 nom. | | •INS will include all of the GN&C flight code. It will receive inputs from the Star Tracker, Radar Altimeter and the velocity radar. It will send RCS command signals to the RCS electronics to drive the jets on & off |
| Star Tracker | 1 | 12 x12 x15 | 8 peak | | • Peak power will be used for IMU allignments (approx. 2 minute/12 hours) |
| Jet Drivers (GD & HI est.) | 17 | 25 x 25 x 12.5 | 20 quiescent 300 peak | | •Actual poweer will be based upon jet driver electronics, soleroid valve characteristics and duty cycle (see GN&C Power Requirements chart) |
| Total | 27 | | 65 quiescent 345 peak | | |

*** single string (zero fault tolerant) military to class B parts**
**costs are for one flight unit**

Nancy C. Smith/EG23/x38275

# GN&C Schematic



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# Rationale for GN&C baseline

- <u>Schedule</u> => requires basically off-the-shelf units
- <u>Cost</u> => limits non-recurring expenses
- <u>Weight & power</u> =>limits the selection of components available for schedule
- <u>Performance</u> => requires accurate components to meet 5km landing accuracy
- <u>Lunar Orbit</u> => requires additionai software and component accuracy
- <u>Jets selected</u> => requires valve drive electronics to handle 1Amp signal

# Future GN&C work

- GN&C requirements
- Weight & power reduction trade studies
- GN&C connectivity (i.e. connection between INS & other components)
- Avionics Integration
- Test program philosophy @ component,subsystem & system level
- Manpower & cost refinement

# GN&C Options

| | components | cost | weight | delivery |
|---|---|---|---|---|
| Baseline | IMU,jet drivers,ST | | <27 | 2 yrs |
| Option 1 | IMU w/ jet driver card, ST | | < 11.6 | 4 yrs |
| Option 2 | IFMU,jet drivers,ST | | <25.7 | 2 yrs > 12/94 |
| Option 3 | IFMU w/ jet driver card, ST | | <9.5 | 4 yrs |
| Option 4 | RFOG IMU, jet drivers,ST | | <19 | 2 yrs >12/93 |

*Extremely ROM, HI is in the process of refining these costs & schedules

Pursuing additional options:

- Synergism with SDIO's Clementine program (March- April 92)
- Vendor survey in CBD (Spring 92)

# GN&C Option #1



| BASELINE | | | | OPTION#1 | | | |
|---|---|---|---|---|---|---|---|
| component | cost | weight (kg) | delivery | component | ROM cost | weight (kg) | estimated delivery |
| H-764 IMU | \$ & \$ NRE (tests for space applications) | 9.1 | 1 year from end of procurement cycle | Modified H-764 IMU w/ jet driver board | \$ & \$ NRE for tests & jet driver board design | 10.6 | 4 years from end of procurement cycle |
| Jet drivers (GD & HI estimate for \$, kg & delivery) | \$ & \$ NRE (redesign for 8 R4Ds & 8R6Cs) | 17.0 | 2 years from end of procurement cycle (18 months redesign & 6 months delta qual tests) | | | | |
| LLNL Star Tracker | \$ & \$ NRE (add 28V, star map) | < 1 | 1 1/2 years from end of procurement cycle | same | \$ & \$ NRE | < 1 | 1 1/2 years from end of procurement cycle |
| Total | \$ & \$ NRE | <27 | 2 years from end of procurement cycle | Total | \$ & \$ NRE | <11.6 | 4 years from end of procurement cycle |

# GN&C Option #2

Artemis

| BASELINE | | | | OPTION#2 | | | |
|---|---|---|---|---|---|---|---|
| component | cost | weight (kg) | delivery | component | cost | weight (kg) | delivery |
| H-764 IMU | $ & $ NRE (tests for space applications) | 9.1 | 1 year from end of procurement cycle | HI IFMU (w/ decreased accuracy due to RLG 1308) | $ & $ NRE (tests for space applications) | 7.5 | 1 year from end of procurement cycle no earlier than 12/94 |
| Jet Drivers (GD & HI estimate for $ , kg & delivery) | $ & $ NRE (redesign for 8 R4Ds & 8R6Cs) | 17.0 | 2 years from end of procurement cycle (18 months redesign & 6 months delta qual tests) | same | $ & $ NRE | 17.0 | 2 years from end of procurement cycle |
| LLNL Star Tracker | $ & $ NRE (add 28V, star map) | < 1 | 1 1 /2 years from end of procurement cycle | same | $ & $ NRE | < 1 | 1 1/2 years from end of procurement cycle |
| Total | $ & $ NRE | <27 | 2 years fromend of procurement cycle | Total | $ & $ NRE | <25.5 | 2 years from end of procurement cycle no earlier than 12/94 |

# GN&C Option #3

Artemis

| BASELINE | | | | OPTION#3 | | | |
|---|---|---|---|---|---|---|---|
| component | cost | weight (kg) | delivery | component | cost | weight (kg) | delivery |
| H-764 IMU | $ & $ NRE (tests for space applications) | 9.1 | 1 year from end of procurement cycle | Modified HI IFMU w/ jet driver board (w/ decreased accuracy due to RLG 1308) | $ & $ NRE for tests & jet driver board design | 8.5 | 4 years from end of procurement cycle finishing no earilier than 12/96 |
| Jet drivers (GD & HI estimate for $, kg & delivery) | $ & $ NRE (redesign for 8 R4Ds & 8R6Cs) | 17.0 | 2 years from end of procurement cycle (18 months redesign & 6 months delta qual tests) | | | | |
| LLNL Star Tracker | $ & $ NRE (add 28V, star map) | < 1 | 1 1/2 years from end of procurement cycle | same | $ & $ NRE | < 1 | 1 1/2 years from end of procurement cycle |
| Total | $ & $ NRE | <27 | 2 years from end of procurement cycle | Total | $ & $ NRE | <9.5 | 4 years from end of procurement cycle |

Nancy C. Smith/EG23/x38275

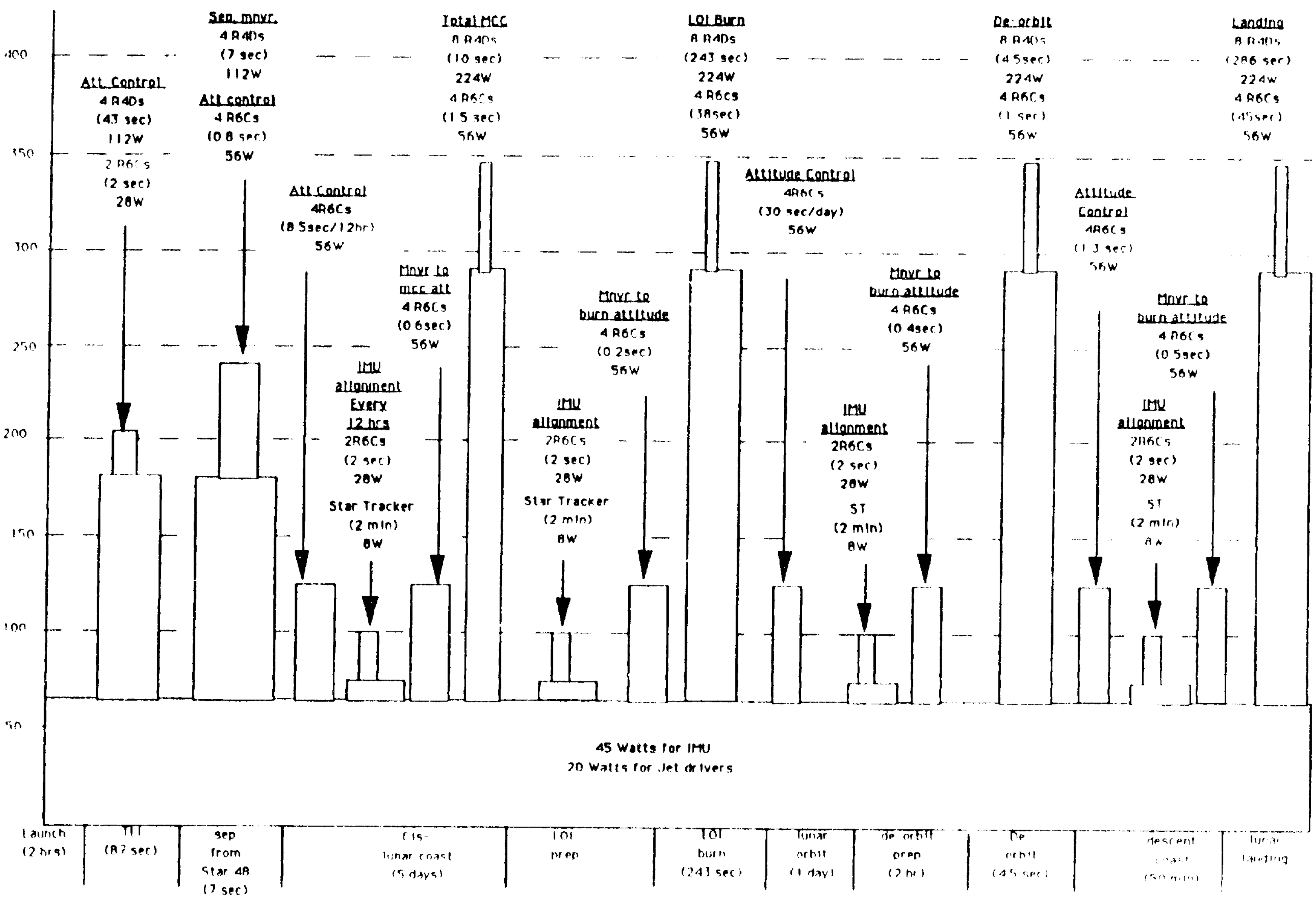
# GN&C Option #4

Artemis

| BASELINE | | | | OPTION#4 | | | |
|---|---|---|---|---|---|---|---|
| component | cost | weight (kg) | delivery | component | cost | weight (kg) | delivery |
| H-764 IMU | $ & $ NRE (tests for space applications) | 9.1 | 1 year from end of procurement cycle | Dual Polarization RFOG IMU w/o processing (would require GN&C flight code to be placed in the GPC & additional work to calculate time delays) | | < 1 | 2 years from end of procurement cycle (no earlier than 12/93) |
| Jet drivers (GD & HI estimate for $, kg & delivery) | $ & $ NRE (redesign for 8 R4Ds & 8R6Cs) | 17.0 | 2 years from end of procurement cycle (18 months redesign & 6 months delta qual tests) | same | $ & $ NRE | 17.0 | 2 years from end of procurement cycle |
| LLNL Star Tracker | $ & $ NRE (add 28V, star map) | < 1 | 1 1/2 years from end of procurement cycle | same | $ & $ NRE | < 1 | 1 1/2 years from end of procurement cycle |
| Total | $ & $ NRE | <27 | 2 years from end of procurement cycle | Total | $ & $ NRE | <19 | 2 years from end of procurement cycle |

# GN&C Power requirements

Artemis




Nancy C. Smith/EG23/x38275

# Artemis
# Common Lunar Lander
# Data System

David Pruett / EK11 / x35269

# Driving Requirements and Assumptions

Artemis

- Developed from program requirements and goals and detailed interviews with subsystem leads
- Data Systems Design Requirements
  - Open loop command and control
    - -- On board stored command sequences and ground initiated commands
    - -- Health and Status Monitoring with operational Mode Control (nominal and backup)
  - Data acquisition, distribution and telemetry formatting
    - -- One sample per second, no data recording
    - -- Telemetry rate of 4.4Kbits/sec (75% of Comm capacity)
  - Timing control
    - -- 1ms resolution
    - -- Accuracy of 5x10-8 sec/sec of error over 6 days (do not meet - see issues list)
- Assumptions
  - Utilize existing Space Qual hardware or Mil spec if it can be qualified to mission levels

# Trades and Analysis

- Command and Control
  - Initial understanding was for no command and control from the ground
  - State Vector update, failure mode reset, and command sequence modification became apparent
  - Level of commanding is minimal, not a design driver

# Trades and Analysis

Artemis

- Data Acquisition and Telemetry
  - Interviewed all subsystem leads to determined data acquisition needs
  - One sample per second required except possibly during landing phase
  - Final discussions determined that landing phase rate could be 1 sample per second also
  - Requirement for recording data during communications loss caused a data recorder to be included in the data system - significant increase in power and weight
  - Subsequent discussions determined that recording during translunar coast was not cost efficient and that recording during landing was not necessary because the vehicle would be in line of sight of Earth at all times during landing
  - Removal of recording requirement removed need for recorder, thus reducing power, weight and cost
  - Removal of higher data acquisition rate during landing simplified the data acquisition system by requiring only one format during the entire mission.
  - Removal of recorder and higher rate on landing also resulted in the reduction of the Comm data link rate from 12 kbps to 6 kbps, an additional savings in power

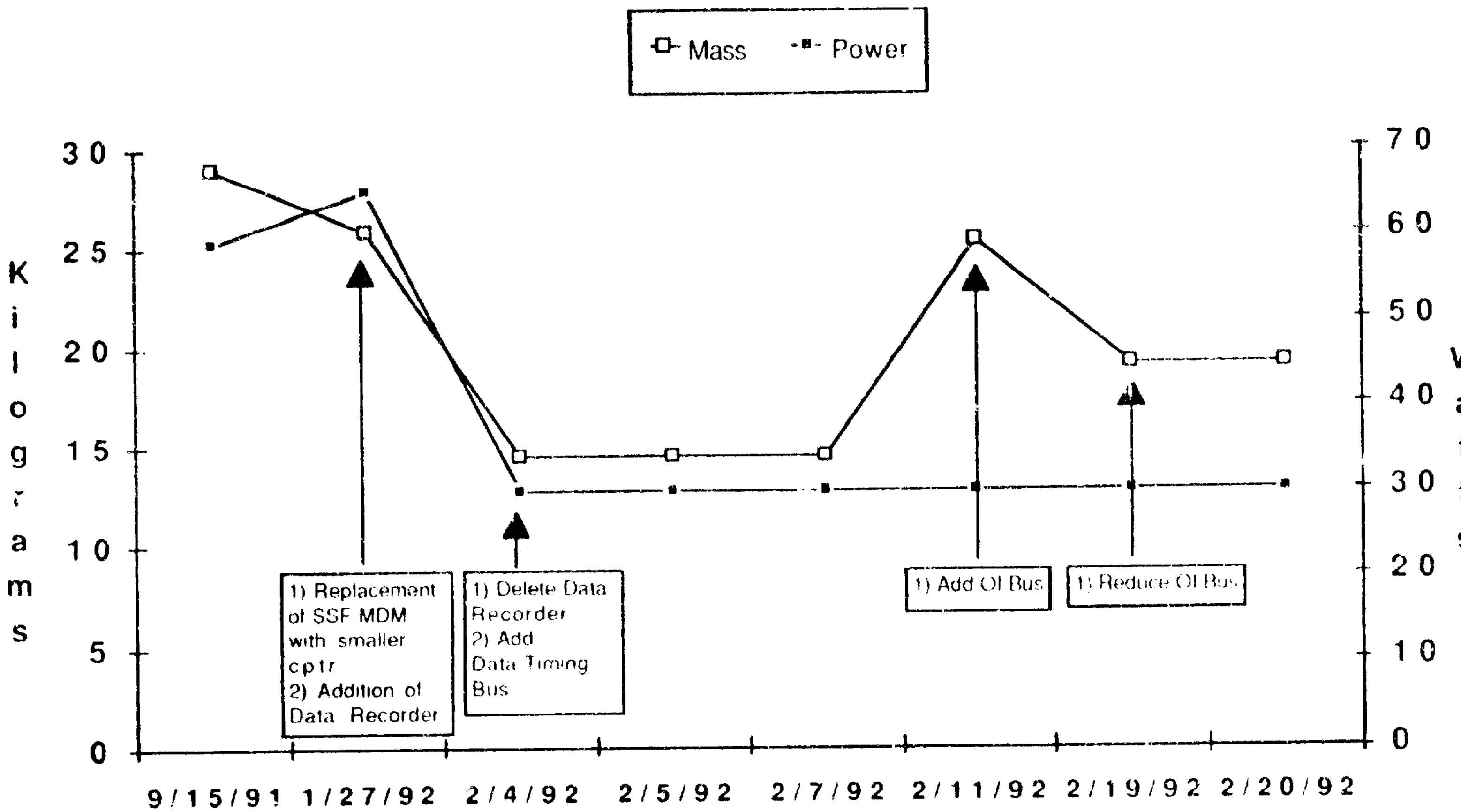
# Trades and Analysis



- Timing
  - GN&C desired a clock accurate enough to allow them to ignore timing error in their guidance calculations
  - This would also have reduced the need for ground processing of state vector data
  - An investigation of timing sources indicated that to get the necessary error, a temperature controlled cesium clock would be necessary - a costly, heavy and power hungry solution
  - Subsequent discussions led to the conclusion that state vector updates would be needed anyway to meet the 5 km targeting error
  - Discussions identified several possible procedural alternatives for a high accurate onboard clock
  - Therefore we only specified a moderately accurate clock that could be put on a card.
  - Final recommendations on how the timing will be handled will be resolved during Phase A

Artemis

## Artemis Data System Mass/Power History



# Data Acquisition Load

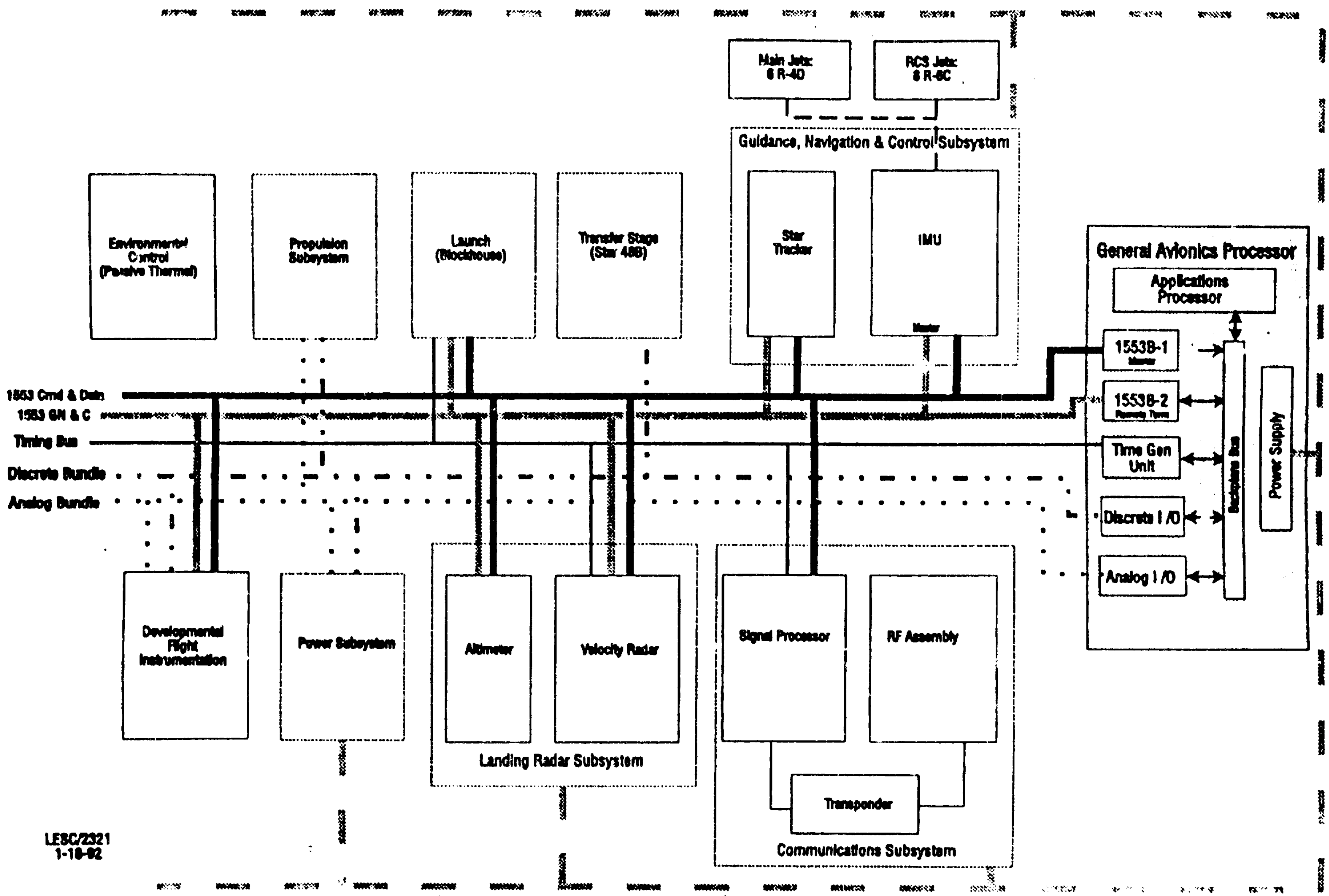
| | Bits/Word | Words | Total Bits/sec |
|---|---|---|---|
| Digital(1553) | 8 | 421 | 3,368 |
| Analog | 8 | 78 | 624 |
| Discretes | 1 | 63 | 63 |
| Total | | 562 | 4,055 |
| With 10% Format Overhead | | 618 | 4,461 |

| Mission Phase | Earth Orbit | TL Coast | Lunar Orbit | Landing |
|---|---|---|---|---|
| Link Rates(Kbps) | 6 | 6 | 6 | 6 |
| % of Link Rate | 75% | 75% | 75% | 75% |

# GAP Loading Estimates

| Function | New Code (SLOCs) | Reuse (SLOCSs) | PROM (KB) | Buffer (KB) | Data (KB) | Proc (KIPS) |
|---|---|---|---|---|---|---|
| Data Serv Mgr | 6000 | 0 | 150 | 333 | 75 | 75 |
| Data System Mgr | 8000 | 0 | 200 | 0 | 100 | 100 |
| OS/AdaRTE | - | - | 30 | 0 | 337 | 337 |
| Command Proc | 4000 | 4000 | 100 | 0 | 50 | 50 |
| Subsys Direction | 1000 | 0 | 25 | 0 | 12 | 12 |
| Gross Size Totals | 19000 | 4 | 505 | 333 | 575 | 575 |
| Design Size | | | 750 | 1000 | 1000 | 1200 |
| % Capacity | | | 67% | 33% | 58% | 48% |

# Common Lunar Lander
# Space Data System Interface Diagram




LESC/2321
1-18-92

# Data System Baseline Architecture

Artemis

- External Interfaces
  - 1553B serial bus for
    - --Subsystem Command and Control (if required by the subsystem)
    - -- Health and Status Data from Subsystem (if required by subsystem)
  - Serial Timing Bus for timing synchronization with GN&C
  - Operational Instrumentation (OI) bundle
    - -- Analog bundle and discrete bundle for sensor and effector input and output

# Data System Baseline Architecture

- Internal Architecture
  - Single String
  - General Avionics Processor (GAP) (128,000 hrs MTBF)

| | |
|---|---|
| EEPROM | >= 750 KBytes |
| RAM | >= 2,000 KBytes |
| Processing Capacity | >= 1.2 MIPs |
| EDAC | for single event upset |
| Interfaces | Dual 1553B |
| | 8 bit Analog Input (79) |
| | 1 bit Digital I/O (63) |
| | Ground Support Equipment (TBD) |

  - Time Generation Unit (TGU) (100,000 hrs MTBF)

| | |
|---|---|
| Accuracy | 2x10-5 seconds of error/sec over 6 day mission (see issues list) |
| Format | IRIG-B, CCSDS or other |
| Resolution | 1ms |
| Interfaces | Timing Bus |

# Critical Path

- Procurement of Development and Flight Hardware
  - Establish Source Evaluation Committee
  - Develop SOW/RFP
    - -- Successor to Requirements Review
  - Vendor evaluations
  - Contractor selections (multiple for detailed analysis)
  - Vendor designs
  - Final Selection
    - Successor to PDR
  - Contract Change Order development and negotiations
  - Hardware development and delivery

# Open Issues

- Timing Requirements
  - GN&C Accuracy requirements (5 x10-8 Sec of error/sec over 6 day mission) drive us to high accuracy clock that is costly in money, weight and power
  - GN&C is analyzing requirement to determine alternative solution
    - -- Time update from ground with state vector
    - -- Less accurate time required without update
- Power and weight reductions
  - Trade to determine if combining Data System, Comm and GN&C processing in one box would be effective
  - Will perform trade discussions during Phase A
- Fault Tolerance and Recovery
  - Current goal is single string
  - Program needs to specify degraded operation modes so that subsystems can design in recovery control
- Integration, Test and Verification
  - Will define in more detail what is required for IT&V during Phase A

# COMMON LUNAR LANDER COMMUNICATION SUBSYSTEM PHASE II DESIGN

BY

EE7/HENRY CHEN

March 10, 1992

# INTRODUCTION



- TCD team effort
  - EE2/ K. D. Mclain
  - EE3/ T. Early
  - EE7/ H. Chen, B. Luu
  - LESC/ K. Byerly
- Requirements
  - Doppler/Tone Ranging for state vector generation
  - Command data on the uplink to provide state vector and timing updates as well as real time commands
  - Telemetry data on the downlink
- Phase II baseline design has been completed
- Power, weight, size, cost, and schedule estimates have been provided to ET

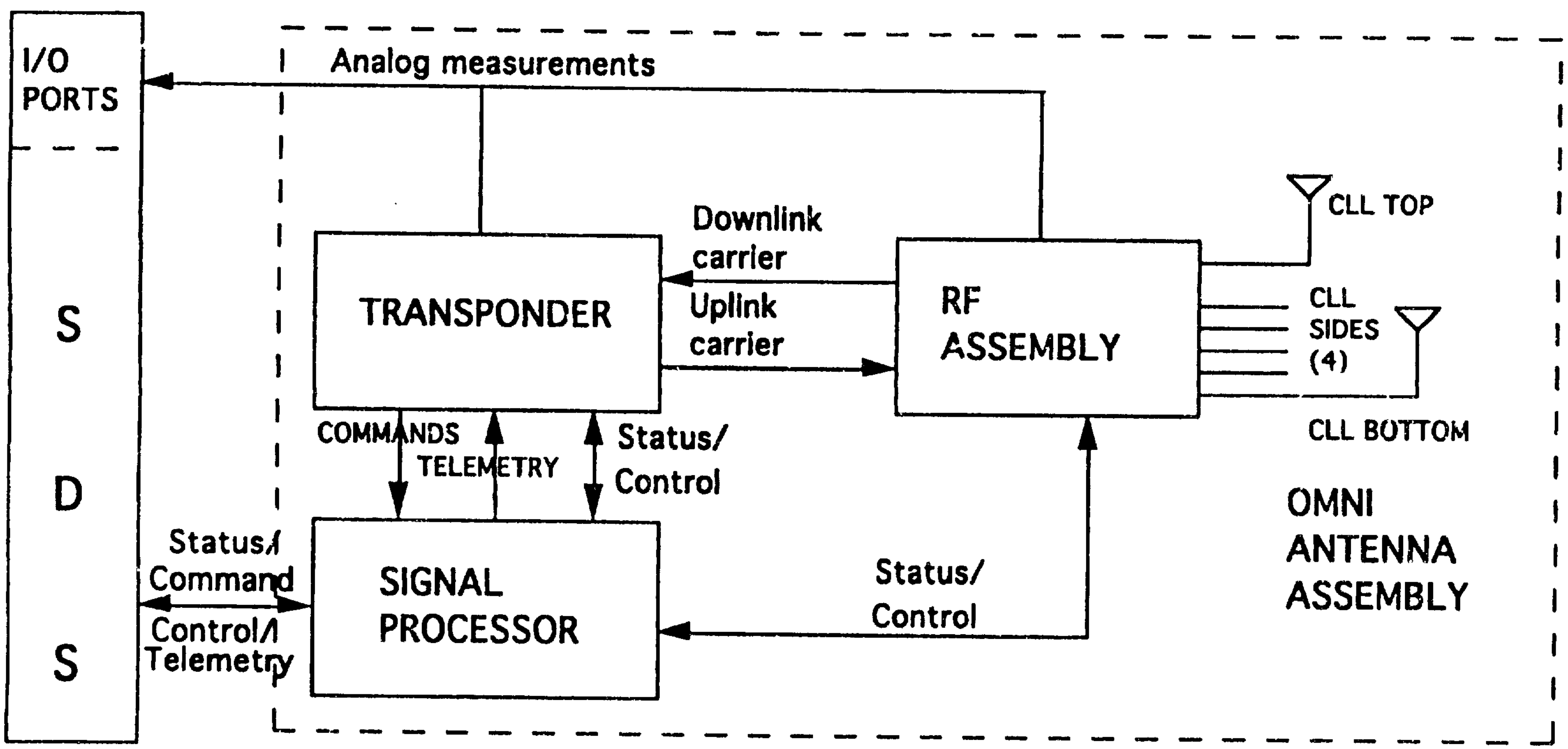
Artemis

# BASELINE DESIGN

- Data Rate:
  - Telemetry: 6Kbps
  - Command: up to 2Kbps
- Deep Space Network (DSN) 26m subnet is used as ground interface
- There are four major assemblies: Antennas, RF Assembly, Transponder, and Signal Processor
- Conical spiral omni directional antennas are used
- RF Assembly performs antenna switching functions
- Transponder performs power amplification, diplexing, modulation/demodulation, synchronization functions, and ranging signal turn-around
- Signal Processor performs convolutional encoding, decryption, and frame synchronization. It also interfaces with the Space Data System for all digital data through 1553B bus
- Single string system with a MTBF of 10,000 hours

Artemis

# CLL COMMUNICATION SUBSYSTEM



# DESIGN CHANGES FROM PHASE I

Artemis

- Data rate for the telemetry is reduced from 11.6 Kbps to 6 Kbps to allow operation with the DSN 26m subnet
- DSN 26m subnet replaces 34m subnet
  - Greater antenna tracking capability
  - Tone ranging which provides compatibility with existing ESTL Communication testbed
- CLL will use DSN 26m subnet up to 3 hours out of every 12 hours
- 10 Watts Power Amplifier (PA) in the RF Assembly is deleted ( net power savings). The PA in the transponder is increased to 5W. The diplexer in the RF Assembly is transferred to the transponder
- A coherent receiver is selected to facilitate antenna selection
- The communication subsystem's control and monitoring functions for all digital data are transferred to the Signal Processor
- The Space Data Subsystem baseline design requires a change to 1553B interface for all digital data

POWER, WEIGHT, AND SIZE FOR CLL COMMUNICATION SUBSYSTEM
(Telemetry, ranging & command option)

| UNIT | WEIGHT | VOLUME | # | Power | VENDOR |
|---|---|---|---|---|---|
| S-Band System using DSN 26 subnet<br>[XPNDR, RF Assembly, Processing module, Antennas(6) and Cables] | | | | | |
| RF Assemblly<br>[w/embedded controller] | 7.94Kg | 7800cc<br>16x20x24 | 1 | 12.7W | custom* |
| XPNDR<br>[Near-Earth XPNDR] | 4.05Kg | 3500cc<br>16x20x11 | 1 | 40.5W(p)**<br>4.5 W(s)** | Motorola |
| Antennas | 1.38Kg | 8640<br>9 cm height<br>*10 cm diameter* | 6 | 0 W | Watkins Johnson |
| Cable | 0.68Kg | 900cc | 1 set | 0 W | GORE |
| Processing Module | 6.0Kg | 8430cc<br>18x18x26 | 1 | 27W | custom* |
| Total | 20.05Kg*** | 29,270cc | | 80.2W(p)**<br>44.2W(s)** | |

* Equipment to be built in house from components with known heritage
** p - peak power, s- standby power
*** with mounting structure 23.2Kg

# BACK-UP MATERIAL

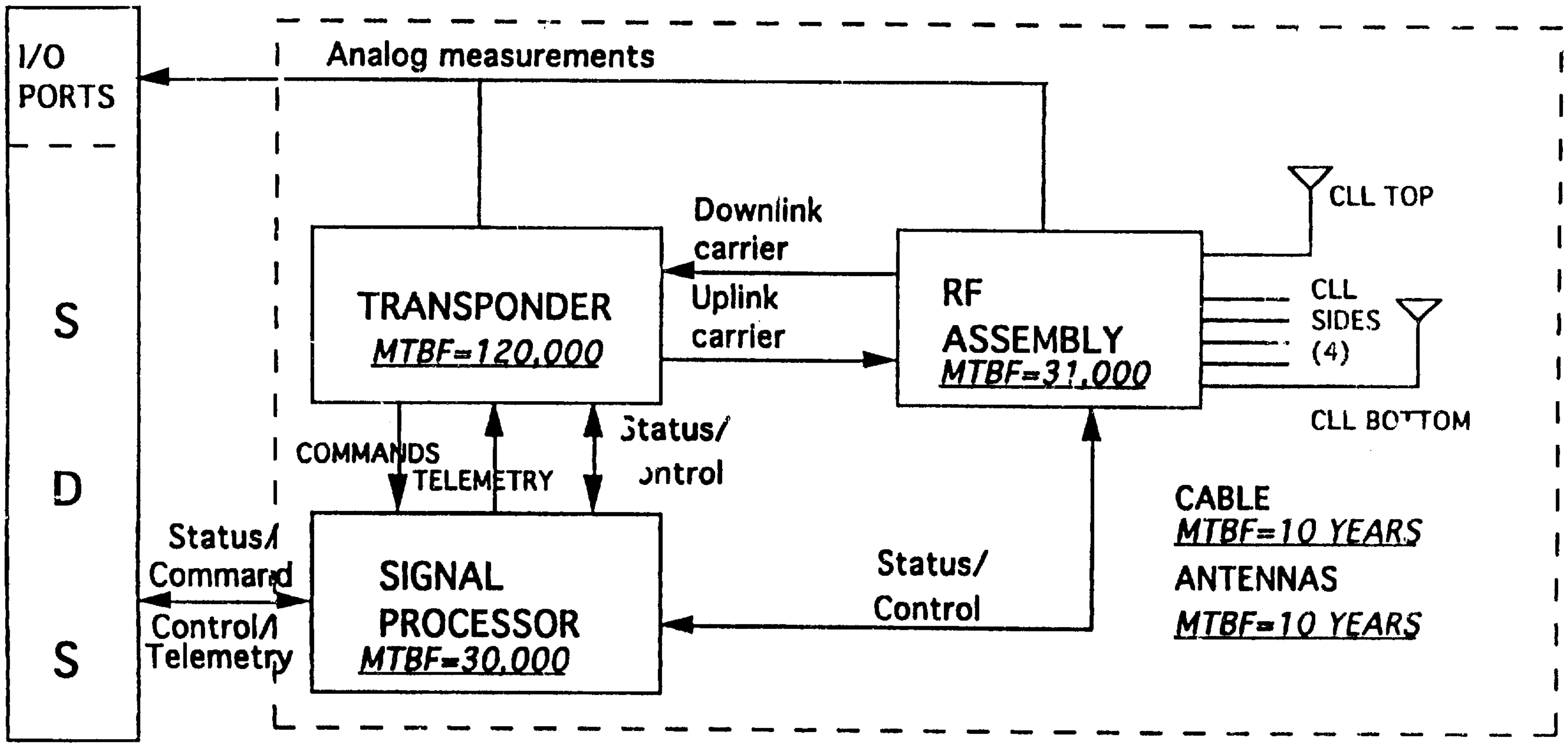
I. COMMUNICATION SUBSYSTEM RELIABILITY
II. DETAILED BLOCK DIAGRAMS
III. LINK MARGIN TABLES
IV. AREAS FOR FUTURE STUDIES



Henry Chen/EE7/x30128

Artemis

# COMMUNICATION SUBSYSTEM RELIABILITY

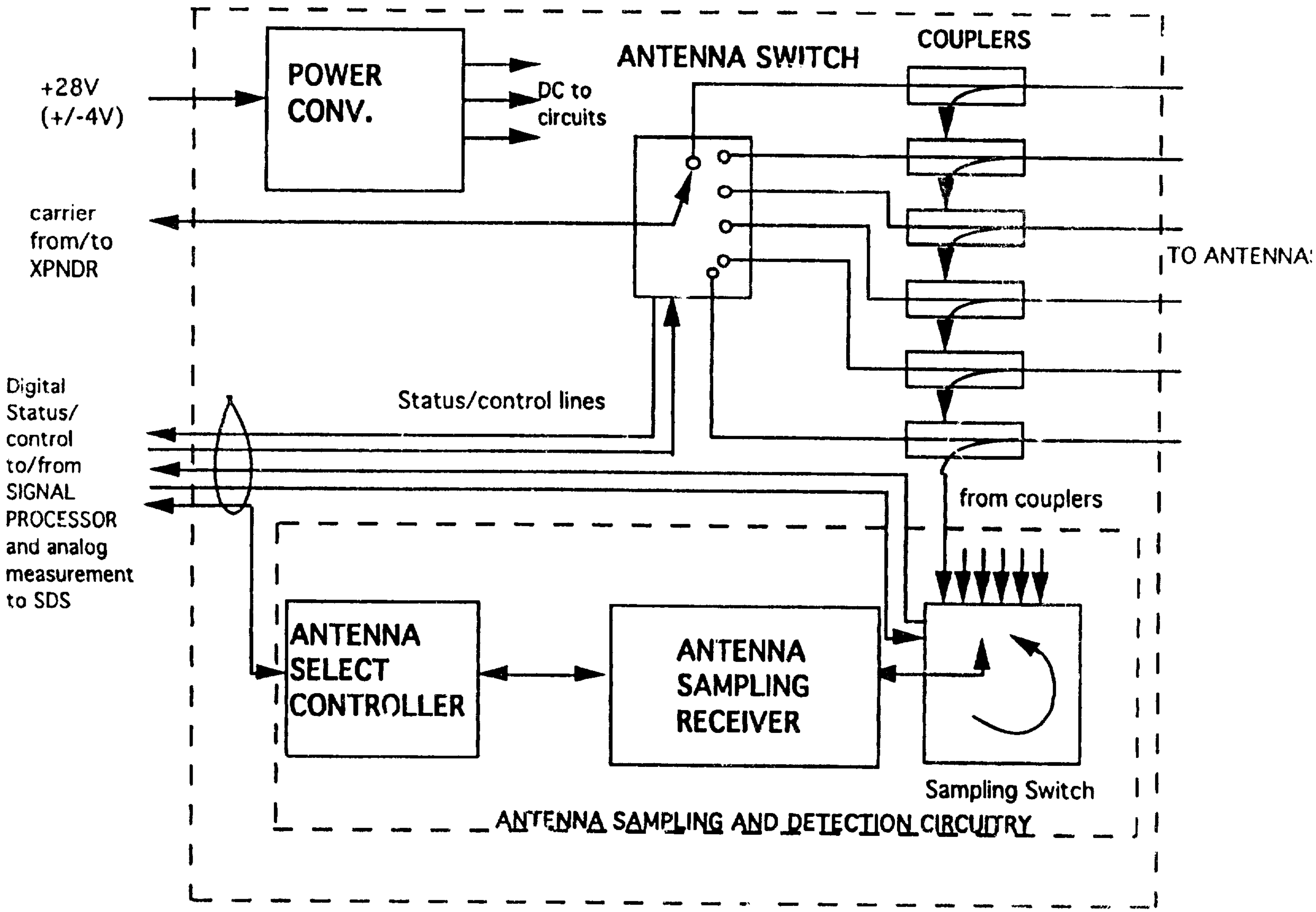



COMMON LUNAR LANDER (CLL)
RF COMMUNICATION SUBSYSTEM

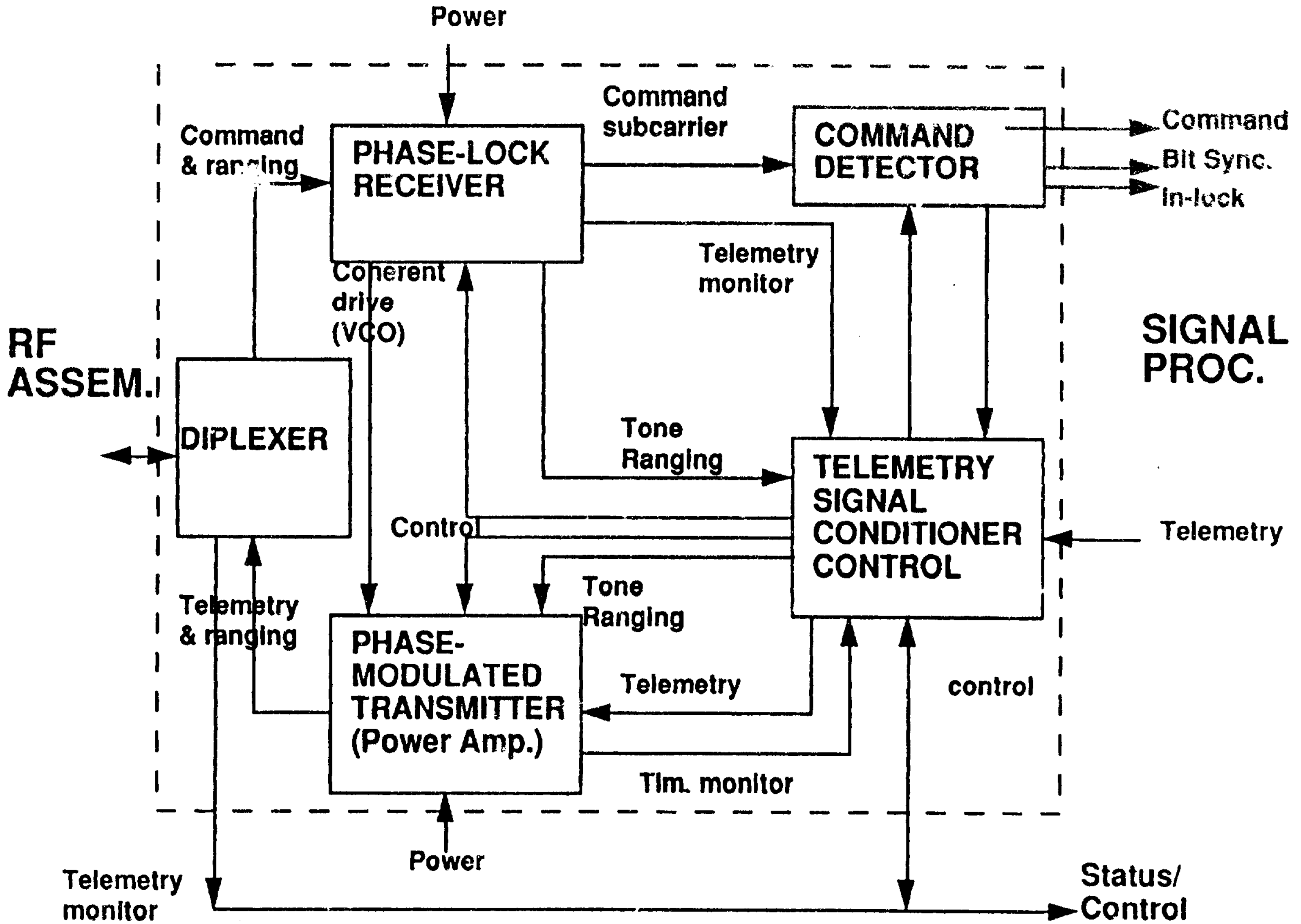
MTBF=10,000 HOURS

Artemis

# CLL RF ASSEMBLY



# NASA NEAR EARTH TRANSPONDER

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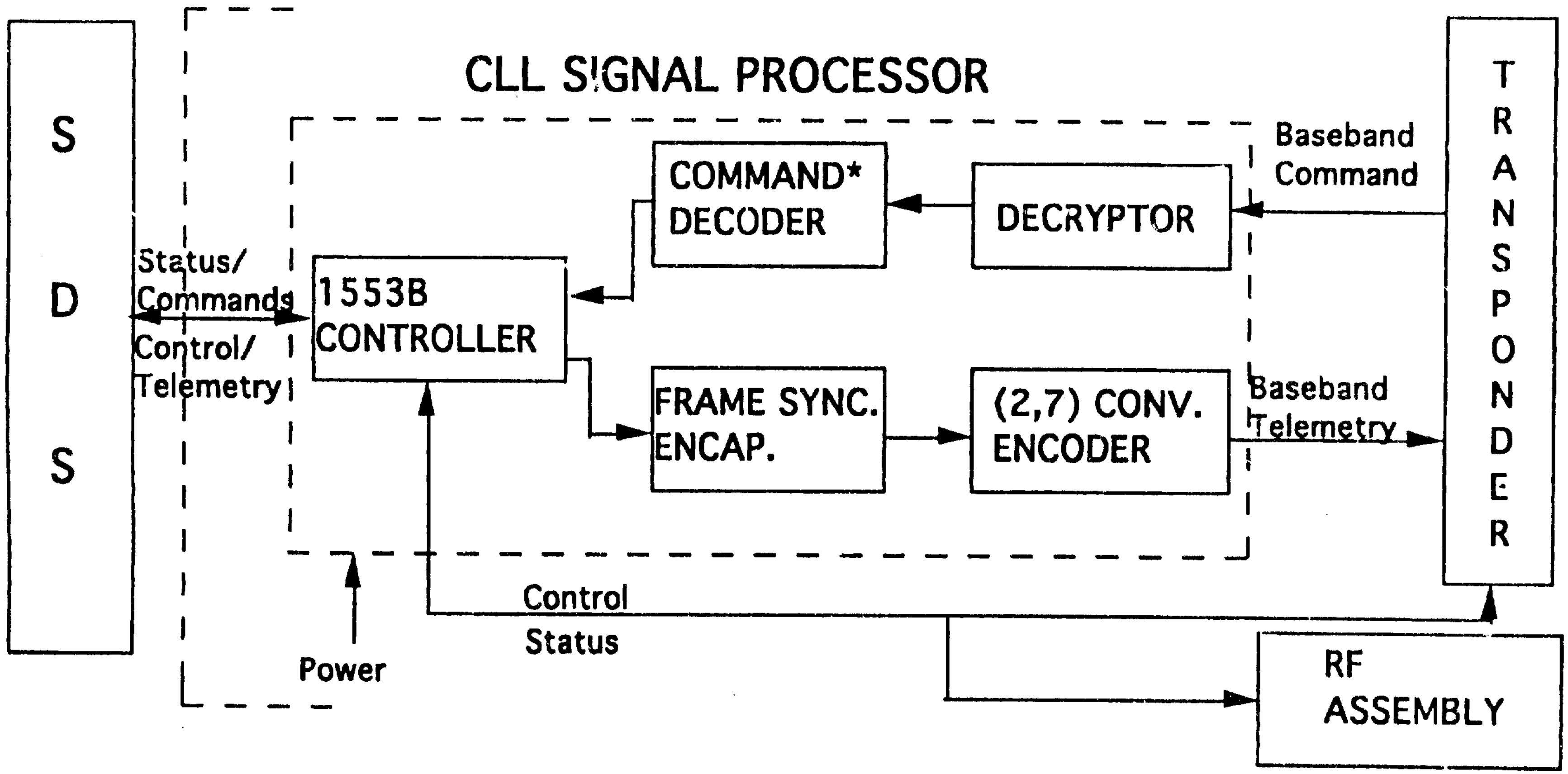
# CLL SIGNAL PROCESSOR MODULE




* The command decoder performs frame synchronization and error detection (if needed).

COMMON LUNAR LANDER TO DSN 26M SUBNET–S BAND; 6W PA, OMNI ANTENNA
RETURN TELEMETRY CHANNEL CIRCUIT MARGIN

| Parameters | Value | Remarks |
|---|---|---|
| 1 Transmit Power, dBW | 7.70 | 5.9 watts |
| 2 Transmit Circuit Loss, dB | -5.90 | Estimate |
| 3 Transmit ant gain, dBic | 2.00 | Omni antenna |
| **4 Transmit EIRP, dBW** | **3.80** | 1 + 2 + 3 |
| 5 Space loss, dB | -211.60 | r = 395000 km<br>f = 2.300 GHz |
| 6 Polarization loss, dB (worst case) | -0.50 | Estimate |
| 7 Pointing Loss | -0.20 | DSN 810-5 Rev. D Auto-track |
| 8 Receive antenna gain | 51.50 | 26M, DSN 810-5 Rev. D,TCI-20<br>(including feedline loss) |
| 9 Total Received Power (Prec) | -157.00 | 4 + 5 + 6 + 7 + 8 |
| 10 Maximum System Temp | 22.86 | DSN 810-5 Rev. D 193 °K<br>Ref 100° LNA at Zenith, 175 °K<br>Horizon Addition 18 °K |
| 11 Boltzman's Constant (dBW/K)/Hz | -228.60 | 1.38E-23 W/Hz/K |
| 12 Noise Spectral Density (No) | -205.74 | 10 + 11 |
| 13 Receiver G/T, dB/K | 28.64 | 8 - 10 |
| 14 Prec/No, dB-Hz | 48.74 | 9 - 12 |
| 15 Modulation Loss | -2.10 | Bsc=0.5 rad Bd=1.0 rad |
| 16 Bit rate bandwidth, dBHz | 37.80 | 6.00 kbps |
| 17 Bit Sync Degradation, dB | -1.00 | DSN 810-5 Rev. D, TLM-50 |
| 18 Coding Gain, dB | 5.30 | (2, 7) Convolutional Code |
| **17 Receive Eb/No, dB** | **13.14** | 14 + 15 - 16 + 17 + 18 |
| 24 Required Eb/No, dB | 9.60 | 1.0E-5 BER |
| 25 LINK CIRCUIT MARGIN, dB | 3.54 | 20 - 24 |

3/5/92

COMMON LUNAR LANDER TO DSN 26 M–S BAND; OMNI ANTENNA, 6W PA
RETURN LINK RANGE CHANNEL CIRCUIT MARCIN

| | Parameters | Value | Remarks |
|---|---|---|---|
| 1. | Transmit Power, dBW | 7.70 | 5.9 watts |
| 2. | Transmit Circuit Loss, dB | -5.90 | Estimate |
| 3. | Transmit ant gain, dBic | 2.00 | Omni antenna |
| 4. | **Transmit EIRP, dBW** | **3.80** | 1+2+3 |
| 5. | Space loss, dB | -211.60 | r = 395000 km<br>f = 2.300 GHz |
| 6. | Polarization loss, dB (worst case) | -0.50 | Estimate |
| 7. | Pointing Loss | -0.15 | DSN 810-5 Rev. D Auto-track |
| 8. | Receive antenna gain | 51.50 | 26M, DSN 810-5 Rev. D,TCI-20<br>(including feedline loss) |
| 9. | Total Received Power (Prec) | -156.95 | 4+5+6+7+8 |
| 10. | Maximum System Temp | 22.86 | DSN 810-5 Rev. D 193 °K<br>Ref 100° LNA at Zenith, 175 °K<br>Horizon Addition 18 °K |
| 11. | Boltzman's Constant (dBW/K)/Hz | -228.60 | 1.38E-23 W/Hz/K |
| 12. | Noise Spectral Density (No) | -205.74 | 10 + 11 |
| 13. | Receiver G/T, dB/K | 28.64 | 8 - 10 |
| 14. | **Prec/No, dB-Hz** | **48.79** | 9 - 12 |
| 15. | Modulation Loss | -14.70 | Bsc=0.5 rad, Bd=1.0 rad |
| 16. | Post det. tone S/C power to noise spectral density, Psc/No, dbHz | 34.09 | 14 + 15 |
| 17. | Required Psc/N0, dBHz | 22.80 | Required for using 500KHz Major Range Tone |
| 18. | **Major range tone margin, dB** | **11.29** | 16 - 17 |
| 19. | Required Prec/No for carrier in-lock , dBHz | 30.00 | Required for medium acquisition |
| 20. | Carrier in-lock tracking margin, dBHz | 18.79 | 14 - 19 |
| 21. | LINK CIRCUIT MARGIN, dB | **11.29** | Smaller of 18 and 20 |

DSN 26M SUBNET TO COMMON LUNAR LANDER, S-BAND
FORWARD COMMAND CHANNEL CIRCUIT MARGIN

| Parameters | Value | Remarks |
|---|---|---|
| 1. Transmit power, dBW | 43.0 | 20.0 KW |
| 2. Transmit ant gain, dB | 51.5 | 26M, DSN 810-5 Rev. D, TCI-20 (including feedline loss) |
| 3. Transmit EIRP, dBW | 94.5 | 1 + 2 |
| 4. Space loss, dB | -210.9 | r=395000 km f=2.120GHZ |
| 5. Polarization loss, dB (worst case) | -0.9 | 2.2 dB (Tx AR) 6.0 dB (Rx AR) |
| 6. Pointing loss, dB | 0.1 | DSN 810-5 Rev. D conscan, TCI-30 |
| 7. Receive ant gain, dB | 2.0 | Omni antenna |
| 8. Receive ckt loss, dB | -5.9 | Estimate |
| 9. Total receive power (Prec), dBW | -121.2 | Sum 4 thru 9 |
| 10. System noise temp, dBK | 30.1 | Ts=1024 K(5.5 dB NF, Ta=290K) |
| 11. Receiver G/T, dB/K | -34.0 | 7 + 8 - 10 |
| 12. Noise Spectral density (No), dBW/Hz | -198.5 | Boltzmann's constant(-228.6) + 10 |
| 13. Modulation loss, dB | -3.8 | Bsc=1.0 rad, Bd=1.0 rad |
| 14. Prec/No, dBHz | 77.3 | 9 - 12 |
| 15. Noise bandwidth, dBHz | 33.0 | 2 0 Kbps |
| 16. Receive Eb/No, dB | 40.5 | 13 + 14 - 15 |
| 17. Theoretical Eb/No, dB | 9.6 | 1.E-5 BER, OQPSK |
| 18. Implementation loss, dB | -2.0 | Estimate |
| 19. Req'd Eb/No, dB | 11.6 | 17 - 18 |
| 20. LINK MARGIN, dB | 28.9 | 16 19 |

3/5/92

# DSN 26M SUBNET TO COMMON LUNAR LANDER, S BAND FORWARD RANGE CHANNEL CIRCUIT MARGIN

| | Parameters | Value | Remarks |
|---|---|---|---|
| 1. | Transmit power, dBW | 43.0 | 20.0 KW |
| 2. | Transmit ant gain, dB | 51.5 | 26M, DSN 810-5 Rev. D, TCI-20 (include feedline loss) |
| 3. | Transmit EIRP, dBW | 94.5 | 1 + 2 |
| 4. | Space loss, dB | -210.9 | r=395000 km f=2.120GHZ |
| 5. | Polarization loss, dB (worst case) | -0.9 | 2.2 dB (Tx AR) 6.0 dB (Rx AR) |
| 6. | Pointing loss, dB | -0.1 | DSN 810-5 Rev. D conscan, TCI-30 |
| 7. | Receive ant gain, dB | 2.0 | Omni antenna |
| 8. | Receive ckt loss, dB | -5.9 | Estimate |
| 9. | Total receive power (Prec), dBW | -121.3 | Sum 3 thru 8 |
| 10. | System noise temp, dBK | 30.1 | Ts=1024 K(5.5 dB NF, Ta=290K) |
| 11. | Receiver G/T, dB/K | -34.0 | 7 + 8 - 10 |
| 12. | Noise Spectral density (No), dBW/Hz | -198.5 | Boltzmann's constant(-228.6) + 10 |
| 13. | Prec/No, dBHz | 77.2 | 9 - 12 |
| 14. | Modulation loss, dB | -3.8 | Bsc=1.0 rad, Bd=1.0rad |
| 15. | Pos. det. tone S/C power to noise spectral density, Psc/N0, dBHz | 73.4 | 13 + 14 |
| 16. | Required Psc/No, dBHz | 22.8 | Estimate for using 500KHz Major Ranging Tone |
| 17. | Major range tone margin, dB | 50.6 | 15 - 16 |
| 18. | Required Prec/No for carrier in-lock, dBHz | 50.0 | Required for fast acquisition |
| 19. | Carrier in-lock tracking margin, dBHz | 27.2 | 13 - 18 |
| 20. | LINK MARGIN, dB | 27.2 | Smaller of 17 and 19 |

3/5/92

Artemis

# AREAS FOR FUTURE STUDY

- Should part of the signal processor's functions be incorporated in the Space Data System (i.e., to eliminate the 1553B interface)?
- Should Space Data System power up/down the communication transmitter and receiver instead of only the transmitter?
- Antenna switching algorithm: coherent receiver vs. line-of-sight calculation
- Is encryption of the command link a program requirement?
- Can the signal go through the DSN in bent-pipe mode?

# TRACKING SUBSYSTEM

## PHASE 2 BASELINE

## COMMON LUNAR LANDER DESIGN REVIEW

March 10, 1992

William X. Culpepper/EE6/x31479

# HISTORICAL PERSPECTIVE

THREE U.S. SPACE PROGRAMS HAVE ACCOMPLISHED PLANETARY LANDINGS

- SURVEYOR
- APOLLO
- VIKING

ALL THREE USED THE SAME TECHNIQUE

- ALTIMETER FOR RANGE TO THE SURFACE
- VELOCITY RADAR TO SENSE VELOCITY ALONG THE MAJOR AXES

ALL THREE SYSTEMS WERE SUCCESSFUL

# MAJOR SYSTEM DRIVERS

FIRST LAUNCH DATE OF JUNE, 1996

- SUBSYSTEM FLIGHT HARDWARE NEEDED BY NOVEMBER, 1994

PERFORMANCE REQUIREMENTS/COMPLEXITY ON THE LEVEL OF SURVEYOR

- THE TRACKING SYSTEM IS NOT A TECHNOLOGY ISSUE

# TRACKING SYSTEM BASELINE

BASED ON THE VIKING PROGRAM INSTRUMENTATION DESIGN.

**RADAR ALTIMETER**

MEASURES HEIGHT OF VEHICLE ABOVE THE SURFACE.

**VELOCITY RADAR**

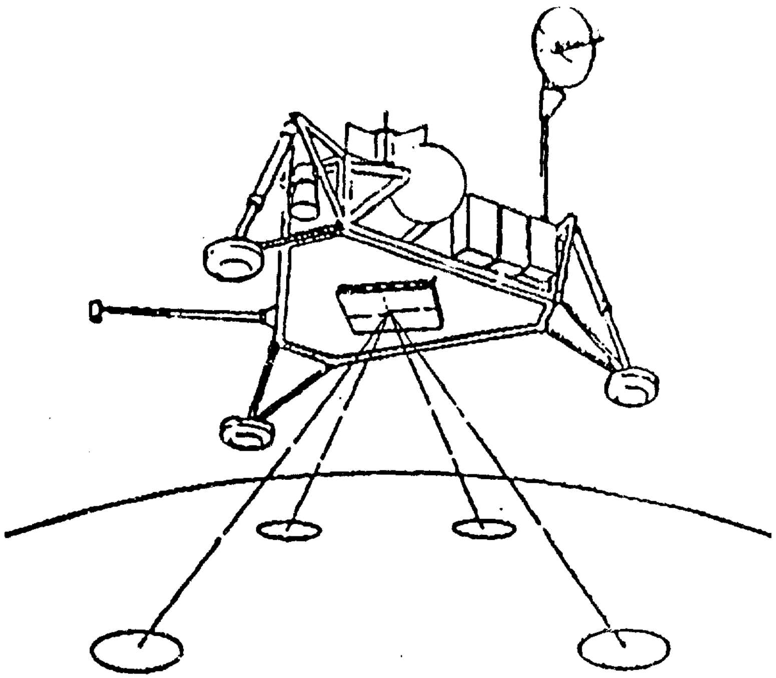
MEASURES 3-AXIS VELOCITIES OF VEHICL ELATIVE TO SURFACE.

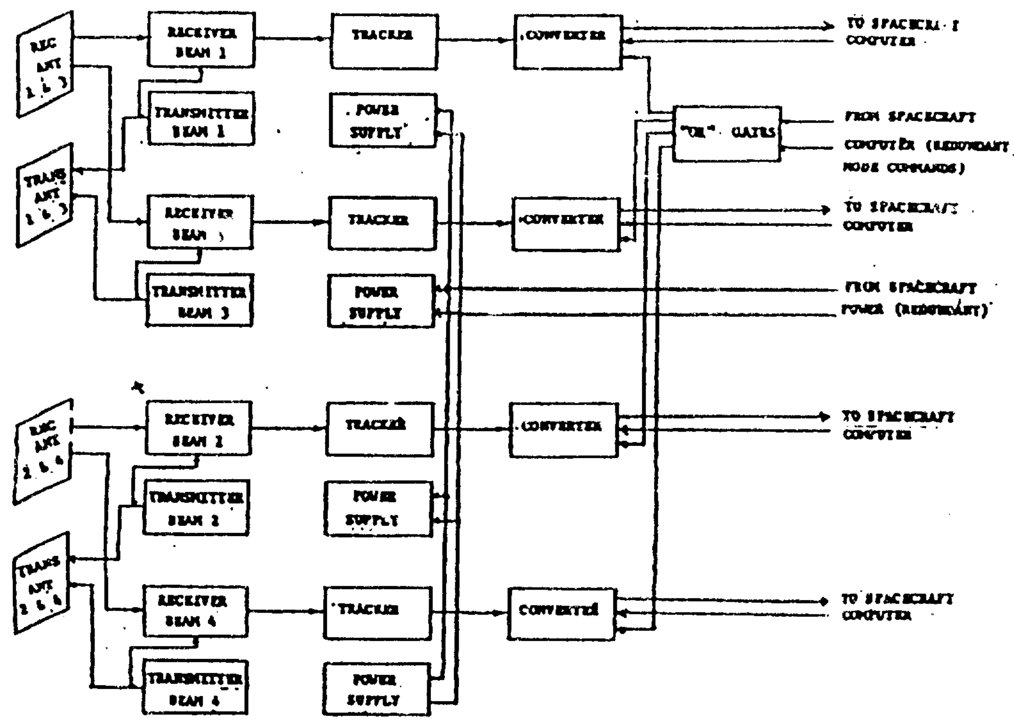
**PERFORMANCE ENVELOPE**

VELOCITY: 0.3 m/s + 3% (V>300m/s)
0.3 m/s + 2% (v<200m/s)

RANGE: 9 m + 5% (R>300m)
1.3 m + 5% (R<300m)

MINIMUM HEIGHT: 30 m.



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# BASELINE UPDATES

RADAR ALTIMETER

MAXIMUM ALTITUDE CAPABILITY DECREASED FROM VIKING REQUIREMENTS.

WEIGHT SAVINGS. POWER SAVINGS.

VELOCITY RADAR

REDUCED TO A THREE CHANNEL SYSTEM FROM A FOUR CHANNEL SYSTEM

25% SAVINGS IN WEIGHT AND POWER. LOSS OF REDUNDANCY CAPABILITY.

PHASE "2" BASELINE WEIGHTS AND POWER

ALTIMETER: 4.9 Kg, 27 W

RADAR: 16.6 Kg, 51 W

SAVINGS

WEIGHT: 6.5 Kg

POWER: 18.5 W

# PROGRAMMATIC CONSIDERATIONS

**PROCUREMENT SCHEDULE**

REDUCED FROM 12 MONTHS TO 9 MONTHS

IMPLIES PRIORITY ACTION AND ON TIME SPECIFICATION DATA.

**HARDWARE DEVELOPMENT SCHEDULE**

REDUCED TO 24 MONTHS FROM 29 MONTHS

INCREASED COST. SOME INCREASED RISK.

**COST**

NON-RECURRING COSTS INCREASED TO $ FROM $

**RELIABILITY**

ORIGINAL MTBF OF 16,400 HOURS REMAINS VALID.

MTBF DRIVEN BY ALTIMETER. RADIATED POWER REDUCTION AIDS MTBF. MTBF FOR FOUR CHANNEL RADAR WAS A SERIAL CALCULATION AND A REDUCTION TO THREE CHANNELS SHOULD IMPROVE MTBF.

PROBABILITY OF SUCCESSFUL OPERATION FOR ONE HOUR IS 0.99939.

# TRACKING SYSTEM CRITICAL PATH

PHASE C/D HARDWARE PROCUREMENT AND DEVELOPMENT SCHEDULE IS 33 MONTHS.

FROM B-LEVEL SPECIFICATIONS TO CONTRACT PLACEMENT IS 9 MONTHS.
THAT INCLUDES WRITING SOW, RFP OR JOFOC, etc.

HARDWARE DEVELOPMENT SCHEDULE IS 24 MONTHS.
THAT INCLUDES QUAL PROGRAM, HELICOPTER FLIGHT TEST, etc.

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# Common Lunar Lander Thermal Control System (TCS)

Steve Rickman/ES3/x38867

## Common Lunar Lander Thermal Control System (TCS) Baseline

- CLL thermal control will be accomplished passively.
- TCS components consist of multi-layer insulation (MLI) blankets, heaters, coatings, and isolators.
- Propellant tanks will be surrounded by MLI blankets and all tanks will be enshrouded in the spacecraft structure. A low emittance coating will cover the inside of the structure to provide further radiative isolation. Tank heaters may not be needed provided that the lunar orbit loiter time is less than about 7 days.
- Heaters will be required to keep the main engine injector flanges above 40 °F. An estimated peak power of 122 Watts (with a 50% duty cycle) is required for the current configuration in a solar inertial attitude.

## Common Lunar Lander Thermal Control System (TCS) (Continued...)

- Propellant line heater power is estimated to be 48 Watts (with a 50% duty cycle).
- Power dissipating avionics boxes will require mounting with a good view to space.
- There may be benefit in using a passive thermal control (PTC) flight mode (Barbeque mode) to reduce heater power requirements.

## Baseline Rationale

- Solar inertial attitude baselined. Current TCS concept is response to the baseline.
- Minimize TCS cost and complexity. Passive TCS is less complex and less costly than active system.
- Passive TCS techniques successful during cruise phase of Surveyor missions.

## Options to Drive Weight Down

- Reduce heater power consumption through isolation, integrated thermal design, and possibly, use of PTC cruise attitude.

## Long Pole in Schedule

- No major schedule, manpower, or cost issues identified.

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# Common Lunar Lander

## Propulsion System Conceptual Design Review

**March 10, 1992**

**Don Hyatt**

Joe Riccio

Dave Amidei

Rich Schoenberg

Todd Lucht

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# Requirements

Delta-V (m/sec)

| | |
|---|---|
| o TLI Completion | 85.3 |
| o Mid-Course Corrections | 30.0 |
| o Lunar Orbit Insertion | 887.0 |
| o DeOrbit | 18.6 |
| o Terminal Descent | 1549.8 |
| o Constant Rate Descent | 285.2 |
| TOTAL | 2855.9 |

Vehicle Dry Mass (kg)

| | |
|---|---|
| o Other Subsystems | 286 |
| o Structure | 72 |
| o Growth | 58 |
| TOTAL | 416 |

Note: 101 kg Jettisoned after DeOrbit

Single Stage

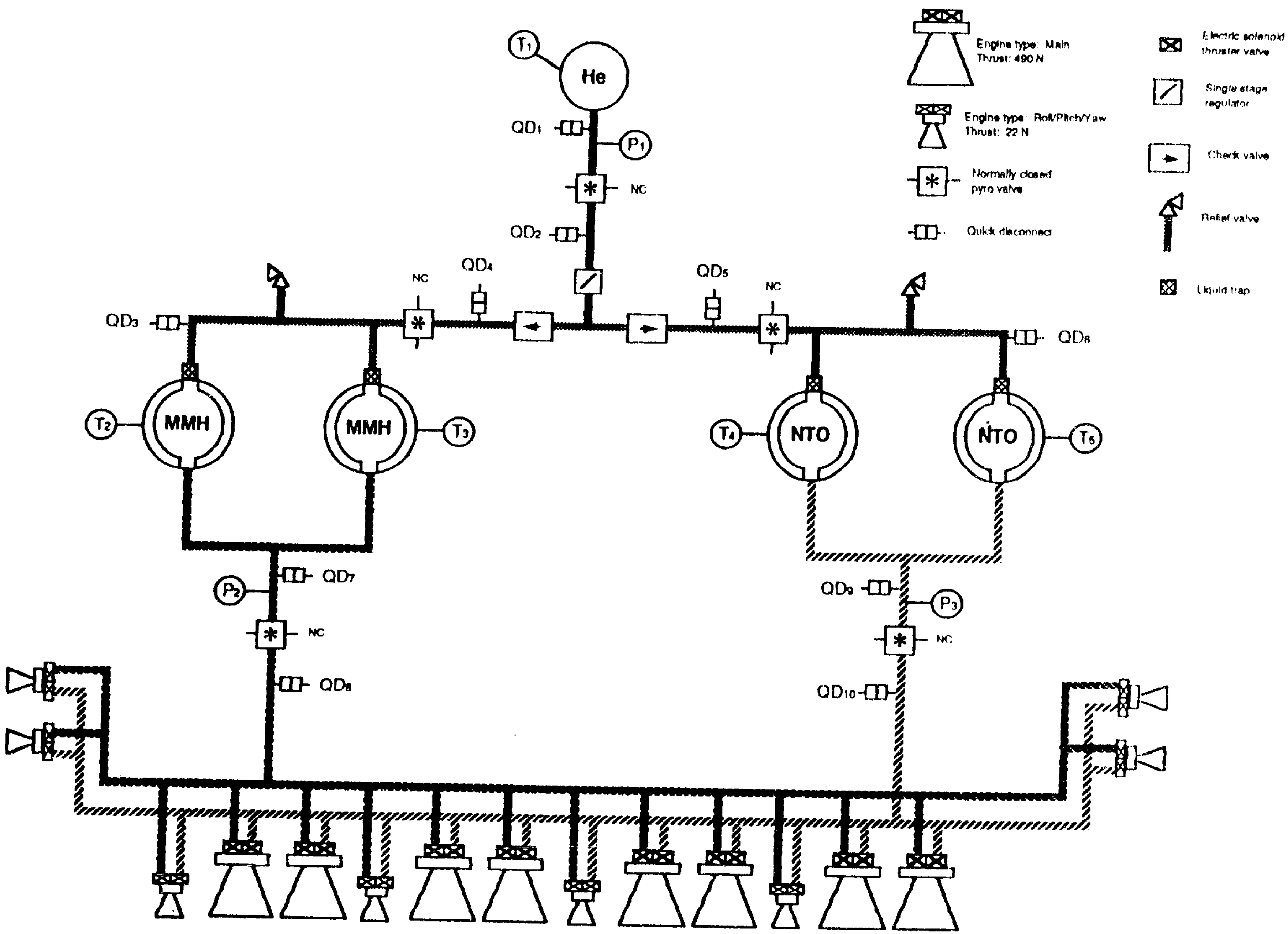
Pulse Mode Thrust Modulation

Four Tank Configuration

Single String

Off the Shelf

# Common Lunar Lander Propulsion System




3/4/92 RJS/JRR

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CLL Propulsion Metric

| | Component | Quantity | Vendor | Mass (lb) | Mass (kg) | Mass tot (kg) | Heritage |
|---|---|---|---|---|---|---|---|
| 1 | Helium tank (19.21" Sphere) | 1 | Brunswick | 19.95 | 9.07 | 9.07 | Intelsat VII |
| | Pyro valve | | | | | | |
| 2 | gas/gas (4500 psi/0.25") | 1 | Unidynamics | 0.50 | 0.23 | 0.23 | |
| 3 | gas/gas (250 psi/0.25') | 2 | OEA | 0.31 | 0.14 | 0.28 | Harpoon |
| 4 | liquid/gas (250 psi/0.75") | 2 | OEA | 1.25 | 0.57 | 1.14 | Viking |
| 5 | Regulator | 1 | Fairchild | 1.10 | 0.50 | 0.50 | Inmarsat/MX |
| 6 | Check valve | 2 | Fairchild | 0.20 | 0.09 | 0.18 | Peacekeeper |
| 7 | Relief valve | 2 | Futurecraft | 0.50 | 0.23 | 0.45 | |
| 8 | Propellant tanks | 4 | PSI/TRW | 15.00 | 6.82 | 27.27 | |
| | Engines# | | | | | | |
| 9 | main | 8 | TMC-R4D | 8.30 | 3.77 | 30.18 | Apollo/RCS |
| 10 | vernier | 8 | TMC-R6C | 1.48 | 0.67 | 5.38 | |
| | Quick disconnects | | | | | | |
| 11 | pressurized gas | 4 | Futurecraft | 0.20 | 0.09 | 0.36 | |
| 12 | pressurized liquid | 6 | Futurecraft | 0.20 | 0.09 | 0.55 | |
| | Transducers | | | | | | |
| 13 | pressure | 3 | Statham | 0.25 | 0.11 | 0.34 | |
| 14 | temperature | 5 | Rosemont | 0.25 | 0.11 | 0.57 | |
| | Filters | | | | | | |
| 15 | prop tank (20 mic/0.5") | 4 | Brunswick | 0.38 | 0.17 | 0.69 | Mariner |
| 16 | pyro outlet (15 mic/0.75") | 2 | Brunswick | 0.93 | 0.42 | 0.35 | Mariner |
| 17 | Lines | | | | | 3.60 | |
| | | | | | | | |
| | Totals | 55 | | | | 81.64 | |
| | | | | | | | |
| | | | | | | | |
| | #R4D: SS Isp=312s @ e=164 | | | | | | |
| | #R6C: SS Isp=289s @ e=100 | | | | | | |
| | | | | | | | |
| | | | | | | | |
| | | | | | EP4/JRR/Feb 28, 1992/Rev 3.3 | | |

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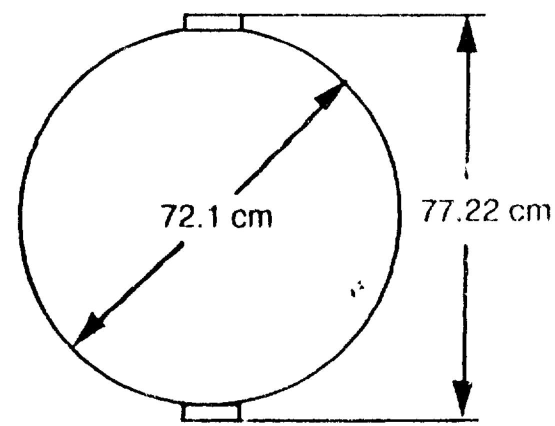
# CLL Mass Statement

| | (lbm) | (kg) |
|---|---|---|
| Fuel Tanks (2) | 30.00 | 13.61 |
| Oxidizer Tanks (2) | 30.00 | 13.61 |
| Pressurant Tanks (1) | 19.95 | 9.05 |
| Engine(s) | 78.24 | 35.48 |
| Lines & Components | 21.50 | 9.75 |
| Fuel, Residual | 14.62 | 6.63 |
| Oxidizer, Residual | 24.12 | 10.94 |
| Pressurant | 5.46 | 2.48 |
| Mounting | 15.99 | 7.25 |
| | ------- | ------- |
| Propulsion, Dry | 239.88 | 108.80 |
| Fuel, Useable | 716.38 | 324.89 |
| Oxidizer, Useable | 1181.88 | 536.00 |
| | ------- | ------- |
| Propulsion, Wet | 2138.14 | 969.69 |

# CLL Propellant Tank

**Propellant Tank Issues**

- Only propulsion system component requiring design, development and qualification testing
- Tanks comprise over 30% of the propulsion system dry mass
- New design based on forging used for APU tanks
- Other tank options explored:
  - Aluminum tanks
  - Diaphragms for propellant management
  - Over-wrapped tanks
- Lead time for tanks 24 months including tank qualification
- $1M non-recurring price
- $115K recurring




**Tank Configuration**

- Ti 6Al-4V Sphere
- 0.58 mm (0.023") Nominal thickness
- 2.07 MPa (300 psia) MOP - 1.5 SF to burst
- Surface tension propellant acquisition
- 7 kg per tank
- 4 tanks per vehicle

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# Propulsion System EMS Inputs

- Engineering Support (16.9 MY)
- Propulsion Breadboard/Development/System Qualification Unit
  - Manpower (15.3 MY)
    - Majority of cost covered under ETB if performed at TTA
    - $1.5M if performed at WSTF
  - Hardware costs ($800K)
- Integrated Test Article for structural analysis ($75K)
- Development vehicle hardware ($550K)
- Component costs for 2 ship sets plus spares ($9.2M)

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## POWER SUBSYSTEM

- Energy Storage and Power Generation
- Electrical Power Distribution and Control (EPDC)
- Pyrotechnics

DESIGN TEAM -- Betsy Kluksdahl - Lead
Bobby Bragg - Energy Storage, Shannan Fisher and Don Allison - Power Generation, Bob Hendrix - EPDC, Darin McKinnis - Pyrotechnics

## SUBSYSTEM DESIGN

- Increased power requirements received from other subsystems
- Power scrub
  - R4D jet driver current decreased from 4 Amp to 1 Amp
  - Delete propulsion tank heater power if lunar loiter less than 7 days
- Thermal analysis of Solar Inertial vs. Passive Thermal Control
  - Propulsion line and engine heater power required
- Baseline B power levels determined
  - Secondary Battery and Solar Array Design

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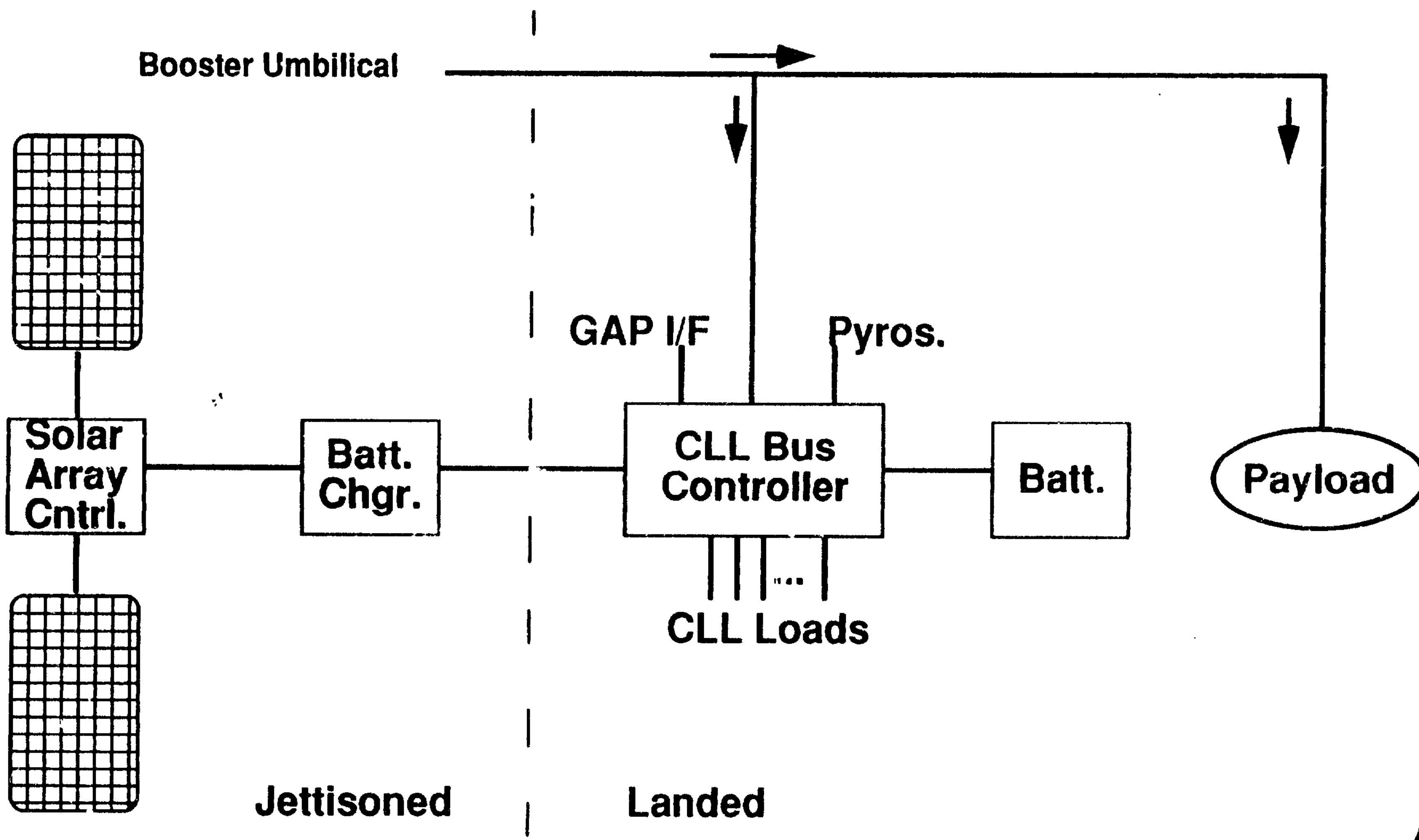
# BASELINE B
# ELECTRICAL POWER SYSTEM




Betsy Kluksdahl/EP5/x36484



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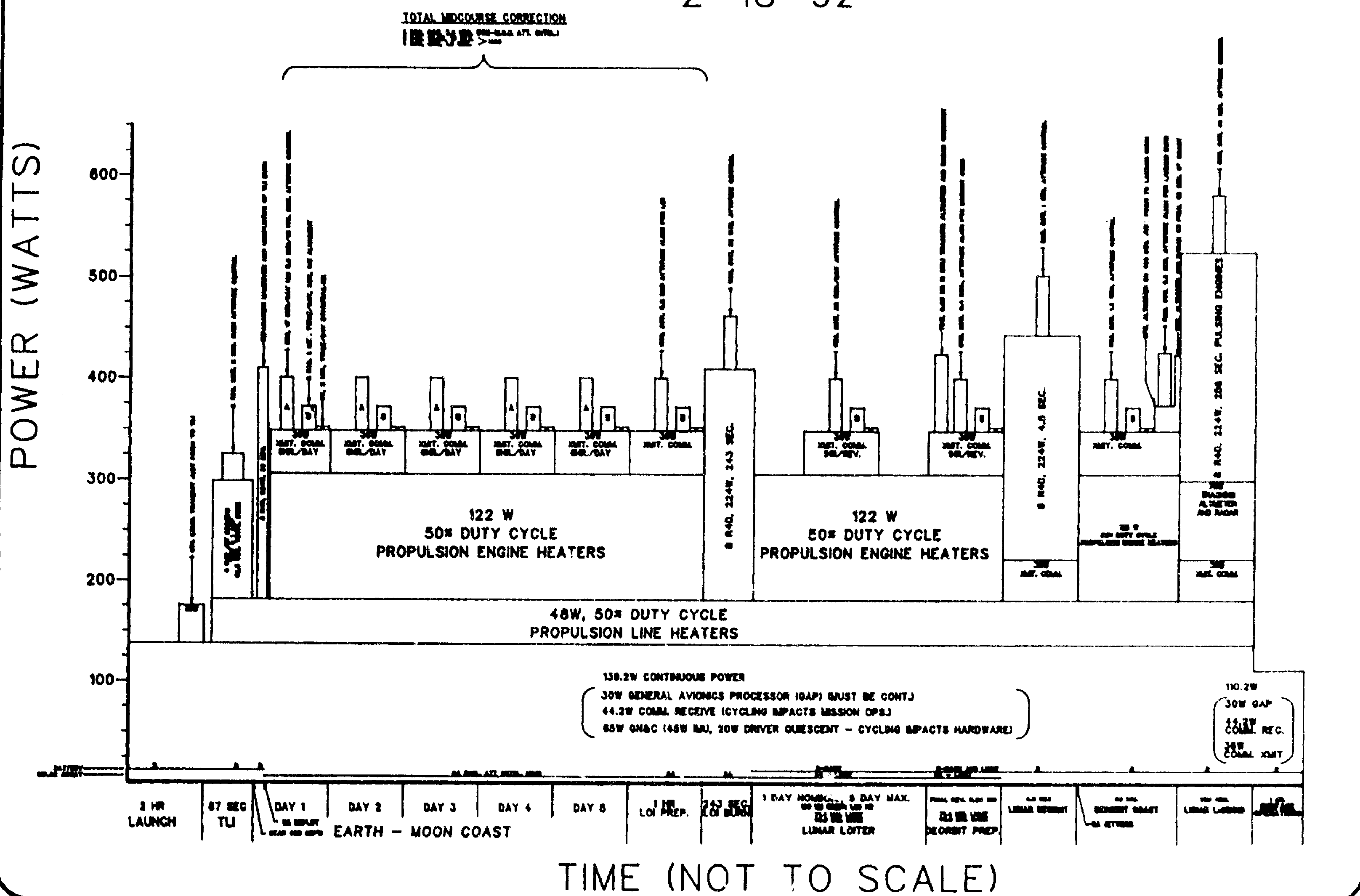
# CLL BASELINE C
# POWER PROFILE
# 2-18-92



# Baseline B Power Subsystem Design for Solar Inertial Orientation



## Jettisoned Prior to Lunar Landing

| | |
|---|---|
| **POWER GENERATION** | |
| 2 Silicon Solar Arrays | 35 kg total (350 W, 5.2 m2 each) |
| **EPDC** | |
| Battery Charger | 9 kg |
| Solar Array Controller | 9 kg |
| Wiring | 4 kg |
| **PYROTECHNICS** | |
| 2 Solar Array Pin Pullers | 2(0.1 kg) |
| 2 Wire Bundle Guillotines | 2(0.77 kg) |
| 12 Explosive Bolts | |
| Solar Array Box | 8(0.11 kg) |
| Star 48B Adapter Ring | 4(0.11 kg) |
| **JETTISON TOTAL** | 60.06 kg |

# Baseline B Power Subsystem Design for Solar Inertial Orientation



## Landed on Lunar Surface

| | |
|---|---|
| **ENERGY STORAGE** | |
| Silver Zinc Secondary Battery | 10 kg |
| **EPDC** | |
| Bus Controller | 18.6 kg |
| Wiring Harness | 15.3 kg |
| **PYROTECHNICS** | |
| 5 Pyrovalves | 5(0.14 kg) [mass carried in propulsion mass statement] |
| 4 Uplock Cutters - leg deploy | 4 (0.26 kg) |
| **LANDING TOTAL** | 44.94 kg |

**TOTAL MASS FOR POWER SUBSYSTEM = 105 kg**

## Design Refinements for Next Phase

- Another power scrub
  - Get power levels for Passive Thermal Control orientation low enough to consider primary battery option again
    - Decrease costs ($3M), MYE, complexity

- Investigate alternate solar array options
  - Solar Inertial orientation

    Alternative to accordion-style array?

    Alternative means of stowage, deployment, placement?
  - Passive Thermal Control orientation

    Body-mounted arrays feasible?

    Mounting schemes to increase surface area for arrays?

END

DATE

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APR 29 1994

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Centimeter

1 2 3 4 5 6 7 8 9 10 11 12 13 14 15 mm

1 2 3 4 5

Inches

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