To: ES63 / McDowell

NASA
National Aeronautics and
Space Administration

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George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama 35812

Lunar Transit Telescope (LTT)

2–m Aperture UV/VIS/IR Telescope

A Feasibility Study
By Program Development

LUNAR TRANSIT TELESCOPE, LTT

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KEY PERSONNEL

ACKNOWLEDGEMENTS

We gratefully acknowledge the support and contributions to this study from NASA Headquarters, the scientific community and research institutions. The study was initiated from NASA Headquarters by Mike Kaplan, Code S. The scientific community, especially John McGraw, U. of Arizona/Steward Observatory, identified and defined the mission and scientific objectives. The optical architecture of the telescope was defined by Dietrich Korsch of Korsch Optics. Definition of the lightweight optical elements was enhanced by Kaman Aerospace Corporation, Pat McMillian, et. al; Roger Angel, U. of Arizona; R. Locke of Kodak Corporation; and Mike Krim of Hughes Danbury.

12/27/91:BGD

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LUNAR TRANSIT TELESCOPE

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SECTION 1

1. INTRODUCTION **PAGE**

1.1 BACKGROUND AND GUIDELINES

LUNAR AND HEO ASTRONOMY STUDIES

As part of the ongoing advanced telescope studies, MSFC has studied several missions over the past two years. Starting in 1989, several candidate missions were identified and inputs made to the Astro Tech 21 program. Subsequently, MSFC was assigned study lead on the 16 m Large Lunar Telescope (LLT). The LLT primary mirror is composed of eighteen clusters of segments, each 4 meters in diameter. One cluster forms the primary mirror of the Cluster Telescope Experiment (CTE) which serves as a test bed for the larger LLT, especially optical components, alignment, transportation systems and lunar operational environment.

The 6 m High Earth Orbit Telescope (HEOT) is also based upon the cluster of segments approach, and is a candidate for the next generation Hubble Space Telescope (HST). All three missions, LLT, CTE and HEOT, require a new Heavy Lift Launch Vehicle (HLLV). The question was posed: What size telescope can be placed on the moon using an existing launch vehicle? This led to the conceptual definition of the 2 m Lunar Transit Telescope (LTT) which is designed for launch on the Titan IV/Centaur. Feasibility reports have been published on each of the listed missions. Assuming authorization by NASA Headquarters, a phase A study on the LTT could be started in early 1992.

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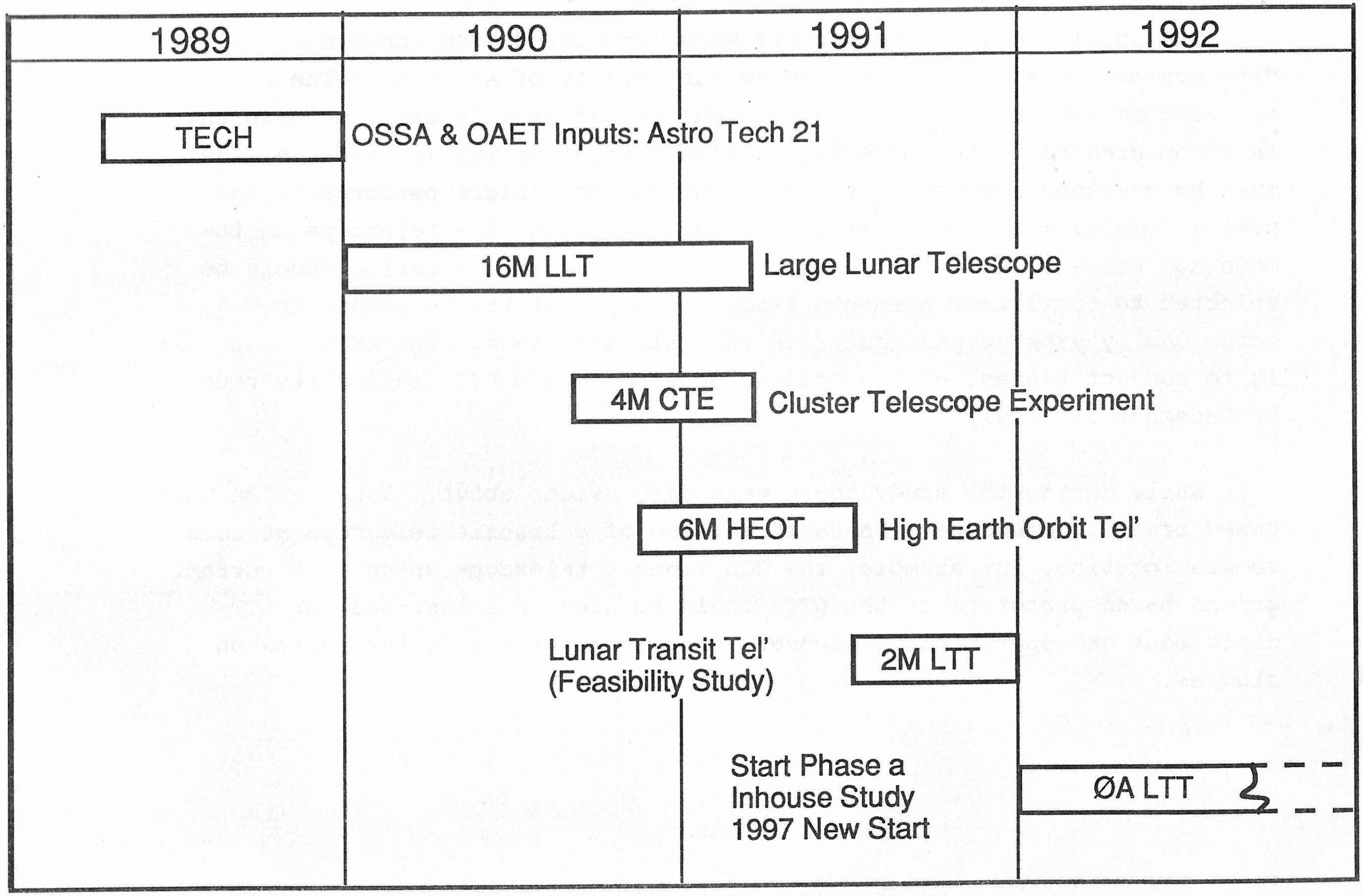
Lunar and HEO Astronomy Studies

CY 1989, 1990, 1991 and 1992



July 10, 1991

LTT INITIAL GUIDELINES

The initial guidelines for LTT were formulated from conversations with Mike Kaplan (Code S) and John McGraw (University of Arizona). The LTT must be launched with an existing launch vehicle and provide early science that is associated with the Space Exploration Initiative (SEI). As such, the LTT must be designed subject to the existing launch vehicle performance and payload constraints. A lunar lander must soft land the telescope on the moon and serve as a foundation for the LTT. The site location should be selected to facilitate a smooth landing and permit the telescope to occasionally view perpendicular to the Galactic plane. The MSFC study team is to conduct trades, define options and complete a LTT feasibility report by December 31, 1991.

Early during the study there were discussions about a lunar vs an Earth based transit telescope. An Earth version of a transit telescope at some remote location, for example, the CCD transit telescope which is a current ground based prototype of the LTT, could be used as a test bed for LTT deployment and operations. However, this is an area left for follow-on studies.

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LTT INITIAL GUIDELINES

1. LAUNCH DIRECT TO THE MOON WITH AN EXISTING LAUNCH VEHICLE

2. DESIGN TO LAUNCH VEHICLE PERFORMANCE

3. MAKE PRIMARY MIRROR AS LARGE AS POSSIBLE SUBJECT TO LAUNCH CONSTRAINTS

4. ASSUME A LARGE CCD DETECTOR WITH HIGH DATA RATES

5. TELESCOPE SHOULD VIEW A SWATH NEARLY PERPENDICULAR TO THE GALACTIC PLANE

6. LUNAR LOCATION AT SOME AN INTERMEDIATE LUNAR LATITUDE

7. LANDER SERVES AS A BASE FOR THE TELESCOPE

8. DESIGN MULTIPLE LUNAR TRANSIT TELESCOPES

9. MODIFY ONE LTT FOR EARTH BASED VERIFICATION

10. CONDUCT TRADES AND EVALUATE OPTIONS

11. COMPLETE FEASIBLITY STUDIES BY NOV. 29, 1991

12. PUBLISH A REPORT BY DEC. 31, 1991. (FACING PAGES)

14. START PHASE A STUDIES, JAN. 1 1992

1.2 MSFC STUDY

LTT 1991 STUDY SCHEDULE

The LTT schedule is keyed to a June 1991 kickoff, a mid-term review to Mike Kaplan (Code S) and John McGraw (University of Arizona) and a final report by Jan 1, 1992. Weekly working group meetings were scheduled and held to discuss progress, problem areas and present current findings. General trades and tasks are listed and a name or names (not shown) was assigned to each. About 15 to 20 individuals have contributed to this LTT study. Some dedicated almost 100% of their time, others about half-time, and still some others were used only as consultants or to answer a specific question.

Most of the technical work was completed by November and a reference LTT defined. During October, a quick study was conducted to determine if a small telescope could be launched with an Atlas IIAS/Centaur. If encouraged by the scientific community and authorized by NASA Headquarters, a phase A study could be initiated in early 1992.

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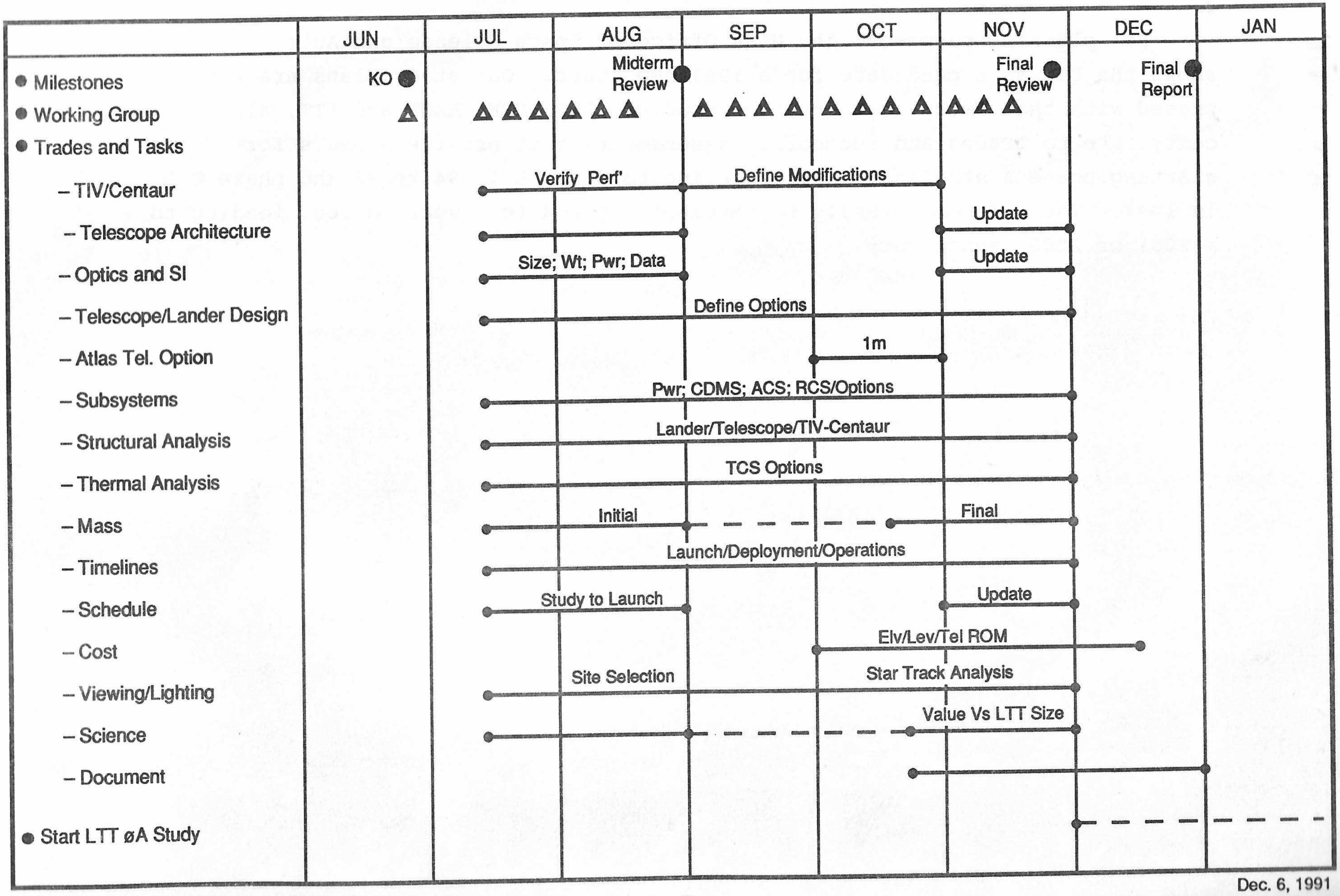
Lunar Transit Telescope 1991 Study Schedule



Dec. 6, 1991

LTT DEVELOPMENT PLANNING

For planning purposes, the NASA Office of Space Science and Astrophysics shows the LTT as a candidate for a 1997 new start. Our study plans are time phased with that date. Our previous studies, LLT, CTE, HEOT and LTT, all contribute to trades and technology assessments that provide a basis for starting phase A studies in 1991, leading to phase B in 94 to 96 and phase C/D in 1997. The LTT could easily be developed in a 4 to 5 year period, leading to a 2001 or 2002 launch date.

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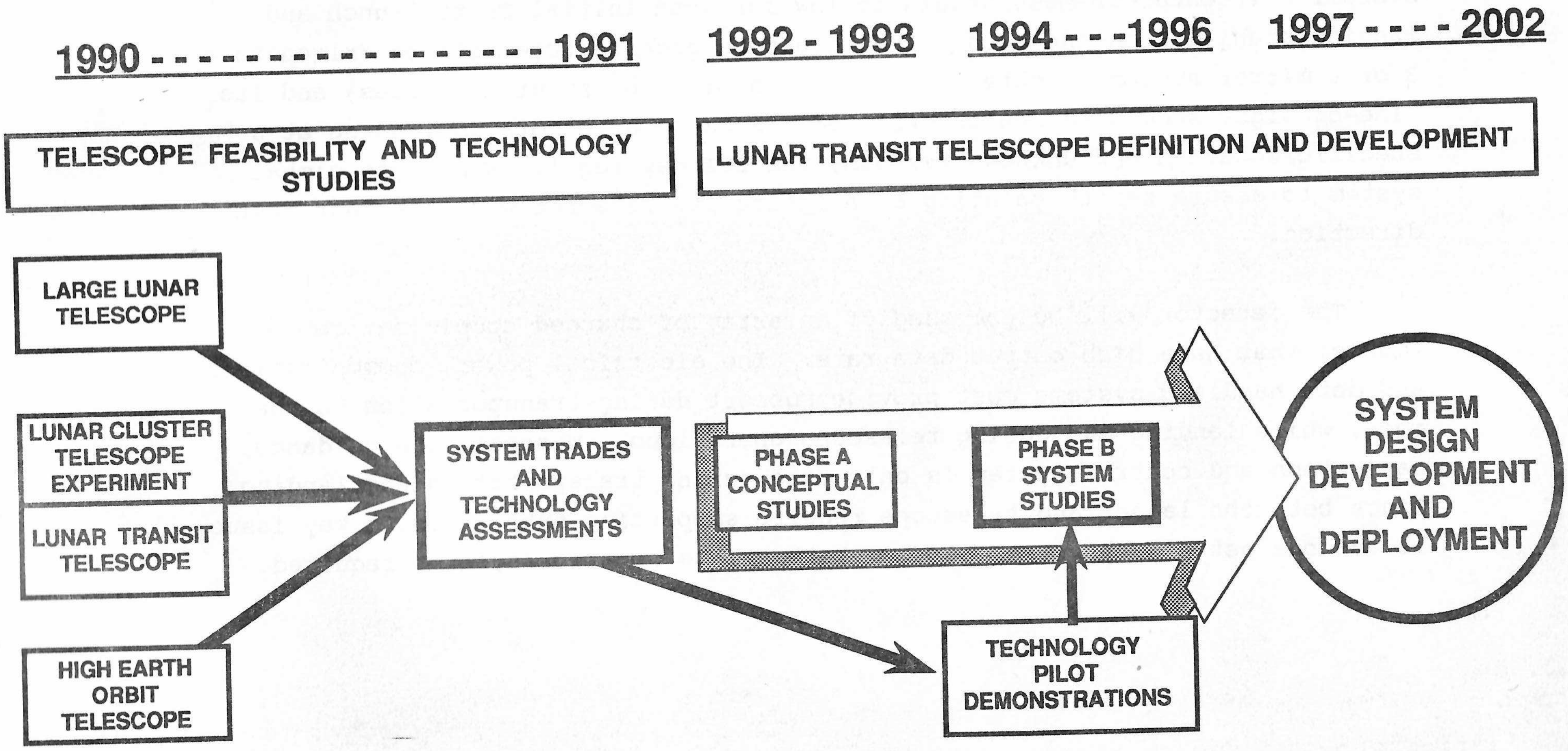
LUNAR TRANSIT TELESCOPE DEVELOPMENT PLANNING

SYSTEM AND TECHNOLOGY STUDIES, DEMONSTRATIONS AND DEVELOPMENT



KEY ISSUES

Several of the key issues to be addressed during the study are listed on the facing page. The major design driver is to make the telescope as large as possible subject to the launch and lunar landing vehicle constraints. The overall LTT center-of-mass should be low for both initial Earth launch and landing stability on the moon. The telescope architecture will be driven to a 3 or 4 mirror system to obtain a wide field-of-view (about 2 degrees) and its line-of-sight should be zenith oriented. Because of errors in landing at a specific location and unknown terrain, the LTT may require some type gimbal system to assure zenith pointing or aligning its detector with the East-West direction.

The detector will be composed of an array of charged couple devices (CCD's) that have high output data rate. The electrical power, communications and data handling systems must provide support during transportation to the moon, while landing and during telescope operations. Whereas, the guidance, navigation and control system is only used during transportation and landing. Since both the lander and telescope require supporting subsystems, a key issue is can one set of subsystems support both or are separate systems required.

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KEY ISSUES

1. INTEGRATED VS SEPARATE LANDER/TELESCOPE VS LAUNCH WEIGHT LIMIT
2. DESIGN FOR LOW CG LOCATION FOR LAUNCH AND LUNAR LANDING
3. SITE LOCATION VS CELESTIAL VIEWING, EARTH COMMUNICATIONS AND SUNSHADE
4. ELECTRICAL POWER SOURCE VS WEIGHT AND SAFETY ISSUES
5. HIGH DATA RATES ASSOCIATED WITH AN ARRAY OF SEVERAL CCD'S
6. THERMAL PROTECTION FROM THE LUNAR DAY/NIGHT EXTREMES
7. EFFECT OF THE LUNAR ENVIRONMENT ON THE TELESCOPE (DUST; RADIATION)
8. COMBINED VS SEPARATE LANDER AND TELESCOPE SUBSYSTEMS
9. OPTICAL SYSTEM TYPE VS MIRROR WEIGHT. 3 VS 4 MIRROR TELESCOPE
10. FIXED VS LIMITED GIMBAL OPTICS
11. DESIGN SELECTION FOR THE METERING STRUCTURE
12. STORABLE VS CRYOGENIC PROPELLANTS (WEIGHT, VOLUME AND PERF')
13. LANDING GEAR DESIGN VS LANDING STABILITY AND LOADS
14. ALIGNMENT OF TELESCOPE, DETECTOR, ANTENNAE AND SUNSHADE

TYPICAL LTT TRADES AND TASKS

During the LTT study, the team conducted trades and complete discipline oriented tasks to define a reference LTT design. The LTT study should center around the preliminary design of the telescope and its lunar lander. The ensuing reference design should serve as a point of departure for additional studies and as a base for comparing new designs. Currently the Titan IV/Centaur launch vehicle is well defined and its performance parameters established. The LTT with its lander must be within the translunar injection weight of the Titan IV/Centaur. Since no lunar landers currently exist, a new lander design that is compatible with the Titan IV/Centaur must be part of the LTT study objectives. One major issue is the degree of integration between the lander and telescope.

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TYPICAL LTT TRADES AND TASKS

1. DETERMINE THE LUNAR ENVIRONMENT
2. TARGETS AVAILABLE: SITE LOCATION
3. LAUNCH VEHICLE PERFORMANCE. EARTH TRANSPORTATION/TRANSLUNAR INJECTION
4. LUNAR LANDER DESIGN INTEGRATED WITH TELESCOPE
5. TELESCOPE ARCHITECTURE AND SCIENCE INSTRUMENTS
6. TYPE OPTICS, MASS AND SIZE
7. LTT REFERENCE DESIGN AND CONFIGURATION OPTIONS
8. SUBSYSTEMS POINT DESIGN AND OPTIONS EVALUATION
9. STRUCTURAL ANALYSIS AND LIGHT WEIGHT MATERIALS EVALUATION
10. THERMAL SIMULATIONS AND CONTROL OPTIONS
11. INITIAL AND FINAL MASS STATEMENTS
12. TIMELINES: LAUNCH, DEPLOYMENT AND OPERATIONS
13. SCHEDULE DEFINITION AND COST ESTIMATES
14. PREPARE BRIEFINGS FOR MANAGEMENT, DOCUMENT STUDY FINDINGS

LTT IS THE INTEGRATED TELESCOPE AND LANDER WITH THEIR SUBSYSTEMS

Candidate transportation options are the Atlas IIAS/Centaur, Shuttle/IUS, Titan IV/Centaur, modified Titan IV/Centaur and a new launch vehicle with upper stage. Of the existing launch vehicles with upper stages, the Titan IV/Centaur has the greatest lift capability. Subsequently, it was selected at the beginning of the study to launch LTT into a translunar injection (TLI) orbit. The mass in TLI includes the propellant required for a soft lunar landing, the inert (dry) lander, the telescope and sub-systems for both the telescope and lander. The LTT mass is everything that lands on the lunar surface. A key issue was, can common subsystems support both the lander and telescope, or must each have its own independent subsystems.

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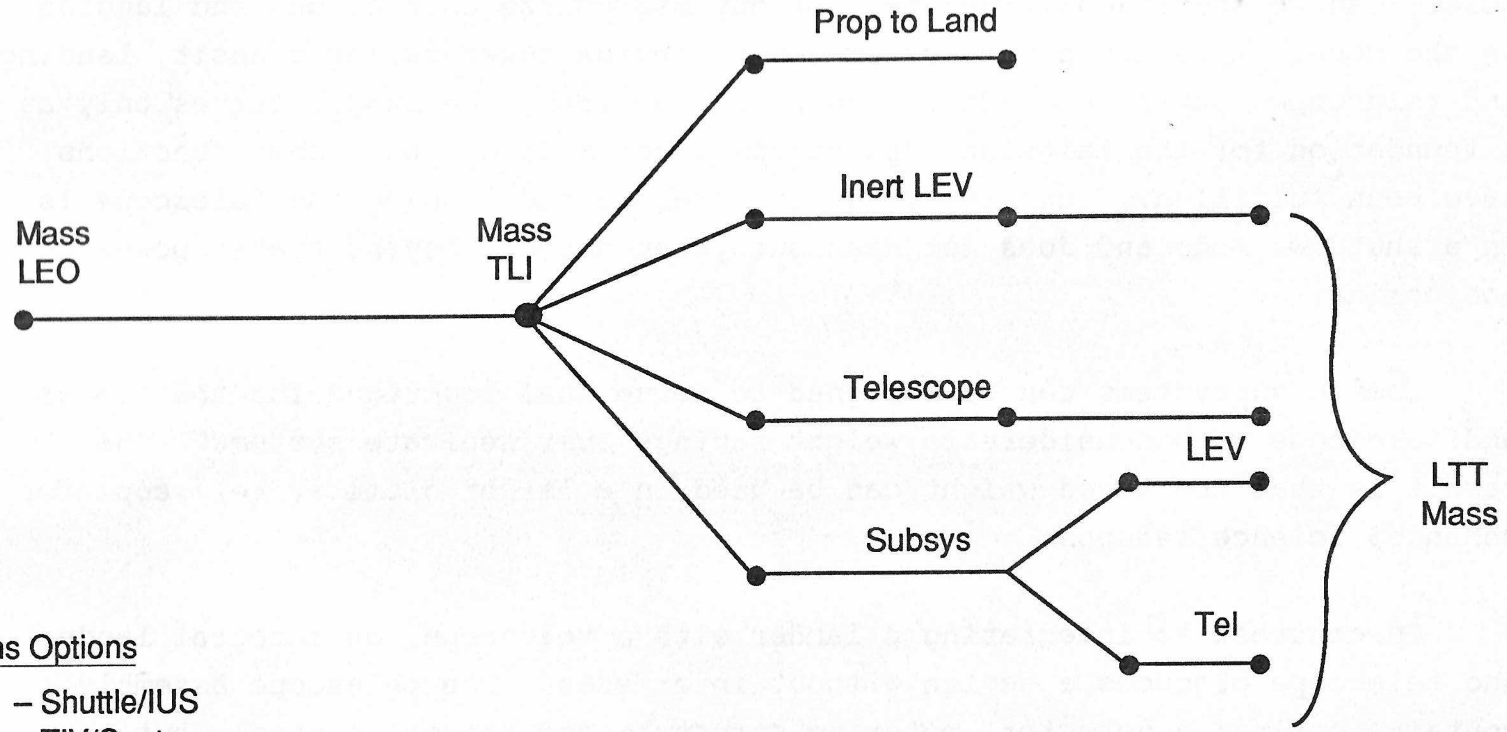
Lunar Transit Telescope is The Integrated Telescope and Lander With Their Subsystems



Trans Options

- Shuttle/IUS
- TIV/Centaur
- TIV + USRM/Modified Centaur
- New NLS With New Upper Stage

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INTEGRATED LUNAR LANDER AND TELESCOPE

A top level assessment of the subsystems requirements of both the lander and telescope indicate that they have similar needs and that common sub-systems could be designed to serve both. During transit to the moon, communication must be maintained with the Earth so that the guidance, navigation and control (GN&C) can be continuously updated for any mid-course corrections and landing on the moon. A common power system could provide power during transit, landing and telescope operation. After landing on the moon, the lander serves only as a foundation for the telescope operations since most of the lander functions have been fulfilled. Conversely, during transit and landing the telescope is in a shutdown mode and does not need subsystem support beyond heater power and monitoring.

Common subsystems can be designed to serve dual functions for the lander and telescope with considerable weight savings over separate systems. The result is that the saved weight can be used in a larger diameter telescope for enhanced science returns.

In contrast to integrating a lander with a telescope, an integral lander and telescope produces a design without interfaces. The telescope assembly contains optics, a detector, metering structure and thermal control, but no subsystems. The lander assembly contains subsystems such as CDHS, GN&C, EPS, propulsion and structure, but no optical components. An integrated telescope and lander is baselined for this study. The total ensuing system is called the LTT.

INTEGRATED LUNAR LANDER AND TELESCOPE

1. LANDER SERVES AS A FOUNDATION FOR THE TELESCOPE
2. COMMON SUBSYSTEMS SERVE DUAL FUNCTIONS
3. EPS AND CDHS ARE ACTIVATED AT LAUNCH SERVES BOTH LANDER AND TELESCOPE
   - DURING TLI AND LANDING SUPPORTS LANDER AND MONITORS LTT
   - DURING TELESCOPE OPERATIONS SUPPORTS LTT ONLY
4. SUPPORTING SUBSYSTEMS CAN BE MOUNTED LOW ON THE CONFIGURATION
   - LOWER CG AT LAUNCH INCREASES TITAN/CENTAUR PERFORMANCE
   - LOWER CG AT LANDING ENHANCES STABILITY
5. CONSIDERABLE WEIGHT SAVINGS OVER SEPARATE SYSTEMS
6. PERMITS A LARGER OPTICAL SYSTEM TO BE LAUNCHED
7. AN INTEGRATED TELESCOPE/LANDER PRODUCES CLEAN INTERFACES
   - THE TELESCOPE ASSEMBLY DOES NOT CONTAIN SUBSYSTEMS
   - THE LANDER ASSEMBLY CONTAINS ALL SUBSYSTEMS AND PROPULSION

ALIGNMENT REQUIREMENTS

Various parts of the LTT have definite orientation and pointing accuracy requirements. The high gain antennae must point toward the Earth with a 1 degree accuracy, the telescope sunshade should be oriented toward the lunar equator (southward for northern latitude sites) with 1 degree accuracy, and the detector array should be aligned east-west with an accuracy in the arcsecond range. Ideally, the telescope line of sight should be aligned with local zenith. There has been considerable discussion about the effects of off zenith pointing, either in the meridian (north-south) or out of the meridian (east-west). However, late in the study, the consensus of opinion is that it is necessary that the LTT points at the planned for declination, its alright if it is leaning east or west, just so it is pointing at the declination for which the focal plane was designed. Some type mechanisms and software are required to counteract alignment errors.

The facing chart shows typical initial conditions that are expected just prior to landing, the slope tolerance that should be built into LTT, the alignment requirements and LTT needs. These are soft and subject to much debate and change as more assessments are made.

ALIGNMENT REQUIREMENTS

* COMPENSATE FOR UNCERTAINTIES IN LUNAR TERRAIN
  - RESOLUTION OF AVAILABLE LUNAR SURFACE PICTURES
  - SIZE OF ROCKS AND BOULDERS
  - DEPTH OF SMALL CRATERS
  - SOFTNESS OF REGOLITH AND DEPTH OF DUST

* OVERCOME EFFECTS OF INITIAL CONDITIONS JUST PRIOR TO IMPACT
  - VERTICAL VELOCITY OF 3 M/SEC (10 FT/SEC)
  - HORIZONAL VELOCITY OF 1 M/SEC (3 FT/SEC)
  - UP TO 0.5 DEGREES ATTITUDE ERROR ON ALL AXES
  - POSITION ERROR OF 100 M (300 FT)

* OVERCOME FORCE AND MOMENTS IMPARTED TO LTT AT LUNAR IMPACT
  - DESIGN FOR SLOPE TOLERANCE UP TO 12 DEGREES
  - ROCKS AND BOULDERS UP TO 1-2 M IN DIAMETER; 0.5 M DEPTH
  - LANDING CAN IMPART ROLL ERROR TO LTT

* ALIGNMENT REQUIREMENTS
  - SUNSHADE APEX MUST POINT SOUTHWARD WITHIN 3 DEGREES
  - DETECTOR ARRAY ALIGNED E-W WITHIN 7 ARCSECOND
  - LINE-OF-SIGHT TOWARD A PREPLANNED MERIDIAN, (SEVERAL ARCMINUTES)
  - EAST-WEST ALIGNMENT OF TELESCOPE, (SEVERAL DEGREES)

* LTT NEEDS
  - COARSE E-W DETECTOR ALIGNMENT MECHANISMS (10-15 DEG RANGE)
  - FINE E-W DETECTOR ALIGNMENT MECHANISMS, 1-2 ARCSEC ACCURACY
  - SUNSHADE ROLL MECHANISM FOR 1 DEG. ACCURACY, 10-15 DEG. RANGE
  - LINE-OF-SIGHT CORRECTION, 1 ARCMIN. ACCURACY, 12 DEG. RANGE

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SECTION 2

2. PROGRAM DESCRIPTION

PAGE

2.1 SCIENCE OBJECTIVES[1]

HOW LTT OPERATES

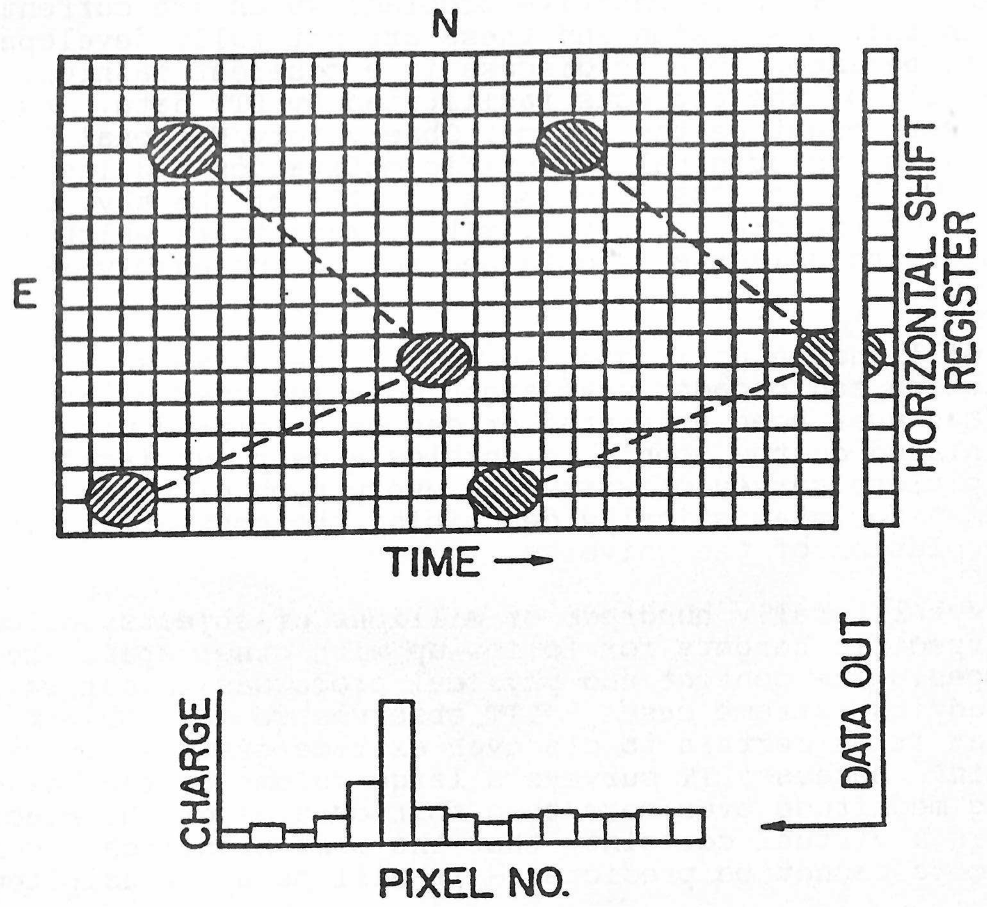
Before describing the issues of the scientific programs addressable by LTT, it is necessary to understand how the telescope operates. The basic concept of operation is schematically illustrated. The telescope takes advantage of the read-out scheme for frame transfer charge couple device (CCD) detectors. The detection technique, known as "time-delay and integrate" (TDI), has been used for years for surveying moving objects, or stationary objects from moving vehicles. For single element detectors, it is also known as the "push-broom" detection scheme. In the LTT, CCD detectors are aligned from east to west in the focal plane and clocked at the apparent sidereal rate. This allows the electrical image formed in the CCD to move virtually synchronously with the optical image as it crosses the focal plane of the telescope. In this technique, the physical length of the CCD determines the integration time per transit and the physical width of the device determines the area surveyed. Numerous constraints apply and will be discussed in the section on the LTT focal plane array.

The LTT utilizes as the "drive" of the telescope the read-out system of a frame transfer CCD. In this read-out system, a clock sequence in the "vertical" (time) direction of the CCD results in the last (westernmost) row of charge being transferred into the horizontal shift register. Simultaneously, every other row is shifted one pixel towards the horizontal shift register (west).

A star field, represented by the similarly hatched and connected images, is shown at two times as it transits on the backside of one CCD. A computer system provides CCD "vertical" clock pulses so that charge moves along the array at the apparent sidereal rate (determined by the moon's rotation rate). As data "fall off" the western edge of the array into the horizontal shift register, they are read out at high speed (much less than one vertical clock period), digitized and stored in computer memory.

[1] AN EARLY LUNAR-BASED TELESCOPE: THE LUNAR TRANSIT TELESCOPE (LTT), Dr. John McGraw, Steward Observatory, University of Arizona

HOW LTT OPERATES



LTT SCIENTIFIC PROGRAMS

This section briefly addresses the types of scientific programs LTT will address. For a number of reasons, this discussion should be taken as indicative only. Only representative problems which are currently familiar are included in this discussion and these are not fully developed. It would, in fact, be impossible to discuss in a coherent fashion, and in limited space, all of the projects facilitated by LTT data. An appropriate paradigm, however, might be the Palomar Observatory/National Geographic Sky Survey (POSS) with its original scientific motivation and its continued exploitation. While the POSS surveyed about 70% of the sky, LTT will survey perhaps only 2% of the sky - but to limiting magnitudes which ensure that a "fair" sample of the Universe is obtained for an incredibly large number of programs.

The LTT will enable an imaging survey of the Universe with higher angular resolution and broader wavelength coverage over a larger fraction of the sky than has ever been attempted or can be attempted until this telescope is placed on the moon. It enables a deep, unbiased, statistically significant complete survey of virtually every type of object. It is an ideal instrument for statistically describing the content, structure, texture and evolution of the Universe.

LTT surveys literally hundreds of millions of objects, allowing selection of specific targets for follow-up with other space and ground based telescopes. The content and physical processes in our Universe are learned by studying extreme cases - LTT observes so many objects so completely that it is certain to discover extreme examples in virtually any physical domain. Because LTT surveys a large volume of the Universe to faint limiting magnitude over more than four octaves of the electromagnetic spectrum, it is a virtual certainty that the most significant contribution of this telescope cannot be predicted - it will be a serendipitous discovery.

LTT SCIENTIFIC PROGRAMS

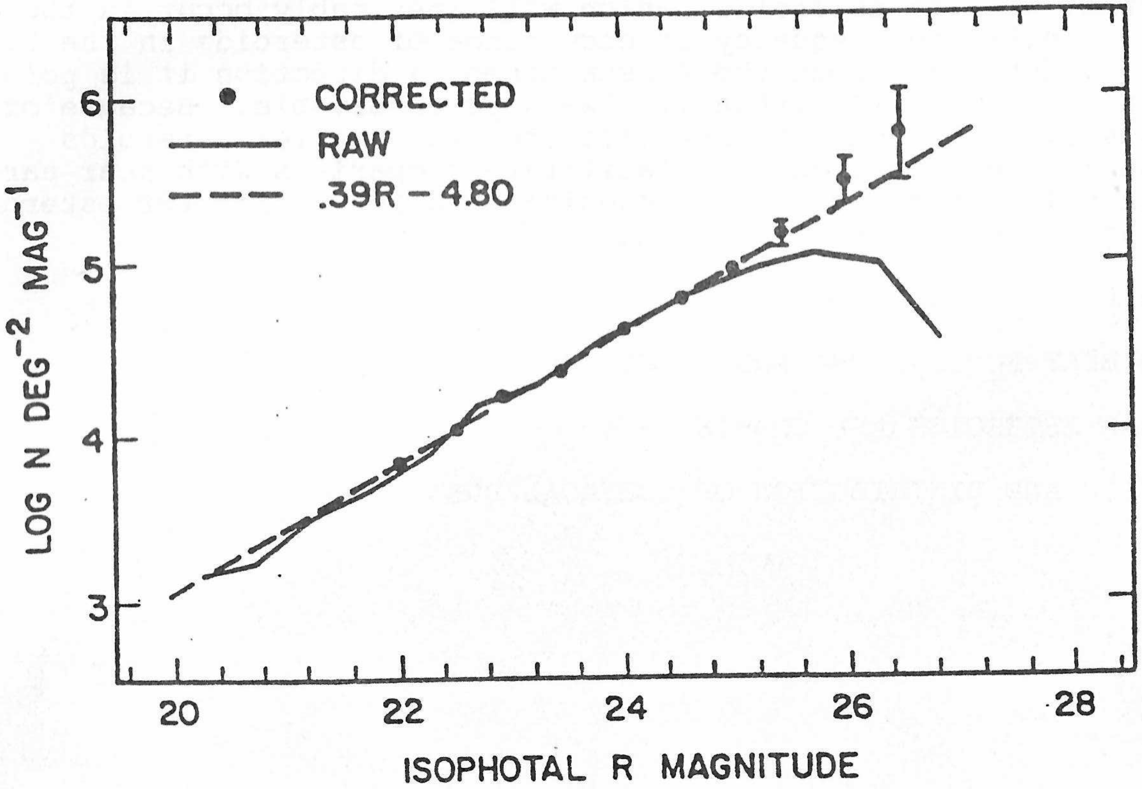
- EXTRA GALACTIC ASTRONOMY
- GALACTIC ASTRONOMY
- SOLAR SYSTEM ASTRONOMY
- RELATIVITY

EXTRAGALACTIC AND GALACTIC ASTRONOMY

At isophotal magnitude R = 25, there are observed approximately 10,000 galaxies per square degree per magnitude interval . The figure, abstracted from Tyson (1988), shows differential number counts for galaxies observed in 12 high galactic latitude fields in a ground-based, deep CCD imaging survey of essentially "blank fields". The LTT survey area is approximately 800 square degrees, over which approximately 30% of the sky will be confusion limited by galaxies in optical and near infrared bandpasses. With 0.1 arcsec pixels, imaging will allow surface photometry and morphological classification for field and cluster galaxies brighter than the confusion limit. Colors will allow a large-scale investigation of galaxy structure and evolution to $z > 1$. With sufficient understanding of evolutionary effects in galaxies, direct geometrical tests of q_0 become possible.

Much effort has been and continues to be expended on the search for the elusive brown dwarfs which star formation and evolution theory predict should exist. The faint end of the stellar luminosity function remains ill defined, again for the reason that it is observationally difficult to unambiguously isolate these objects. LTT will contribute to the search for and definition of the statistics of these faint, red objects. In particular, in the LTT field of view, the faintest stars and brown dwarfs will be visible magnitude (V) in a volume greater than 250 cubic kiloparsecs, that is, in a sphere of radius four kiloparsecs. In addition to optical and near infrared colors, because LTT makes multiple observations of the same objects over an 18.6 year mission, proper motions, derived either from LTT alone or from combinations of surveys, can be utilized as a discriminant for nearby objects. For stars brighter than approximately V = 27, single positional measurement errors are about 1-2 milliarseconds. It will be possible to measure parallaxes for stars at distances of several hundreds of parsecs.

DIFFERENTIAL GALAXY COUNTS



Differential galaxy counts obtained in the R bandpass by Tyson (1988). At an isophotal magnitude of R = 25 we expect approximately 10,000 galaxies per square degree per magnitude. Filled dots represent counts corrected by simulations for undercounts and the dashed line is a continuation of the N(m) relationship found at brighter magnitudes.

SOLAR SYSTEM ASTRONOMY

Nearer still will be asteroids, which will inevitably occur in the LTT field of view. While the frequency of occurrence of asteroids in the LTT field of view is dependent upon the (instantaneous) direction it is pointed, asteroids of some class will virtually always be observable. Because of its faint magnitude limit LTT can observe kilometer and smaller asteroids throughout the asteroid belt and can facilitate comparison with near-earth and other main belt asteroids. Near opposition, a 100 m diameter asteroid will have a magnitude $V \leq 27$ at a distance of 3 AU.

- KUIPER BELT COMET SEARCH
- ASTEROID BELT POPULATIONS AND STRUCTURE
- INDIVIDUAL ASTEROIDS AND COMETS
- COMPOSITION AND DISTRIBUTION OF ZODIACAL DUST

RELATIVITY

Finally, a possible contribution to the test of general relativity is discussed. This suggestion, which originated with an undergraduate student at the University of Arizona, is simply the Eddington experiment revisited - but in a continuous manner. We are currently investigating the potential of this suggestion simply because of all the fundamental theories upon which physics is currently based, general relativity is by far the least firmly tested by observation and measurement. At the limb of the sun, light rays are gravitationally deflected by approximately 1.7 arcsec, with the deflection amplitude decreasing inversely as the impact parameter to an amplitude of about 4 milliarcseconds normal to the earth - sun line. The direction of deflection is, of course, radially outward from the center of the sun. This effect will act as a major perturbation to all-sky astrometry obtained by the LTT. By allowing the LTT field to pass near the sun, repetitive measurements of literally thousands of stars with individual positional uncertainties of 1-2 milliarcseconds can be obtained. Those obtained near solar opposition can readily be compared with those near conjunction in a coordinate system tied to quasars to derive the amplitude and direction of deflection over 2 radians. Effectively, this experiment can repetitively map the solar gravitational potential to relatively large radius to test the radial dependence of the deflection, as well as the amplitude and direction.

Clearly, a discussion of scientific programs enabled by LTT data could be very long and detailed. There are two conclusions, however, to be drawn even from this cursory discussion. The first is that LTT development can readily be justified on the basis of multiple, fundamental scientific programs which are currently in the forefront of astrophysical investigation. The second conclusion is that the elegant simplicity of LTT does not lead to trivial scientific programs. On the contrary, the programs addressable by LTT data are absolutely fundamental, and philosophically shall remain so into the indefinite future, especially in the domain of astrophysics in which the knowledge of objects, structures and processes in the Universe are broadened.

PASSIVE COOLING OF DETECTORS AND STRUCTURE

Silicon CCDs sensitive to ultraviolet and optical radiation, that is, wavelengths shortward of one micron, must be cooled to about 170 K to decrease the number of electrons thermally generated in the device to acceptable limits. Operation of CCDs in the near infrared requires both colder detectors and, as the thermal infrared is approached, a cold, minimally emissive telescope. Simply put, there is a real benefit for LTT to passively operate as cold as possible during the entire lunar day/night cycle. The specific goal is to continue observations, including those in the near infrared to 3.7 microns, during the entire lunar day.

The specific requirements for passive cooling are: 1) For operation of the silicon CCDs, the LTT detector package must be kept at 170 K or colder. 2) For operation of K-band HgCdTe CCDs with a 2.5 micron cutoff, the detector package must be kept at 65 K or colder. The thermal emissivity at 2.5 microns should be minimized. 3) For operation of HgCdTe detectors at wavelengths longward of 2.5 microns, the detector package and telescope must be kept correspondingly colder to obviate the effects of an exponentially rising instrumental background. Again the emissivity must be minimized.

Thermal shielding of the LTT <u>for continuous operation</u> requires that the LTT be at a lunar latitude (north or south) greater than about 6 degrees so that the sun and earth can never pass directly over the telescope. Cooling for continuous operation becomes progressively easier as the latitude increases.

The earth as well as the sun is a possible source of overwhelming light and heat. If the LTT is on or near the equator, it is possible that it will have to be shut down and shielded to protect it during sun transits. A more favorable location might be at an intermediate latitude of 20-30 degrees. At still higher latitudes shielding becomes very much easier.

THE LTT FOCAL PLANE ARRAY

The focal plane of LTT is the heart of the instrument. Its design, implementing a mosaic of frame transfer CCDs, allows the telescope to operate as a superb survey instrument. The size of the focal plane arrays, the drivers leading to specific designs, and alternatives are design trade-offs. 1) The focal plane should subtend at least 2^{o} in the north-south direction to provide the largest reasonable survey area. 2) The pixel resolution should be 0.1 arcsec or better. 3) The wavelength range covered should extend from 0.1 to at least 2 microns. 4) There should be at least one bandpass per octave of electromagnetic spectrum.

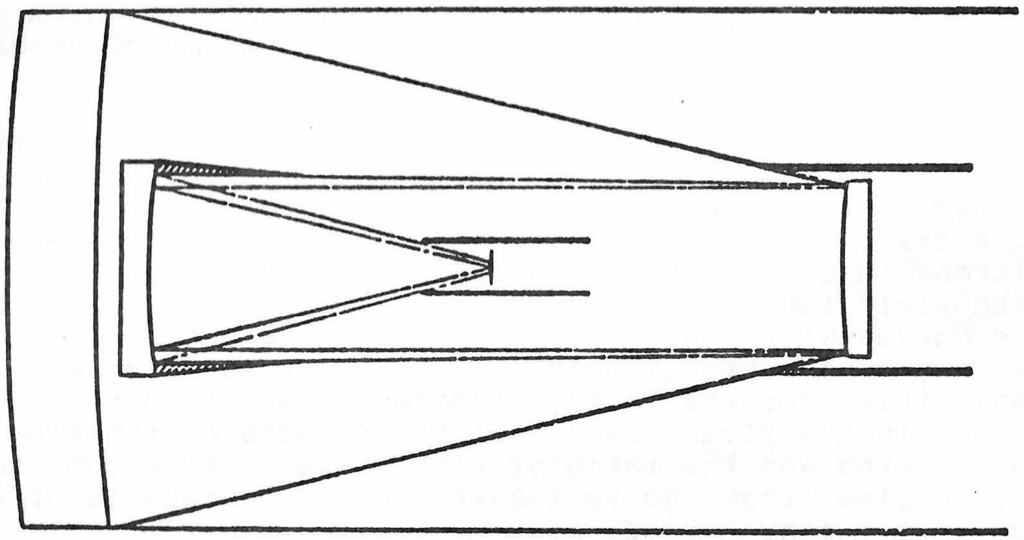
Assume a square focal plane array 2^{o} on a side with 0.1 arcsec pixels, the focal plane will contain 72,000 x 72,000 = 5.184×10^{9} pixels. The final focal plane arrangement including fill-factor, and other considerations such as the volume of "engineering" pixels (underscan, overscan, etc.) will modify this number somewhat, but for "strawman" investigation this value is utilized. The lunar rotation causes an apparent motion of the sky of 0.549 arcsec/s giving a pixel read-out rate of 5.49 s^{-1} for 0.1 arcsec pixels. If there are 72,000 pixels in the north-south dimension and five bandpasses (one per octave), the total data rate from this focal plane is about 2×10^{6} pixels/s or, if two bytes per pixel is assumed then 4 Mbytes/s.

THE PAUL-BAKER SYSTEM

The current CCD transit instrument, CTI, on Kitt peak is a Paul-Baker system. The principle of the Paul-Baker system is easily described and will be readily recognized. The primary is a parabaloid, the converging beam from which is intercepted by a slightly aspheric secondary. For pedagogical purposes, consider the secondary initially to be a parabaloid placed confocal with the primary. The input to the primary is, of course a parallel bundle of rays. The beam returned by the secondary towards the primary in this case is again an aberration-free parallel bundle of rays. This bundle is intercepted by a spherical tertiary with radius of curvature the same as that of the secondary and with secondary and tertiary vertices separated by the common radius of curvature. The tertiary forms a focal plane midway between the secondary and tertiary, but the images have spherical aberration. If the required corrections for spherical aberration are added to the secondary surface, the net result is to return the figure of the secondary more nearly to a sphere with a small aspheric correction in the form of a "turned down edge." The optical system can be thought of as a classical Schmidt telescope of small aperture fed by a much larger parabaloidal primary.

The positive aspects of a Paul-Baker optical system include the fact that it is all-reflective, that it easily provides a wide field of view from a fast optical system, and that the images are exquisite - nearly diffraction limited in the optical. Negative aspects are that there are three reflections (which is probably not a problem when compared to HST current instruments, or other ground-based telescopes, for that matter), it can be hard to baffle, the focal plane is curved (but this can be accommodated by mosaicking) and the central obscuration in some configurations can be high (eg. 21% for CTI) relative to classical Cassegrain systems.

THE PAUL-BAKER SYSTEM



The optical path diagram and baffling for the 1^{o} field of view Paul-Baker system currently in use in the CCD/Transit Instrument (CTI). This system can be thought of as a Schmidt telescope fed by a larger parabaloidal primary.

A WIDE FIELD OPTICAL SYSTEM FOR LTT

The first subsystem of LTT to consider is the optics. The primary requirements for this subsystem of the telescope are: 1) all reflective optics to allow simultaneous observation over the widest possible range of the electromagnetic spectrum; 2) a wide field of view to accommodate the survey nature of this project; 3) a fast optical system to minimize the physical size and mass of the support structure; 4) a system which provides an appropriate match of the focal plane to the sky in terms of resolution, sensitivity and dynamic range.

To meet these criteria, a "strawman" LTT with a Paul-Baker system (Paul 1935, Baker 1969) was initially proposed. This system includes a parabaloidal primary mirror with a two mirror corrector to provide a wide field of view. A f/2.2 Paul-Baker system with a 1° field of view encloses a spot of 15 microns in size. These spots are produced at the best focus averaged over the field for wavelengths between 0.35 and 0.85 microns. The focal surface is curved with radius 3,991 mm. For the CCD transit instrument, CTI, a ground-based telescope with 1.5 arcsec pixels, this curvature is negligible over the field subtended by a single CCD (15 mm diagonal) mosaicked in the focal plane. If the effects of atmospheric seeing can be eliminated and the match of pixel size to telescope resolution accomplished, it is clear that the Paul-Baker design is capable of producing exquisite images from a fast optical system.

Another three mirror optical concept, designed by D. Korsch and described in the next section, was introduced early during the LTT study and is now the baseline optical configuration. This configuration is optimized for maximum field angle, minimal field curvature and distortion and a focal ratio which matches current technology detector design. Greater structural simplicity and compactness is achieved with this configuration because of the focal plane location behind the secondary mirror.

11/26/91

A WIDE FIELD OPTICAL SYSTEM FOR LTT

Detectors:

- Si CCDs (Ultraviolet and Optical)
- HgCdTe with CCD Readout (Infrared)
- 0.1 Arcsecond (4 um size) Pixels
- Diffraction Limited Beyond 2 Microns
- CCDs are Mosaicked in the Focal Plane

Field-of-View:

- All Reflective Optics, Three-Mirror System
- 2 Degree (Wide, ie. N-S) Field of View
- 575 - 720 Sq Degrees Surveyed Each Lunar Day (Typical)
- Precession of Lunar Orbit Widens Observed Strip to 5°
- More Than 75% of Strip Observed Each Lunation

Bandpasses:

5 Bandpasses (R = 0.1 -0.25), One Bandpass Per Octave

Estimated Limiting Magnitudes (S/N = 10, single 360 s integration):

V = 27	Estimates scaled from HST WF/PC and
H = 25	NICMOS calculated performance and
K = 24.5	CTI measured performance. Assumed quantum efficiency of 60%.

2.2 OPTICAL ARCHITECTURE

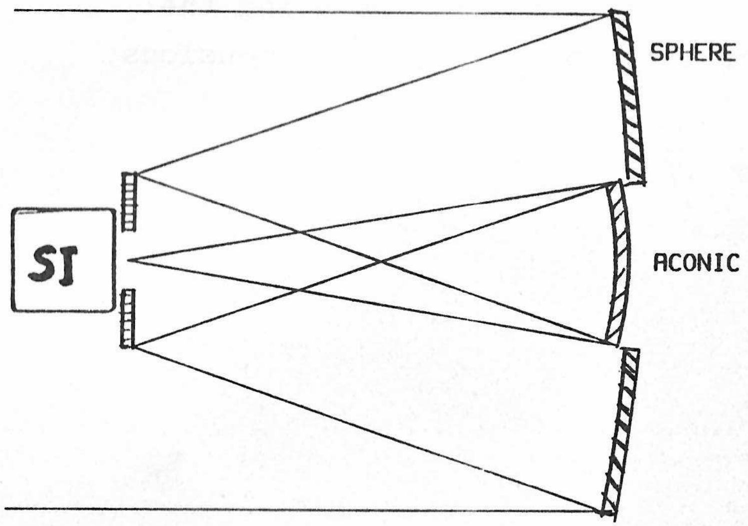
THREE AND FOUR MIRROR TELESCOPES

The optical design of the Lunar Transit Telescope is primarily driven by the requirement for a wide field of view. Two mirror telescopes of the Cassegrain as well as the Gregorian configuration can only be corrected for spherical aberration and coma. The remaining aberrations, specifically astigmatism and field curvature, limit the high resolution field to no more than a few arc minutes. Furthermore, since the use of any refractive corrector which might solve the field of view problem would result in major reduction in throughput and a severe limitation of the spectral range, the only available option is a multi-mirror system. Satisfactory performance can be obtained with both three and four mirror designs.

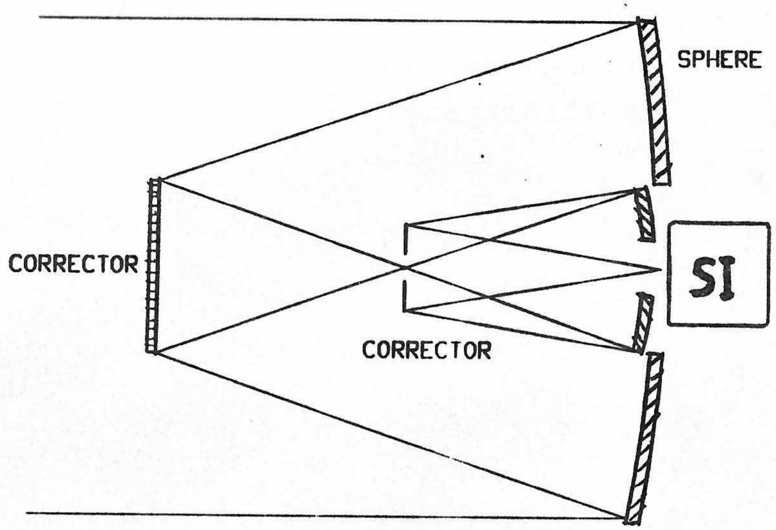
There are two important differences between three and four mirror systems. The first, and most obvious, is the number of mirrors, and therefore also the number of reflections. The second is the location of the focal plane. While in most practical systems the focal plane of a four-mirror telescope is located behind the primary, the focal plane of a three-mirror telescope is located behind the secondary. This may have some advantages for a lunar transit telescope because the focal plane instrument is exposed to the environment. Thermal energy from the electronics can be dissipated easier than in the case where the detectors would be shielded from seeing deep space by the primary mirror. Also, it may be advantageous to avoid secondary galactic cosmic radiation from striking massive structures near the detectors.

7/24/91:BGD

THREE AND FOUR MIRROR TELESCOPE OPTIONS



THREE MIRROR



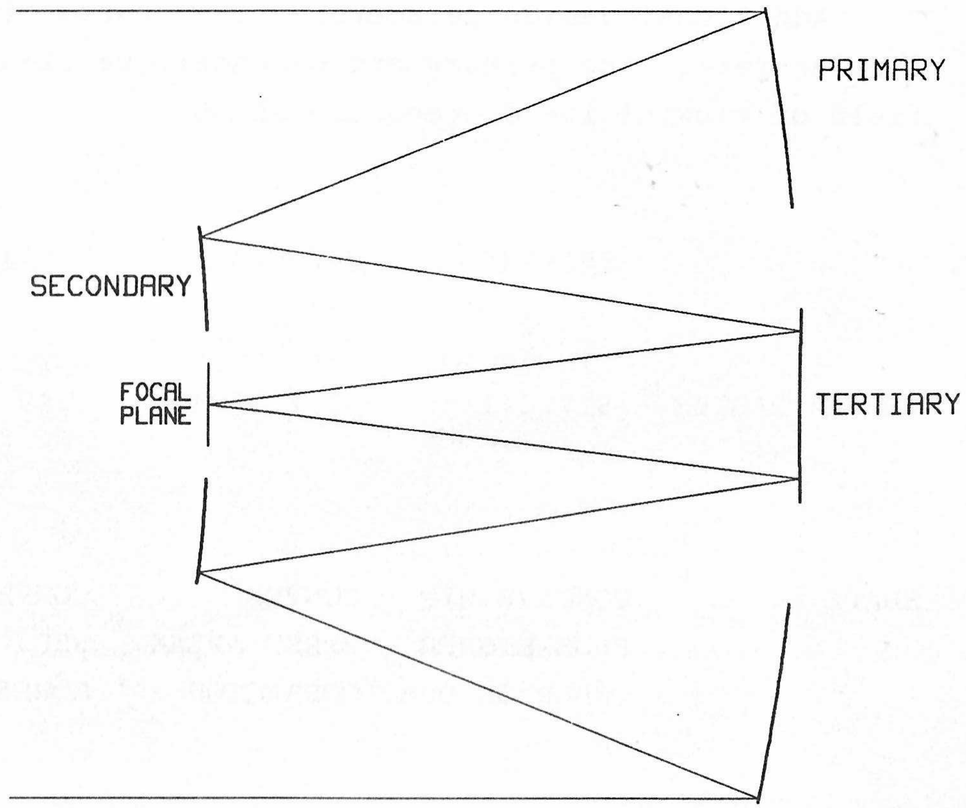
FOUR MIRROR

THREE MIRROR LTT

Because of the higher throughput and the overall greater simplicity, the three-mirror configuration was selected as candidate concept for the LTT. The location of the focal plane in the front of the telescope has the added advantage of facilitating the required cooling for the detector arrays. The basic configuration of the three-mirror LTT, including the arrangement of the primary, secondary, tertiary and the system dimensions, is illustrated.

SYSTEM DIMENSIONS:

PRIMARY O. DIAMETER	:	200cm
PRIMARY I. DIAMETER	:	100cm
SECONDARY DIAMETER	:	90.5cm
SECONDARY HOLE	:	38cm
TERTIARY DIAMETER	:	49cm
MIRROR SEPARATION	:	150cm
BACK FOCAL DISTANCE	:	150cm
SYSTEM FOCAL LENGTH	:	800cm
FOCAL PLANE DIAMETER	:	21cm

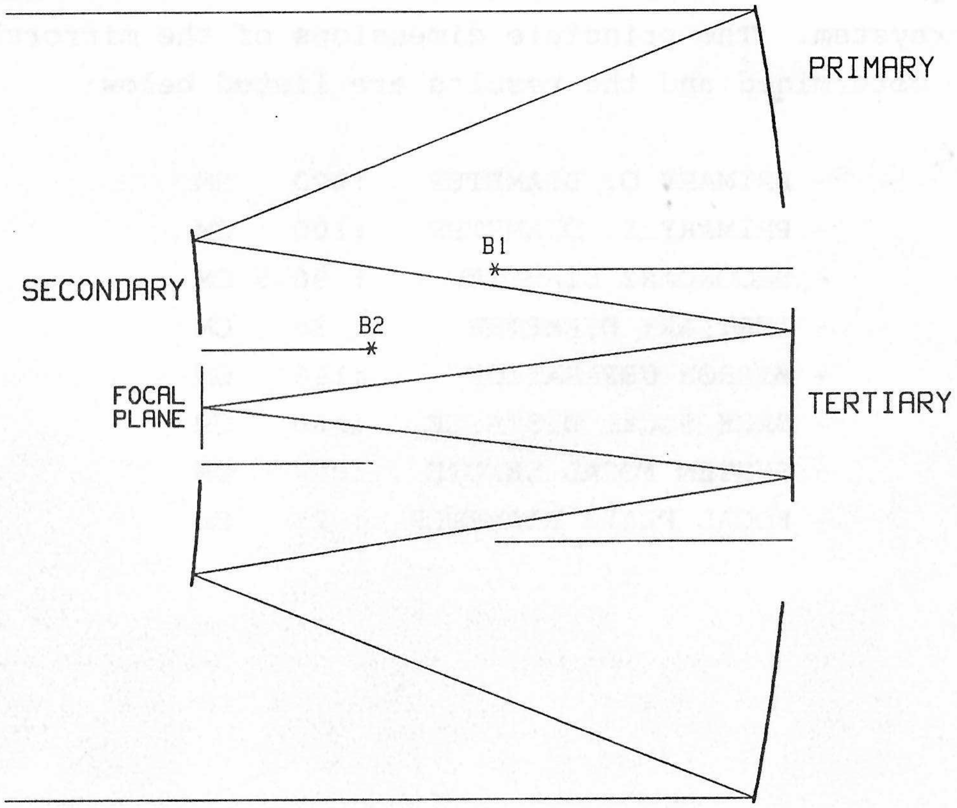


LTT OPTICAL DESIGN CONCEPT

LTT OPTICAL DESIGN PARAMETERS

Additional design parameters for a three mirror telescope are given in tabular form. The primary and secondary baffles for an unvignetted circular field of view of 1.5 degrees are shown.

	PRIMARY	SECONDARY	TERTIARY	FOCAL SURFACE
VERTEX RADIUS (CM)	-517.24146	-362.58947	+1249.97344	+333
SHAPE	CONCAVE HYP PLUS HIGHER ORDER DEFORM.	CONVEX FIRST APLAN. CORRECTOR	CONVEX SEC. APLAN. CORRECTOR	PARABOLIC



PRIMARY & SECONDARY BAFFLE POINTS:
B1: X=34.73cm Z= -75.79cm
B2: X=14.57cm Z=-105.64cm

(origin coincides with the vertices of primary and tertiary)

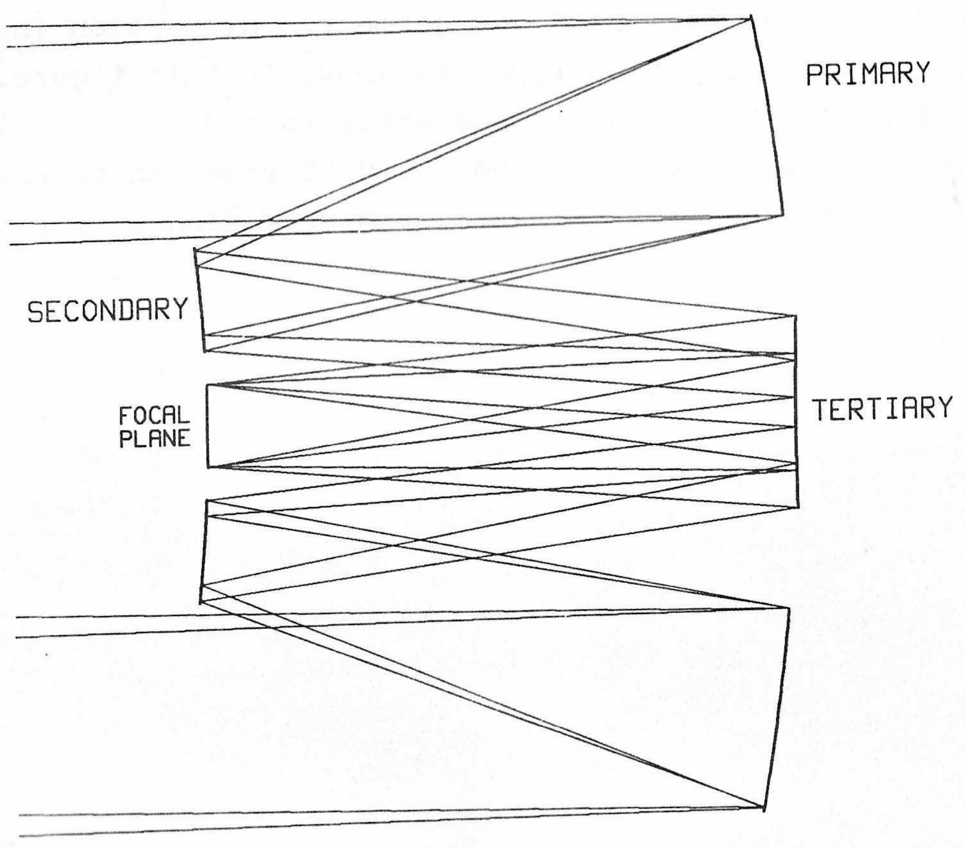
THREE-MIRROR LTT

(UNVIGNETTED AND FULLY BAFFLED FOR 1.5 DEGREES)

FULL-FIELD RAY TRACE THREE MIRROR LTT

A full-field ray trace was made for the 1.5 degree field of view system. The principle dimensions of the mirrors, spacing and lengths were determined and the results are listed below:

- PRIMARY O. DIAMETER :200 CM
- PRIMARY I. DIAMETER :100 CM
- SECONDARY DIAMETER : 90.5 CM
- TERTIARY DIAMETER : 38 CM
- MIRROR SEPARATION :150 CM
- BACK FOCAL DISTANCE :150 CM
- SYSTEM FOCAL LENGTH :800 CM
- FOCAL PLANE DIAMETER : 21 CM



LTT FULL FIELD RAY TRACE

(UNVIGNETTED 1.5 DEGREES)

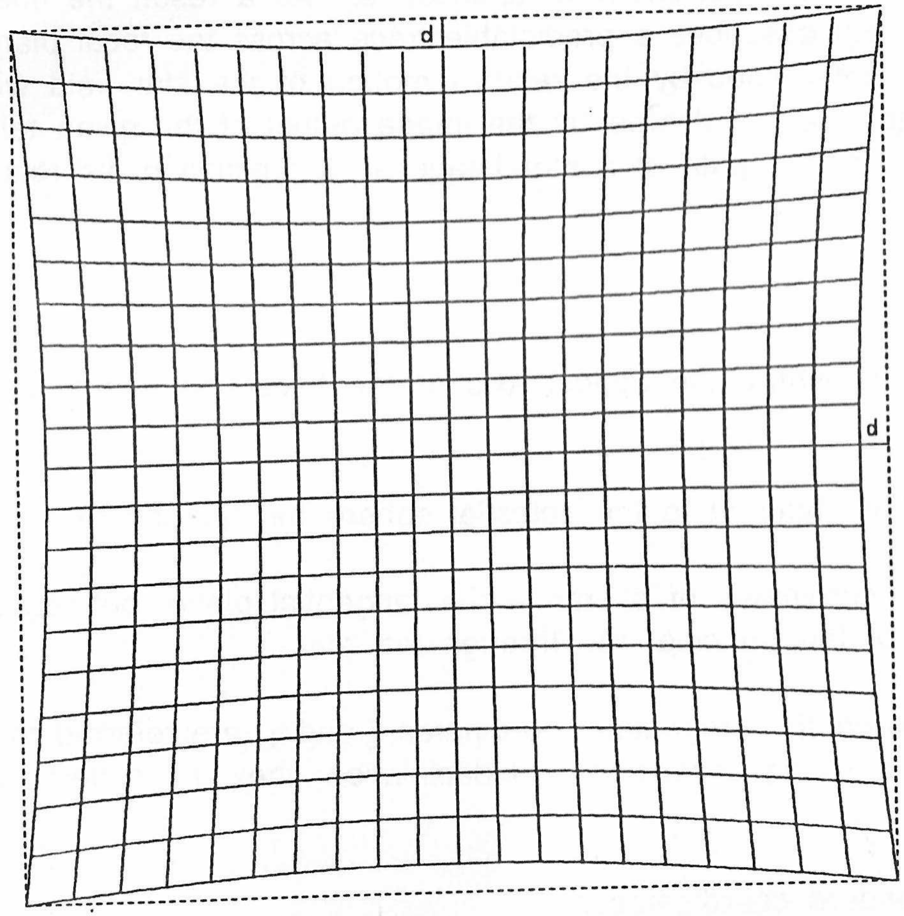
FIELD DISTORTION

An exaggerated illustration of the geometric distortion in the image plane, which is of the pincushion type, is shown in this figure. Preliminary performance optimization and analysis indicates that the residual aberrations at the edge of the field of view can be reduced from 14 to less than 8 micrometers, approximately 1 microradian at a focal length of 8 meters.

DIAGONAL FOV:

1.5deg (21cm)

d= 14 micrm



3MLTT - FIELD DISTORTION

D.KORSCH
10/15/91

THE PATH OF A STAR IMAGE IN THE FOCAL PLANE OF A TRANSIT TELESCOPE

The absence of a tracking mechanism is the one characteristic that distinguishes a transit telescope from most conventional astronomical telescopes. As a result the image of a star is not fixed at one location, but describes a predictable trace across the focal plane. The path of the star image is mainly determined by the relative motion of the star field with respect to the telescope axis, but also to a lesser degree by the image defect of the given telescope, called distortion. The equation for the path of a star image as a function of its relative motion is derived first.

Definitions

Tangential Point: the point where the optical axis of the telescope intersects the celestial sphere.

Tangential Plane: the plane tangent to the celestial sphere through the tangential point.

Tangential Coordinates: coordinates of a star in the tangential plane, defined by the intersection point of the principal ray through the star.

Standard Coordinates: when the tangential coordinates, ξ and η, are referred to axes in direction of increasing ascension and declination, they are called standard coordinates.

The equations for the standard coordinates are*

$$\xi = \frac{\cos\delta\sin(\alpha-A)}{\sin D\sin\delta + \cos D\cos\delta\cos(\alpha-A)} \qquad (1)$$

$$\eta = \frac{\cos D\sin\delta - \sin D\cos\delta\cos(\alpha-A)}{\sin D\sin\delta + \cos D\cos\delta\cos(\alpha-A)} \qquad (2)$$

* Robin M. Green, "Spherical astronomy", Cambridge University Press 1985

whereby A, D are right ascension and declination of the tangential point (optical axis), and α, δ are right ascension and declination of the star.

The image of a star pattern in the focal plane of a distortion-free, flat-field telescope is geometrically similar to the star pattern projected onto the tangential plane. The scale factor is equal to the telescope focal length.

The star traces in the focal plane of a transit telescope are the result of a rotation of the tangential point (optical axis) about the celestial pole. In eqs. (1) and (2) this rotation is represented by the angle A. By eliminating $(\alpha - A)$ from both equations a closed-form relation for the trace of a star in the tangential plane is obtained, namely,

$$\xi^2 = \frac{a}{cd-be}^2 (c^2 - b^2)\eta^2 + 2(bd-ce)\eta + (e^2 - d^2) \tag{3}$$

whereby

$$\begin{aligned} a &= +\cos\delta, \\ b &= +\sin D \sin\delta, \\ c &= +\cos D \cos\delta, \\ d &= +\cos D \sin\delta, \\ e &= -\sin D \cos\delta. \end{aligned}$$

Equation (3) is the equation of a conic section. The vertex equation of a general conic section is of the form

$$\xi^2 = 2\rho(\eta - \eta_o) \quad - \quad (1-\varepsilon^2)(\eta - \eta_o) \tag{4}$$

with ρ being the vertex radius, e being the numerical eccentricity, and η_o being the vertex location on the η-axis.

After expanding eq. (4), and upon comparing coefficients with eq. (3), and setting $\delta = D + \phi_{yo}$, yields;

$$\rho = +\cot(D+\phi_{yo}) \tag{5}$$

$$\eta_o = +\tan\phi_{yo} \tag{6}$$

$$1 - \varepsilon^2 = \cos\phi_y \cos(2D+\phi_{yo})/\sin^2(D+\phi_{yo}) \tag{7}$$

The focal plane coordinates are obtained by multiplying all linear dimension in eq. (4) with the telescope focal length, f. obtaining

$$x^2 = 2r(y-y_o) - (1-\varepsilon^2)(y-y_o)^2 \tag{8}$$

with $x = f\varepsilon$, $y = f\eta$, $y_o = f\eta_o = f\tan\phi_{yo}$ and $r = f\rho = f\cot(D+\phi_{yo})$.

The telescope field angles in directions of declination and right ascension are

$$\phi_y = \arctan(y/f),$$

in particular $\phi_{yo} = \arctan(y_o/f)$ for $x=0$, i.e., when the star crosses the y- (or η-) axis,

and

$$\phi_x = \arctan(x/f),$$

respectively.

Since ϕ_x represents only a very small portion of the conic section in the vicinity of the vertex, the quadratic term in y of eq. (8) may be omitted without loosing noticeable accuracy. That is, instead of eq. (8), the following may be used

$$y = x^2/2r + y_o \tag{9}$$

The first graph shows a series of star traces in the focal plane of a telescope that is free of distortion (Delta in the equation is equal to $-\varepsilon^2$).

If the telescope is affected by geometric distortion, as is almost always the case, this effect must be added to eqs. (8) or (9). Third-order distortion in parametric representation is given by

$$x_d = Ef({\phi^2}_x + {\phi^2}_y)\phi_x \tag{10}$$

$$y_d = EF({\phi^2}_x + {\phi^2}_y)\phi_y \tag{11}$$

with E being the coefficient of distortion, and f being the system focal length. An example of star traces in the presence of distortion is shown in the second graph.

Decl of opt axis : +40.00 deg

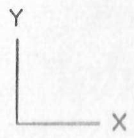
Length of Trace : 20.00 cm

Coeff of Distortion: +0.00

Tel Focal Length : +800 cm

Sag [microm]	Yo [cm]	R [m]	Delta
+538.5	+10.47257	+9.28446	-1.37721
+535.7	+8.37789	+9.33376	-1.38563
+532.9	+6.28331	+9.38337	-1.39414
+530.0	+4.18883	+9.43328	-1.40275
+527.2	+2.09440	+9.48350	-1.41147
+524.4	+0.00000	+9.53403	-1.42028
+521.6	-2.09440	+9.58488	-1.42919
+518.9	-4.18883	+9.63605	-1.43821
+516.1	-6.28331	+9.68754	-1.44733
+513.4	-8.37789	+9.73936	-1.45656
+510.6	-10.47257	+9.79151	-1.46590

$$X^2 = 2R(Y-Yo)-(1+Delta)(Y-Yo)^2$$



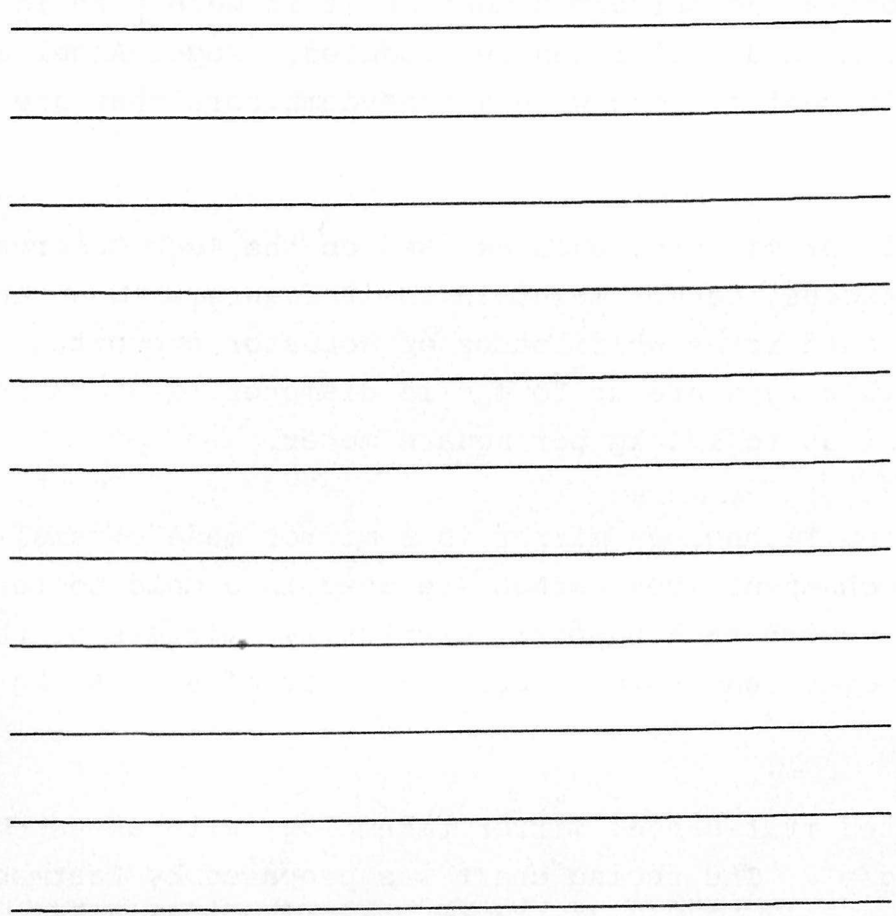
D.KORSCH
11/11/91

Decl of opt axis : +40.00 deg

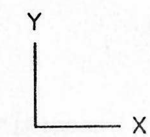
Length of Trace : 21.00 cm

Coeff of Distortion: -0.91

Tel Focal Length : +800 cm



SAG [μm]	Yo [cm]
+610.2	+10.47257
+603.8	+8.37789
+597.4	+6.28331
+591.0	+4.18883
+584.6	+2.09440
+578.2	+0.00000
+571.9	-2.09440
+565.5	-4.18883
+559.2	-6.28331
+552.9	-8.37789
+546.6	-10.47257



D.KORSCH
11/14/91

PREDICTED MIRROR WEIGHT/AREA

The HST is a current example of designs using lightweight mirrors. A honeycomb core is frit or fusion bonded between two thin layers of glass. The HST mirror has an areal density of about 166 kg/m^2. Typically, mirrors of this type are rim supported and perform almost as if it were a solid piece of glass. Mirrors up to 8 m in diameter can be produced. Roger Angel at the U. of Az. has produced spin cast mirrors with a honeycomb core that are in the HST weight range.

Thin glass segments or mirrors, such as used on the Keck Observatory and ESO New Technology Telescope, cannot maintain their figure without active support. They are designed to be whiffletree or actuator supported. Typically, mirrors of this type are up to 4 m in diameter, 5-10 cm thick and have an areal density of 30 to 100 kg per square meter.

A promising emerging technology mirror is a mirror made of small molded segments. Materials such as silicon-carbon are used in a mold to form the mirror surface with honeycomb back support. Typically, mirrors of this type are 5 cm to 1 m in diameter, and have an areal density of 5 to 50 kg per square meter.

The LTT has selected frit-bonded mirror technology with an assumed areal density of 25 to 75 kg/m^2. The facing chart was prepared by Eastman Kodak several years ago. The curve on "predicted IR frit-bonded mirror" may be representative of todays best weights obtainable.

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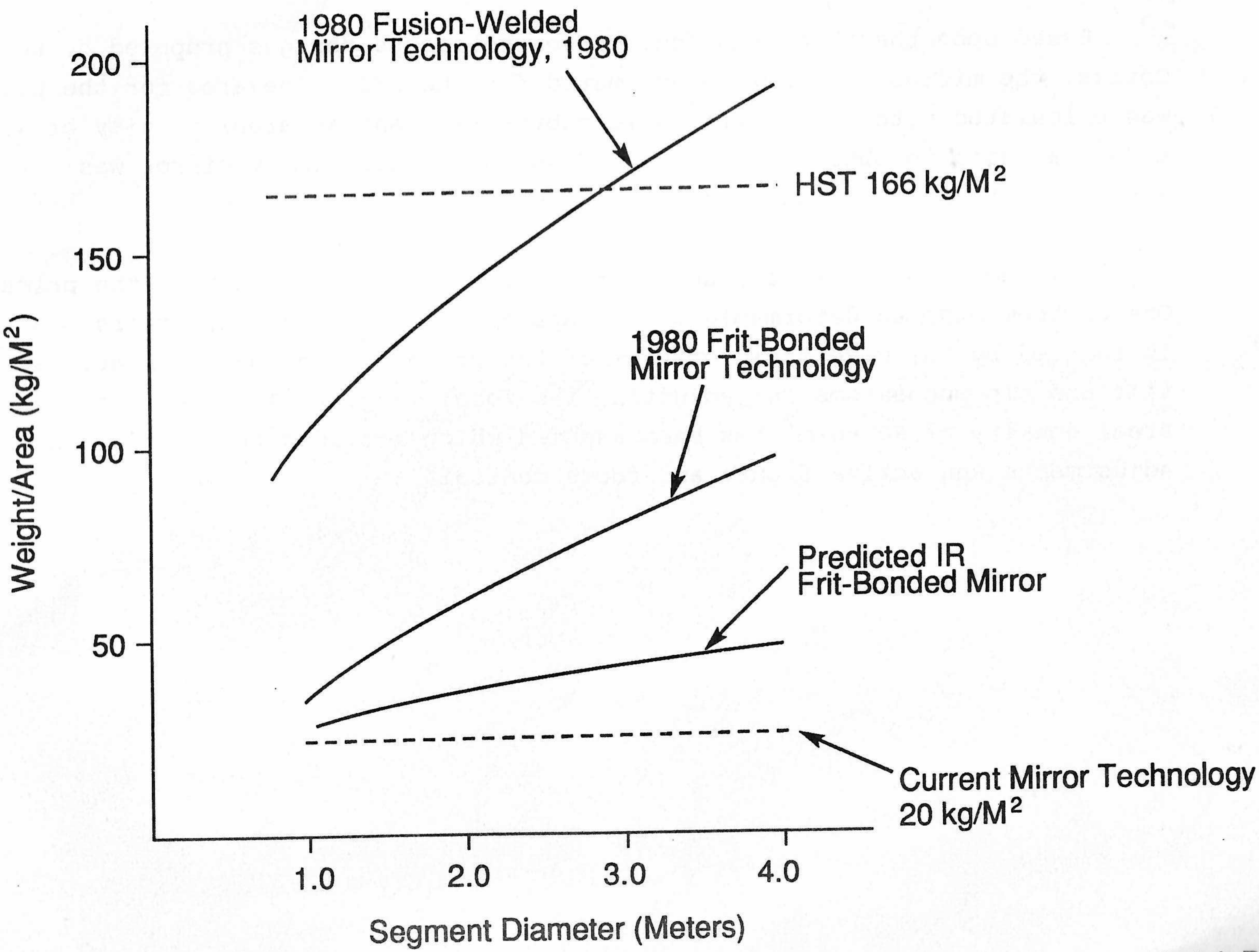
Predicted Mirror Weight/Area



Ref. Eastman Kodak Data

LTT MIRROR MASS ESTIMATES

Based upon the three and four mirror telescope designs proposed by Korsch Optics, the mirror masses were estimated for the LTT. The area for the primary was calculated with the central hole subtracted, and an areal density of 45 kg/m^2 was used to obtain the estimated weight. The primary mirror was assumed to be strong enough to be rim supported.

The secondary, tertiary and quaternary are more complex than the primary. One of them must be deformable or be capable of correcting for errors introduced by the wide field-of-view of the primary, and the secondary may have tilt and tip mechanisms for adjusting the focal length. In either case, an areal density of 60 kg/m^2 has been assumed which includes the mechanisms for adjustments and active figure and focus control.

9/18/91:BGD

LTT MIRROR MASS ESTIMATES

* ASSUMPTIONS

1. SIZES SCALED FROM KORSCH DESIGN SKETCHES
2. MASS CALCULATED WITH HOLE IN PRIMARY; NO HOLE IN OTHER MIRRORS
3. PRIMARY AREAL DENSITY OF 45 KG/M^2
4. SECONDARY, ETC. AREAL DENSITY OF 60 KG/M^2

* **PARAMETERS**

	3- MIRROR			4- MIRROR			
	P	S	T	P	S	T	Q
DIAM, M	1	.35	.325	1	.35	.325	.175
AREA, M^2	.70	.096	.083	.70	.096	.083	.024
MASS, KG	32	6	5	32	6	5	1.44
DIAM, M	2	0.7	0.65	2	0.7	0.65	0.35
AREA, M^2	2.8	0.38	0.33	2.8	0.38	0.33	0.10
MASS, KG	126	23	20	126	23	20	5.8
DIAM, M	3.4	1.2	1.1	3.4	1.2	1.1	0.6
AREA, M^2	8.1	1.1	0.95	8.1	1.1	0.95	0.28
MASS, KG	364	67	57	364	67	57	17

SUN SHADE OPTIONS

Direct sunlight must not enter the telescope aperture. As with most telescopes the LTT cannot take data within about 45 degrees of the Earth's limb or the sun. If the LTT were in the center of the moon near its equator on the Earth side then it would always be pointing to the Earth and no data could be taken. Also the sun would be directly overhead at mid-day and an aperture door must be closed for thermal protection. To preclude Earth shine entering the aperture the LTT should be located on the East or West limb, but the sun would still pass directly overhead. If the LTT were placed at a higher latitude then it would be possible to shield from both the Earth and sun without having an aperture door that closes when the sun comes up.

The left illustration is a fixed truncated light shade with a roll up aperture closure. The truncation angle is a function of the latitude and becomes shorter as the latitude is increased. The aperture closure is for protection during launch to landing, and after landing on the moon is blown off or rolled up and never used again. The center illustration represents a flat, box like closure that rotates 90 degrees to open the aperture and serve as a sun shade. The right illustration represents a circular section that enclosed the telescope at launch and extends upward to serve as a sun shade after landing on the moon. Rails or some type deployment mechanisms are required. If it can be packaged into the payload launch shroud then the fixed truncated sun shade with a roll up closure would be best for the LTT.

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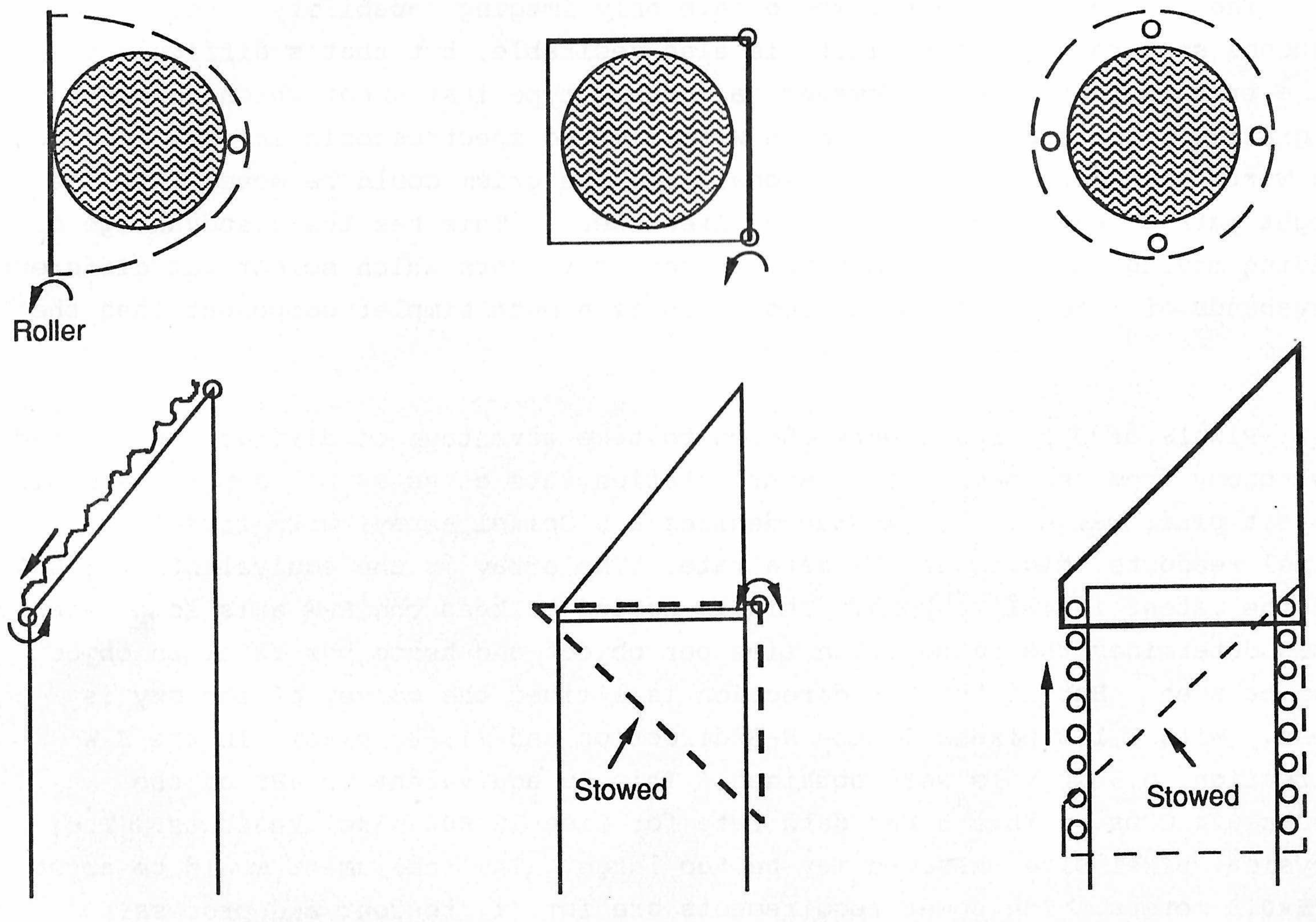
Sun Shade Options



Fixed Truncated w/Roll-up Closure | Rigid Rotatable Flat Closure | Tube Extension Blow-off Closure

2.3 STRAWMAN SCIENCE INSTRUMENTS

TRANSIT TELESCOPE STRAWMAN INSTRUMENT

The simplest instrument would have only imaging capability. For science reasons spectroscopic capability is also desirable, but that's difficult to do in a transit instrument. However, a 'grism' type instrument which consists of a grating prism providing an image with limited spectroscopic information might be worth considering. As an optional mode the grism could be moved into the light path rather than permanently fixed there. This has the disadvantage of adding moving parts. At a minimum, a set of filters which select out different wavebands of light would be needed. This is a much simpler component than the grism.

Pixels of 0.1 arcsec were chosen to take advantage of diffraction limited astronomy from the moon. The lunar rotation rate gives 88 bit/s/pixel assuming 16-bit pixel values. Dr. McGraw desires a 5 Gpixel array, with five 72,000 pixel readouts, giving 31 MHz data rate. The array is the equivalent of 5,000 of the latest 1024x1024 CCDs. This is needed to keep the E-W axis long because this determines the integration time per object and hence how faint an object can be seen. But if the N-S direction is limited the survey of the sky is less. With 7,168 pixels in the N-S direction and 71,680 pixels in the E-W direction, 0.5 Gpixels were obtained. This is equivalent to 490 of the 1024x1024 CCDs, with a 3 MHz data rate for five 72,000 pixel readouts. The physical pixel sizes assumed may be too large. The instrument would be about 0.5x0.5 meters. The power requirements are for (1) readout and processing electronics which require a small amount of power and (2) cooling. Using standard active cooling this focal plane would need about 100 W, but can be averted if passive cooling can be used.

TRANSIT TELESCOPE STRAWMAN INSTRUMENT

Detector:	Focal plane mosaic optical/ultraviolet CCD array 7,168 x 71,680 pixels Broad Band Photometric Filters from 0.1 to 1 microns Wide field grating prism (?)
Size:	Roughly 0.5 to 1.0 metre across (will investigate further)
Power:	Dominated by cooling requirements
Data Rate:	3 Mbit/s
Cooling:	To 170 K
Tradeoffs:	Abandoned near infrared for simplicity (different detectors required) Reduced size of surveyed area

LTT STRAWMAN INSTRUMENT (SAO INPUTS)

Based on conversation with Dr. John Geary, SAO (8/7/91).

Big chips have larger dead space at the edges than smaller ones. Moreover big chips cannot be made with the 'stepper projectors' used for small chips, the design must use the 'whole wafer photolithography' technique instead. This technique can only do 10 micron pixels instead of a couple micron pixels for the projectors. Possibly 8 microns can be made. A 47 mm active area has been demonstrated which corresponds to 5,800 8-micron pixels. It's probably possible to make a strip with no dead area, by mosaicking a number of edge-abuttable ones. At 10 micron pixels a 0.5m x 0.5m focal plane is needed. Only about half of the focal plane can be filled with active area. Assuming a build of 50 CCD's gives a total of 1.6 Gpixel and a true area of about 0.4 x 0.4m.

Temperature: repeated cycling from -70 C to 0 C, for example, would harm the chips. But they can be launched warm and cooled down after landing on the moon, if that's helpful. Cooling is still the big unknown. How is the heat dissipated that is produced by the coolers? Can it be cooled enough to use IR CCDs? Thermoelectric coolers may be the way to go but are inefficient. These are questions requiring examination in further studies.

10/23/91:JM

LTT STRAWMAN INSTRUMENT (SAO INPUTS)

Mass budget:	CCDs and packaging	5 kg
	Electronics	20 kg
	Cooling	50 kg (guess)
	Total	75 kg
Power budget:	CCDs	Negligible
	Electronics	20W
	Coolers	100W
	Total	120W
Temperature:	UV CCDs only	170K
	IR CCDs	65K
	Variation in operational use	<10K (preferred)
	Launch and coast phases	Can be warm; avoid large variations
Data rates:	Require data compression system and high bandwidth, ~50 Mbit	

2 M LTT STRAWMAN SCIENCE INSTRUMENT

Based upon inputs from Dr's John McGraw and Jonathan McDowell, the physical characteristics of a strawman science instrument for the LTT was estimated. The LTT detector is an array of charged couple devices (CCDs). Each CCD is assumed to have 2,048 x 2,048 pixels, and each pixel is expected to be from 5 to 10 micrometers square. A CCD is normally customized for a specific wave length. To span the spectrum from infrared, visible to ultraviolet, the array may be composed of a mix of different CCDs. The North-South dimension of the array determines the percent of the celestial sphere covered and the east-west dimension determines the integration time (time a star image stays on the array).

The physical characteristics of the 2 m LTT science instrument are listed, and are used to drive the system and subsystem support requirements. The physical size is estimated at 0.5 N-S x 0.8 E-W x 0.5 meters with a mass of 75 kg which does not include mechanical coolers or radiators if needed for thermal control. The average support power is 20 W for electronics and 100 W for cooling. Assuming 5 N-S readouts of 88 bits/s/pixel gives a data range of about 31 Mbps. If a 10 to 1 compression ratio can be obtained then the average data rate to be transmitted back to Earth is about 3 Mbps.

9/24/91:JM

2 M LTT
STRAWMAN SCIENCE INSTRUMENT

ASSUMPTIONS:

1. MOSAIC ARRAY OF OPTICAL/ULTRAVIOLET CCD'S

2. 72,000 N-S X 72,000 E-W PIXELS (MAY BE SMALLER IN N-S)

3. 1,200 CCD'S, EACH WITH 2,048 X 2,048 PIXELS (5.2 BILLION PIXELS)

4. 5 N-S READOUTS OF 88 BIT/S/PIXEL GIVING 31 MBPS

CHARACTERISTICS:

1. SIZE, 0.5 S-N X 0.8 E-W X 0.5 METERS

2. MASS ESTIMATE 75 KG
   - CCD'S AND PACKAGING 5 KG
   - ELECTRONICS 20 KG
   - COOLING EST. 50 KG

3. PASSIVE COOLING OF DETECTOR IF POSSIBLE
   - 170 K FOR UV CCD'S
   - 65 K FOR IR CCD'S
   - LESS THAN 10 K VARIATION DURING OPERATION

SUPPORT REQUIREMENTS:

1. AVERAGE DATA RATE OF 31 MBPS BEFORE COMPRESSION

2. AVERAGE POWER: 120 W: 20 W ELECTRONICS + 100 W COOLING

3. STABILITY, TBD. FOV SWATH SHOULD BE PERPENDICULAR TO THE GALACTIC PLANE

4. CAN BE WARM DURING LAUNCH AND COAST PHASES BUT AVOID LARGE VARIATIONS

5. DETECTOR ARRAY MUST BE ALIGNED WITH E-W LUNAR ROTATION

1 M LTT STRAWMAN SCIENCE INSTRUMENT

The instrument for the 1 m LTT is about 1/4 the number of CCD contained in the 2 m LTT. The 2 m case had a 35 x 35 array of CCDs which is too large for the 1 m case, the array size blocks some of the 1 m telescope's aperture. An 18 x 18 CCD array has been assumed for the 1 m LTT and the physical characteristics scaled downward from the 2 m instrument.

The physical characteristics for a CCD array sized for the 1 m LTT is 0.3 x 0.3 x 0.3 m with a 40 kg mass. The readout rate is 16 Mbps, and assuming a 10 to 1 data compression ratio the average data rate returned to Earth is 1.6 Mbps. The average power needed during operations is 65 W of which 50 W is for cooling. The estimated weight does not contain a mechanical cooler or radiators.

9/24/91:BGD

1 M LTT
STRAWMAN SCIENCE INSTRUMENT

ASSUMPTIONS:

1. MOSAIC ARRAY OF OPTICAL/ULTRAVIOLET CCD'S
2. 36,864 N-S X 36,864 E-W PIXELS (MAY BE SMALLER IN N-S)
3. 324 CCD'S, EACH WITH 2,048 X 2,048 PIXELS (1.4 BILLION PIXELS)
4. 5 N-S READOUTS OF 88 BIT/S/PIXEL GIVING 16 MBPS

CHARACTERISTICS:

1. SIZE, 0.3 S-N X 0.3 E-W X 0.3 METERS
2. MASS ESTIMATE 40 KG
   - CCD'S AND PACKAGING 5 KG
   - ELECTRONICS 10 KG
   - COOLING EST. 25 KG
3. PASSIVE COOLING OF DETECTOR IF POSSIBLE
   - 170 K FOR UV CCD'S
   - 65 K FOR IR CCD'S
   - LESS THAN 10 K VARIATION DURING OPERATION
   - CAN BE WARM DURING LAUNCH AND COAST PHASES, AVOID LARGE VARIATIONS

SUPPORT REQUIREMENTS:

1. AVERAGE DATA RATE OF 16 MBPS BEFORE COMPRESSION
2. AVERAGE POWER: 65 W: 15 W ELECTRONICS + 50 W COOLING
3. FOV SWATH SHOULD BE PERPENDICULAR TO THE GALACTIC PLANE.
4. DETECTOR ARRAY MUST BE ALIGNED WITH E-W LUNAR ROTATION (3 ARCSEC)

LLT INSTRUMENTATION

The LTT should contain instrumentation to monitor its own environment and for housekeeping functions. There are many unknowns in the lunar environment that may effect telescope operations such as micrometeoroid impacts, dust from secondary emission of meteoroids, dust from solar levitation, cosmic radiation, temperature extremes, etc. Housekeeping functions include side to side temperature variations, bulk temperatures, optical train alignment and adjustment, detector cooling and alignment, subsystems status, etc. Without going into the details required to define instrumentation for such measurements, a gross estimate was made to serve as a place holder.

The sensors will be located at various points on the LTT both internal and external. The electronics for supporting the sensors and interfacing with the C&DH subsystem are best located on the lander and have a volume of 10x10x20 cm^3. The estimated instrumentation mass is 50 kg and the average power and data rate are 50 W and 8 kbps, respectively.

LTT INSTRUMENTATION

* PURPOSE:
  - MONITOR AND CONTROL TELESCOPE FUNCTIONS
  - MONITOR THE LUNAR ENVIRONMENT
* TYPE SENSORS:
  - TEMPERATURES ON SUN SIDE, DARK SIDE AND INSIDE THE TELESCOPE
  - OPTICAL TRAIN: FOCUS, ADJUSTMENT, AND E-W ALIGNMENT
  - ENVIRONMENT: RADIATION, METEOROIDS, DUST, ETC.
* SUPPORT REQUIREMENTS:
  - LOCATION: MISC FOR SENSORS. WARM ELECTRONICS ON THE LEV
  - VOLUME: SENSORS (TBD); ELECTRONICS 10 X 10 X 20 CM^3
  - AVERAGE POWER: 50 W (TBD)
  - AVERAGE DATA RATE: 8 KBPS
  - MASS ESTIMATE: 50 KG

2.4 SITE SELECTION

LTT SITE SELECTION

A suitable site for the placement of the LTT will be chosen using several criteria. The most critical criteria are: geologically suitable for remote landing, direct line of sight to Earth, and near a lunar limb. Other considerations are: closness to future lunar activity, east vs west limb, and latitude.

For a remotely landed vehicle it is desirable to choose a site as level as possible, free of craters and boulders that make the landing more complex. The better the landing site the higher the probability of success. Identification of such a site will require more detailed mapping and exploration of the lunar surface prior to final site selection.

Direct line of sight to the landing site is deemed very desirable. This will aid in the actual landing of the vehicle by enabling mission controllers to participate. The line of sight site will simplify communications. Any site on the backside will require communication satellite(s) or relay stations which makes the communications activity more complex and costly. As the complexity increases the probability of equipment failure increases.

The LTT line of sight should be pointed along the local vertical. However, if the site is at the equator and on the near side the Earth shines directly into the telescope. Also near noon on each lunar day the sun would shine directly down the telescope. To alleviate both earth and sun shine problems the LTT site should be at a higher latitude and near the limb. A site on the lunar equator will provide maximum celestial viewing. The other extreme, a site at a lunar pole, provides only a single spot on the celestial sphere. Due to more favorable sun lighting constraints (see celestial viewing from the Moon) at higher latitudes a site 30-40 degrees north or south of the equator seems desirable. Preference for particular celestial targets can influence the latitude of the site selected.

The above criteria and considerations will determine the lunar site for the LTT. The relative weight to give each criteria has not been determined at this time. Other criteria and considerations not yet identified will also be used in selecting the lunar LTT site. In conclusion, using the known criteria and considerations a LTT site near the east or west lunar limb and a latitude of 30-40 degrees is recommended.

12/16/91:BGD

SITE SELECTION CRITERIA

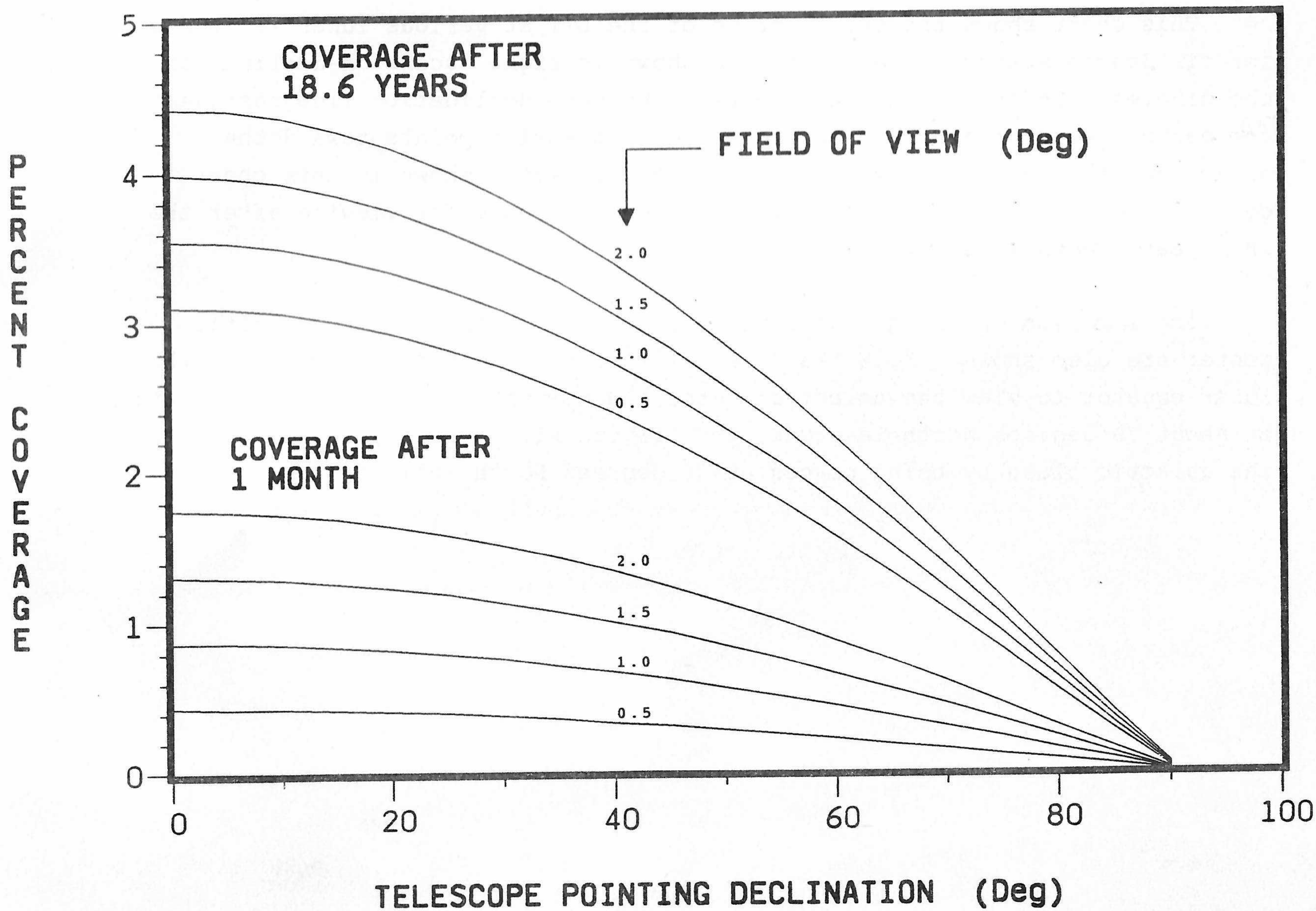
* AVOID EARTH SHINE
* CONTINUOUS COMMUNICATIONS WITH THE EARTH
* AVOID SUNLIGHT IN THE TELESCOPE APERTURE
* PROVIDE VIEWING PERIODS NEAR PERPENDICULAR TO THE GALACTIC PLANE
* SMOOTH, STABLE LANDING TERRAIN: MINIMUM SLOPE, ROCKS AND CREVICES
* AVOID DUST FROM OTHER LUNAR ACTIVITIES
* MINIMIZE MICROMETEROID FLUX
* VISIT OR REPAIR FROM FUTURE LUNAR OUTPOST

LTT PERCENT CELESTIAL SPHERE COVERAGE

As the moon turns on its axis, the LTT field of view sweeps across the sky in a narrow swath. In one month, the Moon completes one revolution. The percentage of the celestial sphere covered by the LTT's field of view after one month of lunar rotation is shown as a function of pointing declination in the lower four curves of this figure. In the MSFC design, the LTT field of view is 1.5 degrees and the planned pointing declination is between 30 and 40 degrees. This yields slightly more than 1 percent of the sky viewable in one month.

The moon's orbit plane precesses about the ecliptic pole completing a 360 degree cycle in 18.6 years. The ascending node of the lunar equator on the ecliptic plane remains aligned with the descending node of the lunar orbit plane on the ecliptic plane. This means that the lunar spin axis "cones" about the ecliptic pole completing one cycle in 18.6 years. This "wobble" in the Moon's orientation causes the LTT field of view swath to see a larger area of the sky over an 18.6 year period. The upper four curves in the figure show the increased sky coverage obtained after an 18.6 year period. For a 40 degree pointing declination, slightly more than 3 percent of the sky can be observed after 18.6 years.

LTT PERCENT CELESTIAL SPHERE COVERAGE



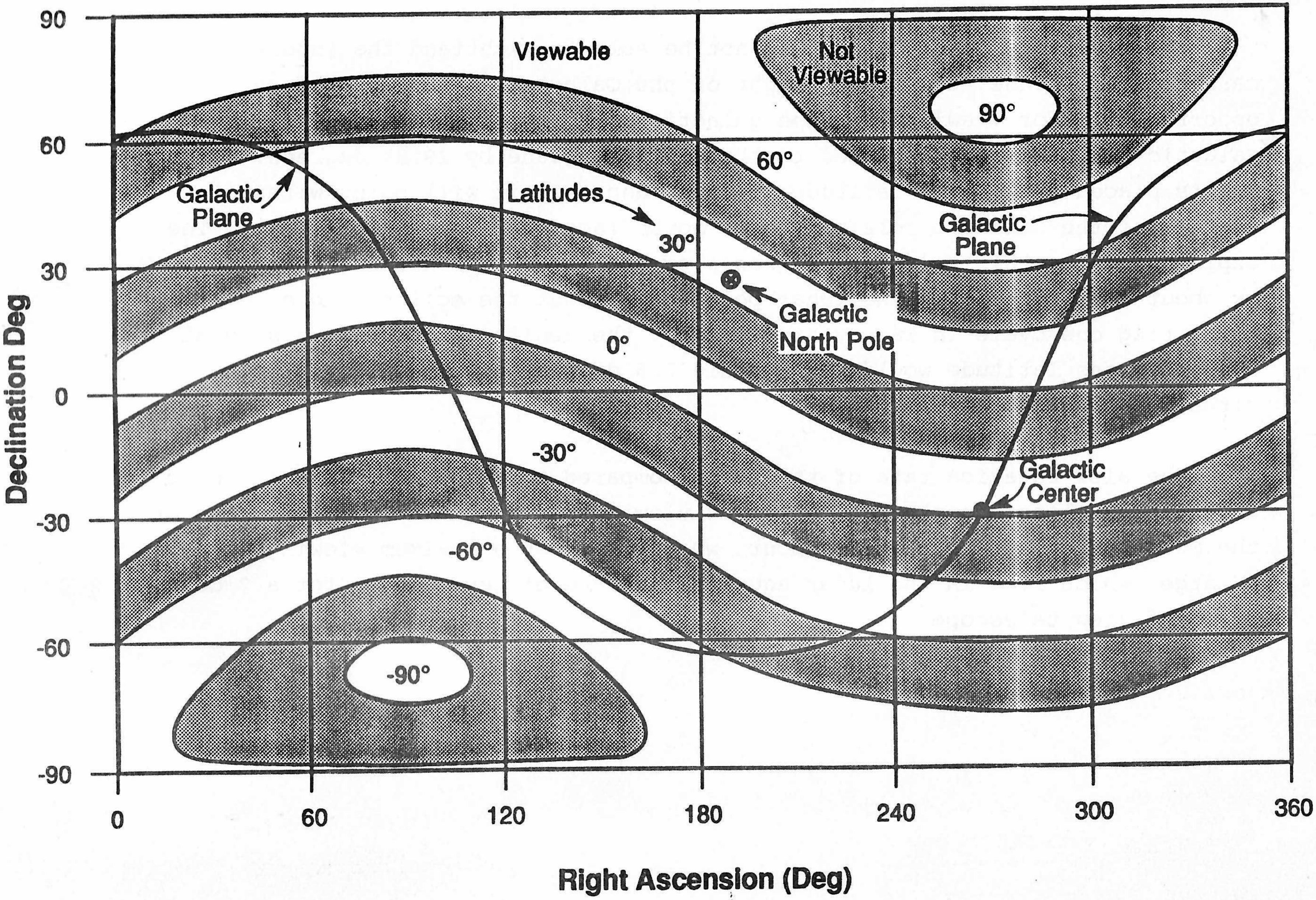
CELESTIAL SPHERE COVERAGE AT VARIOUS LATITUDES

This chart shows the sky coverage of the LTT at various lunar latitudes for six degree swaths. These data are shown in right ascension-declination coordinates. In this coordinate system, the zero declination line represents the earth's equatorial plane and zero right ascension points toward the direction of Vernal Equinox. The sky coverage swaths shown in this chart depict those areas of the sky that have been available for viewing after the 18.6 year precessional cycle.

The location of the galactic plane and the direction of the galactic center are also shown. Note that the LTT must be placed fairly close to the lunar equator to view the galactic center. Since the galactic pole is located at about 30 degrees North latitude, the LTT can view a belt perpendicular to the galactic plane by being placed at 30 degrees North latitude.

1-2826-0-56D

Celestial Sphere Coverage at Various Latitudes. Field of View: 6 Deg From Zenith



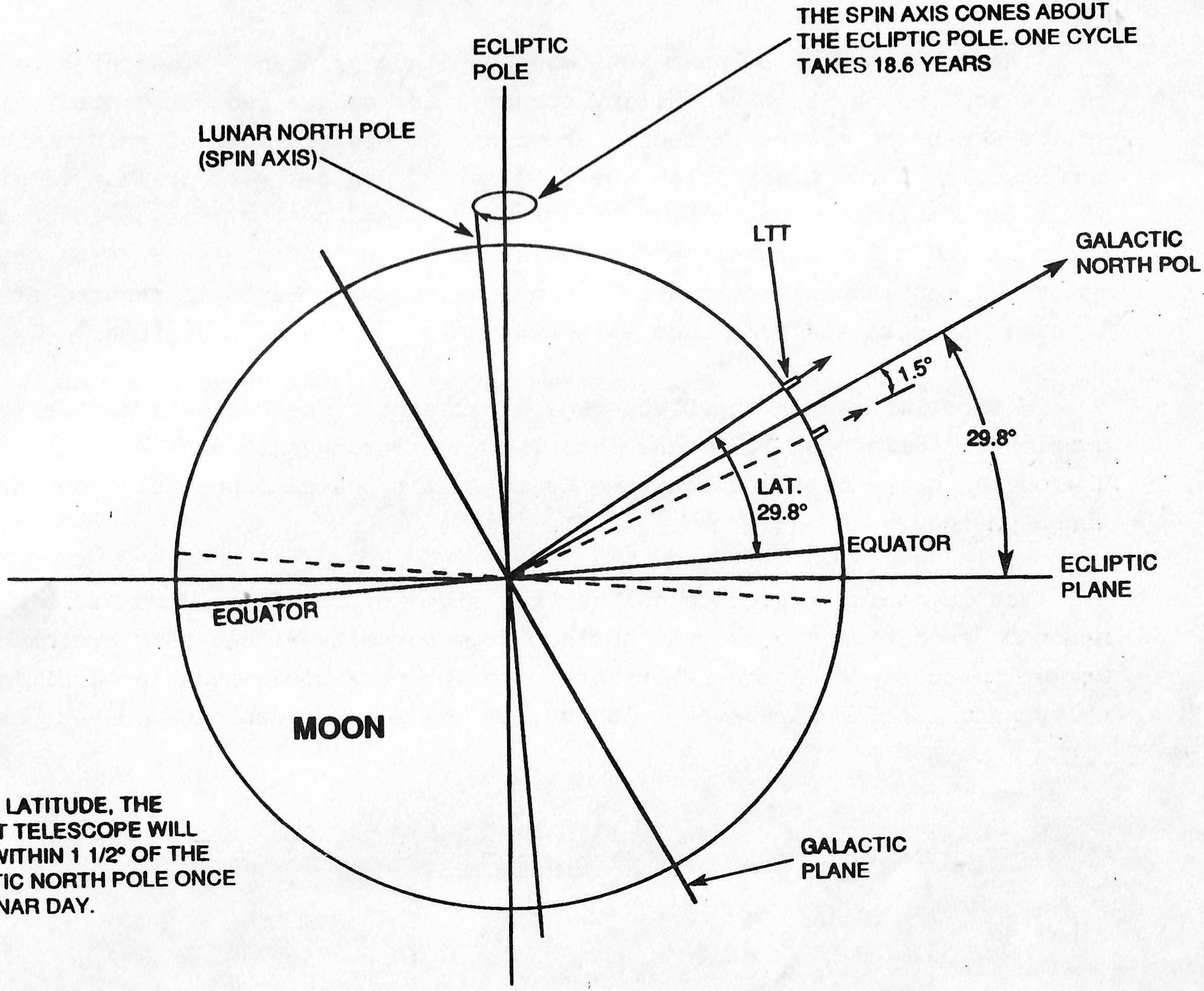
LUNAR TRANSIT TELESCOPE LATITUDE PLACEMENT FOR GALACTIC POLAR VIEWING

The sensitive LTT sensor would not be able to withstand the interference caused by the dense, bright starlight of the Galactic plane. Therefore, opportunities for viewing near the galactic pole should be provided. The galactic north pole is inclined to the ecliptic plane by 29.84 degrees. If the LTT is placed at a lunar latitude of 29.84 degrees, it will point within 1.5 degrees of the Galactic pole once each month (assuming zenith pointing). The explanation for this is that the Moon's equator is inclined from the ecliptic by about 1.5 degrees and the lunar pole cones about the ecliptic pole completing one cycle in 18.6 years. Hence, the zenith vector from a site at 29.84 degrees latitude would pass within 1.5 degrees of the Galactic pole direction once each month.

The slow rotation rate of the moon, compared to the earth, offers the advantage of long, uninterrupted target viewing periods. The rotation rate of the moon (sidereal) is 0.55 deg/hour, which provides a maximum viewing time for a target which lies in the lunar equatorial plane of seven hours for a 2 degree field of view telescope.

1-1693-1-32D

Lunar Transit Telescope Latitude Placement for Galactic Polar Viewing



AT 29.8° LATITUDE, THE TRANSIT TELESCOPE WILL POINT WITHIN 1 1/2° OF THE GALACTIC NORTH POLE ONCE PER LUNAR DAY.

EARTH VISIBILITY FROM THE MOON

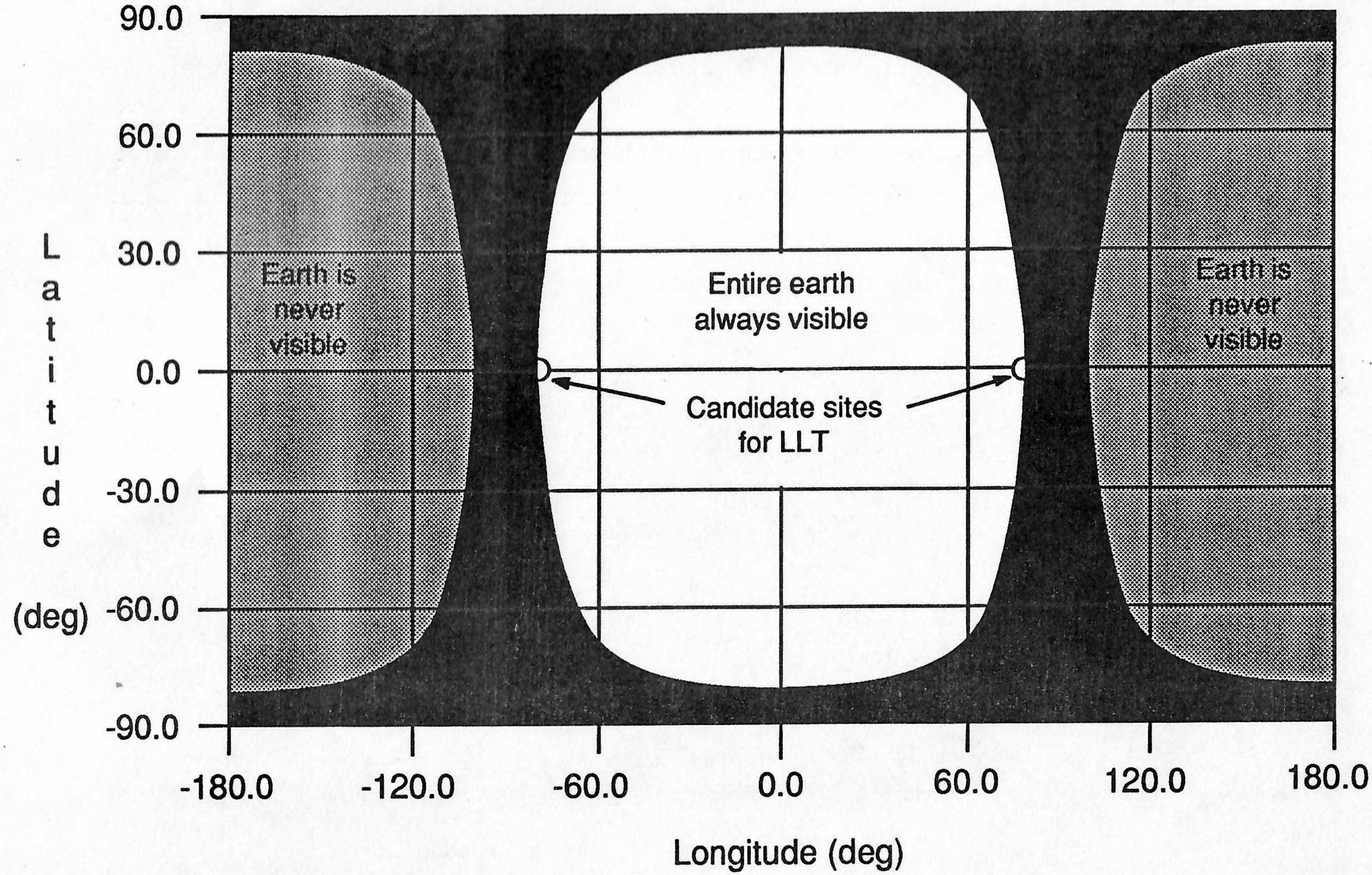
This chart shows a lunar latitude-longitude grid and illustrates regions on the Moon where Earth visibility occurs. Inside the central contour, the entire Earth can always be seen. Shown at the western edge of this contour is the baseline Large Lunar Telescope (LLT) at 80 degrees west or east longitude. It is desirable to locate the Lunar Transit Telescope (LTT) at 30 to 40 degrees north latitude for viewing perpendicular to the galactic plane. From this latitude, continuously available communications with Earth is assured at all times as long as the longitude is between plus or minus 70 degrees.

A site located in the black region can expect the Earth to partially (or completely) disappear below the lunar horizon during part of the month. Therefore, communications with the Earth will experience some blackout during these periods.

The remaining region is on the "far-side" of the Moon. The Earth can never be seen rising above the horizon from any site within this region. Communications with Earth, therefore, are not possible without an orbiting relay satellite, lunar-based relay system, or a remote antenna.

1-1174-1-21D

Earth Visibility From The Moon

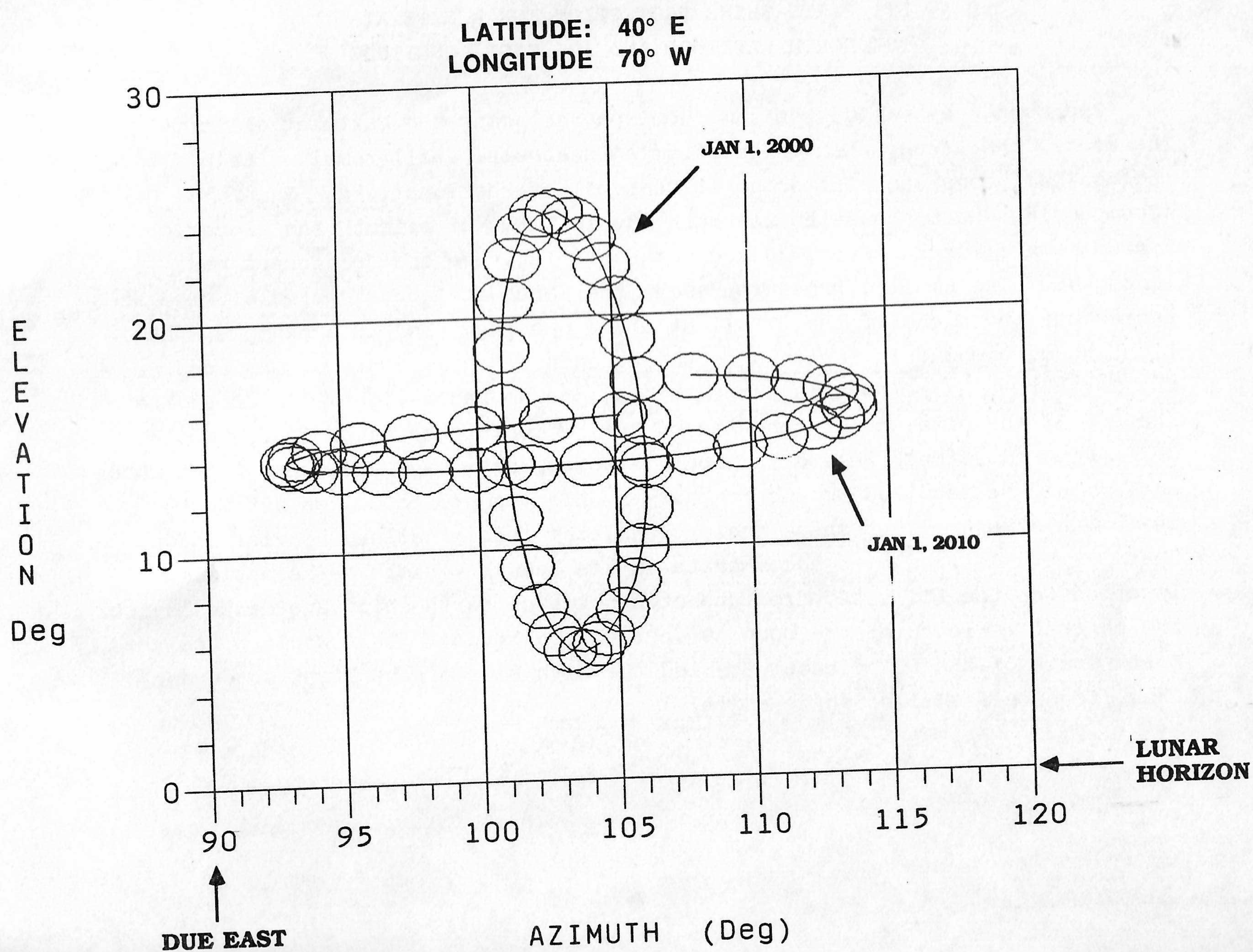


EARTH POSITION AS SEEN FROM LUNAR SITE

This chart shows a view of Earth motion as seen from the telescope site located at 40 degrees north latitude and 70 degrees west longitude. The view is looking due east on the lunar horizon. Each small circle represents the Earth position in azimuth and elevation. The position of the Earth is indicated in one-day increments for 360 days for the years 2000 and 2010.

It is seen from this chart that the Earth will always be confined to a relatively small region of the sky as viewed from the LTT at 70 degrees west longitude on the equator. This apparent motion of the Earth is caused by the optical librations in latitude and longitude.

EARTH POSITION AS SEEN FROM LUNAR SITE

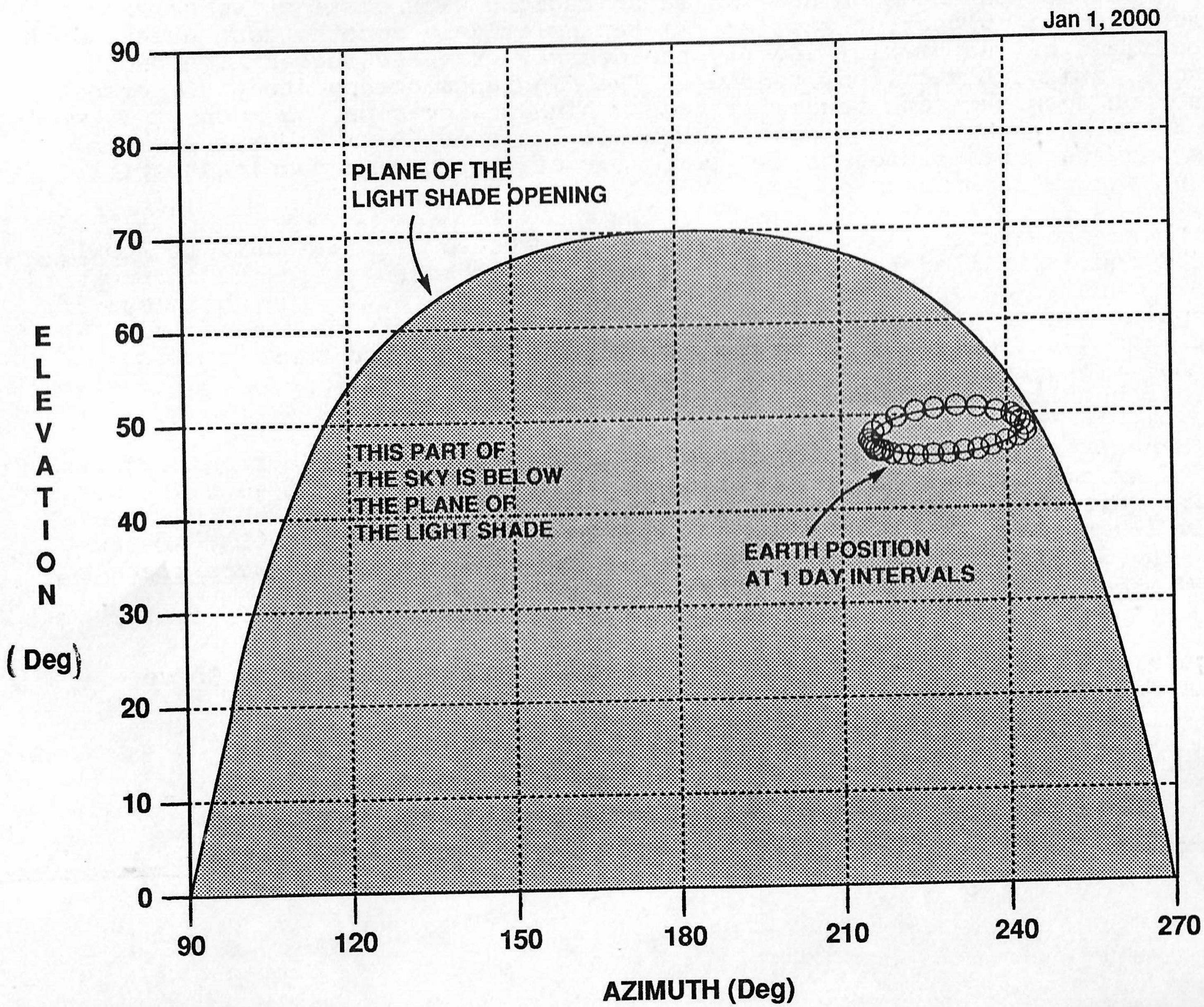


LTT EARTH-SHINE PROTECTION FOR A SITE AT 30° NORTH LATITUDE AND 30° EAST LONGITUDE

The Moon keeps about the same hemisphere facing the Earth at all times. Therefore, the Earth, when viewed from a lunar site, will remain within a small region (half cone angle of about 11 degrees) of the lunar sky. The accompanying chart shows the southern sky in terms of azimuth and elevation as viewed from a lunar site at 30 degrees north latitude and 30 degrees east longitude. The small circles represent the Earth at 1-day intervals. The arc represents the plane of the LTT light shade opening. For this example, the light shade opening is inclined to the horizontal by 70 degrees. As long as the Earth remains below this arc in elevation, no Earth-light can enter the shade. If the site were farther east, the region of the Earth motion would shift west in azimuth (elevation would also decrease) and a steeper light shade angle would be required to ensure that the LTT was protected against Earth shine. A steeper light shade angle translates into a taller, heavier shade that is more difficult to accommodate in the launch vehicle. The optimum location for the LTT site, from the standpoint of Earth-shine avoidance is near the central meridian of the Moon (0 degrees longitude). This would place the Earth south of the LTT directly behind the high side of the light shade and, thus, require a smaller shade angle.

1-2765-1-66D

LTT Earth Shine Protection for a Site at 30° North Latitude, 30° East Longitude



THE LUNAR SURFACE[1]

The surface of the Moon has two major regions with distinctive selenogic features and evolutionary histories (1) the relatively smooth, dark areas which Galileo first called "maria" and (2) the densely cratered rugged highlands (uplands), originally called "terrae". The highlands occupy about 83 percent of the Moon's surface and generally have a higher elevation, as much as 5 km above the mean radius. In places, the maria lie about 5 km below the mean radius and are concentrated on the near side of the Moon, a fact that still lacks an adequate explanation.

The surface is strongly brecciated or fragmented. This mantle of weakly coherent debris, called the regolith, consists of shocked fragments of rocks, minerals, and glass spherules formed by a meteorite impact. The thickness of the regolith is highly variable and depends on the age of the underlying bedrock and the location with respect to nearby craters and their ejecta blankets. In general, the maria are covered by 3 to 16 m of regolith, whereas the older highlands have developed a "soil" at least 10 m thick.

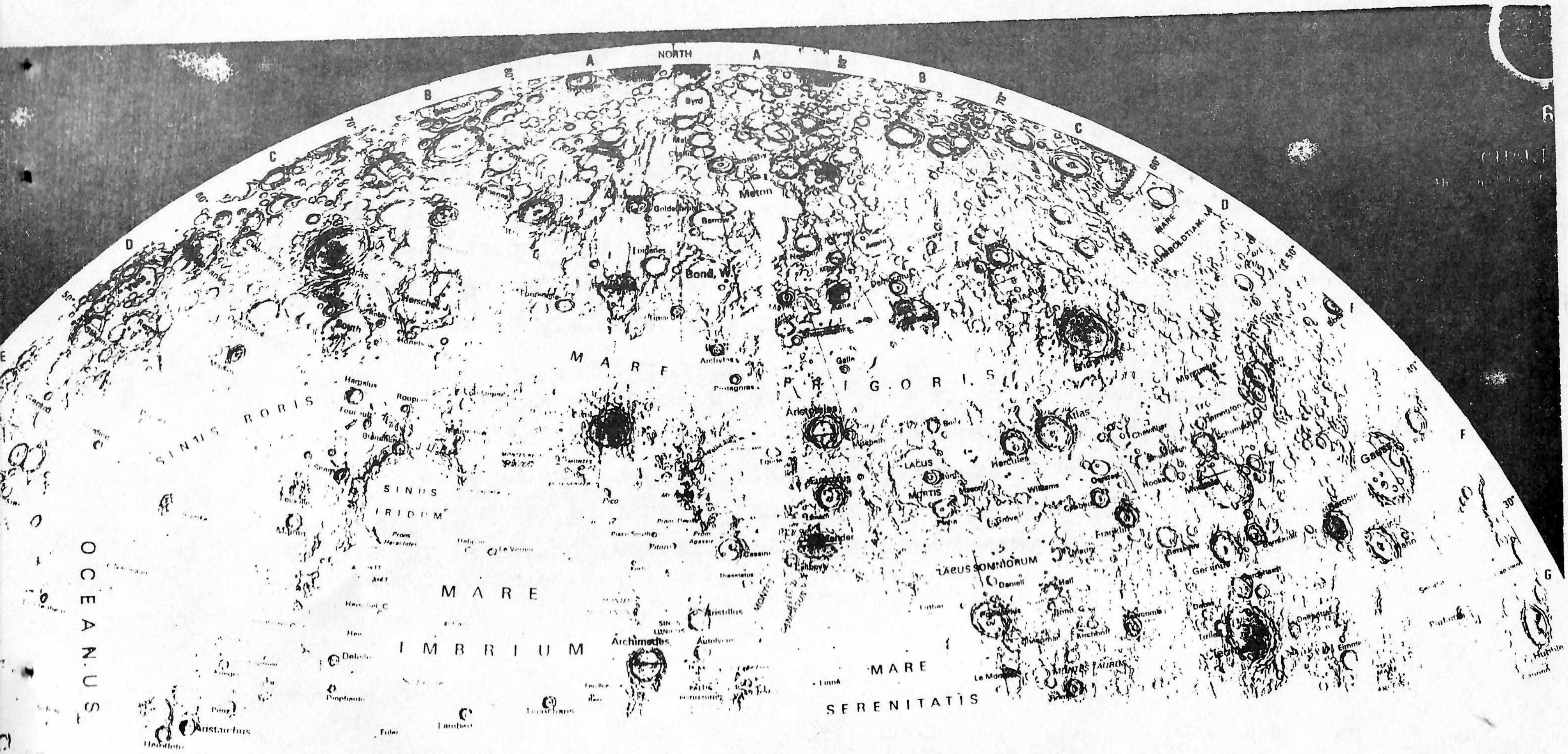
Much of the information regarding the lunar surface is the result of six manned landings on the Moon. Summaries of the selonographic of each of the Apollo landing sites are contained in a series of NASA Special Publications (Apollo 11, SP-214; Apollo 12, SP-235; Apollo 14, SP-272; Apollo 15, SP-289; Apollo 16, SP-315; and Apollo 17, SP-330). The Apollo missions were preceded by a series of unmanned Surveyor missions (NASA SP-184, 1969).

1 Space and Planetary Environment Criteria Guidelines for Use in Space Vehicle Development. 1982 Revision, Volume 1, NASA TM 82478

12/2/91:BGD

THE LUNAR SURFACE

NEAR SIDE OF THE MOON, NORTHERN HEMISPHERE

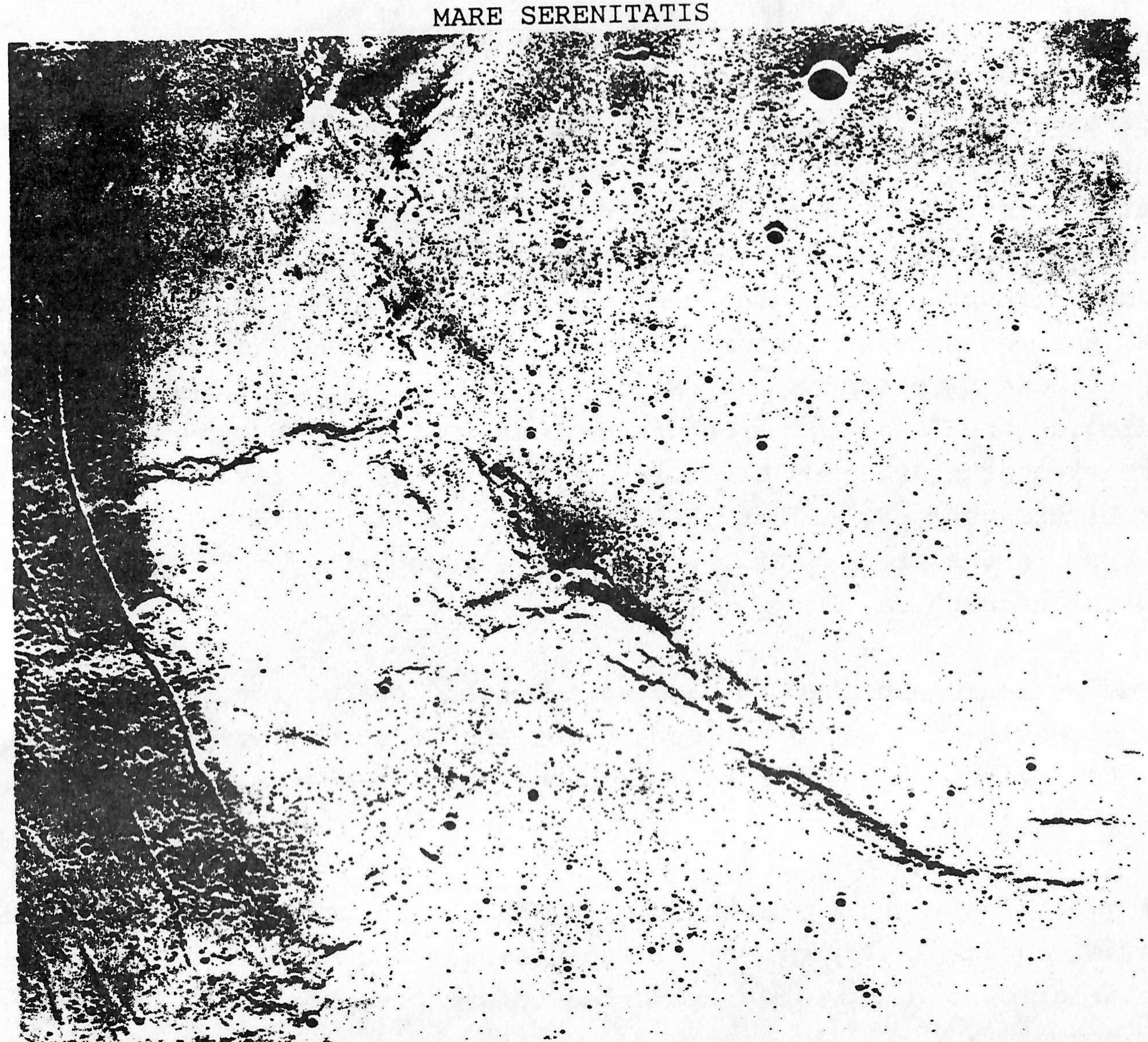


MARE SERENITATIS

The lunar maria are concentrated in topographic basins on the near side of the Moon. The average normal albedo of the maria is a rather low 7 percent. Some maria occur within large circular impact basins such as Crisium, Serenitatis, and Imbrium, whereas others, like Oceanus Procellarum, occupy irregular depressions. The lunar maria are vast plains composed of basaltic lava flows that were erupted after the highlands and the impact basins formed. The basalt in the middle of the basins may be several kilometers thick but is still only a small part of the crust that is approximately 60 km thick.

The maria include narrow ridges, several kilometers wide, with sinuous outlines that extend discontinuously for great distances and, in some cases, transect highland surfaces as well. "Wrinkle" ridges, as they are sometimes called, are believed to be formed by compression as mare basins subsided, adjusting to the load of accumulating basaltic lavas.

MARE SERENITATIS



Wrinkle ridges and lunar rilles are prominent features in this photograph of the margin of Mare Serenitatis. The linear rilles are on an older surface that was partially buried before the ridges formed (AS-17-601).

LIKELY SITES FOR LTT

Berosus crater and Mare Crisium have been suggested as likely sites. Both are surrounded by extensive broken terrain and can be reached only by a dedicated direct landing mission. This is not a problem for the LTT, but could be if man wanted to pay it a later visit. Based on trade studies that have been done, it appears that the best compromise location for the LTT is the central portion of Mare Imbium (42^{o} N Lat., 15^{o} W long.) or the northern region of Mare Serenitatis (30^{o} N Lat., 15^{o} E Long.). Another option, depending on the final location of the Lunar Outpost, is offered by the far western reaches of Oceanus Procellarum. For instance, a site near the small crater Lichtenberg (32^{o} N Lat., 68^{o} W. Long.) will provide significant advantages in the predictability of surface topography, structure and mechanical properties.

The maria is expected to have an average slope of 1 to 2 degrees with many rocks and boulders. Based on mapping for Apollo sites, several areas are mapped to 1-2 meters areal resolution, boulder and crater diameters, and about 1/2 meter depth resolution.

A great deal more in-depth information should be developed and interpreted on the selenograhpic topographical, operational and trafficability characteristics of many different sites before the process of final choice can fairly begin.

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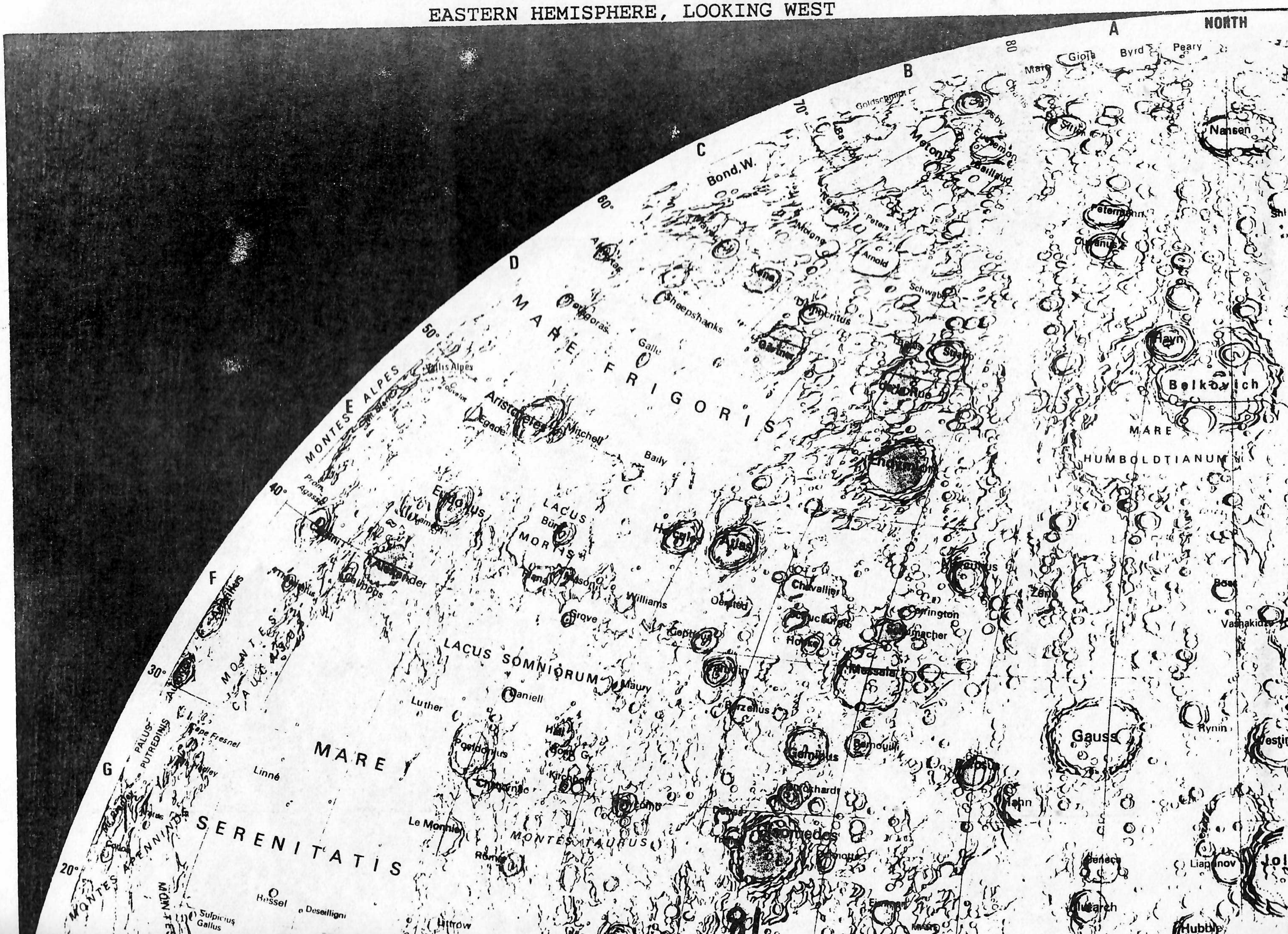
EASTERN HEMISPHERE, LOOKING WEST



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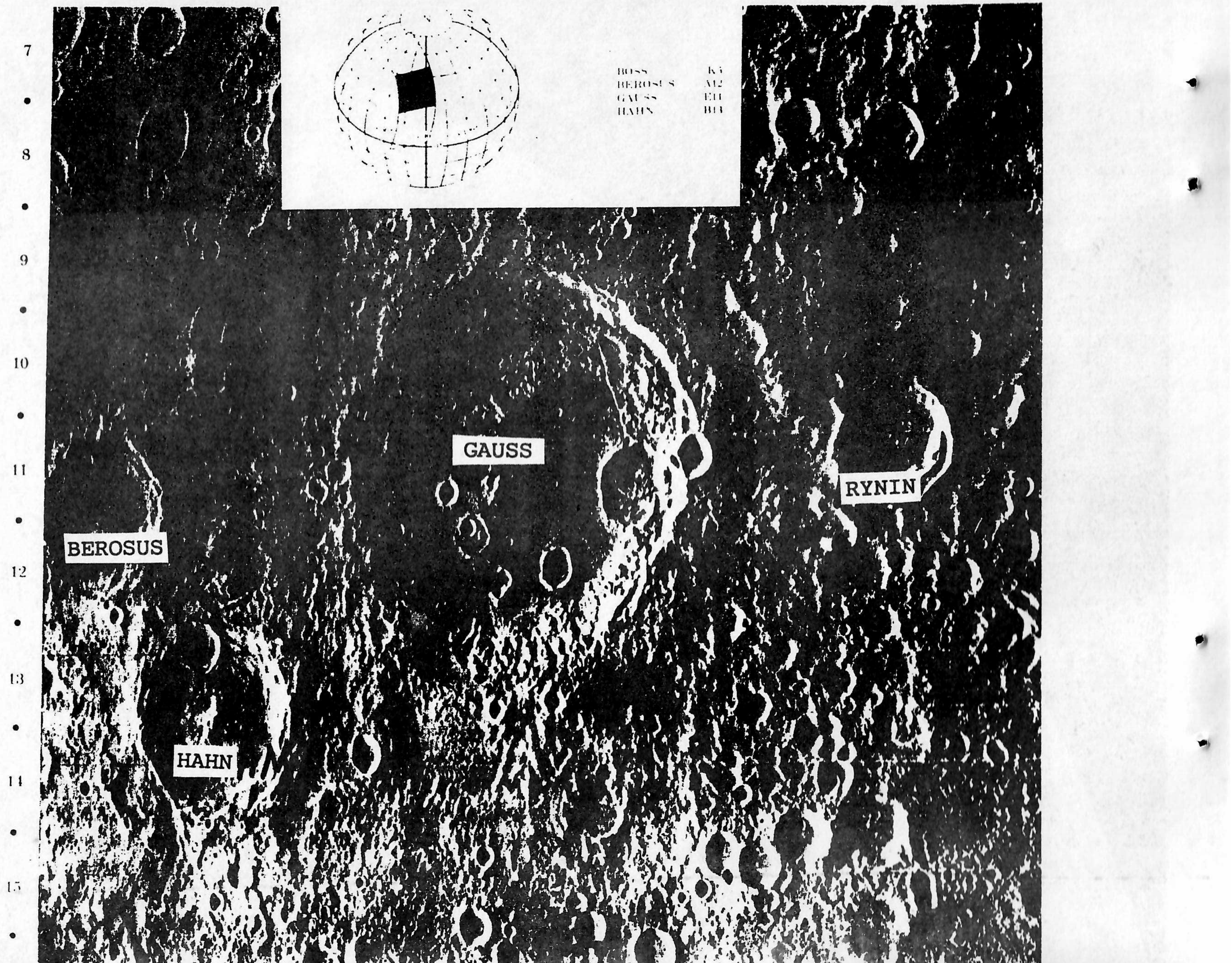
DETAIL OF GAUSS AREA



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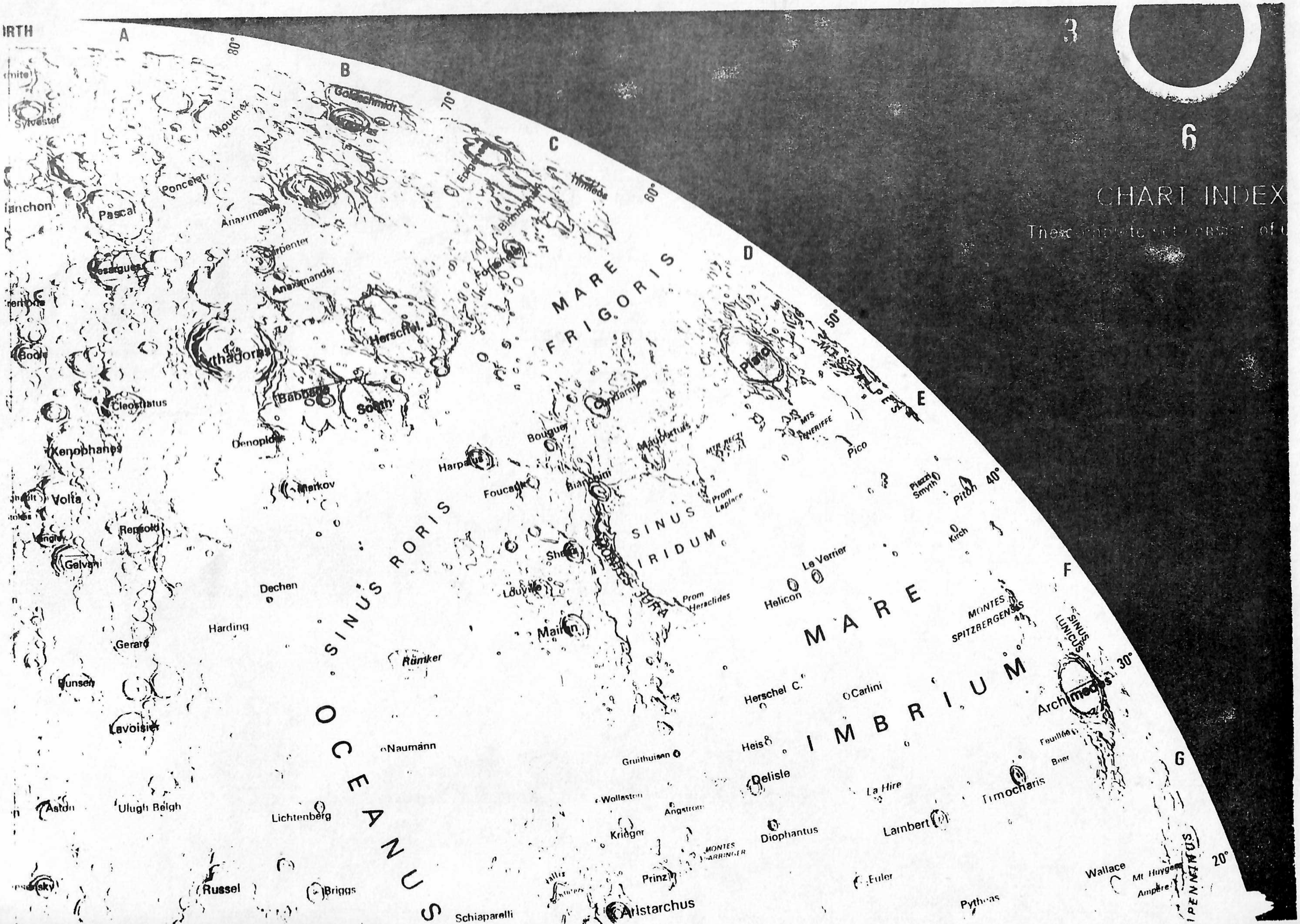
WESTERN HEMISPHERE, LOOKING EAST



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SECTION 3

3. LTT REFERENCE CONFIGURATION **PAGE**

3.1 LAYOUTS

LTT DESIGN OVERVIEW

The preliminary design of a Lunar Transit Telescope (LTT) is presented. The design is based on a maximum use of existing technology so that the spacecraft can be made operational by the year 2001 or 2002.

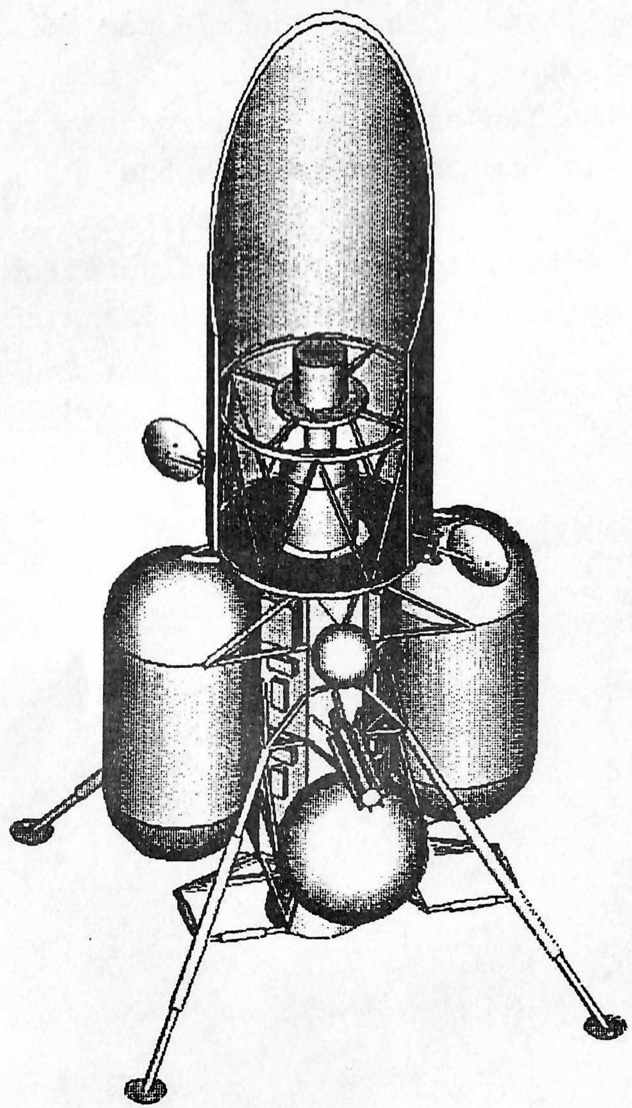
In the process of conceptualizing a LTT, a number of pertinent design considerations emerged for evaluation and discrimination. These included, but were not limited to, launch vehicle selection, the decision to use either cryogenic or storable propellants, and selection of the optical configuration.

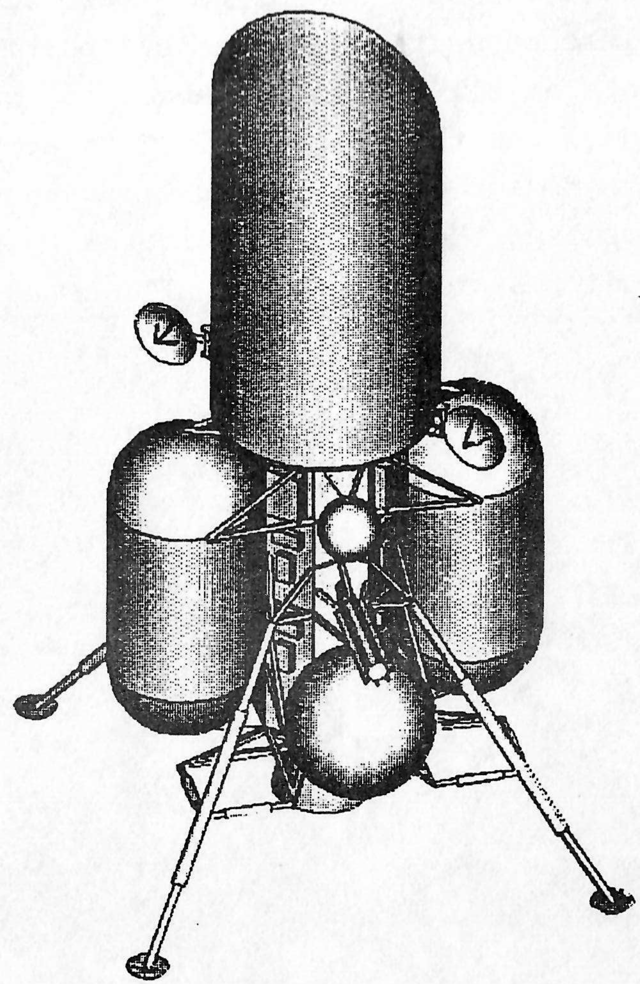
These decisions have considerable effect on the performance and quality of the design, and in some cases are so strongly influenced by the specific application, that final selection or deletion can only be made when evaluated for that singular case.

The desired early launch date with low program risk dictated that only existing launch vehicles and technology be considered. Trades in the propulsion discipline indicated that it favored the cryogenic system to satisfy the scientific communities' desire for a 2 m diameter class mirror.

1-2618-1

Lunar Transit Telescope



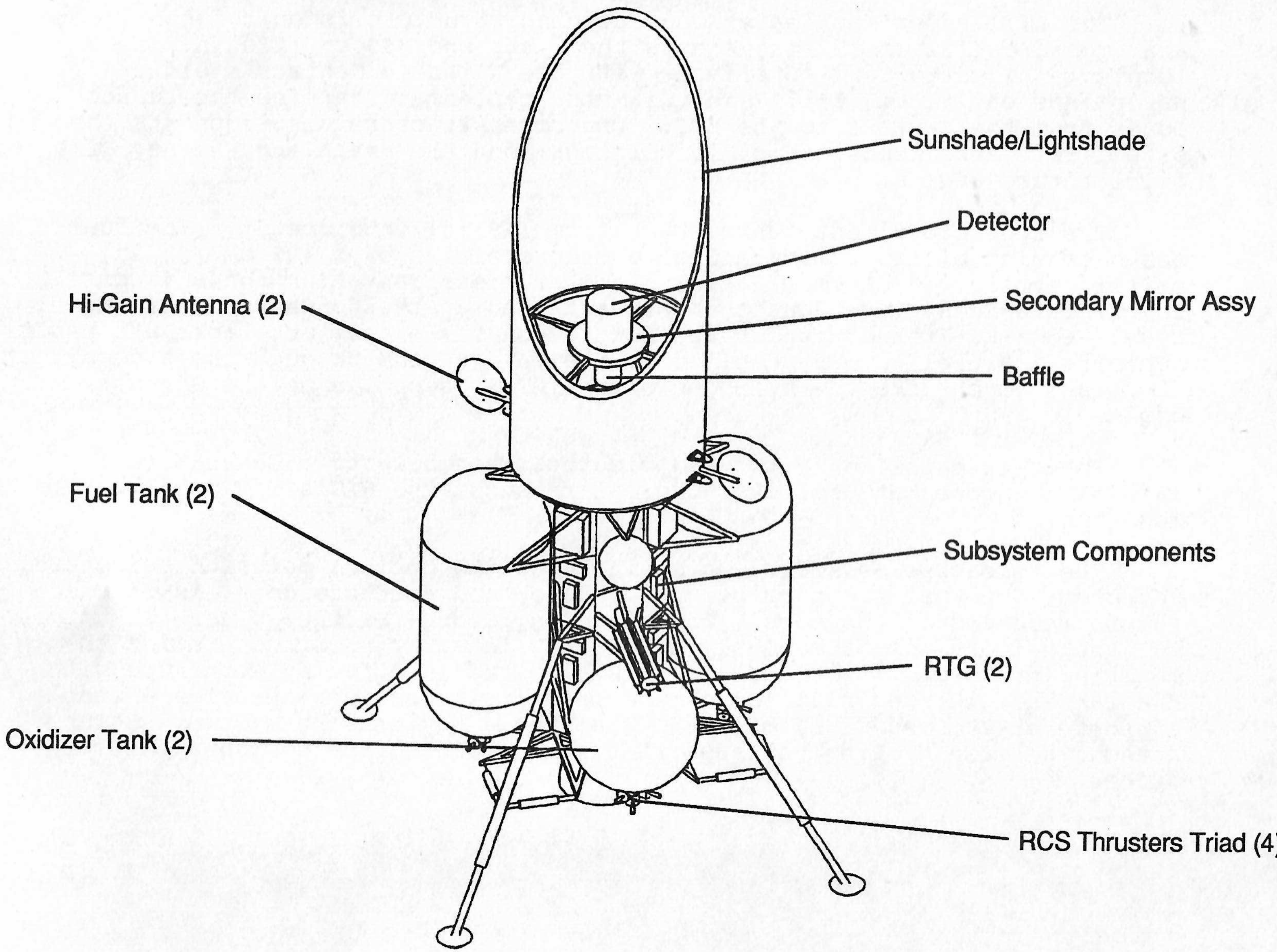


LTT CONFIGURATION LAYOUT

The general configuration of the LTT assembly is shown on the facing page with the key structural elements identified. The structure can be considered as a composite of two sections: the forward portion is the telescope section and the aft portion is the lander. The lander assembly contains all the subsystems. The subsystems are common to both the lander and the telescope. This approach permits clean interfaces and significant weight savings. Mounting subsystems low on the configuration increases the payload delivered to orbit by the launch vehicle and enhances landing stability of the LTT on the moon.

1-2759-1-65D

Lunar Transit Telescope
Three Mirror Configuration



LTT CONFIGURATION

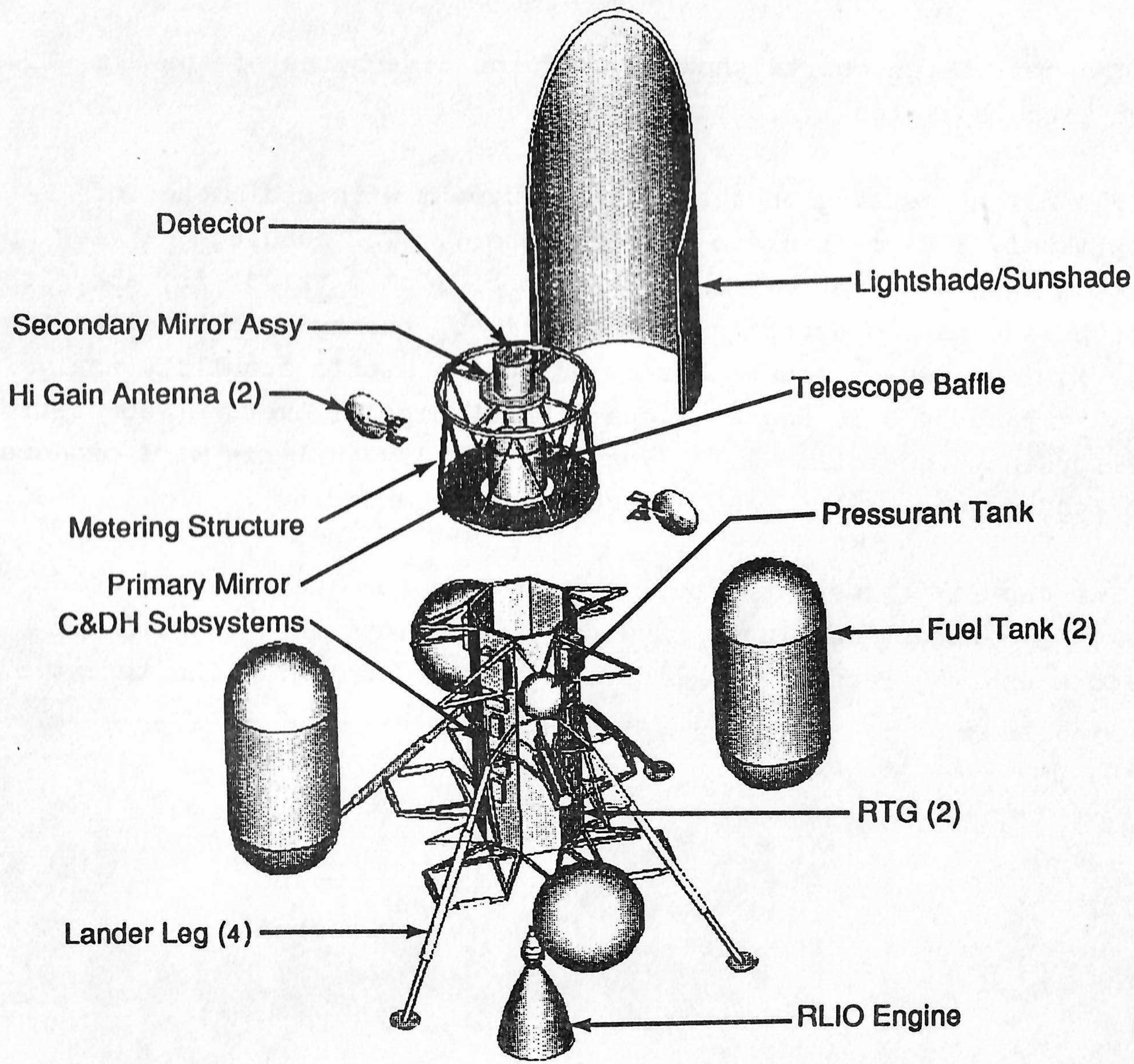
The LTT lander's primary load carrying structure is an octagonal core that measures 132 cm (52 in) across the flats and 330 cm (130 in) in length. This structure interfaces with the Titan IV/Centaur's eight hardpoints on 284 cm (111.77 in) diameter center and carries the launch loads from the Centaur to the LTT. The core structure also supports the propellant tanks, subsystems, lander legs, and the Pratt and Whitney RL10 axial thrust engine.

The four propellant tanks are 148 cm (58 in) in diameter. The fuel tanks consist of two hemispherical domes separated by a 132 cm (52 in) barrel section. A 61 cm (24 in) diameter pressurant tank and a 48 cm (19 in) diameter RCS propellant tank are attached at the forward end of the core. Four RCS thruster pods are located on the propellant tanks aft bipod supports. Propellant is supplied to the main engine through the sidewall of the tanks, much like the hydrogen on the ET is supplied to the Shuttle engines.

Power is supplied by two Radioisotope Thermoelectric Generators (RTG's), and one battery. For thermal reasons, the RTG's may require boom mounting.

The telescope assembly, which includes the optics, metering structure, detector, internal light baffles, sunshade, and aperture door, is mounted to the forward end of the core. The main ring serves as the support structure for the primary mirror assembly and as the integrating structure for the metering truss and lightshade. It is also the interfacing structure with the lander. The key function of the metering truss is to provide a stable support for the secondary mirror. The single bay truss has struts which attach the main ring to the support ring containing the secondary mirror spider.

LTT Configuration

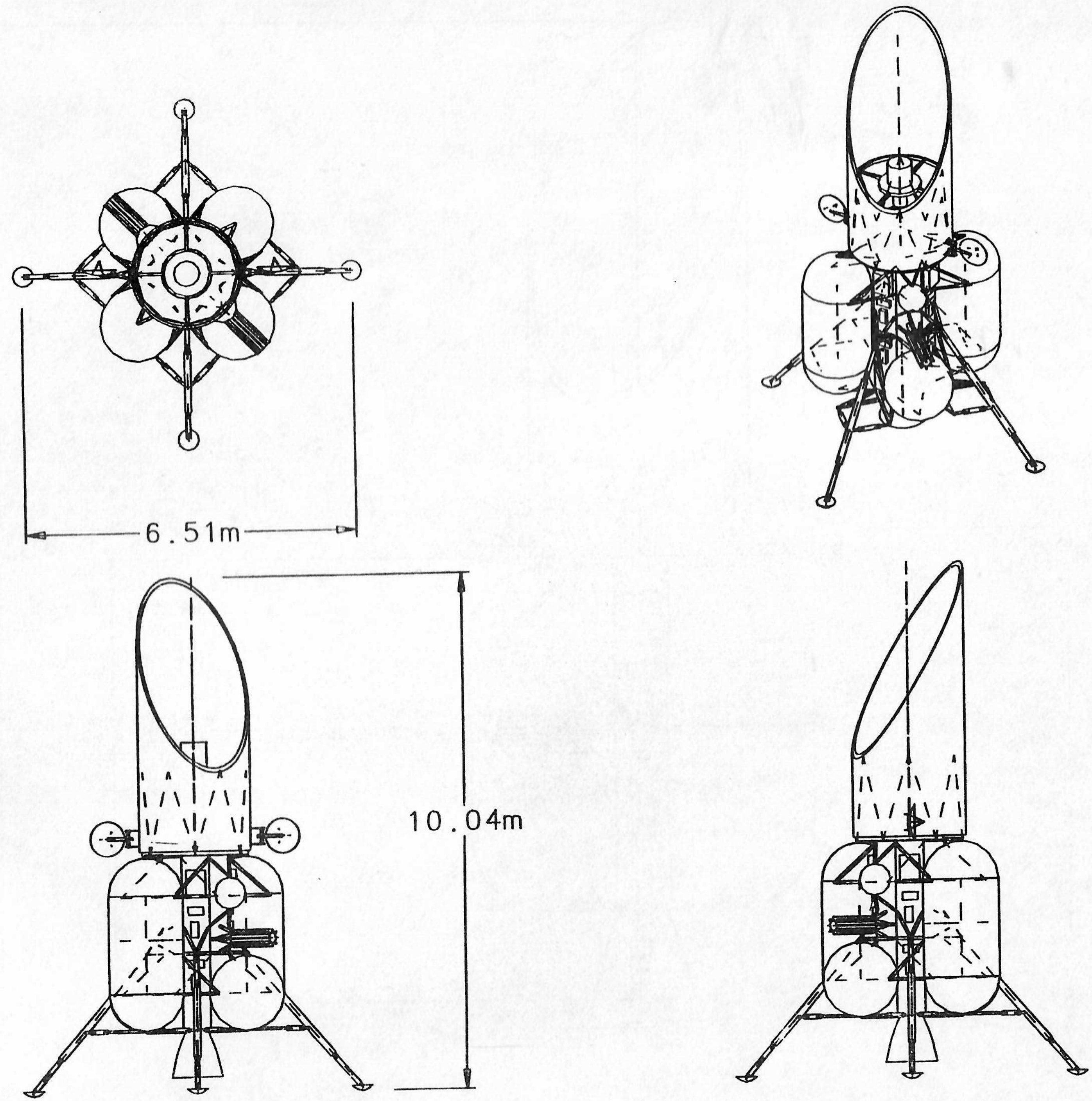


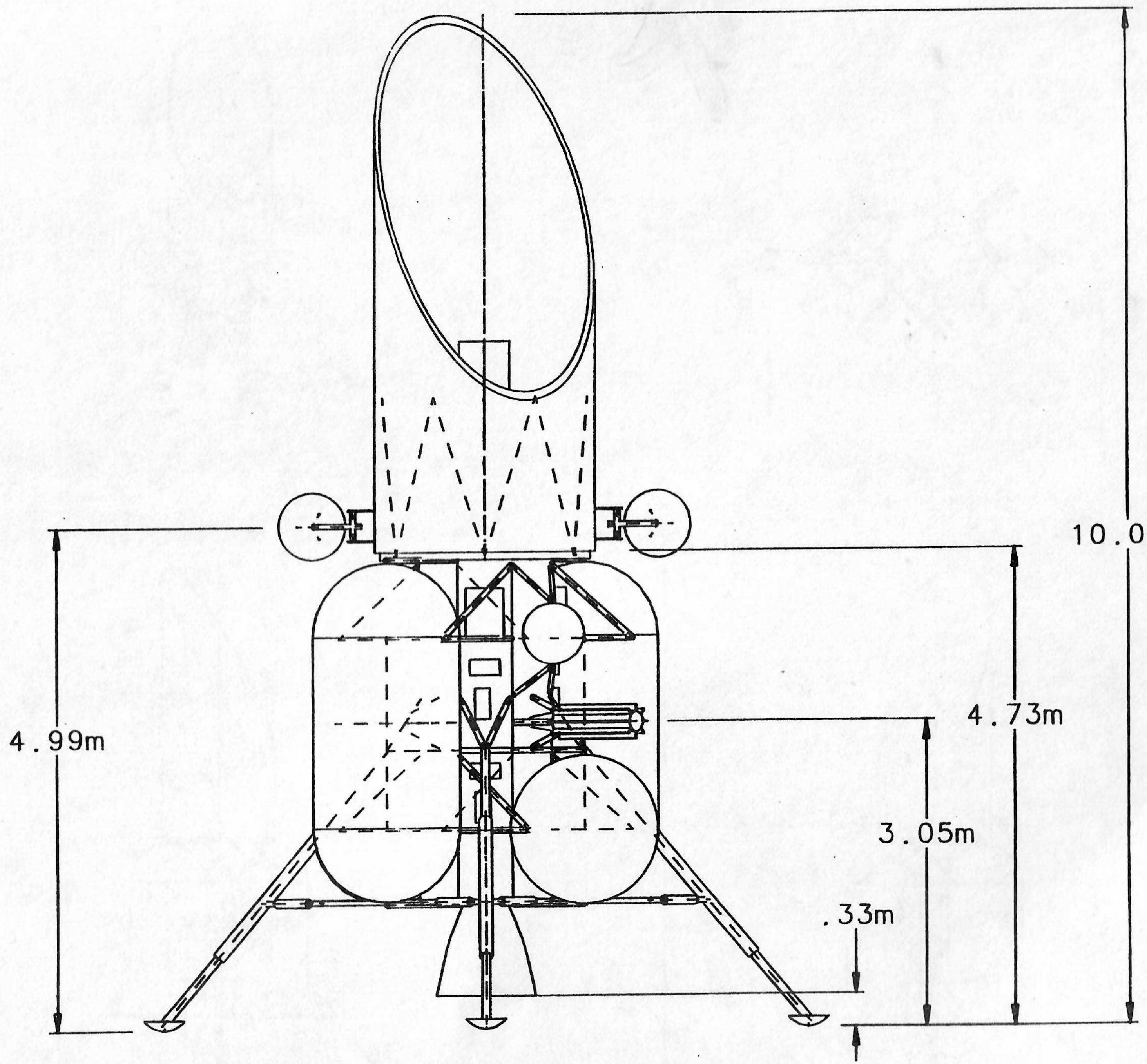
DIMENSIONS OF THE LTT

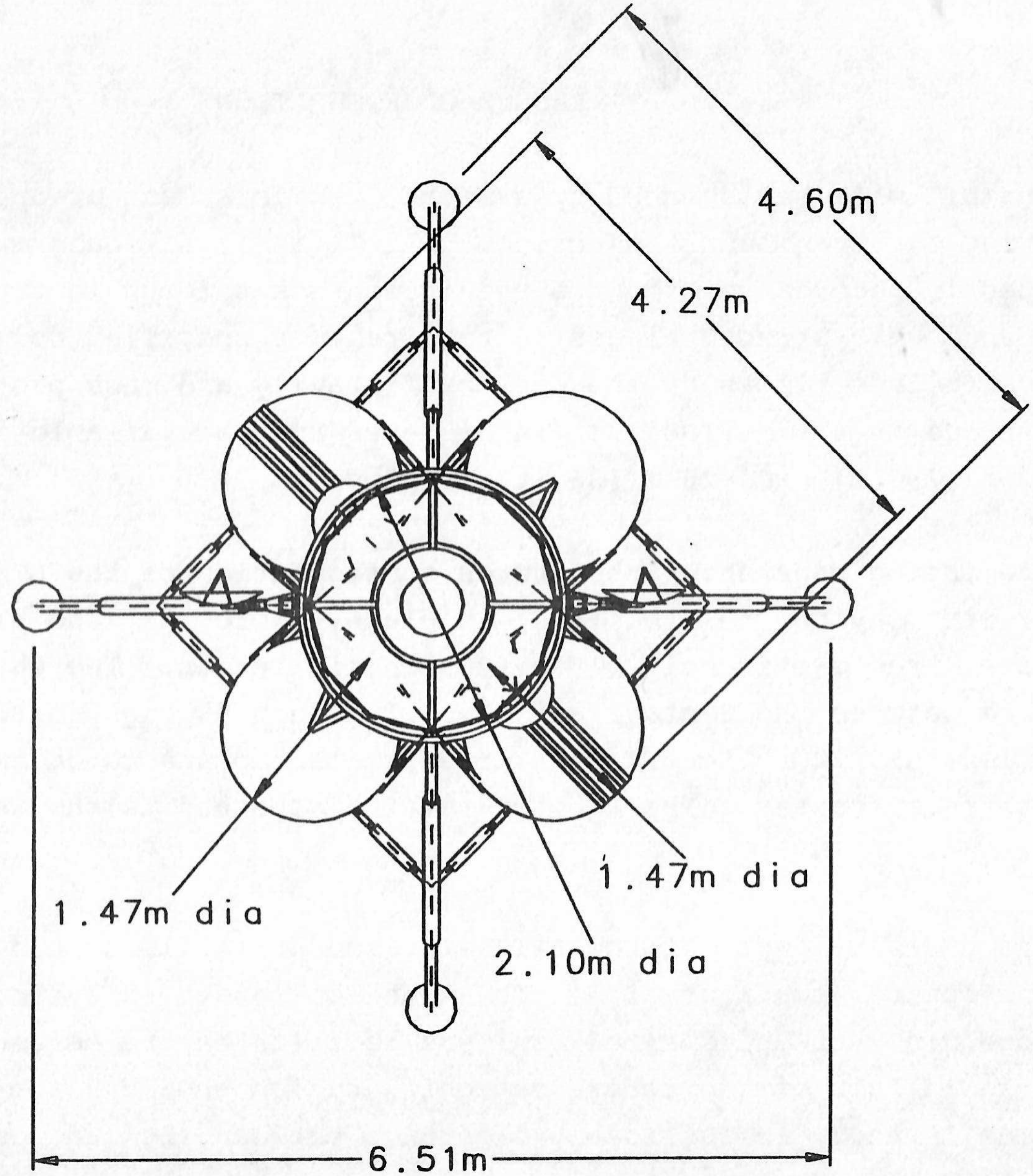
The next three charts show some of the dimensions of the 2 m LTT for launch with the Titan IV/Centaur.

The height setting on the moon is 10.04 m with a diameter of approximately 4.27 m (tank to tank diameter). The landing gear leg spread is 6.51 m with a center of mass distance of 3.47 m (11.37 ft) above the regolith (see mass properties). The ratio of the cg height to landing gear radius is 1.07, which means that based upon a static stability analysis the LTT could land on a 35 degree slope without tipping over. (See static stability in this section.) A dynamic stability analysis will produce less slope landing capability (such an analysis is underway).

The central thrust structure is designed for the expected loads during launch, RL10 thrust and lunar landing, and to interface between the telescope and the lander with minimum structure weight. The thrust structure is the LTT integrating structure between the propulsion system, landing gear and telescope.


6.51m
10.04m


10.04m
4.73m
3.05m
.33m
4.99m


4.60m
4.27m
1.47m dia
1.47m dia
2.10m dia
6.51m

LTT LAUNCH CONFIGURATION

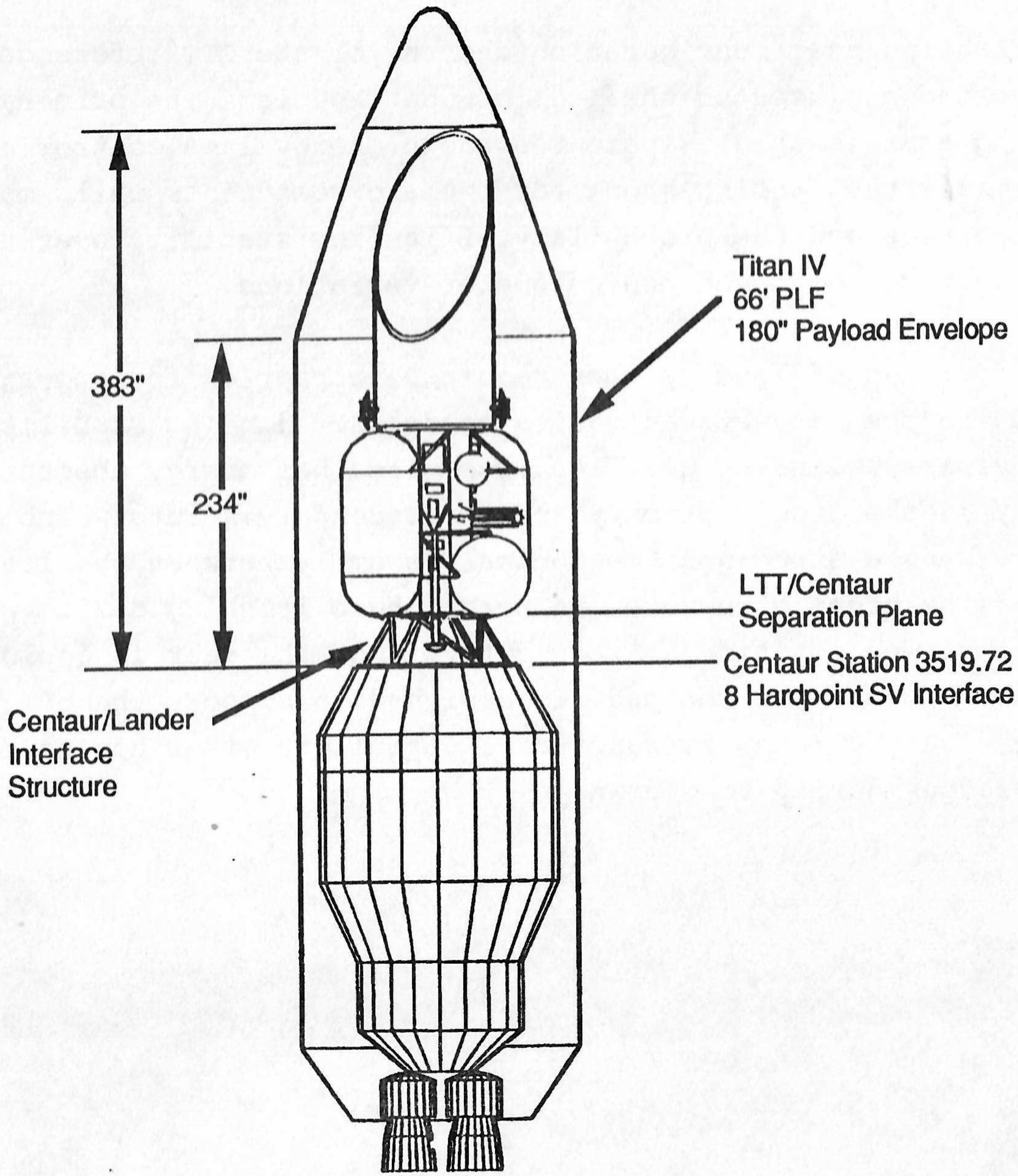
The LTT will be launched by a Titan IV with a Centaur upper stage performing the translunar injection. All Titan IV payloads must be developed in accordance with the requirements set forth in the Titan IV Users Handbook. Structural design requirements specified for Titan IV payloads include limits on the center of gravity and mass properties. The Titan shroud is 4.6 m (180 in) in diameter and is available in 20 m (66 ft), 23 m (76 ft), and 26 m (86 ft) lengths.

The facing page shows the launch configuration of the LTT on the Centaur with payload fairing (PLF) in place. Note that the LTT with its interface structure is only 9.8 m (383 in) in length. The physical interface between the Centaur and the LTT occurs at eight hardpoints located at station 3519.72. The LTT/Centaur separation plane is on the LTT side of the interface adapter. The adapter remains attached to the Centaur after separation.

The overall length of the LTT is driven primarily by the propellant tank and optical configuration. The sunshade length is a function of telescope diameter and desired latitude location of the telescope. A dynamics analysis of the reference configuration should be performed to determine if there is sufficient clearance between the sunshade and shroud. Should this be a concern, the larger 23 m shroud could be utilized to increase clearance around the sunshade.

1-2613-1-59D

Lunar Transit Telescope Launch Configuration

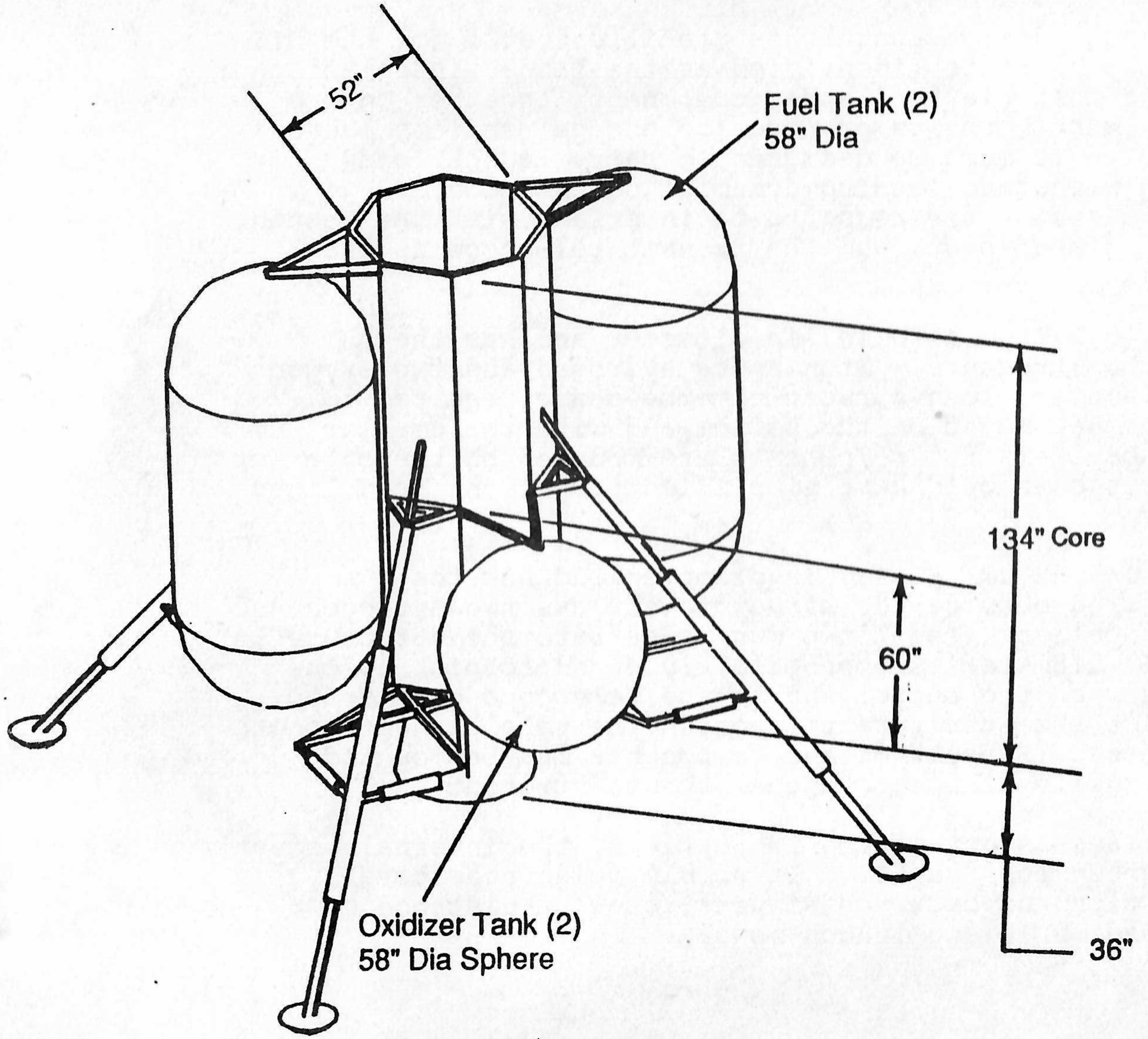


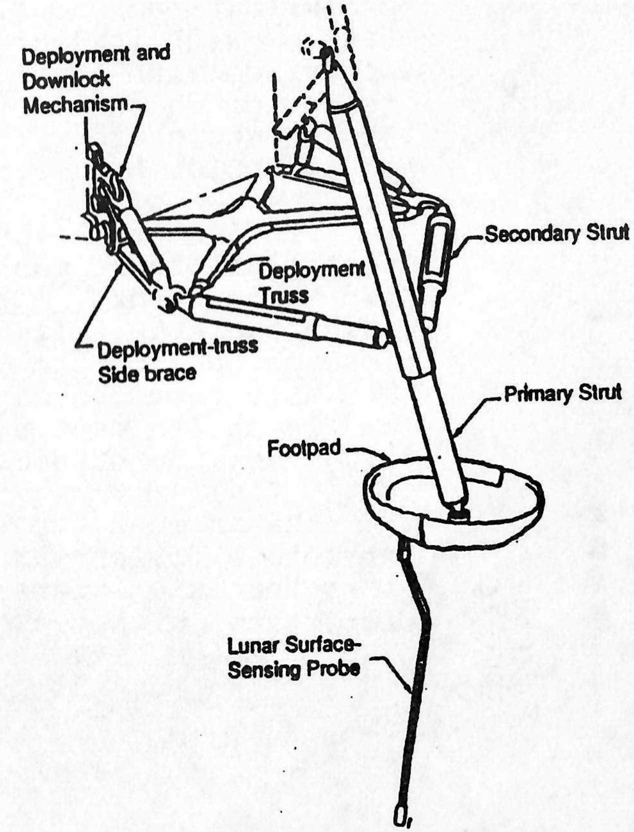
LTT LANDING GEAR

The landing gear configuration chosen for the LTT reference design is similar to the one used on the Apollo Lunar Module. The primary function of the landing gear is to dissipate touchdown energy in a controlled fashion that minimizes the landing shock for onboard components while maximizing ground clearance and the probability of landing stability over the range of possible surface and touchdown parameter variations.

The four legged landing gear requires retraction for stowage during launch due to the large tread radius needed for landing stability. Each of the four separate landing gear leg assemblies has energy absorption capability in the single primary and two secondary struts. The deployment truss serves as a structural-mechanical assembly between the landing gear struts and the stage structure. A pyrotechnic latching device retains the leg in the stowed position. After activating, the gear is deployed and locked into place. The footpad is designed to support the LTT on the lunar surface and should maintain functional capability after having impacted rocks or ledges during touchdown.

Lunar Transit Telescope Lander Configuration





Landing Gear

EXTERNAL VS. INTERNAL THRUST STRUCTURE OPTION

Two completely different design approaches are illustrated for the LTT thrust structure, which may be better identified as the lander integrating structure. This structure must tie the lander components together into a design that is compatible with both its payload (telescope) and its launch vehicle (Titan IV/Centaur). It must be designed to carry launch loads, thrust of the RL10 and to withstand landing dynamics on the moon. Additional structures, not shown, are required to interface with the Centaur payload adapter, the RL10 gimbal post, and the transit telescope as the payload.

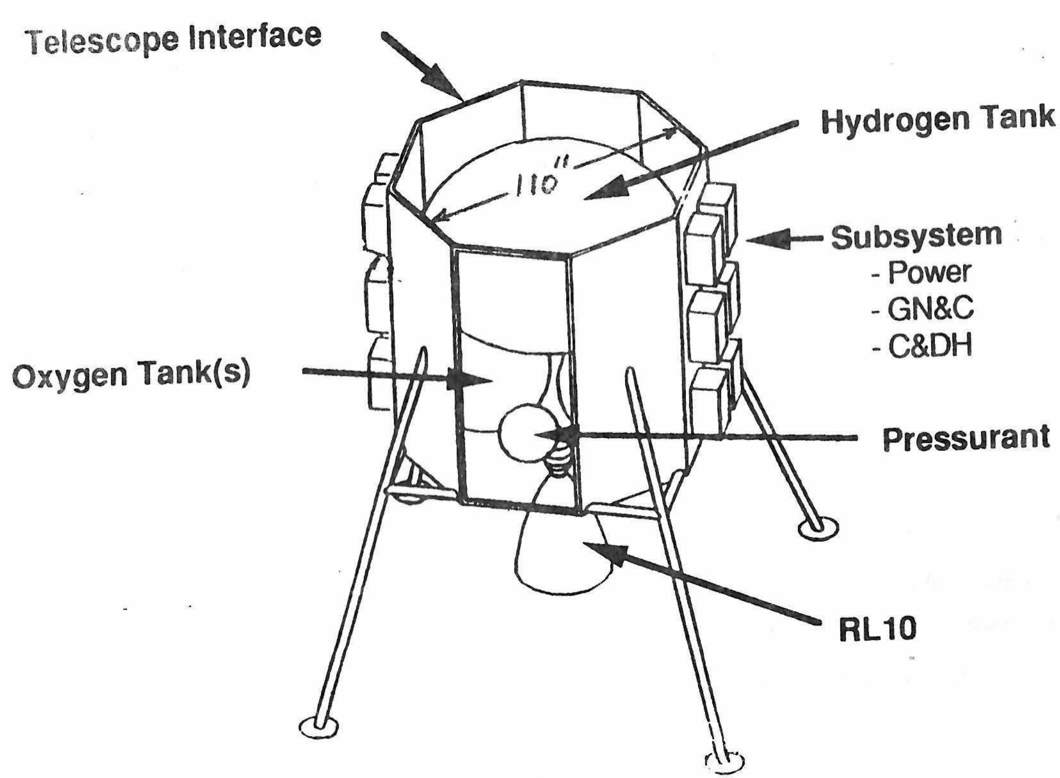
The external option is 279 cm (110 in) in diameter and has the propellant tanks inside the structure. It has one hydrogen and two oxygen tanks. With its large diameter, long structure members are required to interface with the RL10 gimbal mount on the bottom and with the smaller telescope on the top. Subsystems and components are mounted on the exterior of the structure. This option should have simplified propellant feed lines to the RL10.

The internal option is 132 cm (52 in) in diameter and has the propellant tanks cantilevered outside the structure. It has two hydrogen and two oxygen tanks. The propellant feed lines must penetrate the structure to reach the RL10. Its small diameter is appropriately sized to pick up the load paths from the Centaur on the bottom and to the telescope on the top. The RL10 is embedded inside the structure to shorten the total lander length and lower its center of mass. Subsystems and components may be mounted either on the outside or inside of the octagonal thrust structure.

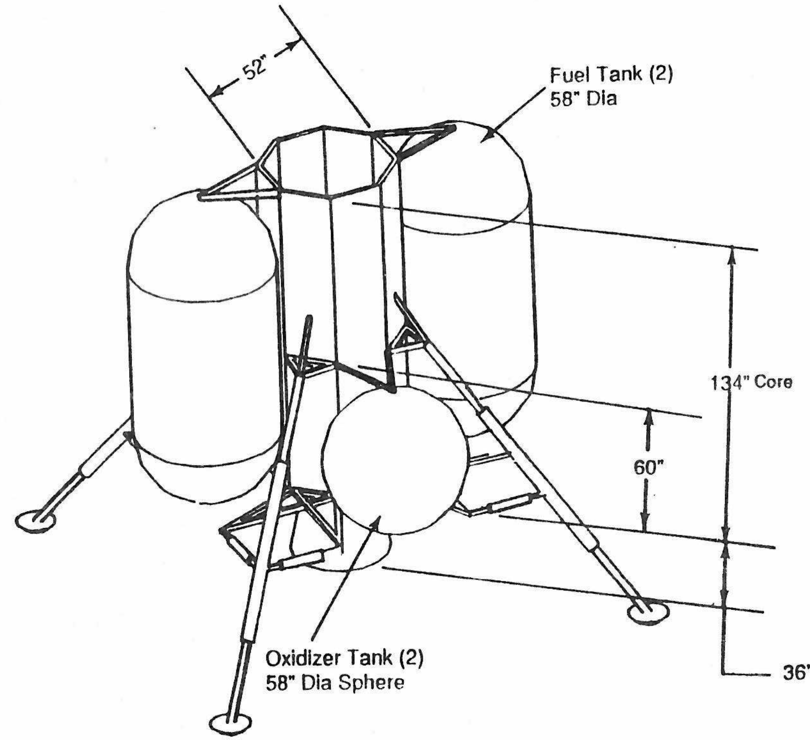
Based on a top level assessment of the two options, the internal thrust structure option was selected for the LTT. It should weigh considerably less than the external option and has much simpler, shorter distance interfaces with the payload, RL10 and launch module.

12/9/91:BGD

EXTERNAL VS INTERNAL THRUST STRUCTURE OPTION



EXTERNAL



INTERNAL

LTT TANK OPTIONS

The propellant tank configuration has a major influence not only on the LTT's propulsion and thermal control system, but also on its structure. This is primarily due to the relatively large mass of propellant (about equal to the inert mass of the lander/telescope) and the physical dimensions of the RL10 engine.

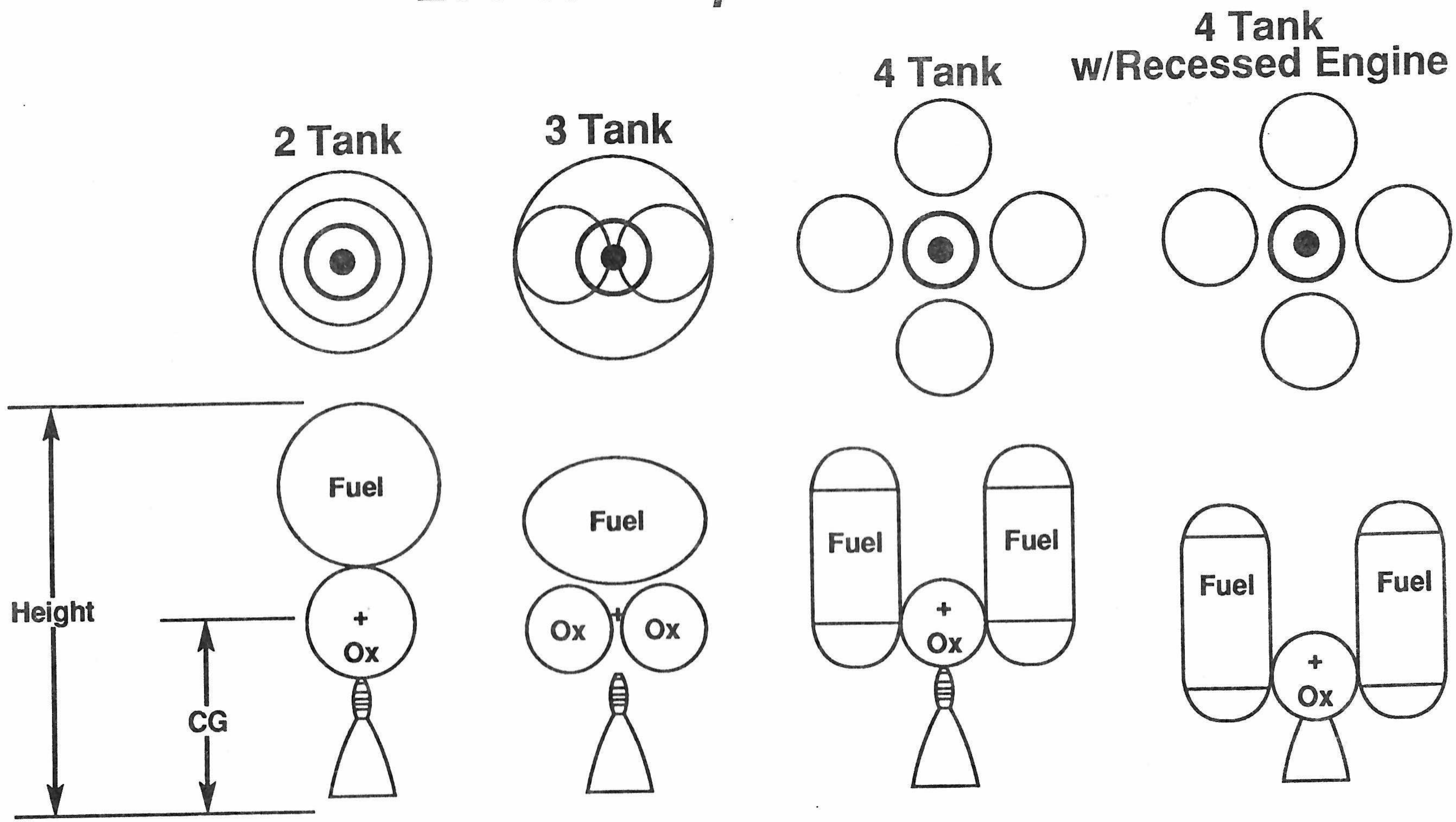
The design requirements for mounting a single large tank or a number of smaller ones typically have a significant influence on the structural design of the spacecraft. Usually the number of tanks and associated hardware should be minimized to reduce heat leakage to a cryogenic system.

The RL10 engine is approximately 178 cm (70 inches) in length. Unless the tank layout lends itself to embedding the engine, a relatively large system cg offset will occur. This can also be overcome with a multiple or toroidal tank configuration with a recessed engine but there are problems associated with their use.

The facing page depicts some of the tank configurations studied. Future work should focus on a toroidal tank option and the possible problems associated with multi-tank vs single tank options.

1-2622-1-61D

LTT Tank Options



	2 Tank	3 Tank	4 Tank	4 Tank w/Recessed Engine
Propellant 7069 (lbs)				
Mixture Ratio 5:1				
Height (in)	233.5	199.3	200.5	162
CG (in)	112.2	103.0	99.0	65.6
# of Tanks	2	3	4	4
Surface Area (Fuel) (in^2)	29740	30143	42679	42879
Surface Area (Oxidizer) (in^2)	13525	17077	17077	17077

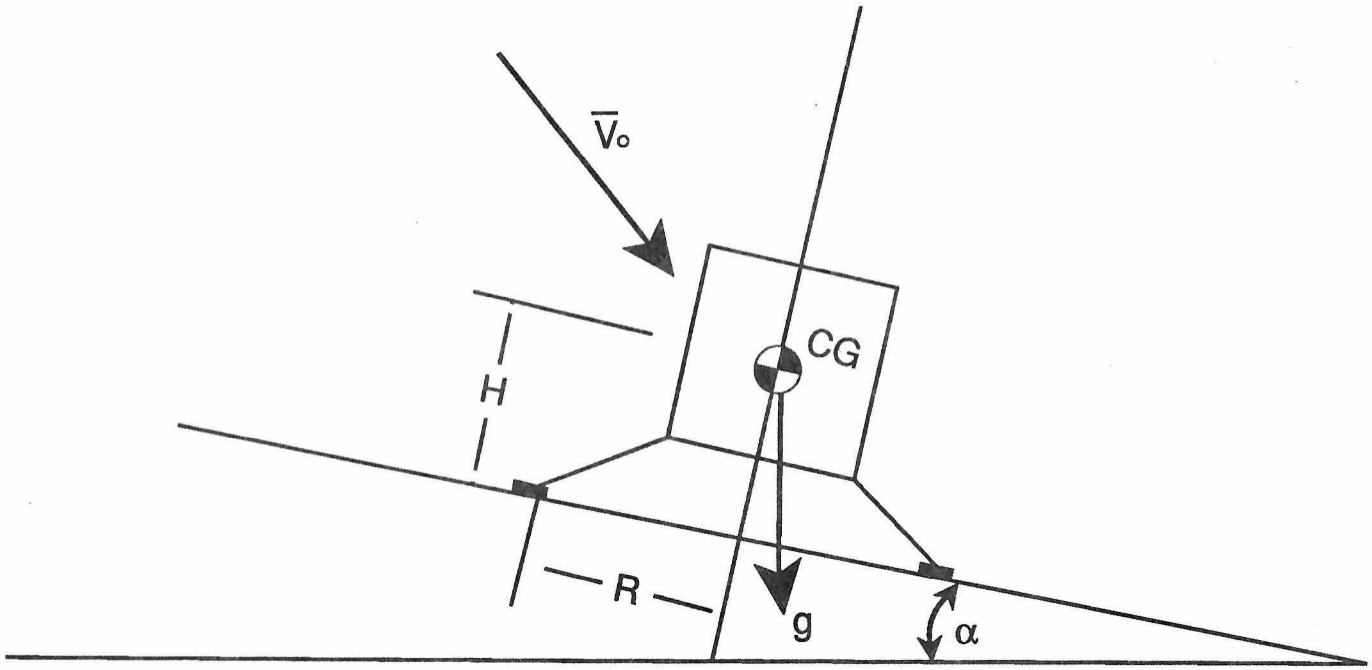
LTT SOFT LANDING ANALYSIS

The LTT should be designed to land on the moon in relatively unknown terrain. Even though the selected landing site may be level, errors can and probably will occur that causes the landing to be on a slope, or one leg on a boulder or in a small crater that produces a tilt of the LTT. For slope tolerance, the LTT cg should be low and its landing legs wide apart. In general, static instability occurs whenever the cg falls outside an imaginary line drawn from zenith that passes through the lowest lander foot.

The legs and landing gear must be robust enough to overcome the effects of initial impact, forces, and moments generated by the dynamics of landing including the unknown variations in the lunar terrain. Assuming an initial side velocity just prior to landing and that one foot impacts first, a roll can be introduced in the LTT that causes an East-West misalignment of the telescope, even though no roll error existed just prior to impact. Secondary bouncing is also a possibility that causes misalignment. In addition to overcoming initial conditions, landing dynamics and terrain uncertainties, the CCD detector array, telescope sun shade and Earth pointing antennaes require precise orientation after landing.

1-1889-1-37D

LTT Soft Landing Analysis



$\overline{V}_o$	Initial Velocity
α	Slope
g	Gravity
H	CG Height
R	Landing Gear Radius

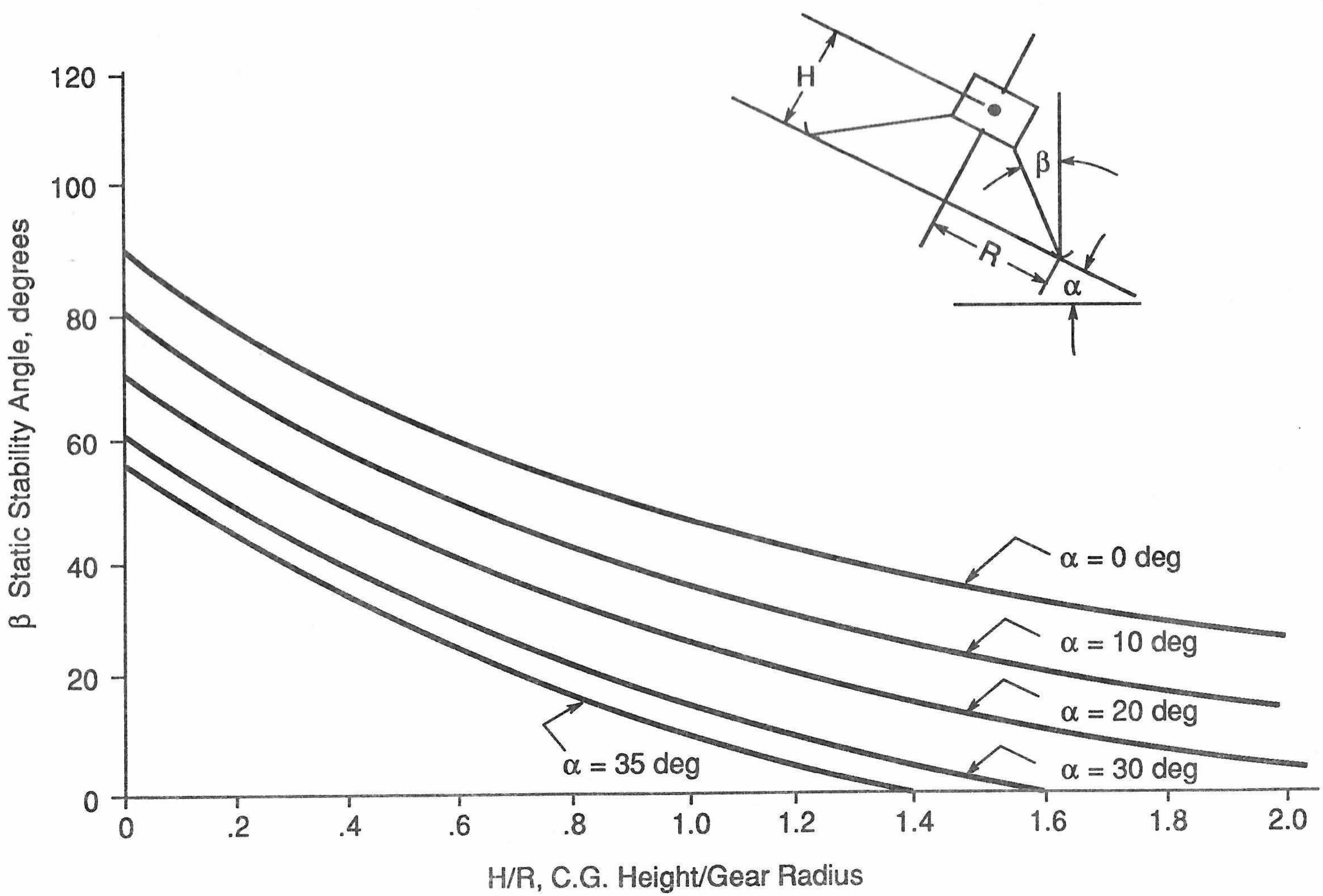
STATIC STABILITY OF LTT[1]

The static stability of the spacecraft was analyzed for surface slopes which varied from zero to 35 degrees. The variation of the static stability angle beta with center of gravity height to landing-gear radius ratio, H/R, is presented on the facing page. It is evident that as the H/R ratio increased the static stability angle, beta, decreases. At a surface slope of 35 degrees the static stability angle becomes zero for a value of H/R = 1.42. Therefore, to prevent the spacecraft from statically tipping over on a surface slope of 35 degrees an H/R of less than 1.42 must be incorporated into the final design of the LTT.

[1]Charles G. Richards, Thomas Perkins, et al

1-2501-1-56D

Variation of the Static Stability Angle with C.G. Height to Landing Gear Radius Ratio



LTT POINTING MECHANISMS

For most optical applications it is desired that the mounting surface be fixed in space and "vibration free". Neither of these two criteria can be absolutely achieved. Tilt to some degree is unavoidable considering the uncertainties in the landing dynamics and surface conditions at the chosen site.

Transient tilts can be induced by changing thermal conditions and permanent tilts may result from the telescope "settling" in the lunar soil. Since the detector is configured to a particular latitude, tilt is a critical factor and a compensation system must be provided for the LTT.

The need for an alignment/pointing system was established late in the preliminary design base. Further work in this area will be performed during follow-on studies. The facing page identifies and evaluates several candidate alignment options.

1-3124-1-32T

LTT Pointing Mechanisms

PASSIVE

- **Gravity Alignment**
  - Great for Pointing Toward Local Vertical, However no Compensation for Landing Latitude Errors
  - Need Focal Plane Array and Sunshade Alignment
  - Accuracy Function of Manufacturing Tolerances, Thermal Distortions, Friction, etc.

ACTIVE

- **3 Axis Pointing Mechanism**
  - Increased Length
  - Minimize Moving Mass
- **Hexapod**
  - Increased Length
  - Minimize Moving Mass
  - Limited Roll for Focal Plane Array and Sunshade Alignment
  - 6 DOF (only Require 3 DOF)
- **Leg Leveling**
  - Mechanism in Each Leg
  - Maximize Moving Mass
  - Need Focal Plane Array and Sunshade Alignment

LTT POINTING OPTIONS

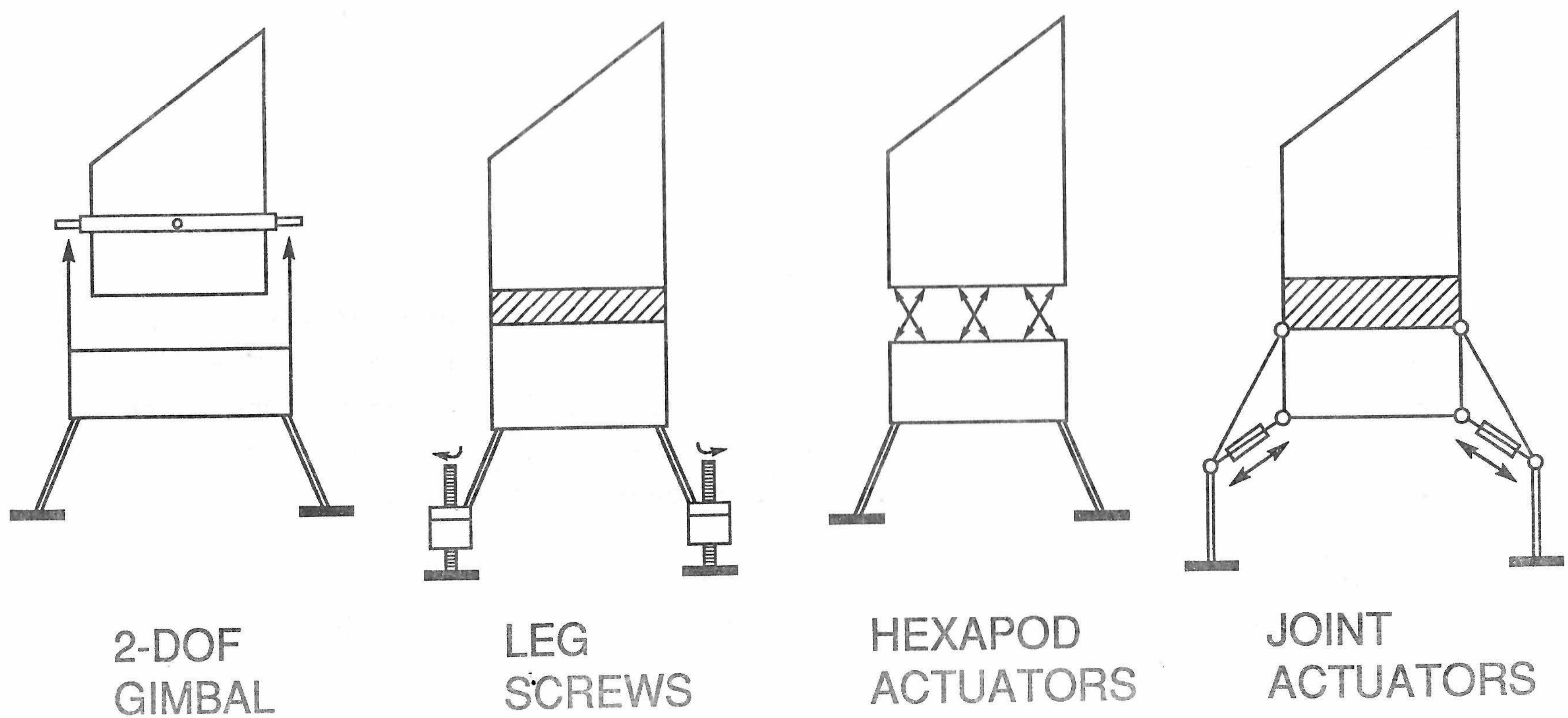
Late in the study (see section 5.3) it was determined that when viewed from mid-latitudes on the moon the stars make a curved track on the science instrument detector at the focal plane of the LTT. If the CCD array, and possibly the pixels of individual CCDs are customized for a specific latitude, then the rows of pixels will be curved for the star track at that latitude. If because of GN&C errors, landing dynamics, or unknowns in the selected landing site the LTT does not land with its telescope pointing to zenith, then its line of sight must be corrected by some type of mechanism. Some of these are illustrated on the facing page. At approximately 40 degrees latitude, the telescope line of sight must point to zenith with an accuracy of 5 arcminutes in its meridian plane and several degrees perpendicular to the plane. Therefore, the pointing or leveling mechanisms should correct for a few degrees error and be accurate to within about 1 arcminute.

Four pointing options are shown: a 2 DOF gimbal, screws on the lander legs, an arrangement of six linear actuators in a hexapod configuration, and linear joint actuators between each leg and the lander body. Both the 2 DOF gimbal and hexapod require a separate telescope that is moved relative to the lander base. Their range of movement can be many degrees, much more than is needed, using known mechanisms. It was decided that a large range of movement would not be necessary and that a separate telescope and lander joined together by mechanisms would be much heavier than an integrated LTT.

Both the leg screws and the joint actuators are capable of a few degrees of telescope motion and should be adequate for the LTT. Because of its simplicity and low weight, the leg screws are recommended for the LTT. The next chart shows the range of displacement required for both coarse and fine pointing. With a four legged lander, level sensing and leveling is done in two steps: 1) level across opposite diagonal legs and 2) then level across the diagonal of the two remaining legs. Note that the second leveling will not change the first, where as with a three legged lander the process is much more complex.

1-2658-1-61D

LTT Pointing Options

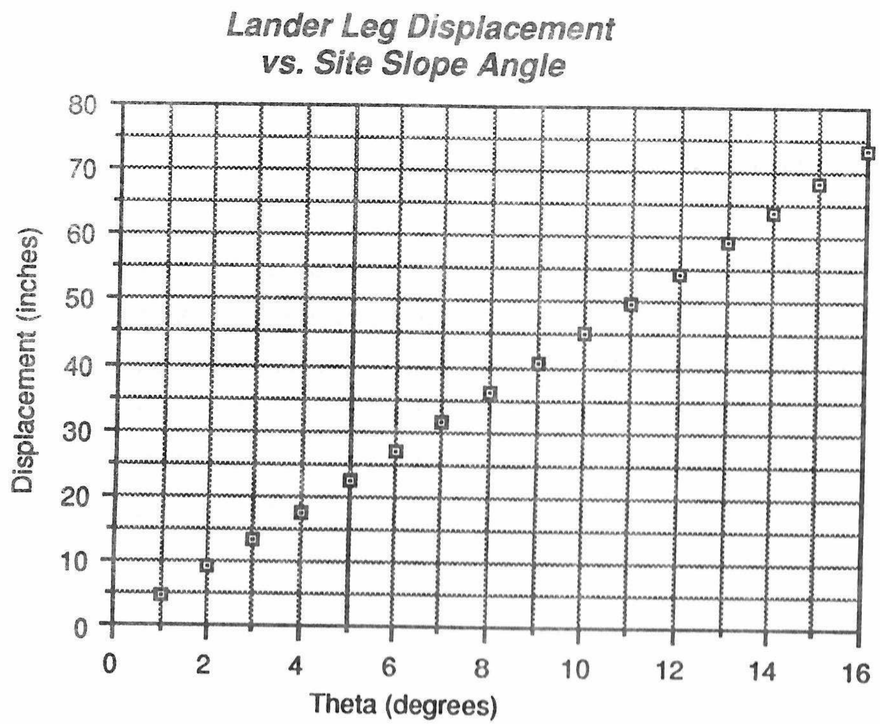


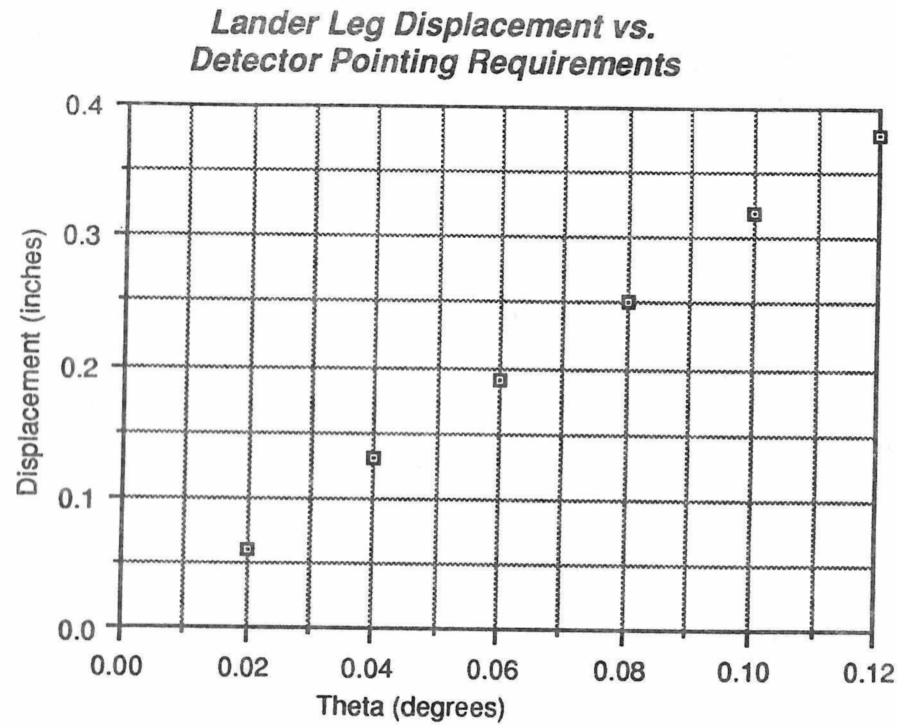
LANDER LEG DISPLACEMENT AND POINTING

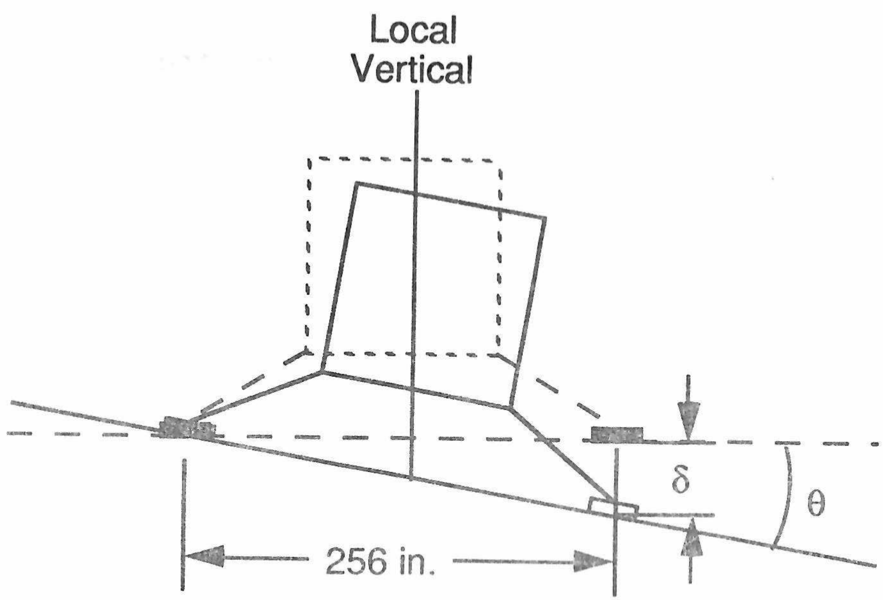
These two graphs show the lander leg displacement required to move the telescope line of sight relative to local vertical (zenith). One graph is for coarse pointing (degrees) to the zenith and the other is for fine pointing (arcmimutes), in each case the leg displacement is for the movement of only one leg. The first graph shows the required leg displacement to point the telescope to local vertical as a function of the site slope. The distance of 6.5 m (256 in) represents opposite diagonal legs. Even though the LTT landing site is expected to be level, landing dynamics and navigation errors could cause it to land, say, on a slope of up to 4 degrees. A leg screw displacement of about 46 cm (18 in) would be required for correction.

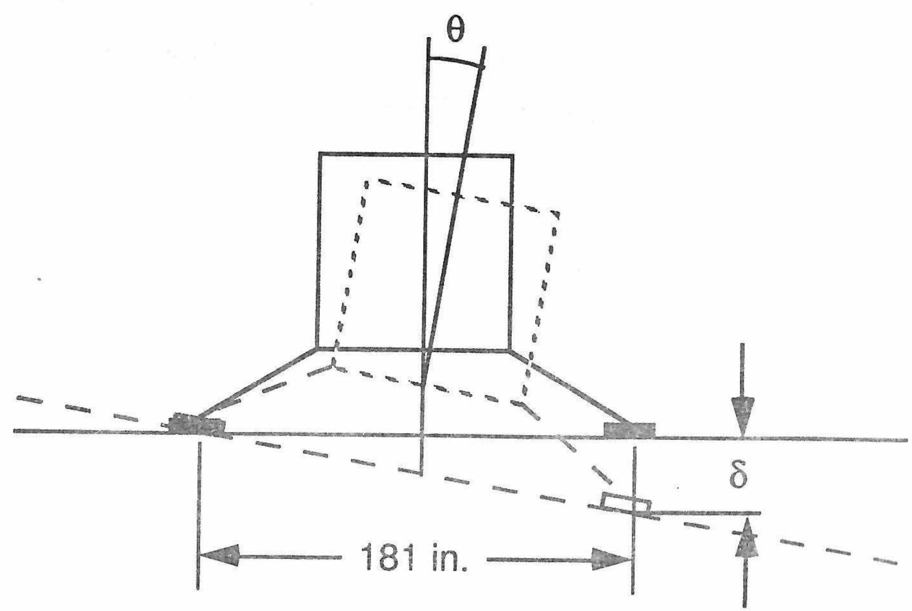
The second graph illustrates the pointing sensitivity of the telescope detector to differential length changes in the legs. The landing gear legs were assumed to contact the surface in pairs and the distance of 4.6 m (181 inches) represents the distance between each leg on the perimeter. The sensitivity for fine pointing needs to be in the arcminute range. The chart illustrates that 5 arcminutes (0.08 degrees) can be achieved by a leg adjustment of 0.64 cm (0.25 in) on the two lower or upper legs.

This orientation represents a "worst case" design scenario. The displacement assumes one leg is non-operational or represents the total displacement required.


Lander Leg Displacement
vs. Site Slope Angle
Displacement (inches)
0
10
20
30
40
50
60
70
80
0
2
4
6
8
10
12
14
16
Theta (degrees)


Lander Leg Displacement vs.
Detector Pointing Requirements
Displacement (inches)
0.0
0.1
0.2
0.3
0.4
0.00
0.02
0.04
0.06
0.08
0.10
0.12
Theta (degrees)


Local
Vertical
δ
θ
256 in.


θ
δ
181 in.

3.2 MASS AND INERTIA

LTT MASS STATEMENT

The LTT integrated system mass statement contains the masses for the telescope, the lander and the subsystems which support both the telescope and lander. The total LTT system dry mass includes a 20 percent engineering contingency. The performance capability of the Titan IV/Centaur to TLI is 6,364 kg (14,000 pounds). The LTT at this phase of the study is weight critical; therefore, included at the bottom of the mass statement is the delta between the launch vehicle performance and the LTT system launch mass.

The total mass at liftoff which is inserted into a translunar insertion orbit is 6,327 kg (13,913 lb). The propellant for midcourse correction and lunar landing is 3,115 kg (6,853 lb) plus 168 kg (368 lb) residuals or 52 percent of the mass. The payload which includes the inert lander and telescope is 3,042 kg (6,692 lb) or 48 percent.

LUNAR TRANSIT TELESCOPE

MASS STATEMENT 12/19/91	(LBS)	(KG)
OPTICS (2m prim.,90cm sec.,49cm tert.)	373	169
SCIENCE INSTRUMENT	165	75
SENSORS/ELECTRONIC MONITORING	110	50
MIRROR SUPPORT STRUCTURE	386	175
SUNSHADE/SHELL/APERTURE COVER	280	127
METERING STRUCTURE	66	30
POINTING/ALIGNMENT MECHANISMS	65	29
THERMAL CONTROL SYSTEM	459	208
ELECTRICAL POWER SYSTEM	436	198
INTEGRATION,ELECTRICAL	386	175
COMMUNICATIONS & DATA MANAGEMENT SYS.	200	91
GUIDANCE, NAVIGATION & CONTROL SYS.	257	117
PROPULSION SYSTEM	1077	488
REACTION CONTROL SYSTEM	100	45
CORE/THRUST STRUCTURES	331	150
TANK SUPPORT,MM SHEILDING,& MISC.MECH.	154	70
LANDING GEAR (INCL.DEPLOYMENT MECH.)	302	137
CENTAUR INTERFACE STRUCTURE	331	150
CONTINGENCY (20%)	1096	497
TOTAL DRY MASS	**6573**	**2981**
PROPELLANT (USABLE MAIN & RCS)	6853	3108
RESIDUALS,BOILOFF,PRESSURANT	368	167
TOTAL LTT SYSTEM MASS	**13794**	**6256**
CENTAUR STRUCTURAL LIMIT	**14000**	**6349**
PAYLOAD AVAILABLE	**206**	**93**

LTT LAUNCH CONFIGURATION (MASS PROPERTIES)

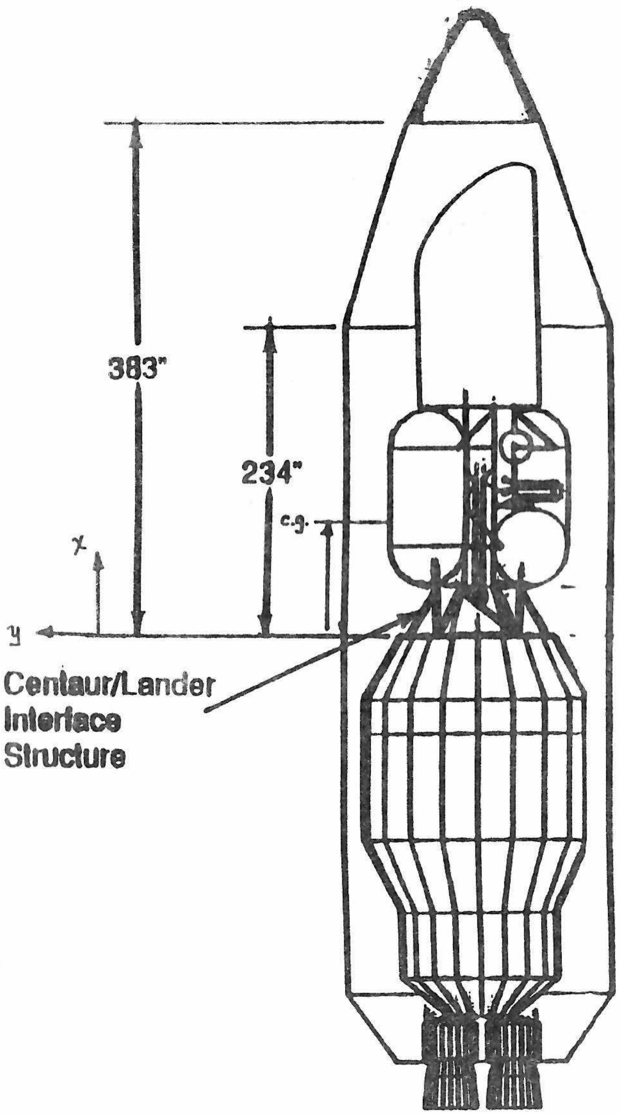
The mass properties for the LTT in the stowed configuration were generated for analysis of the launch and orbital conditions. Analysis of the launch conditions for the integrated LTT system was critical due to the structural limit of the Centaur at liftoff.

The mass properties are referenced from the right handed coordinate system shown to the left of the configuration with the system origin located at the center of the Titan IV/Centaur interface plane. The cg location is critical both at launch and during lunar landing. At launch the Titan IV/Centaur performance is related to the cg distance above the Centaur interface structure. During lunar landing a low cg produces greater landing stability and permits landing on a greater slope.

The longitudinal cg for the LTT at launch is 2.66 m (8.74 feet) above the interface plane. This low cg location is well within the limits for the Titan IV/Centaur, see section 5.1, and means that it can launch more than the critical mass of 6,364 kg (14,000 pounds). The LTT may not be as weight critical as it first appears, based on shroud length, payload center of mass, and both Centaur and Stage II structural limits.

The mass includes an engineering contingency of 20 percent as applied to all system dry masses.

LUNAR TRANSIT TELESCOPE LAUNCH CONFIGURATION



LUNAR TRANSIT TELESCOPE MASS PROPERTIES AT LAUNCH 11/21/91

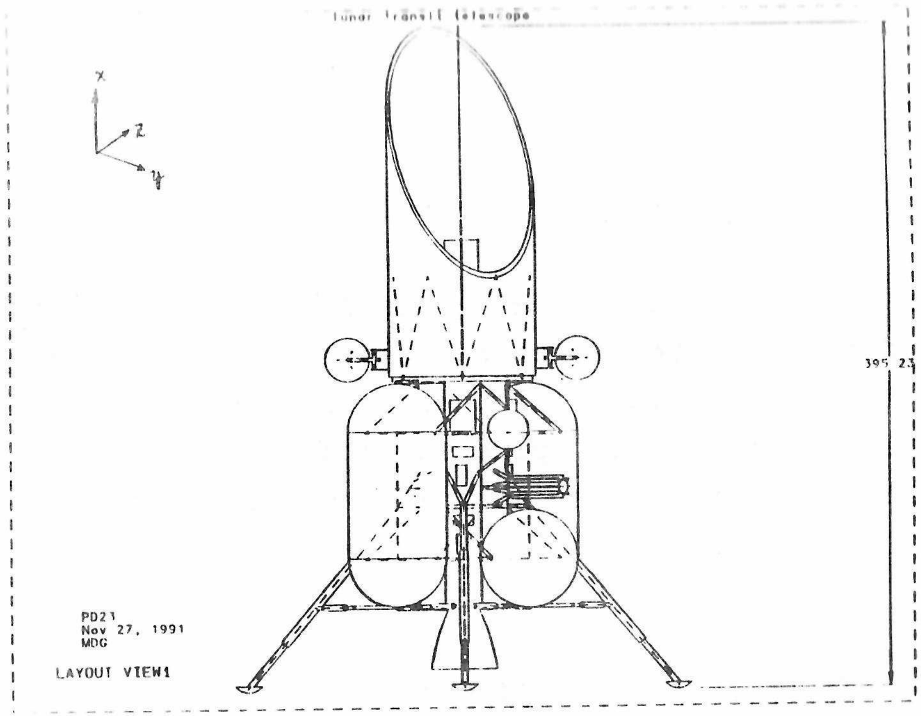
DESCRIPTION	LAUNCH WEIGHT (LBS)	XCG (FT)	YCG (FT)	ZCG (FT)	IXX SLUGS FT^2	IYY SLUGS FT^2	IZZ SLUGS FT^2
					INERTIAS REFERENCED FROM INDIVIDUAL ITEM CG		
1.PRIMARY/SUPPORT/T	709	15.50	0.00	0.00	127	64	64
2.SECONDARY MIRROR	50	22.00	0.00	0.00	7	4	4
3.SI/MONITOR/RADIAT	285	22.50	0.00	0.00	7	7	5
4.SUNSHADE/SHELL/TC	265	22.70	0.00	0.00	95	182	182
5.METERING STRUCTUR	66	18.70	0.00	0.00	24	20	20
6.CORE SYSTEMS	2269	9.50	0.00	0.00	171	823	823
7.LOX TANK	76	6.50	4.80	0.00	9	9	9
8.LOX TANK	76	6.50	-4.80	0.00	9	9	9
9.LH2 TANK	196	9.70	0.00	4.80	34	78	78
10.LH2 TANK	196	9.70	0.00	-4.80	34	78	78
11.RL10 ENGINE & HS	682	5.00	0.00	0.00	31	74	74
12.LANDING GEAR	302	7.00	0.00	0.00	191	151	151
13.CENTAUR INTERFAC	331	2.00	0.00	0.00	403	223	223
14.APERTURE COVER	84	25.50	0.00	0.00	16	8	8
15.BLANK	0	0.00	0.00	0.00	0	0	0
16.LOX	2969	6.20	0.00	0.00	195	195	195
17.LOX	2969	6.20	0.00	0.00	195	195	195
18.LH2	594	9.40	0.00	4.80	49	211	211
19.LH2	594	9.40	0.00	-4.80	49	211	211
20.RCS PROPELLANT	85	13.50	0.00	0.00	1	1	1
* CONTINGENCY	1115						
TOTALS	13913	8.74	0.00	0.00	3195	12982	1192[illegible]

* 20% CONTIN THRU ITEM #15 THEN 0 % CONT THRU ITEM # 20

LTT MASS PROPERTIES ON THE LUNAR SURFACE

The cg and inertias of the LTT integrated system were developed for analysis of the vehicle dynamics at landing, and for the telescope pointing analysis. The right handed coordinate system is the same as the reference for launch. The origin is located at the center of the plane defined by the deployed landing legs with the x-axis along the centerline of the telescope. An engineering contingency of 20 percent is included in these calculations.

The total dry weight on the lunar surface is 2,867 kg (6,307 lb), 45.3 percent of the launch weight. Note that with empty tanks that the cg has increased to 3.47 m (11.37 ft) above the landing gear feet. However, the landing gear leg spread is 6.51 m (21.4 ft). Based upon a static stability analysis the LTT could land on a slope of up to 35 degrees without tipping over.



LUNAR TRANSIT TELESCOPE MASS PROPERTIES ON LUNAR SURFACE 11/26/91

DESCRIPTION	LAUNCH WEIGHT (LBS)	XCG (FT)	YCG (FT)	ZCG (FT)	IXX SLUGS FT^2	IYY SLUGS FT^2	IZZ SLUGS FT^2
					INERTIAS REFERENCED FROM INDIVIDUAL ITEM CG		
1.PRIMARY/SUPPORT/T	709	15.50	0.00	0.00	127	64	64
2.SECONDARY MIRROR	50	22.00	0.00	0.00	7	4	4
3.SI/MONITOR/RADIAT	285	22.50	0.00	0.00	7	7	5
4.SUNSHADE/SHELL/TC	265	22.70	0.00	0.00	95	182	182
5.METERING STRUCTUR	66	18.70	0.00	0.00	24	20	20
6.CORE SYSTEMS	2269	9.50	0.00	0.00	171	823	823
7.LOX TANK	76	6.50	4.80	0.00	9	9	9
8.LOX TANK	76	6.50	-4.80	0.00	9	9	9
9.LH2 TANK	196	9.70	0.00	4.80	34	78	78
10.LH2 TANK	196	9.70	0.00	-4.80	34	78	78
11.RL10 ENGINE & HS	682	5.00	0.00	0.00	31	74	74
12.LANDING GEAR	302	7.00	0.00	0.00	191	151	151
13.BLANK	0	0.00	0.00	0.00	0	0	0
14.APERTURE COVER	84	25.50	0.00	0.00	16	8	8
15.BLANK	0	0.00	0.00	0.00	0	0	0
* CONTINGENCY	1051						
TOTALS @ SURFACE	6307	11.37	0.00	0.00	1372	7867	7659

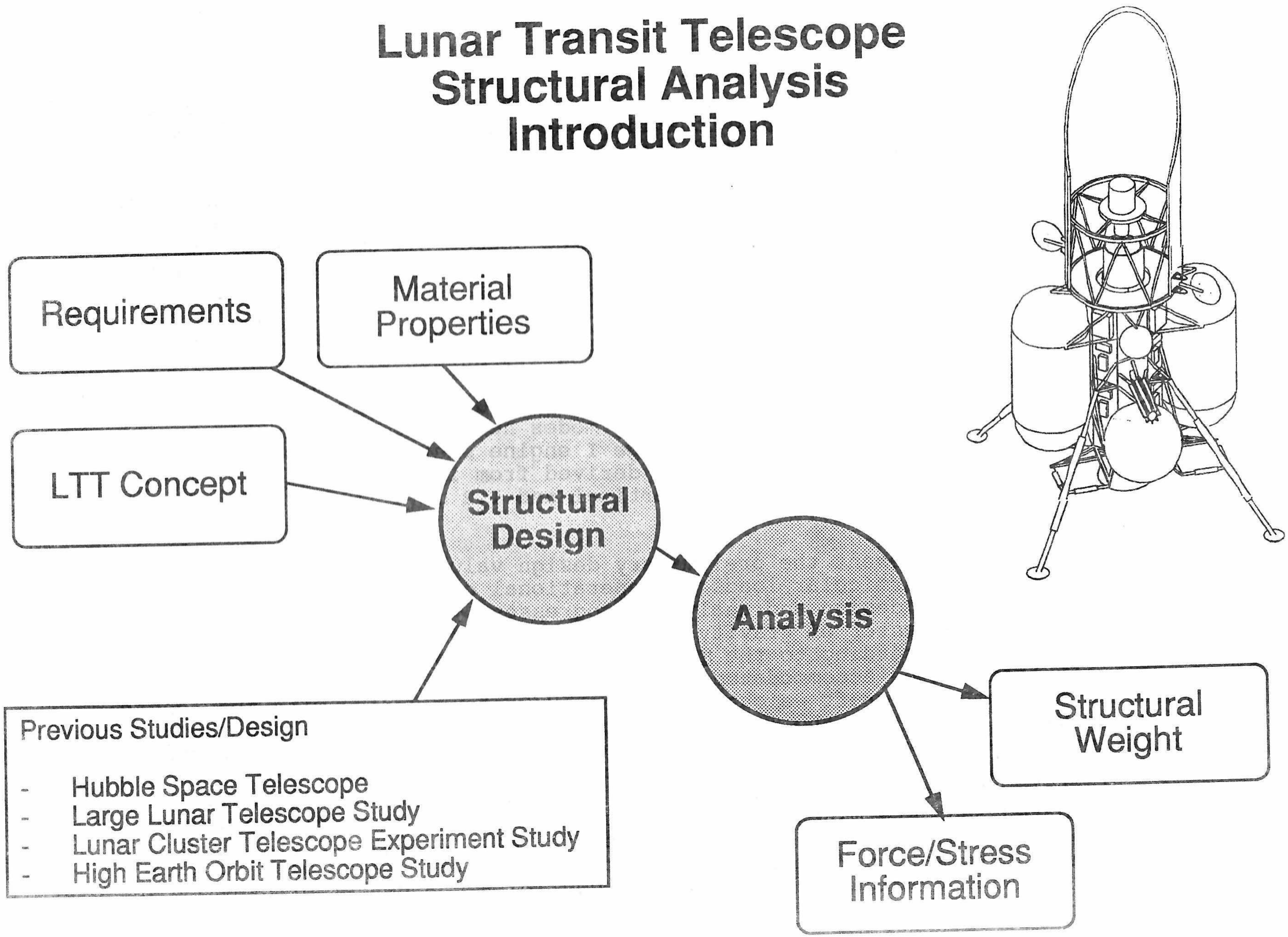
3.3.STRUCTURAL ANALYSIS

INTRODUCTION

The goals of the structural analysis for the Lunar Transit Telescope (LTT) were to determine the structural requirements imposed by launch and landing, select candidate materials, and define concepts. Previous telescope studies, as well as the Hubble Space Telescope, have influenced the design concepts for LTT. Material from sources outside of NASA were invaluable for the selection of alternative configurations for the primary mirror support structure.

Structural analysis of the LTT design consisted of primarily analysis methods to determine rough component sizes and masses. In some instances, spreadsheets were developed in Microsoft Excel on the Macintosh IIfx computer to aid with more complex sets of equations and variables. All of this analysis, plus engineering judgement, contributed to the mass estimates that can be found in the system mass statement.

Lunar Transit Telescope
Structural Analysis
Introduction



STRUCTURAL REQUIREMENTS

Three major categories of structural requirements have been defined for the LTT: load factors for the Titan IV launch vehicle during various phases of flight, frequency constraints to avoid coupling of the LTT with the launch vehicle, and loads that are expected for landing on the lunar surface.

The maximum compressive load defined for the Titan IV is 6.5 g (axial) at Stage I burnout. The maximum lateral load occurs at lift-off with a value of 3.5 g. Lateral loads at the extremities of the spacecraft may exceed the load factors defined at the center of gravity. The Titan IV Users Handbook defines a relationship to establish lateral loads as a function of the longitudinal distance from the spacecraft / launch vehicle interface. In addition to the launch loads, the LTT must maintain the frequency constraints that are defined for the Titan IV vehicle. Lateral modes below 2.5 Hz can affect launch vehicle control system performance. Modes between 6 and 10 Hz will tune with lift-off overpressure excitation. High air loads gust response has been observed for spacecraft with first bending modes near 4 Hz. Axial frequencies between 17 and 24 Hz will respond to Stage I engine shutdown thrust pulsation. The lunar landing loads are derived from initial conditions just prior to touch down: 3 m/s (10 ft/s) vertical and 1.2 m (4 ft) per second horizontal per Apollo data.

These loads represent the preliminary design values for use in the analysis. Mission particular loads and operational deflection requirements must be added as the design and analysis progresses. Centaur structural capability and the center of gravity constraints also weigh heavily in the analysis of LTT. According to the Titan IV Users Handbook, the maximum spacecraft that can be accommodated is 5,215 kg (11,500 lb). Data indicates that the maximum spacecraft weight for Titan IV/Centaur varies with the length of the payload faring and the location of the center of gravity above the 8 point interface. However, a Titan IV/Centaur payload that is currently under development weighs 6,440 kg (14,200 lb) with a center of gravity of approximately 100 inches above the 8 point interface. This configuration has been evaluated and determined to have positive margins. Proposals to upgrade the Centaur to a 6,803 kg (15,000 lb) capability with a high center of gravity have been made to Martin Marietta, manufacturer of Titan. According to data on hand, LTT with a weight of 6,349 kg (14,000 lb) and a center of gravity of 254 cm (100 in) or less should be able to use the Titan IV/Centaur as defined today.

Lunar Transit Telescope
Structural Requirements

Load Factors for Titan IV (Titan IV User's Guide, June 1987)

Event	Axis	Max/Min Values (g's)
Lift-off	Axial	3.2, 1.2
	Lateral	± 3.5
	Torsion	± 0.05/in
Max Air Loads	Axial	3.3 C / 0.9 C
	Lateral	± 3.0
	Torsion	± 0.05/in
Stage I Burnout	Axial	2.5 T, 6.5 C
	Lateral	± 1.5
Stage II Burnout	Axial	2.5 T, 5.5 C
	Lateral	± 1.5

C = Compression, T = Tension

Frequency Requirements (Titan IV User's Guide, June 1987)

- Maintain first lateral mode above 2.5 Hz
- Avoid lateral modes between 6 and 10 Hz, first bending mode near 4 Hz, axial frequencies between 17 and 24 Hz

Lunar Landing Loads (from 7/25/91 working group meeting)

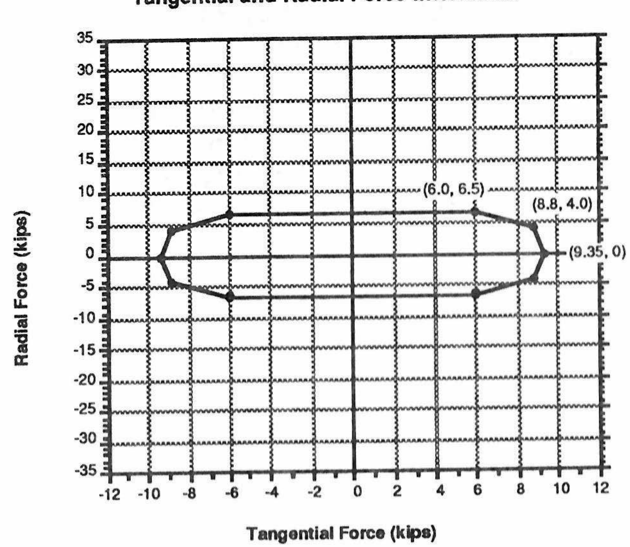
- 10 ft/sec vertical, 4 ft/sec lateral (Apollo)

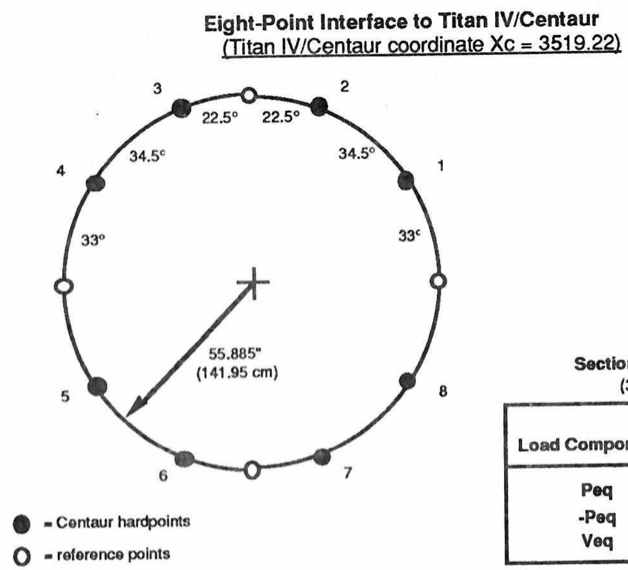
CAPABILITY OF 8-POINT INTERFACE TO CENTAUR

Centaur has two payload interface rings, an eight-point interface on top of the Centaur with a 142 cm radius and a twenty-two point interface on the outer perimeter near the bearing reactors.

The Centaur eight-point interface is the likely attachment for the LTT to the upper stage. These hardpoints are limited in their load carrying capability. The figures show the interactions between the tangential, radial, and axial force components for the interface. Structural analysis for LTT was not of sufficient detail to establish the reactions at the hardpoints. Further analysis will determine the values for LTT, and those values will then be plotted against the allowables for design evaluation.

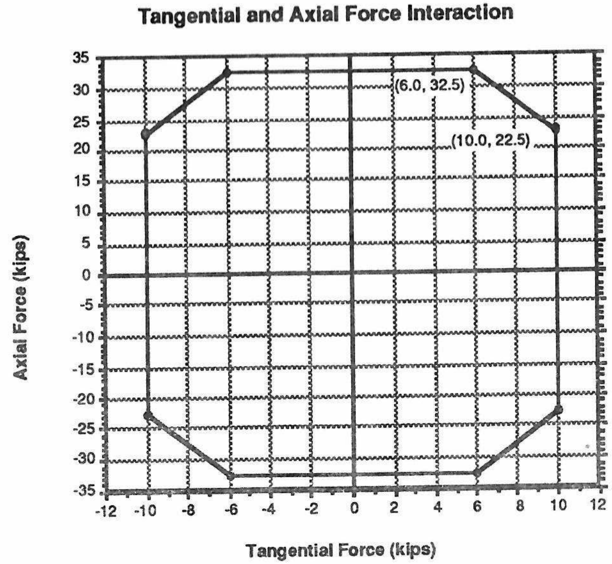
Lunar Transit Telescope
Capability of 8-Point Interface to Centaur

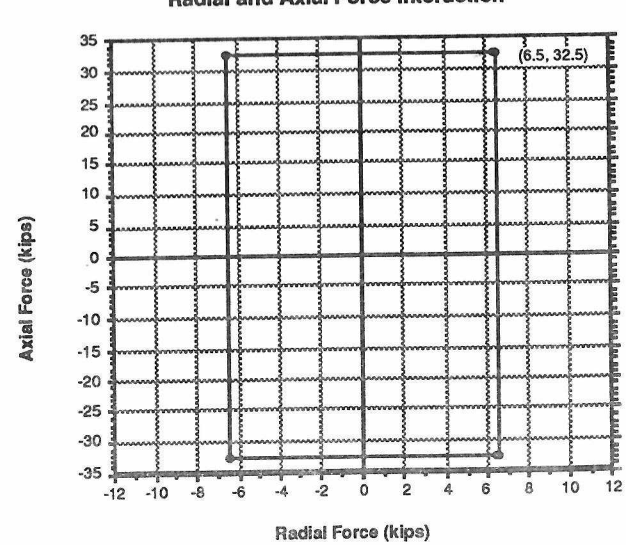




Section Load Comparison
(3519.22 Interface)

Load Component	Test (kips)	PDLC (kips)
Peq	239.0	202.0
-Peq	205.2	180.6
Veq	44.3	36.4





REFERENCE: "Titan IV Centaur: Program Overview", Martin Marietta, September 15, 1989

MATERIALS SELECTION

As a ground rule, technology and advanced development items were to be limited for LTT. Materials selection was no exception. Materials for the telescope structure are required to have low density, low thermal expansion over a wide temperature range, adequate strength, low to nonexistent off-gassing characteristics when exposed to the vacuum of space, and to be available and easily produced in the desired form. Metal matrix composites are currently under development and would have been an excellent choice for the telescope structure. However, these materials are still in the early stages of development and would not be fully characterized when required by the program.

More traditional materials were selected for the telescope structure. None of the materials that have been tentatively selected- Aluminum, Beryllium, Graphite/Epoxy, or Invar- possess all of the desired characteristics for the metering structure or related structure on the telescope. It is possible to design an athermal structure for the metering structure from a combination of Invar and Aluminum. This can also be accomplished with Graphite/Epoxy with metal joints, perhaps Aluminum or Titanium. One drawback of using Invar in space applications is the high density of the material, about four times that of Aluminum. For the LTT, a portion of the structure is designated as being fabricated from Invar, but these items are of low volume and therefore not a threat to the overall mass requirements.

Off-gassing has been identified as a possible hazard to the telescope optical system. The use of metallic components should preclude any difficulties, but Graphite/Epoxy or other composite materials may require special attention, such as shielding of the optics from the structure.

Lunar Transit Telescope
Materials Selection

Room Temperature Properties of Selected Materials

Material	Type	Density (kg/m^3)	Fx comp. strength (MPa)	E c (GPa)	Max. Use Temp. (°K)
Aluminum	2219-T81	2823.4	330.9	74.46	≈450 °
Beryllium	I-250 (structural grade)	1850	≈ 544.7	303	1558° *
Beryllium	O-50 (optical grade)	1850	≈ 172	303	1558° *
Graphite/Epoxy	GY70/934, ±12°	1688.5	498.5	268.2	400 °
Invar	36	8080	≈ 248.2	≈ 144	477.6 °

* melting point.

Coefficient of Thermal Expansion (ppm/°K) of Selected Materials

Material	Type	125 K (-234.7 °F)	220 K (-63.7 °F)	294.3 K (70 °F)	366.5 K (200 °F)
Aluminum	2219-T81			22.14	22.86
Beryllium	I-250 (structural grade)			11.3	
Beryllium	O-50 (optical grade)			11.4	
Graphite/Epoxy	GY70/934, ±12°			-0.54	
Invar	36	0.001772	0.001772	0.001772	

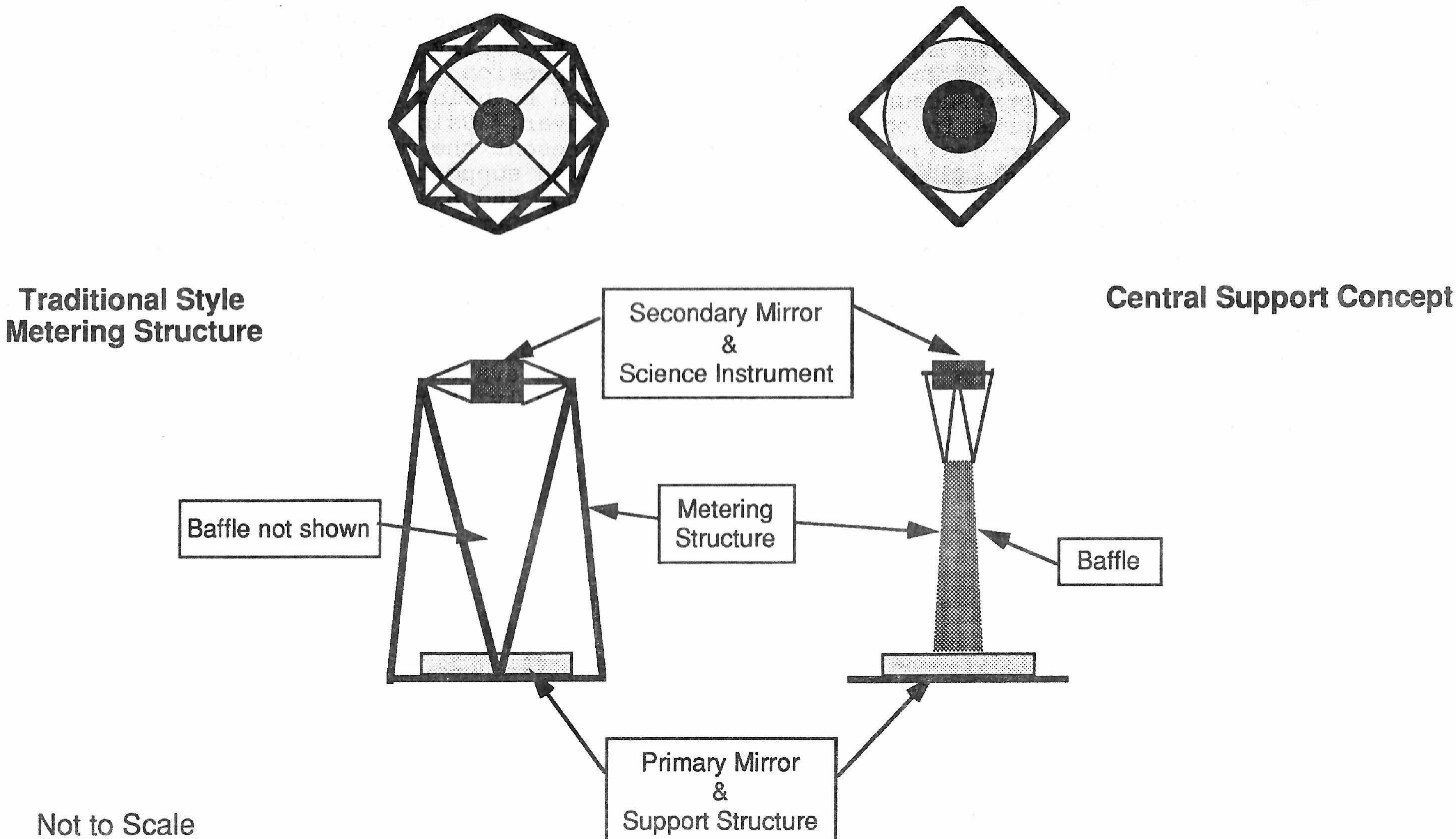
METERING STRUCTURE DESIGN OPTIONS

Several design options were examined for the LTT metering structure. One of the chief design drivers was to minimize blockage of the light path through the optical train. Most modern astronomical telescopes have a metering structure consisting of tubular truss members with tension rods and turnbuckles to support the secondary mirror. For earth-based telescopes, mass is generally not a driver, except for the pointing system. Being a moon based telescope, the LTT must be carefully designed to minimize mass while maximizing performance. A structure similar to that used in the Apache Point Observatory, near Alamogordo, New Mexico, would have a mass of about 160 kg using reasonable member sizes.

Since the bottom of the sunshade opening is below the level of the secondary mirror, a portion of the structure could be illuminated by the sun, while the remainder is in the shade, causing a thermal gradient across the structure. Although it would be impossible to prevent any thermal gradient in the structure, it can be minimized by a centrally mounted support. This concept utilizes a baffle that is designed as a structural support for the secondary mirror and instrumentation. According to the optical baffling analysis, the design could be a cylinder or a frustum as long as the height does not exceed the baffle point. Another desire, from a baffling standpoint, is to have the thickness of the material at the baffle point as small as possible.

Other designs were considered such as an athermalized structure similar to the Charged Couple Device Transit Instrument (CTI) on Kitt Peak, a Graphite/Epoxy truss and ring frame design like the Hubble Space Telescope, and a tetrapod design with members that extend from the outer diameter of the primary mirror directly to the outer diameter of the secondary mirror.

Lunar Transit Telescope
Metering Structure Design Options



Not to Scale
Schematic Representation of Systems

METERING STRUCTURE CONFIGURATION

The central support option was chosen for its simplicity and the advantages mentioned earlier. For the analysis, the structure was considered to be a frustum of the dimensions shown in the figure, with high aspect ratio members to support the secondary mirror and instrumentation on the frustum. For analysis, the entire structure was considered to be fabricated from Invar 36. Invar is a material that is commonly used in thermally stable structures because of the near zero coefficient of thermal expansion. In a detailed design, the metering structure will be athermalized by using carefully designed aluminum parts in conjunction with the Invar. Using a safety factor of 1.4 and an assumed design load of 50 kg to represent the mass of the secondary mirror, instrument package, and associated support structure, the frustum was analyzed for elastic stability, bending, and axial compression using equations from Roark's Formulas for Stress and Strain. A wall thickness of 1 mm was assumed to begin the analysis and was found to be adequate for the conditions examined. Member sizes for the connection of the frustum and secondary mirror support were assumed to be 5 mm x 20 mm. An evaluation of the normal modes of the structure was not performed for this study. Detailed analysis of the entire telescope will be necessary to determine if there are difficulties meeting the frequency requirements. An evaluation of the thermal expansion and thermal stress will be necessary to fully characterize the structure.

Lunar Transit Telescope
Metering Structure Configuration

Metering Structure Analysis

Considered Titan and Centaur Loads

Structure of Invar 36

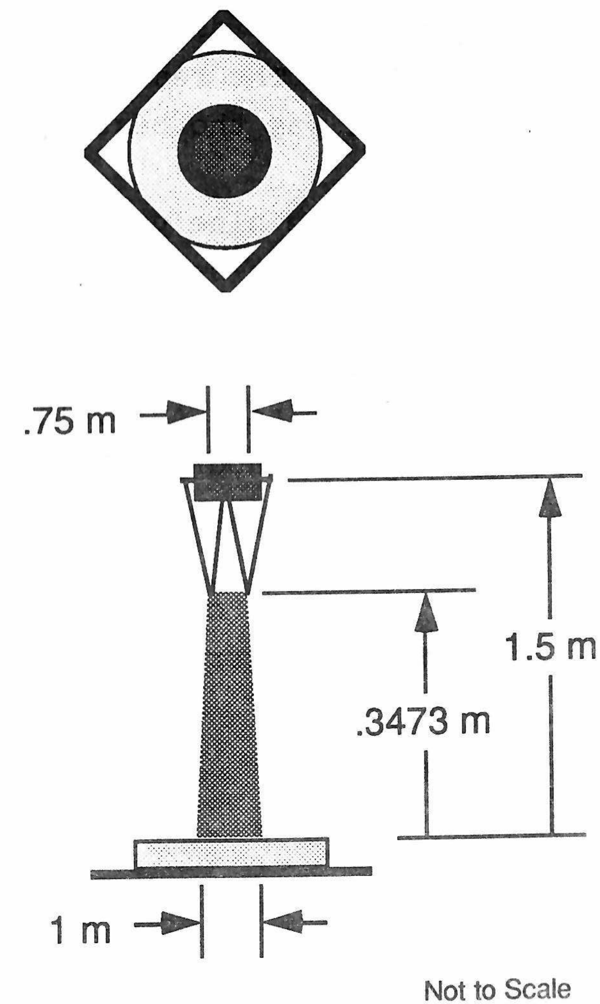
Considered thin shell frustum with high aspect ratio members to connect to secondary mirror

Safety factor of 1.4

Loads for analysis of 50 kg

Performed simple buckling, bending, and axial stress analysis using equations from Roark's <u>Formulas for Stress and Strain</u>

Metering Structure Mass = <u>30 kg</u>



12/13/91:SS

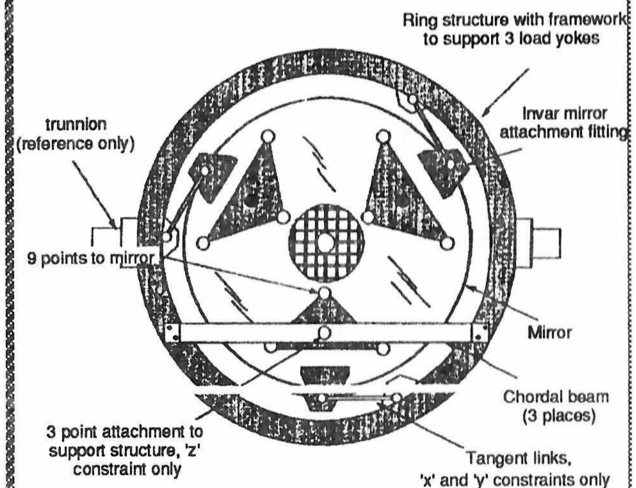
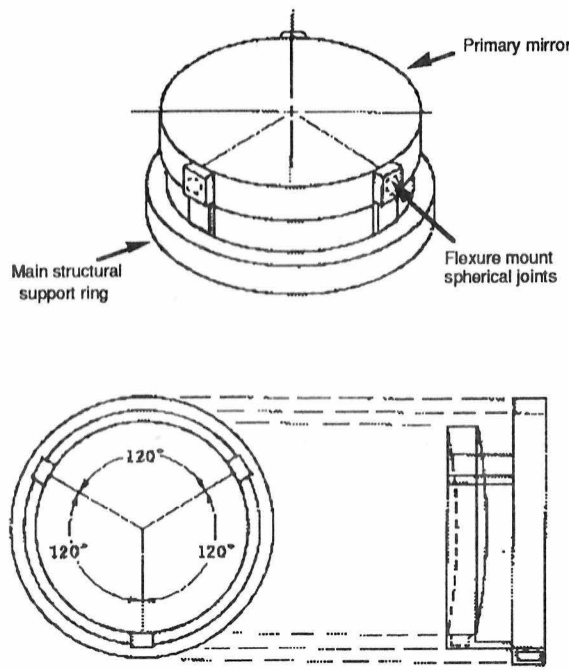
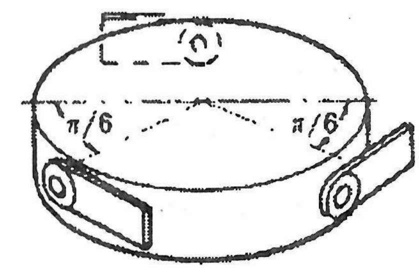
PRIMARY MIRROR SUPPORT STRUCTURE OPTIONS

Several options have been discussed for the primary mirror support structure. The tangential band mount and the axial bar mount concepts were two of the designs that were considered in the Photoheliograph Definition Study. The results of that study determined that the three point axial bar mount was the best choice for that instrument.

The nine point mounting concept is an embellishment of the three point mount. It seems that deflections in the mirror are too great with the mounting on the perimeter. The situation is improved by moving the mounting location inward. Doubling the number of supports also decreased the deflection. An evaluation of the nine point, or Hindle, mount by Mike Krim of Hughes Danbury Optical Systems indicates that this approach is worthy of consideration for use with structurally efficient mirrors.

These concepts must be assessed in detail for applicability to the LTT primary mirror.

Lunar Transit Telescope
Primary Mirror Support Structure Options

9-Point flat mounting concept	3-Point equally spaced, Axial bar mount	3-Point equally spaced, Tangential band mount
		
REFERENCE: Krim, Mike, "Siderostat Mirror Assembly: Preliminary Design Concept Description", Hughes Danbury Optical Systems, Danbury, CT, November 1989.	**REFERENCE:** *Photoheliograph Definition Study* , Volume 2, Book 1, "150-Centimeter Photoheliograph for LSO Mission", Prepared for NASA/George C. Marshall Space Flight Center, NASA contract NAS8-28147, Final Report, ITEK: Optical Systems Division, Itek 72-8212-3, January 8, 1973, p 5.2.	**REFERENCE:** *Photoheliograph Definition Study* , Volume 2, Book 1, "150-Centimeter Photoheliograph for LSO Mission", Prepared for NASA/George C. Marshall Space Flight Center, NASA contract NAS8-28147, Final Report, ITEK: Optical Systems Division, Itek 72-8212-3, January 8, 1973, p 6.3.

SUNSHADE/SHELL

The sunshade and shell structure are vital to the performance of the LTT. Their function is to prevent sunlight and stray light from illuminating the optical system, and to provide a support for the thermal protection of the telescope. For the analysis, the structure was considered to be a truncated cylindrical thin shell fabricated from aluminum. The geometry of the structure was determined by a viewing analysis and thermal considerations. The structure must support approximately 95 kg consisting of an aperture cover, insulation, fabric, meteoroid protection, and miscellaneous mechanisms. Using a safety factor of 1.4 and 6.5 g compressive load from Titan IV/Centaur requirements, the structure was evaluated for axial compression and elastic stability using equations from Roark's <u>Formulas for Stress and Strain</u>. Dynamic analysis of the structure was not performed for this study. The wall thickness was determined to be extremely small, so a minimum thickness of approximately 1 mm was selected. The calculated mass of the entire structure is 89.2 kg, not including the insulation, etc.

Lunar Transit Telescope
Sunshade / Shell

Sunshade / Shell Analysis

Considered Titan and Centaur Loads

Structure of Al 2219

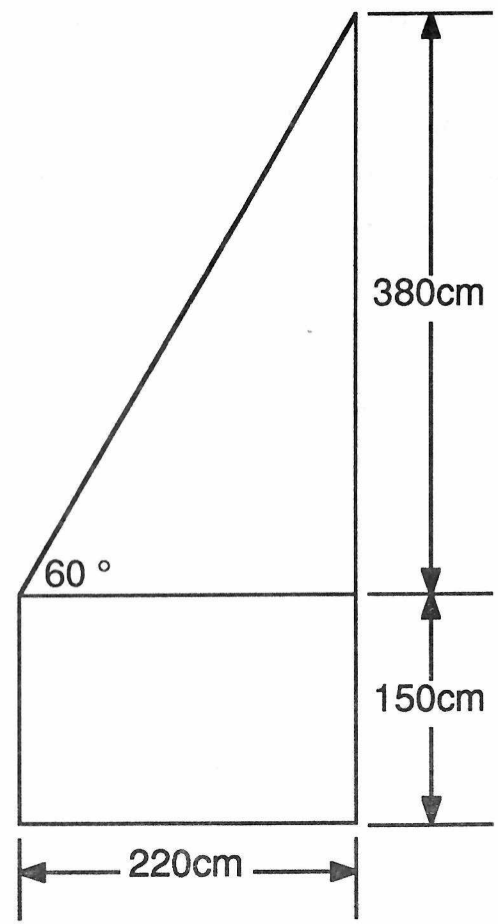
Safety factor of 1.4

Considered cylindrical thin shell

Supports 95 kg

Analyzed for axial compression and elastic stability using equations from Roark's Formulas for Stress and Strain

Sunshade Mass = 41.4 kg
Shell Mass = 47.8 kg



Side View

TELESCOPE / LANDER INTERFACE

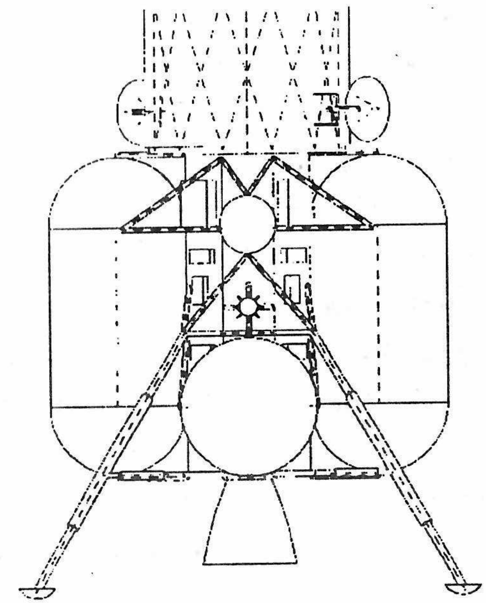
The interface structure between the telescope and lander was not designed for this study. A concept for this structure might be a series of Weldalite^{TM} 049 struts that connect the core structure on the lander to the ring that will be required to support the primary mirror support structure. Late in the study, pointing requirements were defined for the telescope. A mechanism must be designed to allow the optical system to be adjusted as a result of landing errors. The configuration of the primary mirror support structure and the lander will also play major roles in the design of the interface structure. Requirements for the structure will be a safety factor of 1.4 and loads as defined by the Titan IV/Centaur .

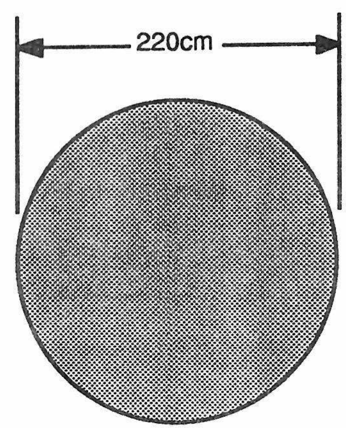
Lunar Transit Telescope
Telescope / Lander Interface

- Configuration of telescope/lander interface has yet to be determined

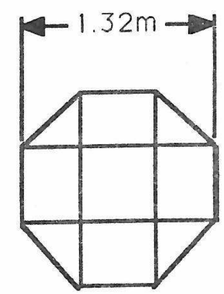
- Design pending better definition of structural components:
  - Pointing mechanism (to compensate for landing at a tilt)
  - Primary mirror support
  - Lander

- Initial parameters
  - Weldalite™ 049 struts
  - Safety factor of 1.4
  - Axial Acceleration of 6.5 g compression
  - Contingency factor for miscellaneous structure of 15 %





Bottom frame/plate of telescope



Top frame of lander

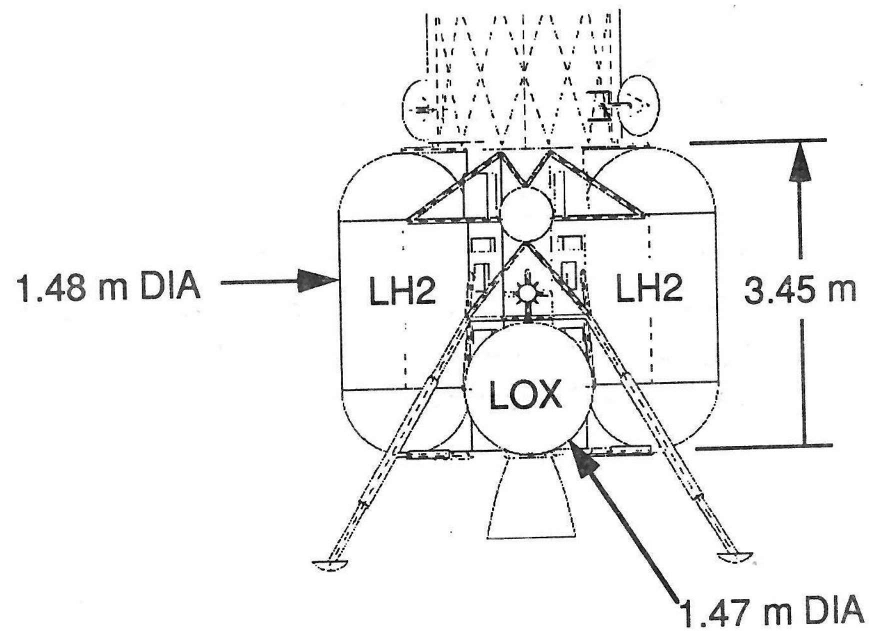
CRYOGENIC TANKAGE SIZING

A Microsoft Excel spreadsheet that was used for tank sizing in a previous study was modified to use for the LTT lander tankage. Two configurations were examined: the 2 m diameter system launched on the Titan IV/Centaur and a 1 m diameter system proposed for an Atlas IIAS launch vehicle. The configuration of the tank set consists of two cylindrical liquid hydrogen tanks and two spherical liquid oxygen tanks. The material chosen for the construction of the tanks was Weldalite™ 049, an aluminum-lithium alloy with high strength and low density. Given propellant loads,tank pressure of 0.34 MN/m^2 (50 psi), a safety factor of 1.4, and an axial acceleration of 6.5 g in compression, the tank wall thickness was determined using traditional pressure vessel analysis techniques. A minimum wall thickness was defined as approximately 1 mm and was used for the mass estimates.

Lunar Transit Telescope
Cryogenic Tankage
Sizing

- Developed spreadsheet to size tanks for propellant load and tank diameter
  - 2 m diameter primary mirror on Titan IV Centaur
  - 1 m diameter primary mirror on Atlas IIAS

- Input parameters
  - Weldalite 049™ Aluminum-Lithium (UTS 90 ksi)
  - Tank pressure 50 psi
  - Safety factor 1.4
  - Axial acceleration 6.5 g compression
  - Minimum wall thickness 1.016 mm
  - Additional structure 10% of total weight



Tank Mass for Lander

	LH2	LOX	Total (4 tanks)
2 m primary mirror	58.1 kg	21.8 kg	$\boxed{159.8\ \text{kg}}$
1 m primary mirror	18.3 kg	9.8 kg	56.2 kg

CORE STRUCTURE AND CENTAUR INTERFACE

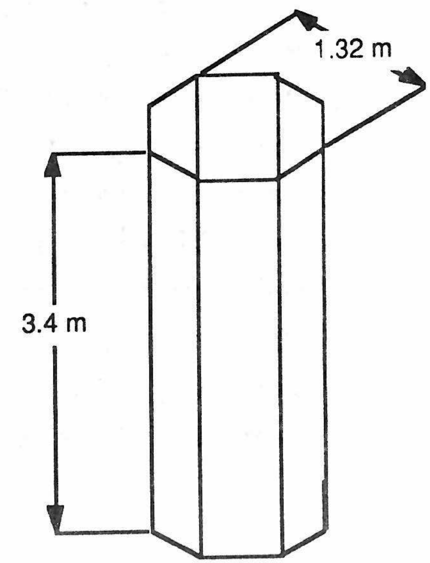
The core structure of the vehicle was studied with respect to loads experienced during the use of the Titan and the Centaur. The structure is of Aluminum 2219, and a safety factor of 1.4 was applied with respect to the ultimate strength. The structure was treated as a thin shelled hollow cylinder that was loaded as a cantilever. The required equivalent thickness was determined based on the loads on the structure. The mass of the structure is 150 kilograms.

The effect of the Titan and Centaur loads were considered on the structure of the LTT and Centaur interface. The interface structure is comprised of tubular struts of Aluminum 2219. A safety factor of 1.4 was applied to the buckling load to size the structure. The mass of the structure is 150 kilograms.

Lunar Transit Telescope
Core Structure and Centaur Interface Structure Analysis

Core Structure Analysis

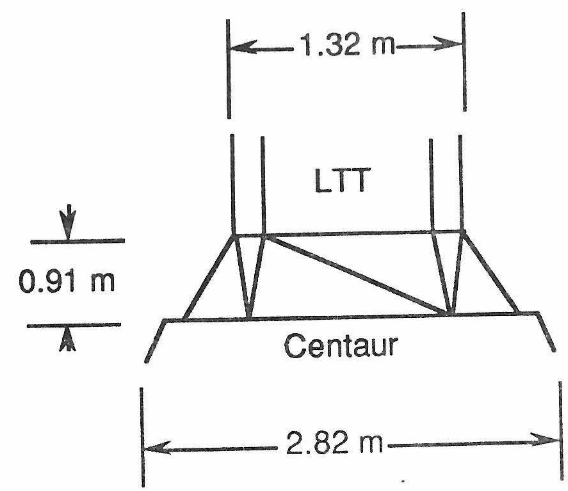
- Considered Titan and Centaur Loads
- Structure of Al 2219
- Safety factor of 1.4 on the Ultimate Strength
- Performed simple analysis
  - Considered the structure as a thin shelled hollow cylinder
  - Treated the structure as a cantilever beam



Core Structure Mass = 150 kg

LTT/Centaur Interface Analysis

- Considered Titan and Centaur Loads
- Structure of Al 2219
- Considered tubular struts
- Safety factor of 1.4 on the buckling load



LTT/Centaur Interface Mass = 150 kg

LANDING GEAR ANALYSIS

The landing gear configuration was studied with respect to its landing environment. The structure is of Beryllium-Aluminum. The mass of the landing gear was determined by considering the landed mass of the Apollo Lunar Excursion Module (LEM). The ratio of the landing gear mass to the total landed mass of the Apollo LEM was .032. Since the geometry of the LTT landing gear is similar to the geometry of the LEM landing gear, the ratio of the LEM mass to the landing gear mass was used to estimate the LTT landing gear mass. The landing gear mass was modified to account for the differences in the geometry of the LTT as compared to the LEM configuration. The landing gear mass is 137 kilograms.

Lunar Transit Telescope
Landing Gear Analysis

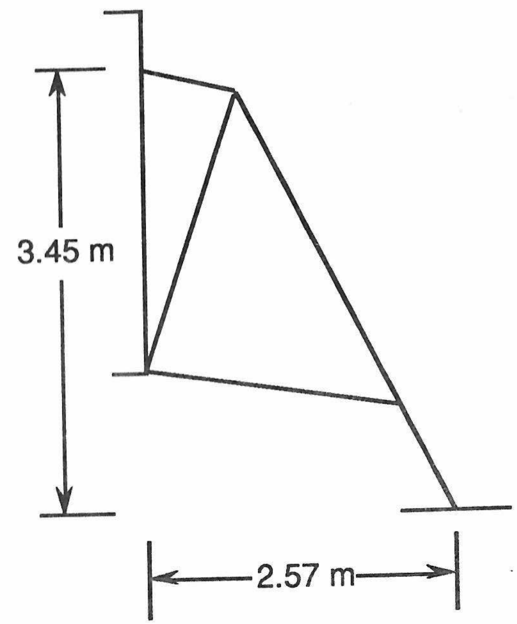
Landing Gear Analysis

Considered the landing environment

Structure of Beryllium - Aluminum

Considered the mass of the Apollo LEM landing gear

Used the landing gear mass to total landed mass ratio of the Apollo LEM



Apollo LEM Landed Mass = 6893 kg

Apollo Landing Gear Mass = 220 kg

Apollo Landed Mass / Landing Gear Mass = 0.032

LTT Landing Gear Mass = 102 kg

Adjusted the mass to account for geometry differences

$\boxed{\text{Landing Gear Mass = 137 kg}}$

STRUCTURES MASS SUMMARY

The mass estimates that were determined by the analysis support the overall mass statement for LTT. Detailed design and analysis will be used to provide higher fidelity estimates in the future. The design of secondary structure will also affect the mass estimates as design progresses.

Lunar Transit Telescope Structures Mass Summary

Part	Material	Mass (kg)
Metering Structure	Invar 36 and Aluminum	30.0
Primary Mirror Support	Invar 36 and Aluminum	175.0
Telescope/Lander Interface	Weldalite™ 049	
Sunshade	Aluminum 2219	41.4
Shell	Aluminum 2219	47.8
Tanks	Weldalite™ 049	159.8
Core	Aluminum 2219	150.0
LTT/Centaur Interface	Aluminum 2219	150.0
Landing Gear	Beryllium-Aluminum	137.0

- Structural details and secondary structure design will affect these mass estimates

STRUCTURAL ANALYSIS CONCLUSIONS / SUMMARY

In conclusion, the major structural components of the LTT were analyzed for a limited set of loading conditions. Elements of the structure that were evaluated in this analysis included the metering structure, sunshade, shell, core, interface to Centaur, and the lander legs. The primary mirror support structure and the interface structure between the telescope and lander were examined conceptually for the LTT study. A dynamic analysis of the system for the determination of natural frequencies was not performed as a part of this study, but will be a necessary element of the detailed analysis to follow. Prior to any extensive structural analysis effort for LTT, primary and secondary structure must be detailed and deflection requirements for all phases of launch, lunar landing, and operation must be determined. The structural mass estimates contained herein represent a fair evaluation of the LTT as defined in this study.

Lunar Transit Telescope
Structural Analysis
Conclusions / Summary

- Major structural components of LTT have been evaluated for limited set of load conditions
- Natural frequencies were not determined for the LTT system as a part of this study
- Structural design must be detailed prior to extensive analysis.

  Need:
  - Deflection requirements
  - Detailed load set definition including launch, lunar landing, and operational loads
- Materials selected represent current/emerging technology (No advanced development foreseen for materials)
- Structural mass estimates fairly represent the current configuration

3.4 THERMAL ANALYSIS AND DESIGN

LTT THERMAL ANALYSIS & DESIGN REQUIREMENTS

LTT optics temperatures of 150 Kelvin (K) or less are desired for several hours per Lunar cycle for IR observations at wavelengths to 3.5 um. Because there is no requirement for mirror temperatures significantly below 150 K, the current design philosophy is to reduce the total temperature excursion of the optical components as much as possible, while maintaining optics temperatures at or below 150 K for a significant portion of the Lunar night. This philosophy of keeping the telescope temperature as stable as possible rather than reaching temperatures as low as possible will result in less thermal distortion of the mirrors and metering structure. Colder mirror temperatures could be obtained during the Lunar night by removing some shielding and insulation, but this would result in higher temperatures during the Lunar day and a much larger total temperature excursion for the optics.

Operating temperature requirements for the Optical/UV and IR CCDs used in the focal plane array are shown on the opposite page. The preferred temperature for the optical/UV devices is approximately 170 K, but there are indications that they may be useable to 80 K if appropriate correction factors can be determined. This would limit the need for heating that array during the Lunar night. Operating and non-operating temperature limits are also presented for subsystems equipment that has been defined.

LTT Thermal Analysis/Design

Requirements

Optics:

Several hours below 150 K per lunation (McGraw & McDowell)
Maximum temperature excursion of ≈ 100 K per lunation?
Entire optical system ≈ isothermal (Korsch & McGraw)

Focal plane:

Optical/UV CCDs -- 77 K to 210 K, prefer 180 K max. (McGraw)
IR CCDs -- 50 K to 80 K (McGraw), prefer 65 K (McDowell)

Subsystem Equipment Temperature Requirements:

C&DH:	-25 to +85°C	
Battery:	0 to 25°C operational, less than 180°C non-operational	
GN&C:	IMU	-10 to 50°C operating
	Star tracker	-5 to 40°C operating, -30 to 60°C non-operating
	Sun sensor	-107 to 85°C
	Sun sensor elec.	-30 to 85°C
	Landing radar	-24 to 60°C operating, -30 to 70°C non-operating
	Control elec.	-24 to 61°C operating, -40 to 80°C non-operating
	Video camera	-24 to 61°C operating, -40 to 80°C non-operating
Propulsion:	Pressurant	21.4°C ±25°C
	RCS thrusters	10°C to 50°C
	Hydrazine	5°C to 44°C
	RCS valves	10°C to 44°C
	Solenoid valves	-54°C to 72°C

THERMAL CHARACTERISTICS OF THE LUNAR SURFACE

A number of sources describe the thermal characteristics of various locations on the Lunar surface. The optical properties, soil thermal conductivity, and soil specific heat used in this thermal analysis are shown in the figure, along with the sources. Also shown are a range of predicted and measured temperatures for the Lunar surface and soil temperatures at various depths. The Apollo 11 temperatures, and those of Wesselink, Jaeger, and Linsky are predictions, the others are measured directly or inferred from measurements of infrared or microwave radiation from the Lunar surface. Note that, although the Lunar surface temperature varies widely, the soil temperature is nearly constant a short distance below the surface.

Sources for Lunar Surface Temperatures in Table:

1. Apollo 11 - Cremers, Birkebak, and White, 'Lunar Surface Temperatures at Tranquility Base,' AIAA paper 71-79, 1971.
2. Apollo 12 - Cremers, Birkebak, and White, 'Thermal Characteristics of the Lunar Surface Layer,' International Journal of Heat and Mass Transfer, May 1972.
3. A. J. Wesselink, 'Heat Conductivity and Nature of the Lunar Surface Material,' Bulletin of Astr. Institutes of the Netherlands, X, 351-363, 1948.
4. J. C. Jaeger, 'The Surface Temperature of the Moon,' Austrian Journal of Physics' A6, 10-21, 1953.
5. J. L. Linsky, 'Models of the Lunar Surface Including Temperature Dependent Thermal Properties,' Icarus 5, 606-634, 1966.
6. E. Pettit and S. B. Nicholson, 'Lunar Radiation and Temperature,' Astrophysics Journal 71, 102-135, 1930.
7. W. M. Sinton, 'Temperatures on the Lunar Surface,' Physics and Astronomy of the Moon, edited by Z. Kopal, pp. 407-428, Academic Press, New York, 1962.
8. J. M. Saari, 'The Surface Temperature of the Antisolar Point of the Moon,' Icarus 3, 161-163, 1964.
9. L. D. Stimpson and J. W. Lucas, 'Revised Lunar Surface Thermal Characteristics Obtained from the Surveyor V Spacecraft,' AIAA paper 69-594, 1969.

LTT Thermal Analysis/Design

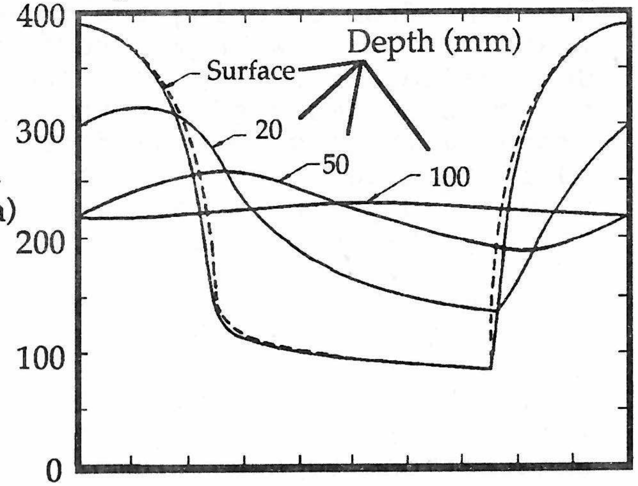
Thermal Characteristics of the Lunar Surface

Measured & Predicted Lunar Surface Temperatures

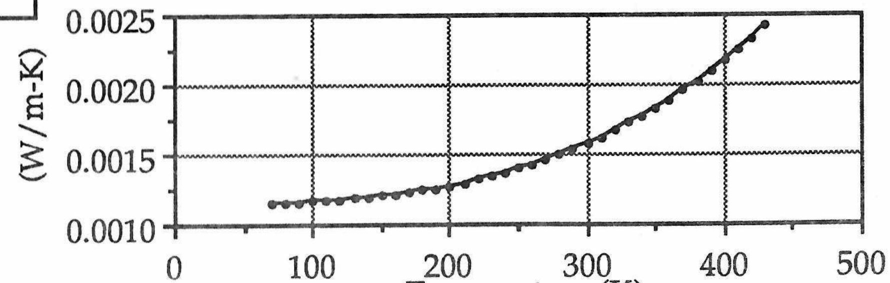
Source	Noon (K)	Sunset (K)	Midnight (K)	Sunrise (K)
Apollo 11	395	152	101	92.9
Apollo 12	389	134	95	86.1
Wesselink	370	144	98	90
Jaeger	368	178	97	89
Linsky			98	89
Pettit & Nicholson	374		120	
Sinton	389	181	122	109
Saari			104	
Stimpson & Lucas	386-390	140-200	100-112	

"Thermal Characteristics of the Lunar Surface Layer" by Cremers, Birkebak, & White, *International Journal of Heat & Mass Transfer*, May 1972.

Temperatures at Various Soil Depths



Soil Thermal Conductivity: Temperature dependent

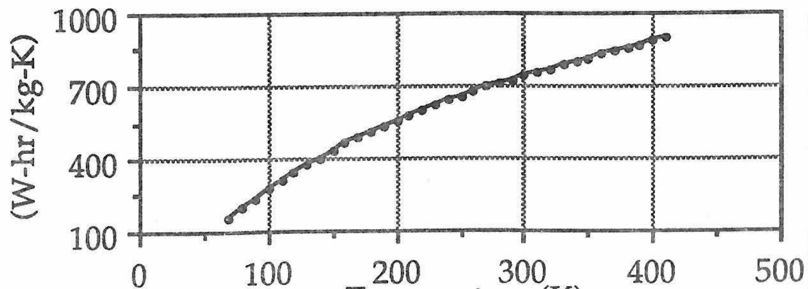


Optical Properties:

Absorptivity (α) = 0.93
Emissivity (ε) = 0.95
"Study of the Lunar Environment" by Tatom, Ramakrishna, & Vaughn, September 1966

Soil Specific Heat: Temperature dependent

"Temperatures and Thermophysical Properties of the Lunar Outermost Layer" by Jones, Watkins, & Calvert.



PD22

LTT THERMAL ANALYSIS & DESIGN ISSUES

The LTT concept incorporates a sunshade designed to prevent direct sunlight from entering the telescope and heating the mirrors or surfaces (baffles, etc.) with a direct view to the mirrors. The sunshade may be fixed, deployable, or articulated (adjusts shape depending on location of the sun), but must be designed for the latitude of the LTT landing site and oriented properly (tall side toward the south for northerly sites). A small additional deployable skirt or ground shade may be of significant benefit in reducing daytime temperatures by reducing radiative heat transfer between the LTT and the Lunar surface.

Reduction of cryogen boiloff during launch and transit to the moon and prevention of condensate on tanks before launch is accomplished with foam and multi-layer insulation (MLI). The foam insulation eleminates the need for a helium purge, although a nitrogen gas purge will still be requires after the liquid hydrogen is loaded. Tank support structure and other insulation penetrations such as plumbing lines and insulation pins should be minimized to reduce propellant boiloff.

Scientific instrument and subsystem equipment thermal control will be passive to the extent possible. Each component will have an integral radiator if heat rejection is required, and appropriate insulation, thermal coatings, and small heaters if needed. The IR CCDs need to operate below 80 K, which may be impossible to achieve without active cooling. This decision depends almost entirely on the heat rejection requirement for the CCDs.

The two RTGs reject 4,100 W_t each at a temperature of about 580 K. These will be located out of the field of view of temperature sensitive optics and detectors. Reflective shields will also be used between the RTGs and the LTT vehicle. Future analyses will determine if extendable booms are required to move the RTGs further from the LTT after landing. A base heat shield (BHS) is required to protect lander components from the engine exhaust plume heating during descent and landing.

LTT Thermal Analysis/Design

Issues

Optical System thermal control

Function: Prevent sunlight from entering telescope or heating objects in the FOV & reduce heat transfer between LTT & surface

Considerations:
- Sunshade shape/size depend on LTT latitude & orientation, need to control sunshade orientation
- Tanks & heat shield may obstruct ground view sufficiently
- Deployable fabric sunshade and/or ground shade are options

Instrument/Subsystems

Function: Maintain components within allowable temperature range

Considerations:
- Passive thermal control preferred
- Individual instrument radiators for cooling
- Active cooling may be required for IR CCDs
- Heaters needed for some components (lunar night)

Cryogenic fluid thermal protection

Function: Reduce boiloff of LH_2 & LO_2 during transit, prevent condensation before launch

Considerations:
- Multi-layer insulation (with foam on hydrogen tank) is method of choice
- Insulation penetrations (struts, etc.) should be minimized
- MLI weight should be traded against reductions in boiloff and tank weight

Power/Propulsion elements

Function: Prevent overheating of components by elements of power or propulsion systems

Considerations:
- RTGs selected run hot (>500 K) and dissipate ≈4100 W each continuously
- RTGs must be out of FOV of temperature sensitive components (esp. cold optics & CCDs)
- RTGs need good view to space
- Base heat shield needed to protect against engine exhaust during descent/landing

PD22

TRADE STUDIES

The first trade study examined the effect of sunshade shell MLI thickness on the total temperature excursion (difference between maximum and minimum temperatures) of the primary mirror. The shell was assumed to be made from 0.102 cm (0.04") Al and covered with MLI. For the purposes of this comparison, the insulation on the underside of the primary mirror was kept constant at 1.27 cm (0.5") MLI. Runs were made for MLI thicknesses between 0.0254 cm (0.01") (A2 layers) and 3.81 cm (1.5") on the shell.

To provide cooler daytime temperatures and smaller total temperature excursion for the mirror, a double shell sunshade concept was also considered. This concept would provide increased resistance to heat penetration, as well as more area to radiate heat from the telescope. As before, the mirror insulation was kept at 1.27 cm (0.5") MLI.

Another concept considered to reduce mirror temperature excursion was a small ground shade, deployed from the upper surface of the lander, to minimize telescope radiative interchange with the lunar surface. Results were compared to results from the case in which a simple lander body shape was used alone to block some of the telescope view to the surface.

A trade study was also done to determine the degree of thermal isolation which will be required for the mirror. The mirror support structure will provide undesirable heat conduction paths between the primary mirror and the sunshade. This study attempted to quantify the effect of conductance through the structure on primary mirror temperature excursion.

Simple radiation and heat transfer models were constructed using TRASYS and SINDA to represent the LTT geometry. These models were intended to indicate bulk temperatures for major components of the LTT and are not sufficiently detailed to indicate spatial temperature gradients in the optics.

LTT Thermal Analysis/Design

Trade Studies

Sunshade Concept

- MLI thickness on single shell sunshade
- Effectiveness of double shell sunshade

Ground Shade Concept

- Lander body only vs. additional deployable shade

Primary Mirror Support Method

- Conductance of mirror support vs. temperature response

Analysis methods

- SINDA and TRASYS models of LTT shape (40 to 60 node models)
- Scaling from previous designs for tank insulation
- Mass estimates based on previous design or hand calculations

Limitations

- Coarse thermal models indicate bulk component temperatures only, no spatial gradients available
- Effect of RTGs on local temperatures not examined
- Propellant boiloff estimates should be refined

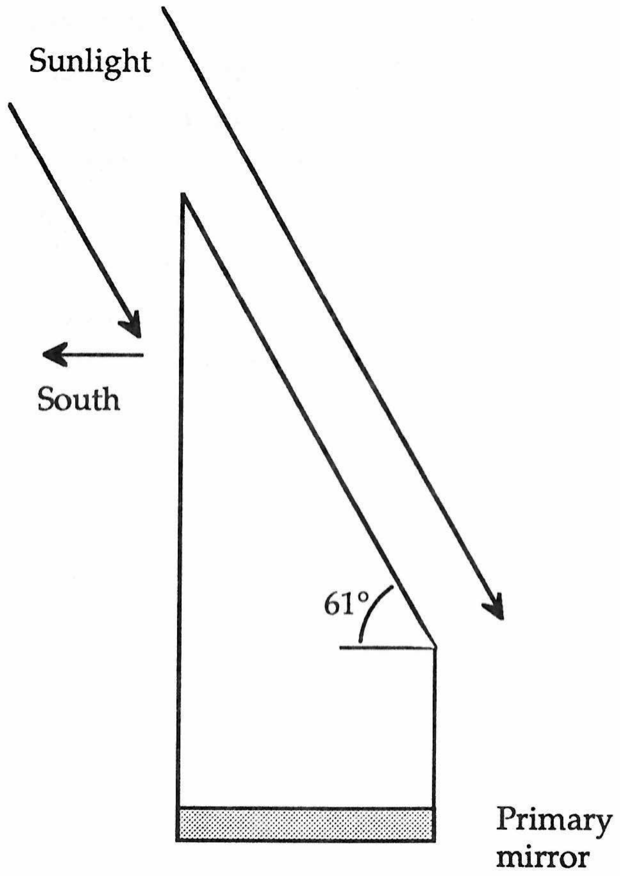
BASIC SUNSHADE CONCEPT

The purpose of the LTT sunshade is to protect the telescope optics from direct sunlight and keep them relatively cool at as stable a temperature as possible. Because of the wide temperature swings on the lunar surface and long days (672 hrs.) on the lunar surface, the latter task was particularly challenging.

The LTT concept incorporates a sunshade designed to prevent direct sunlight from entering the telescope and heating the mirrors or surfaces (baffles, etc.) with a direct view to the mirrors. The sunshade may be fixed, deployable, or articulated (adjusts shape depending on location of the sun), but must be designed for the latitude of the LTT landing site and oriented properly (tall side toward the south for northerly sites). To determine the slope and height required for the sunshade, a landing latitude of 30 N was assumed. This necessitated a sunshade angle of 61 degrees as shown. This sunshade angle is also steep enough to eliminate the possibility of "earthshine" entering the telescope. Since the current telescope line of sight pointing requirements are about 1 arcminute, it was assumed that the sunshade will be properly oriented once the telescope is in operation. Orientation of the sun shade does not have to be as accurate as line of sight pointing, may be within a few degrees of the noon sun.

LTT Thermal Analysis/Design

Trade Studies -- Basic Sunshade Concept



- Assumed 40° N landing site
- Interior baffling may be required
- Sunshade may be deployable
- Sunshade assumed to be light shell covered with MLI
- Sunshade may also serve to block Earthshine from LTT

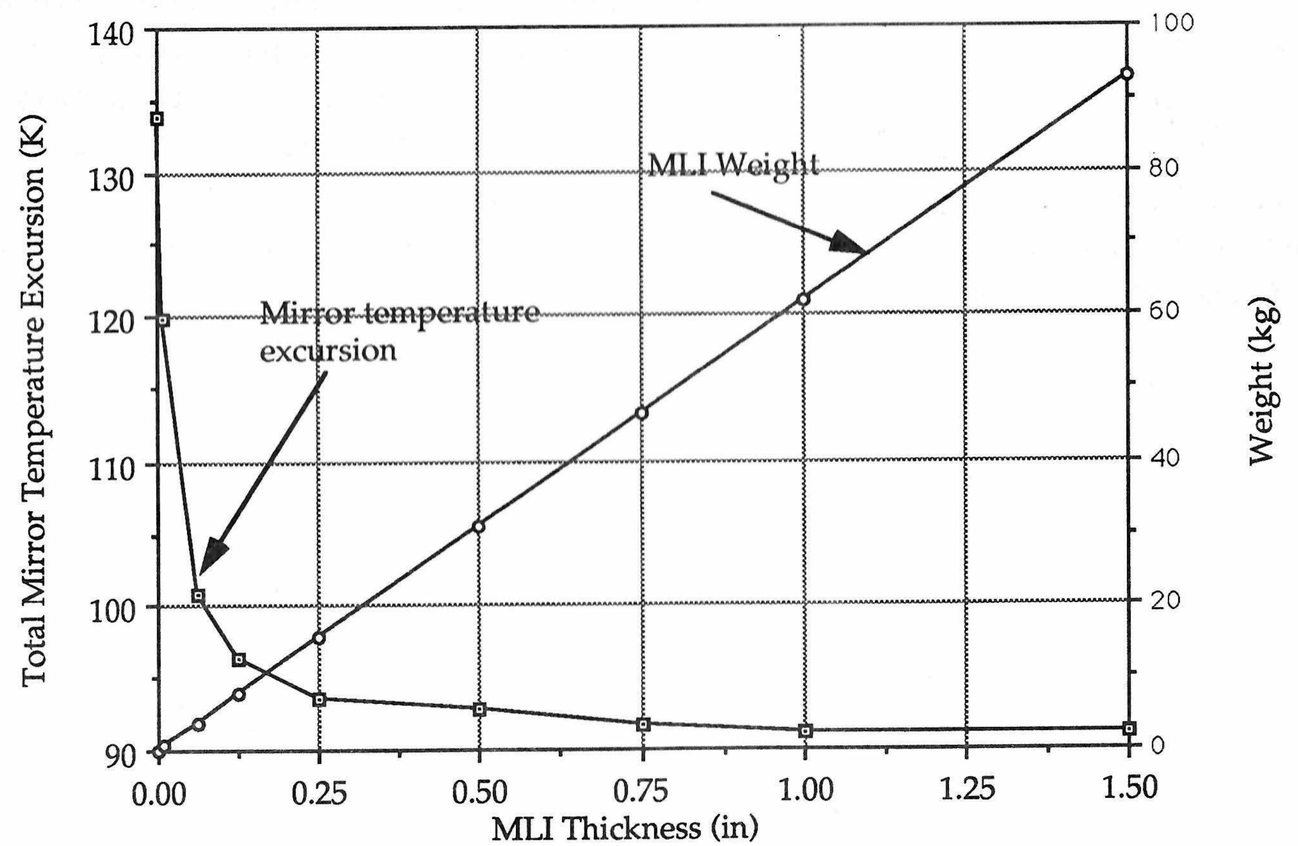
Direct sunlight cannot reach optics due to sunshade shape & orientation

SINGLE-SHELL SUNSHADE, RESULTS OF MLI THICKNESS STUDY

For the purposes of this comparison, the insulation on the underside of the primary mirror was held constant at 1.27 cm (0.5") MLI. Runs were made for MLI thicknesses between 0.0254 cm (0.01") (2 layers) and 3.81 cm (1.5") on the shell. As shown in the figure, there is little decrease in mirror temperature excursion for thicknesses greater than 0.635 cm (0.25"), but a significant weight gain for thicker blankets; therefore, 0.635 cm (0.25") was chosen as the nominal MLI thickness. MLI areal density was assumed to vary with thickness in a linear manner. This is not strictly true since the weight of pins, seams, supports, etc. do not vary much with thickness, but it is sufficient for an initial comparison.

LTT Thermal Analysis/Design

Trade Studies -- MLI Thickness Results for Single Shell Sunshade

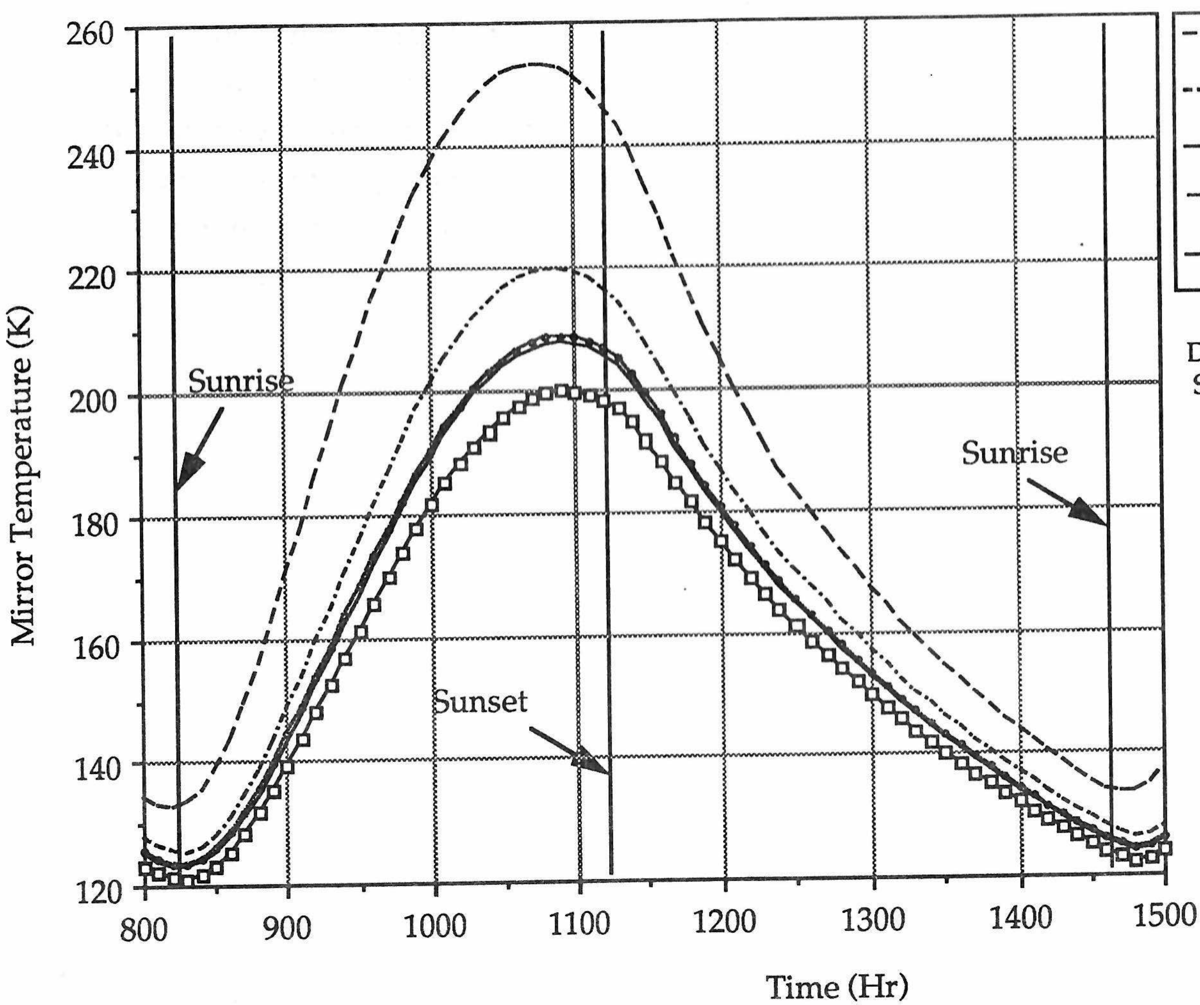


MIRROR TEMPERATURES WITH DOUBLE SHELL SUNSHADE

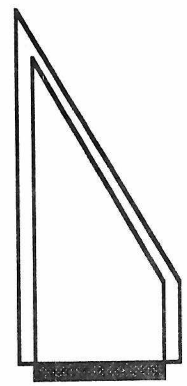
Since 0.635 cm (0.25") MLI seemed to be adequate on the single-shell telescope, this thickness was also used for the double-shell trade study. Results for several configurations are shown in the figure. The single shell cases with near zero and 0.635 cm MLI are included for comparison, as well as cases where 0.635 cm MLI was used only on the inner shell, and then only on the outer shell. The dual-shell sunshade with 0.635 cm MLI on the outer shell was found to provide a significant improvement over the comparable single-shell case, but the increased performance may not justify the added mass of the additional shell.

LTT Thermal Analysis/Design

Trade Studies -- Double Shell vs. Single Shell Sunshade



DS - Double Shell
SS - Single Shell



α=.22, ε=.82

Double-Shell Sunshade
Transit Telescope

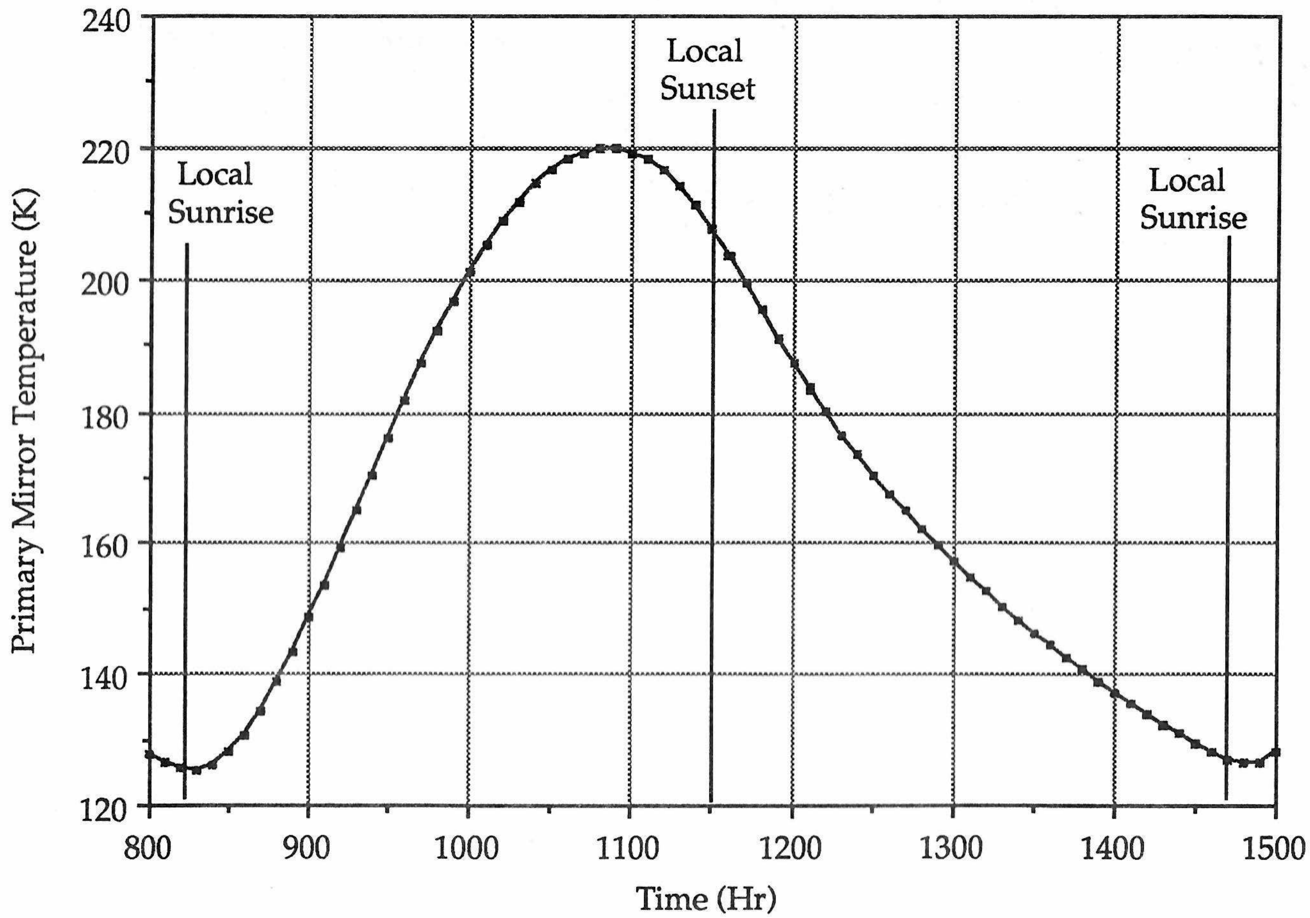
BASELINE SUNSHADE PERFORMANCE -- SINGLE SHELL, 0.25" MLI

A double shell sunshade provides the best thermal performance, however weight constraints may force reliance on a single-shell sunshade. Preliminary results indicate that a thin layer of MLI on the LTT shell and sunshade, as well as thermal isolation of the primary mirror, will be sufficient to provide approximately 150 hr. below 150 K and limit primary mirror temperature swing to less than 100 K per Lunar day as indicated. Although the primary mirror temperature fluctuates between approximately 120 K and 220 K, there are no sharp temperature changes which would cause sudden expansions/contractions in the optics.

LTT Thermal Analysis/Design

Trade Studies -- Baseline Sunshade Performance

Recommend single shell sunshade with 0.25" MLI

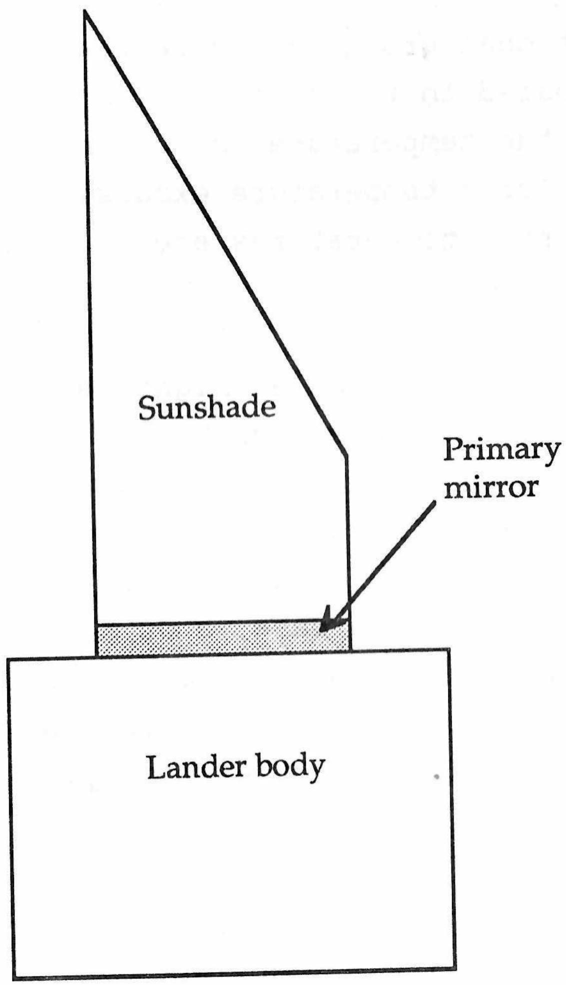


GROUND SHADE CONCEPTS

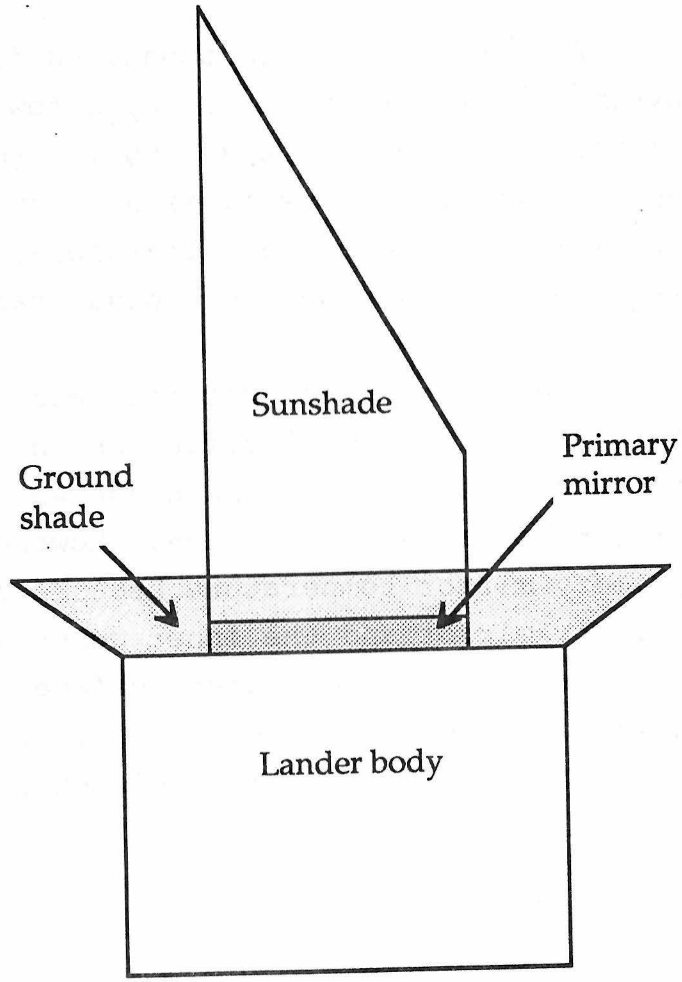
Another concept considered to reduce mirror temperature excursion was a small ground shade, deployed from the upper surface of the lander, to minimize telescope radiative interchange with the lunar surface. The figure shows one concept for such a ground shade. Any performance improvement for this deployable shade must be weighed against its additional mass. This concept was compared to the basic design in which only the lander body (tankage, engine, heat shield, etc.) partially blocks the telescope view to the Lunar surface.

LTT Thermal Analysis/Design

Trade Studies -- Ground Shade Concepts



Lander body (representing tanks & base heat shield) obstructs view to nearby surface



Shade blocks view of shell/sunshade to larger area of Lunar surface

PD22

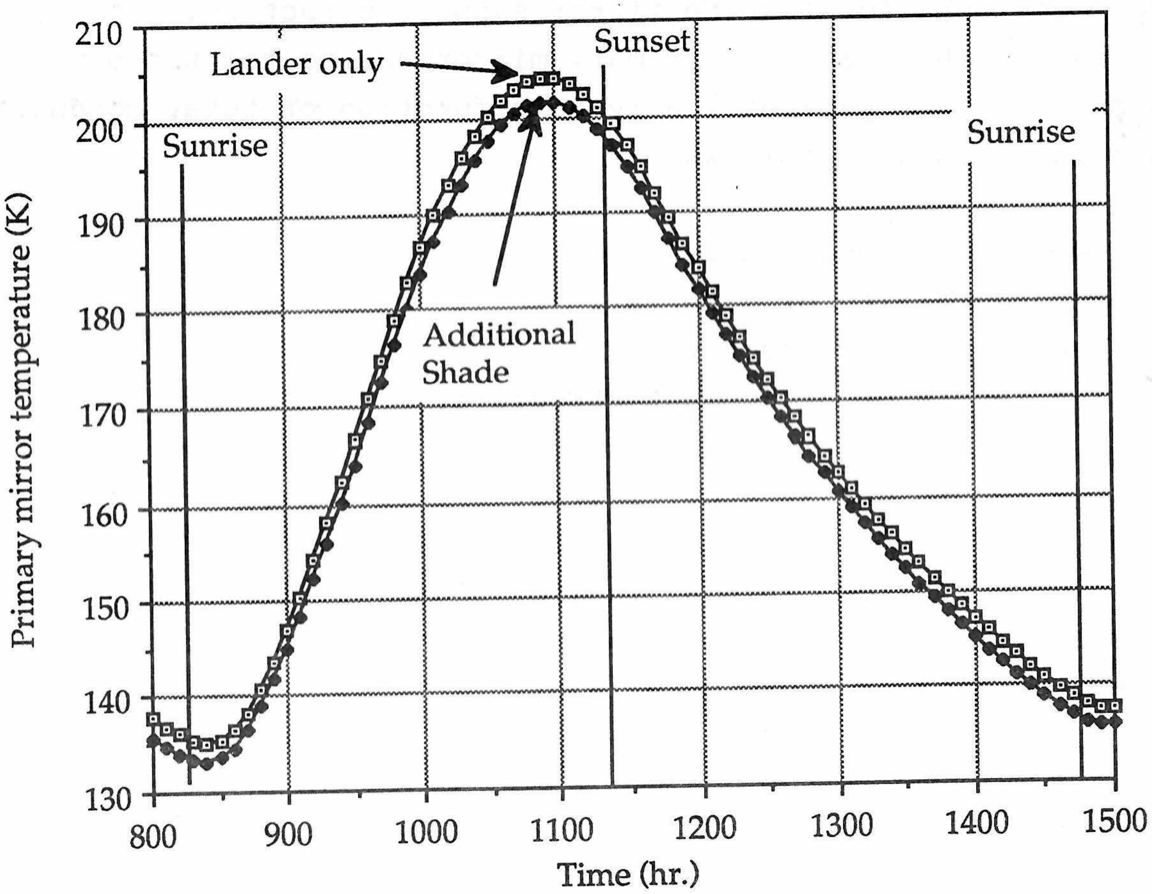
RESULTS OF GROUND SHADE STUDY

As shown in the accompanying figure, the additional ground shade reduces overall mirror temperature by a few degrees as compared to the configuration containing only the lander body. However, because the temperature reduction applies to night as well as day temperatures, the mirror temperature excursion is virtually unchanged. Therefore, unless lower mirror temperatures are required, the deployable ground shade is not advantageous.

The ground shade material was cooler than the lunar surface at night, but this had only a small effect on the already low mirror and inner sunshade temperatures. The ground shade was also cooler than the lunar surface because of its low solar absorption. However, this did not have a large effect on primary mirror temperature because the outer sunshade is insulated from the primary mirror and the inner sunshade surface by MLI. In this case, a 30 K decrease in sunshade outer surface temperature (at some locations) resulred in a mirror temperature decrease of less than 5 K. In addition to the insulation most of the heat transfer between the inner sunshade surface and the primary mirror is via radiation also reduced the impact of a cooler sunshade.

LTT Thermal Analysis/Design

Trade Studies -- Ground Shade Results



CONDUCTANCE OF CANDIDATE MIRROR SUPPORT STRUCTURE MATERIALS

Invar 36, graphite/epoxy, aluminum 2219, and beryllium are some of the materials under consideration for the mirror support structure. The heat leak between the telescope shell and the primary mirror due to the support structure has been treated parametrically as a function of total conductance. The conductance is defined as follows:

$G = kA/l$, where G = conductance

k = thermal conductivity of material

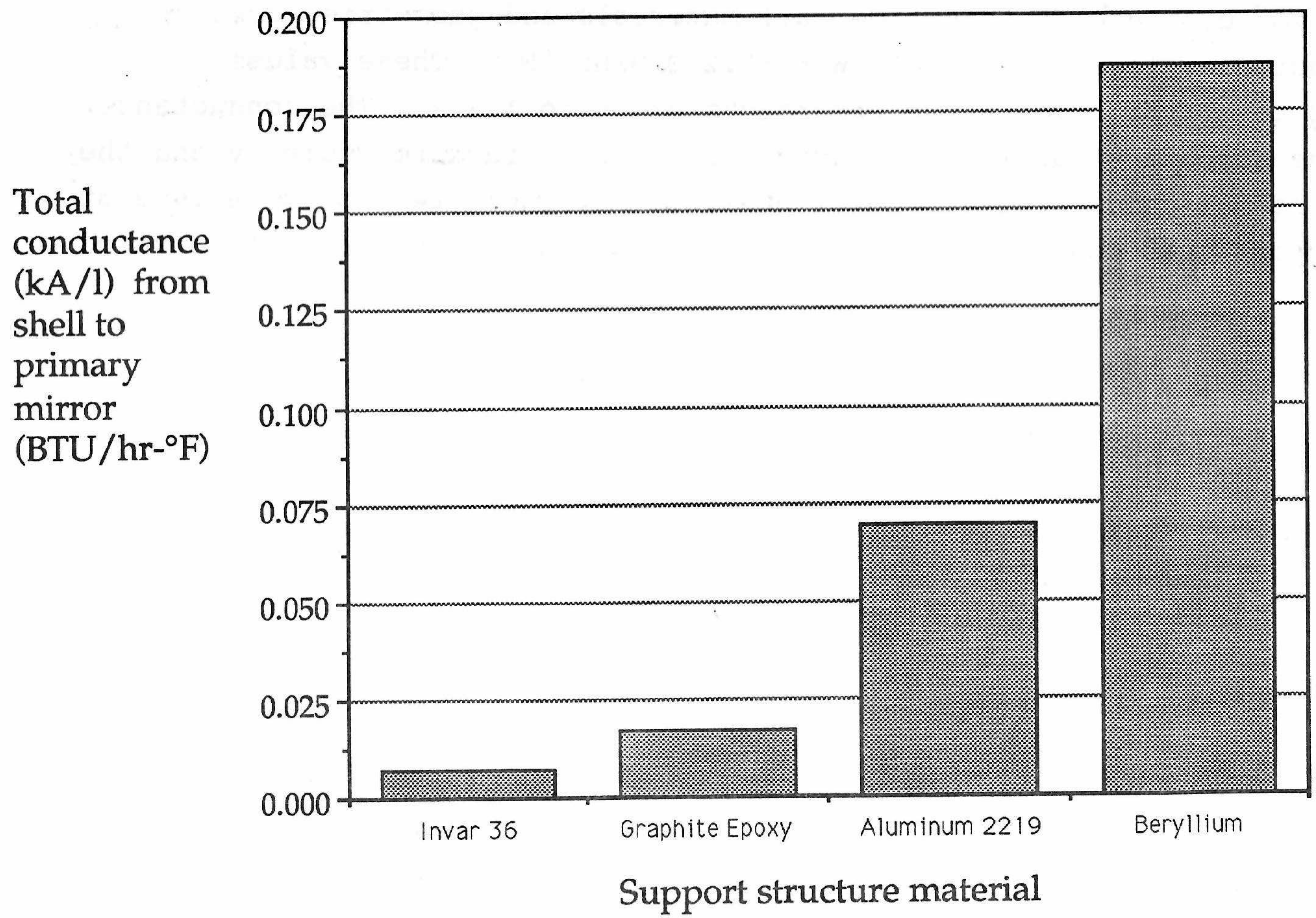
A = cross section area (effective)

l = effective length of attachment structure.

The effective length was 2.54 cm (1") for all materials. Effective cross section area was 0.0968 cm^2 (0.015 in^2) for Invar 36, 0.0671 cm^2 (0.0104 in^2) for graphite/epoxy, 0.1077 cm^2 (0.0167 in^2) for aluminum 2219, and 0.1452 cm^2 (0.0225 in^2) for beryllium. The graph shows the conductance for support structures made from each of these materials.

LTT Thermal Analysis/Design

Trade Studies -- Conductance of Candidate Mirror Support Structure Materials

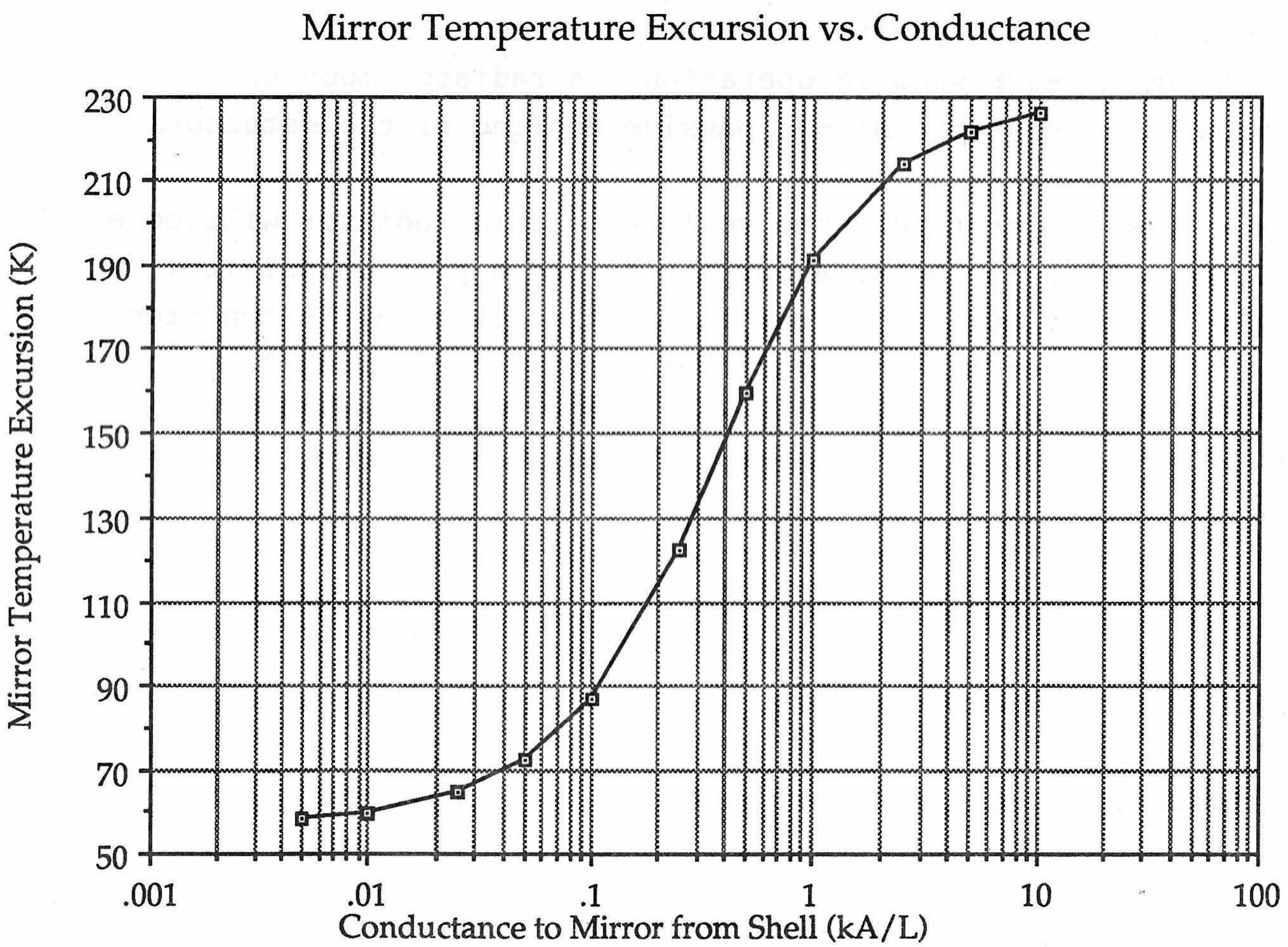


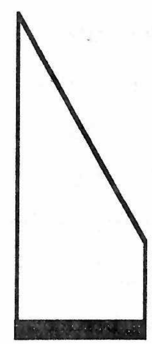
MIRROR TEMPERATURE EXCURSION VS. CONDUCTANCE

The accompanying figure shows the effect of varying conductance on the total primary mirror temperature excursion. Although the support structure design is not complete at this time, all materials and geometries considered have conductances less than 0.1055 W/K (0.2 BTU/hr-^{o}F). These values correspond to mirror temperature excursions of 60 to 115 K. The conductance through the support structure is a function of the structure geometry and the material thermal conductivity. Later analyses may indicate that some type of launch support that could be retracted after landing is preferable to a fixed support.

LTT Thermal Analysis/Design

Trade Studies -- Mirror Support Conductance Results





α=.22, ε=.82

Single-Shell Sunshade
Transit Telescope

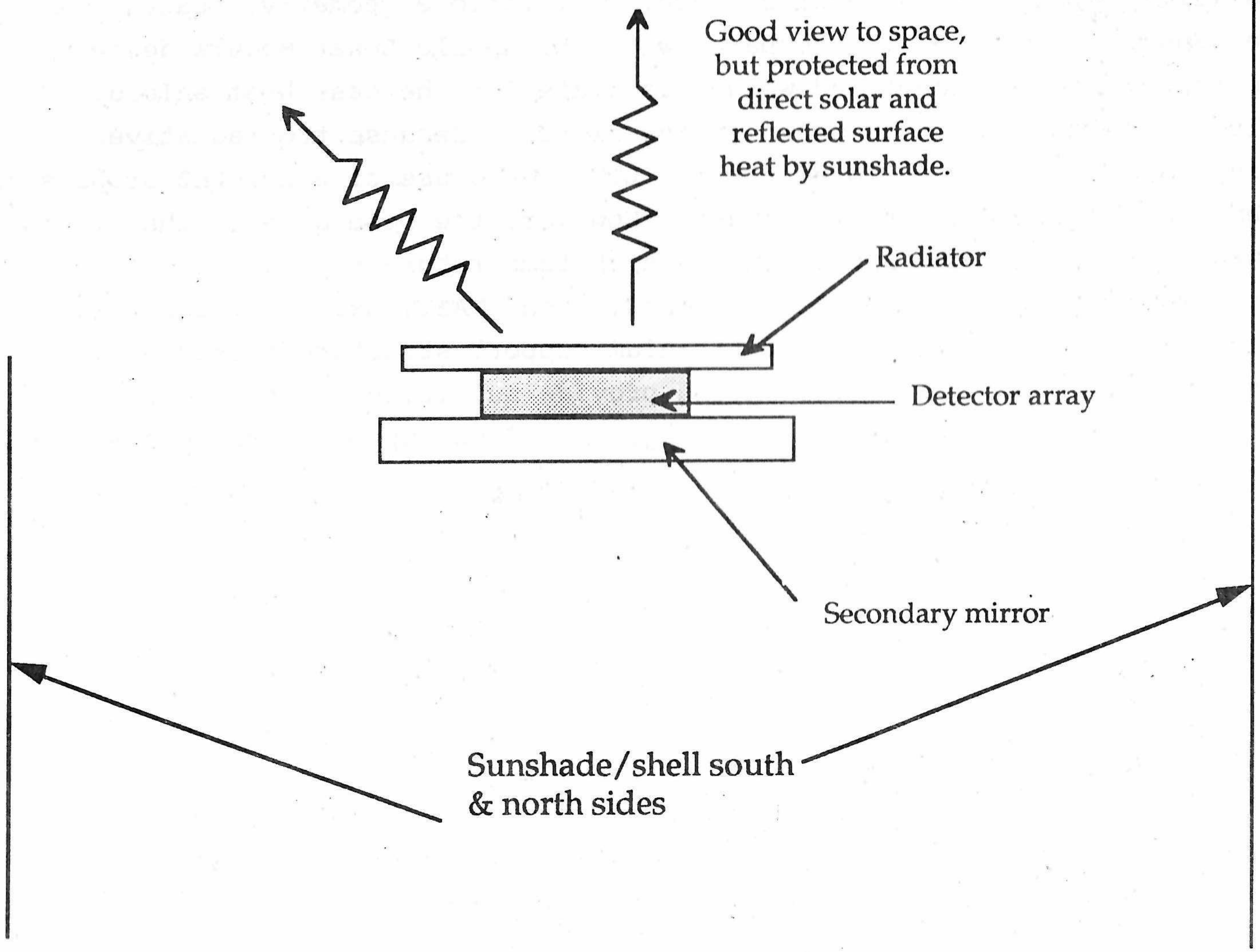
DETECTOR ARRAY RADIATOR

The proposed LTT focal plane detector array will contain UV/visible CCDs as well as IR CCDs. The visible/UV CCDs operate at temperatures as high as 180 to 210 K and may perform satisfactorily as cold as 80 K, but the IR CCDs should be kept at or below 80 K when in operation. A radiator mounted on top of the array as shown in the sketch allows passive cooling of the detectors.

To achieve very low temperatures, the surface of this radiator will be a second surface mirror with a low solar absorptivity and a high emissivity. The low absorptivity, second surface mirror allows the CCD array to function in spite of sunshade damage or misalignment that might occur during landing. In this case, the radiator would maintain a relatively low temperature even with some direct solar impingement. Separate radiator heat conduction paths will be incorporated to obtain minimum temperatures for the IR CCDs without excessively cooling the visible/UV CCDs. The location behind the secondary mirror provides shielding from surface radiation and a good view to deep space, so the radiator should have consistently low temperatures. However, radiator and CCD temperatures are dependant on heat load which is not well defined. This is especially critical when operating at low temperatures where the heat rejection capability of the radiator is approximately 1 W.

LTT Thermal Analysis/Design

Detector Array Radiator

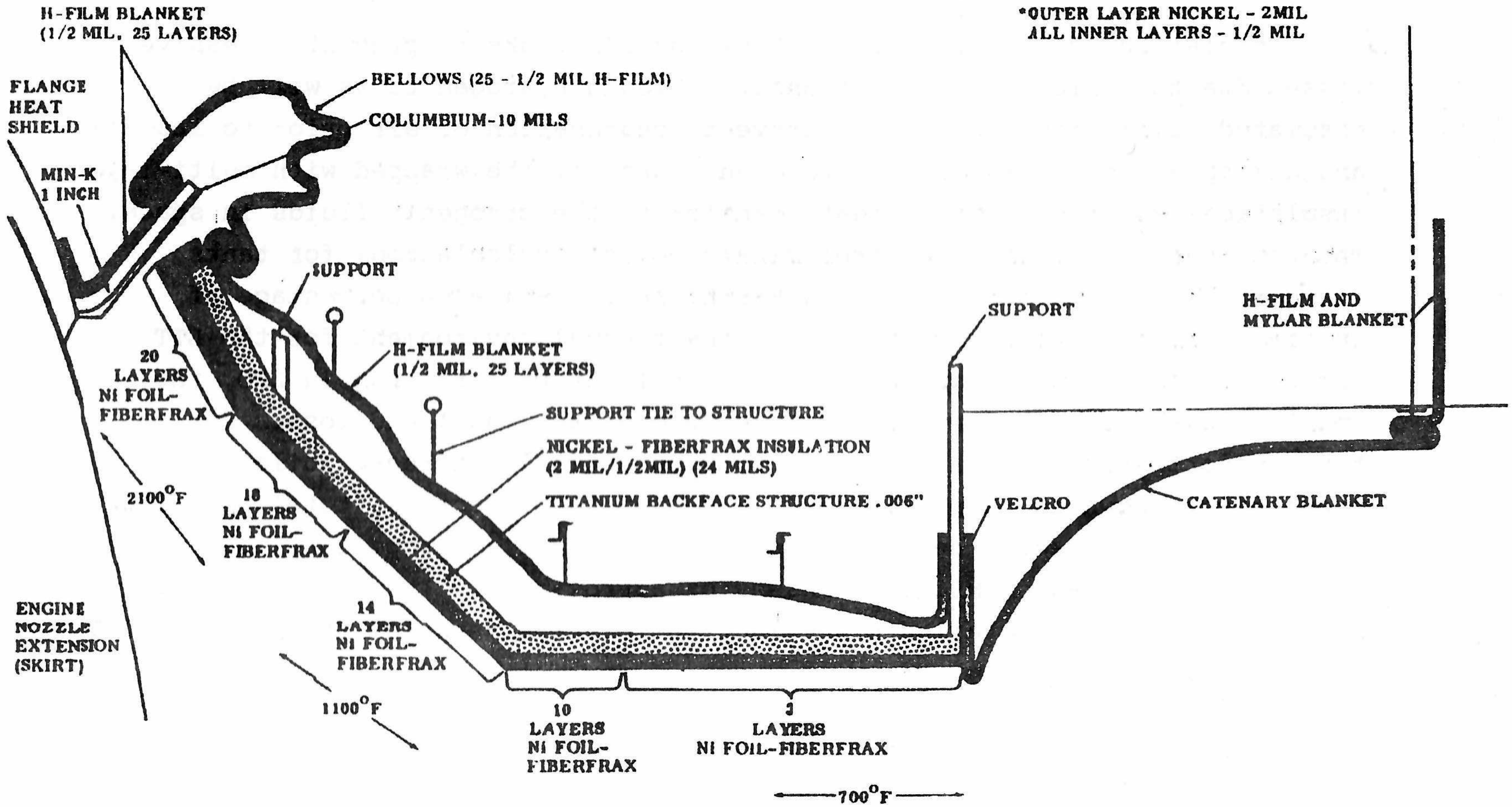


BASE HEAT SHIELD ASSESSMENT

A base heat shield (BHS) is necessary to protect the LTT vehicle from the engine exhaust during descent and particularly near landing when the exhaust plume will impinge on the lunar surface. LTT vehicle geometry, heating rate and exhaust temperature were compared with the Apollo Lunar Module descent stage to develop a concept and weight estimate for the base heat shield. The Apollo BHS cross section is shown in the sketch. Because the radiative heating rate will be much lower for the LTT due to use of a LO2/LH2 propulsion system, the BHS need not be as robust. However, the base area of the LTT is somewhat larger than the Apollo LM. A high temperature blanket insulation, such as Tailorable Advanced, Blanket Insulation (TABI) using silicon carbide fabric, placed over a lightweight titanium support structure backed with MLI to prevent soak-back to the LTT should provide sufficient protection. The estimated weight for such a BHS is 56 kg, including support structure and MLI.

LTT Thermal Analysis/Design

Base Heat Shield Assessment



	Apollo LM Descent Stage	LTT
BHS Area (sq. ft.)	134	150
Radiative heating rate (BTU/Sq. Ft.-sec.)	10-15	≈1
Maximum exhaust gas temperature	>2100°F	1580°F
BHS weight (lb)	165	

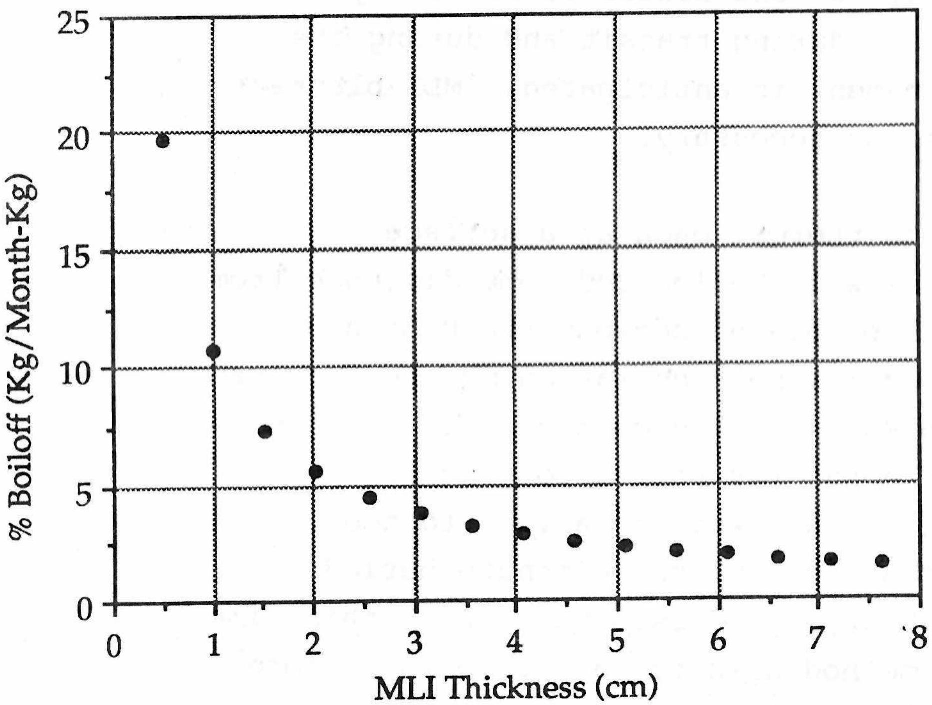
PROPELLANT BOILOFF ESTIMATES -- LEO

Insulation is required for the LH2 and LO2 tanks to prevent excessive losses due to boiloff while in transit. Liquid hydrogen tanks will be insulated with sprayable foam to prevent condensation of air prior to launch and during ascent. Oxygen and hydrogen tanks will be wrapped with multi-layer insulation (MLI) to minimize heat transfer to the cryogenic fluids in space. The accompanying graphs show preliminary boiloff calculations for tanks (larger than those for LTT) in low Earth orbit (LEO) as a percentage of initial weight over one month. While the thermal environment for the LTT during transit should result in lower propellant boiloff than in LEO, the smaller tank sizes (larger surface area/propellant mass) and losses due to tank support structure are expected to substantially increase boiloff over the predictions shown in these figures. Based on these factors, total propellant boiloff is estimated to be less than 5% of the initial propellant mass. These estimates are based on MLI thicknesses of approximately 1 cm. Future efforts should include more detailed propellant boiloff analyses for the specific LTT configuration.

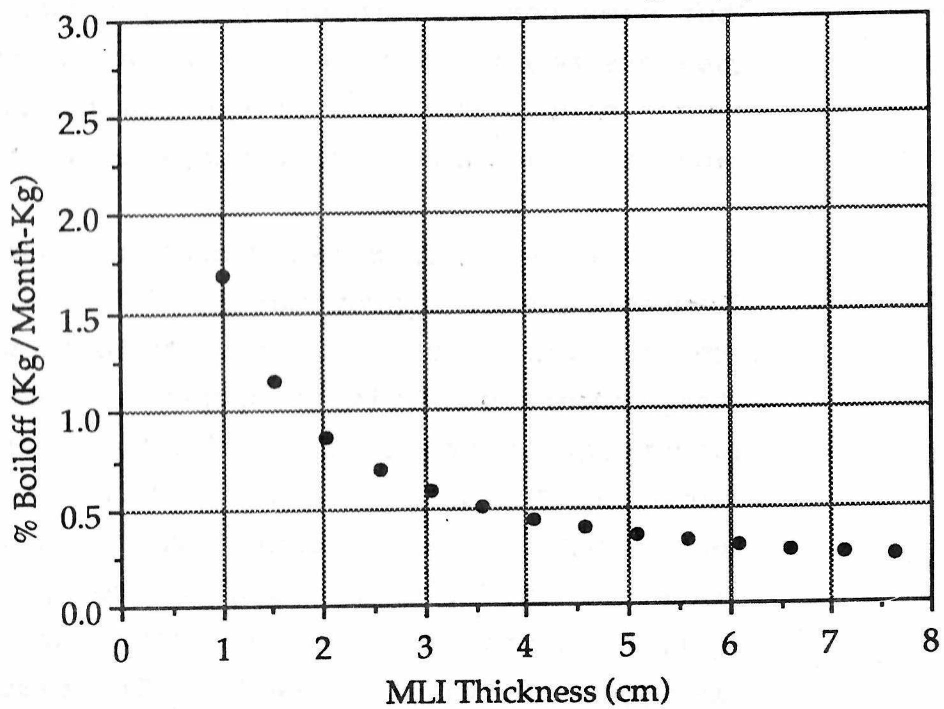
LTT Thermal Analysis/Design

Propellant Boiloff Predictions -- LEO

Boiloff estimate for a 3094 kg Hydrogen tank



Boiloff estimate for a 18570 kg oxygen tank



- LTT tanks are considerably smaller than those shown
- These predictions underestimate boiloff due to heat leak through tank support structure
- These predictions do not include losses during ascent
- Boiloff is presented as a percentage of initial mass per month (28 days)
- Considering all information, total boiloff estimate is 5% of initial mass between launch and landing

PD22

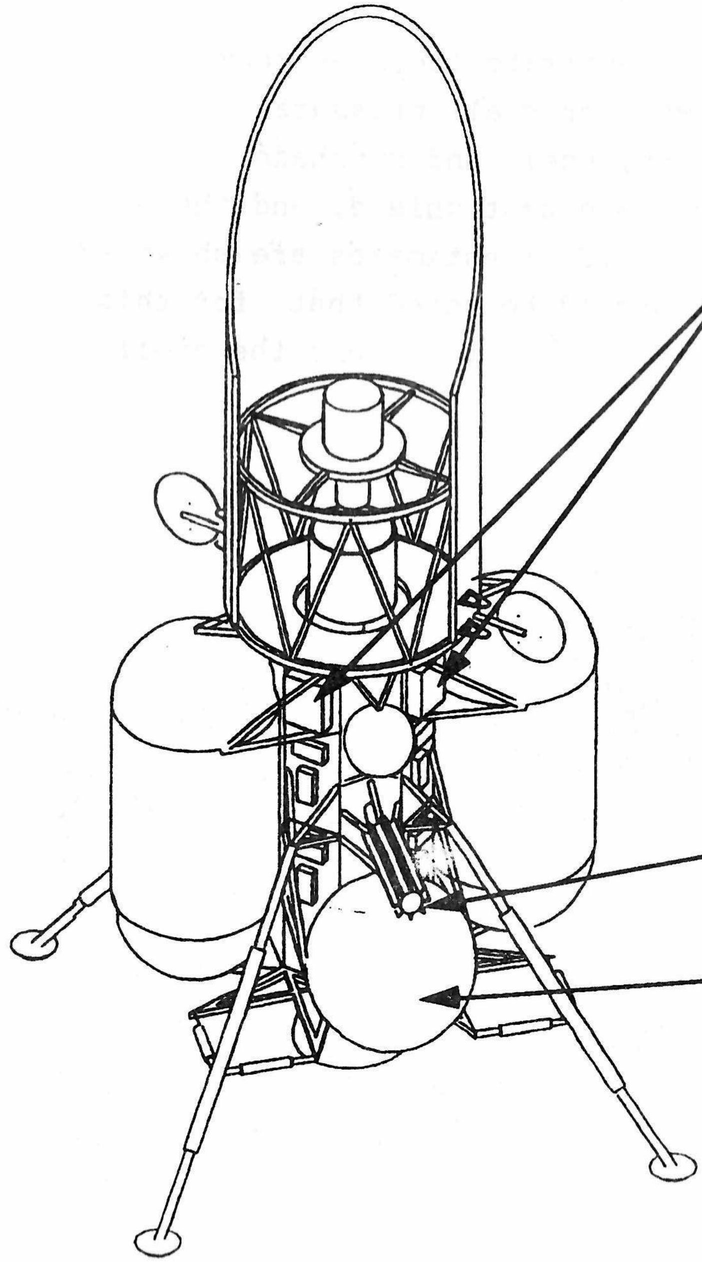
SUBSYSTEM THERMAL CONTROL

Electronic equipment will rely on passive thermal control as much as possible. The LTT does not have a centralized, active thermal control system, but each box will incorporate its own radiator and heater if necessary. Small heaters may be required for some components during transit and during the Lunar night, but no active cooling requirement is anticipated. MLI blankets and thermal control coatings will be used as necessary.

The two RTGs reject about 4,100 Watts thermal each at a surface temperature of approximately 580 K. These will be located some distance from the telescope and reflective shields will be placed between the RTGs and temperature sensitive components. Some additional insulation may be required on cryogen tanks in the vicinity of each RTG. Active cooling will be required for the RTGs before launch, but no requirement for active cooling is anticipated during ascent. The RTGs will be allowed to radiate to the interior of the shroud and to the spacecraft during the 4 minute period between launch and shroud separation. The energy dissipated during this time is approximately 0.55 kW-hr. The exact method used to cool the RTGs before launch is not specified at this time, but will be similar to that being developed for the CRAF and Cassini spacecraft scheduled for launch on Titan IV/Centaur vehicles.

LTT Thermal Analysis/Design

Subsystem Thermal Control



Electronics Components will be insulated, with a portion of the box used as a radiator. Heaters will be included as necessary for transit and night operation. Components are mounted on central core and radiators will face away from the spacecraft.

RTGs Reflective shield will be used between RTG and lander and extra insulation may be used on liquid oxygen tank.

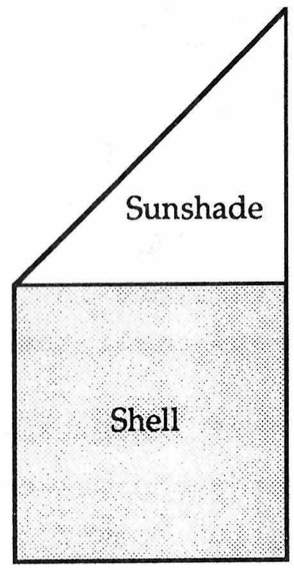
MASS ESTIMATE

The preliminary thermal control system mass estimate includes foam and MLI for two LH2 tanks, MLI for two LO2 tanks, MLI for small pressurant and propellant tanks, radiators, subsystem insulation, shell and sunshade MLI covering, reflective shields near the RTGs, base heat shield, and thermal surface treatments (paint, foil, etc.). The individual estimates are shown in the table, with the TCS total of 208.4 kg. It should be noted that, for this estimate, the sunshade is the angled portion of the LTT 'tube' and the shell is the cylindrical portion.

LTT Thermal Analysis/Design

Mass Estimate

Lander Components	Weight Estimate (kg.)
Hydrogen tank MLI	30.2
Hydrogen tank foam	20.0
Oxygen tank MLI	13.0
Pressurant tank insulation	1.2
Propellant tank insulation	1.2
Insulation for subsystems	4.5
Avionics radiator(s)	15.1
Base heat shield	56.0
Surface treatments	6.8
Misc. TCS	12.0
Lander Total =	160.0
Telescope Components	
Instrument radiator	4.5
Sunshade fabric	7.1
Shell insulation	14.7
RTG shields	6.0
Insulation for subsystems	2.3
Avionics radiator(s)	7.6
Misc. TCS	6.1
Telescope Total =	48.3
LTT Total	**208.3**



TECHNOLGOY REQUIREMENTS/OPTION

Although the proposed thermal control system for the LTT (1cm MLI = 24 layers) appears modest, MLI performance should be consiered a technology issue. The Centaur upperstage, with 2-3 layers of MLI, represents the current flight proven state-of-the-art in cryogenic MLI applications. On-going CFM Technology and Advanced Development progams underway at MSFC and LeRC indicate these insulation systems, while reasonably understood and effectve under ideal conditions, are not particularly robust. MLI blankets are sensitive to small imperfections in fabrication, assembly, lay-up and performance varies between lage and small tanks.

Cryocooler refigeration development should also be maintained as a detector thermal control option. Development issues include reliability, vibration, power requirements, and performance.

LTT Thermal Analysis/Design

Technology Requirements/Options

- **Multilayer Insulation Systems (MLI)**
  - Performance
  - Reliability

- **Refrigeration / Cryocoolers**
  - Reliability
  - Vibration
  - Power Consumption
  - Performance

SUMMARY

Analyses conducted to date indicate that the LTT thermal control concept is feasible, and the stated requirements can be met. However, additional analyses must be conducted to define the thermal control system needed for the detectors. There is some likelihood that a refrigeration system will be required for the IR CCDs during their operation. A single shell sunshade with no deployable ground shade is the recommended concept at this time. Analyses indicate that the primary mirror will be 150 K or colder for over 100 hours per Lunar cycle and the total temperature excursion of the primary mirror will be 100 K or less.

LTT Thermal Analysis/Design

Summary

- LTT thermal control requirements appear feasible
- No more than 0.25" MLI is needed on sunshade/shell
- Double shell sunshade probably not worth weight penalty unless lower temperatures are required
- Deployable ground shade provides little benefit for optics temperatures
- Primary mirror will be below 150 K for 100 to 150 hours per Lunar cycle
- Primary mirror temperature will vary between 120 K and 220 K each Lunar cycle
- Further study is needed to define thermal control system for detectors and subsystem components
- A refrigeration system may be required for IR CCDs, depending on focal array heat load
- MLI performance and cryocooler refrigeration needs further technology development.

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SECTION 4

4. SUPORTING SYSTEMS AND SUB-SYSTEMS **PAGE**

4.1 ATTITUDE CONTROL

LTT LANDING CONTROL ISSUES

The ability to meet alignment requirements for the telescope boresight, detector plane, sunshade, and the communications antenna depends on the orientation produced from landing conditions. Apollo-mission landing gears with crushable aluminum honeycomb shock-absorbers have been baselined to absorb the impact energy and provide landing stability, but the orientation errors they produce are dependent on the particular lunar terrain encountered at landing. Current data is available only for selected sites, and a wider survey of data with more resolution may not become available. Hence one of the landing control issues is the onboard site-surveying and selecting equipment. One choice is a high resolution autonomous hazard detection and avoidance system that collects and analyzes critical surface data early enough before landing to select suitable sites, and provide navigation information to an engine/thruster command system that maneuvers the lander to a suitable location. Another choice is to omit cameras and surveying equipment, and design a robust system that can withstand the worst landing sites. We have selected an on-board video camera that provides real-time feedback to Earth, and permits ground-controlled maneuver commands to select a landing location.

LTT Landing Control Issues

* Location : Knowledge of Terrain -- Maria = Aver. 1°- 2° Slope, Many rocks/boulders
  (Approx. 30° to 40° Latitude) Terra = Aver. 20° Slope, Few rocks

  Apollo and Surveyor Data *Limited* to *Specific Locations*

  No Lunar Surface Mapping Missions Currently *Funded* or *Baselined*

* Priority of Landing Accuracy Vs. Weight and Power Allocations

  Telescope/Lander Leveling vs. Landing on Flat Terrain -- Slope Tolerances?

  Real-time video feedback to earth, fuel/weight issues - best trajectories

  Autonomous Hazard Detection and Avoidance System, Development by JSC for Mars and Lunar Missions -- High Power requirements vs.detailed data

  Antenna Direction Tolerances -- Slope or Rocks Complications in Precision Landing, East/West Pointing

APOLLO LUNAR EXCURSION MODULE (LEM) CHARACTERISTICS

LTT landing velocity and slope bounds have been set at those used for Apollo 11. Both Apollo and Surveyor landing data have indicated that the lunar surface is much firmer and denser than estimated in the 1960's, and that footpad penetration should be measured in inches rather than feet. Analysis of Surveyor data produced kinetic friction coefficient estimates of 0.67, and 0.4 for Apollo 11. These estimates are used in the landing dynamics simulations.

Apollo Lunar Excursion Module (LEM) Characteristics

Apollo 11 Design

Maximum Verticle Velocities -- <= 7 ft/sec

Maximum Horizontal Velocities -- = 4 ft/sec

*(If 7 ft/sec <= Vv <= 10 ft/sec,
Vh = 40/3 - 4/3 Vv ft/sec)

Footpad Penetration -- <= 24 in.

Coeffecient of Sliding Friction = .4-1.0

Engine Cutoff -- Probes on three of the
lander legs (5.6 ft.)
Manual engine shut down

Pad or Probe Landing Mode

Slope -- <=12°

Apollo 11 Results

-Y Velocity -- 2.1 ft/sec

-X Velocity -- 1.7 ft/sec

Slide distance 18-22 in.

Footpad Penentration -- .5-1.5 in.

Coeffecient of Sliding Friction = .4

Pad Landing Mode

LEV CONTROL RECOMMENDATIONS

A gimballed, throttleable main engine provides primary pitch and yaw control, and can accomodate lateral errors in center of gravity (cg) location and various distances between the gimbal point and the cg. The reaction control system configuration uses twelve thrusters in four triads. Four of the thrusters point in the aft direction to provide velocity control during the landing phase. The other eight thrusters are mounted in 90 degree pairs at locations 90 degrees apart around the LTT circumference in such a manner to avoid plume impingement on the landing legs. These thrusters provide pitch, yaw, and roll control during coast, and maneuver and roll control during main engine burns. The RCS also provides roll control, and supplemental pitch and yaw control, during landing.

LEV CONTROL RECOMMENDATIONS

* MAIN ENGINE
    - THROTTLEABLE (16,500 TO 4,000 LBF)
    - GIMBAL CAPABILITY (6 DEGREES)
    - USE FOR MIDCOURSE CORRECTIONS AND LUNAR LANDING

* REACTION CONTROL SYSTEM
    - MINIMUM THRUST OF 25 LBF
    - MINIMUM CONFIGURATION OF 2 CLUSTERS OF THREE THRUSTERS
    - USE FOR:
        # COAST ATTITUDE CONTROL (0.5 DEG DB: 13 KG/3 DAYS)
        # MANEUVERS (0.1 DEG/S: 0.4 KG/MANEUVER)
        # TRIM DURING DESCENT
        # ROLL DURING ALL POWERED FLIGHT
        # DAMP SEPARATION TRANSIENTS (1 DEG/S P,Y: 0.6 KG)
        # ORIENTATION PRIOR TO MAIN ENGINE BURN AT LANDING

LTT RCS PROPELLANT ESTIMATES

Using preliminary mass and inertia estimates, and a specific impulse of 180 seconds, RCS propellant from TLI stage separation to lunar touchdown has been estimated. These estimates include stabilization after tipoff, attitude acquisition maneuvers, a maximum three-day coast, and attitude control during descent and touchdown.

LTT RCS Propellant Req'd
(I_{SP} = 180 Sec)

Damp Separation Transients (1deg/sec 2 Axes)	.6 kg
Maneuver For Attitude Update (2 Stars)	.8
Maneuver To Preferred Attitude	.4
Maneuver For Attitude Update (2 Stars)	.8
Maneuver for Midcourse #1	.4
Maneuver to Preferred Attitude	.4
Maneuver For Attitude Update (2 Stars)	.8
Maneuver For Midcourse #2	.4
Maneuver To Preferred Attitude	.4
Maneuver For Attitude Update (2 Stars)	.8
Coast ATT Control 3 Days	13.0
Maneuver To Descent Burn Attitude	.4
Roll Control During Midcourse #1 (.1% of ΔV)	.2
Roll Control During Midcourse #2 (.1% of ΔV)	.2
Control During Decent Burn (.1% of ΔV)	5.5
Control During Touchdown Burn (.5% of ΔV)	2.5
	30.6 kg
25% Contingency (Control)	7.6
	37.2 kg

•RCS Requirements Will Change When Mass/Inertias Are Updated

LBB, PD12
8/7/91

GN&C EQUIPMENT FOR LUNAR LANDER ON LTT

The guidance, navigation, and control system must provide for initial stabilization of the LTT after separation from the TLI stage, attitude hold during coast and burns, attitude updates, state vector updates, maneuvers for attitude updates and possibly thermal control, mid-course corrections, and descent burns including retargeting, throttling and touchdown. Equipment required includes an inertial measurement unit, star trackers and sun sensors, control electronics to interface with the RCS and RL10 actuators, radar for altitude and velocity determination with respect to the lunar surface, and a video system for site observation prior to touchdown. Once on the lunar surface the control system must accommodate leveling and reorientation of the telescope to arcminutes accuracy to meet science requirements and to orient the sunshade to protect the optics. The CCD located at the focal plane of the telescope must be rotated to a fixed position within arcseconds accuracy to ensure that astronomical images move across the CCD appropriately.

GN&C for Lunar Lander for Lunar Telescope

Requirement

- o Provide GN&C from TLI to lunar orbit
- o Provide GN&C for deorbit/braking & landing on unprepared surface
- o Assumes State Vector Updates from ground

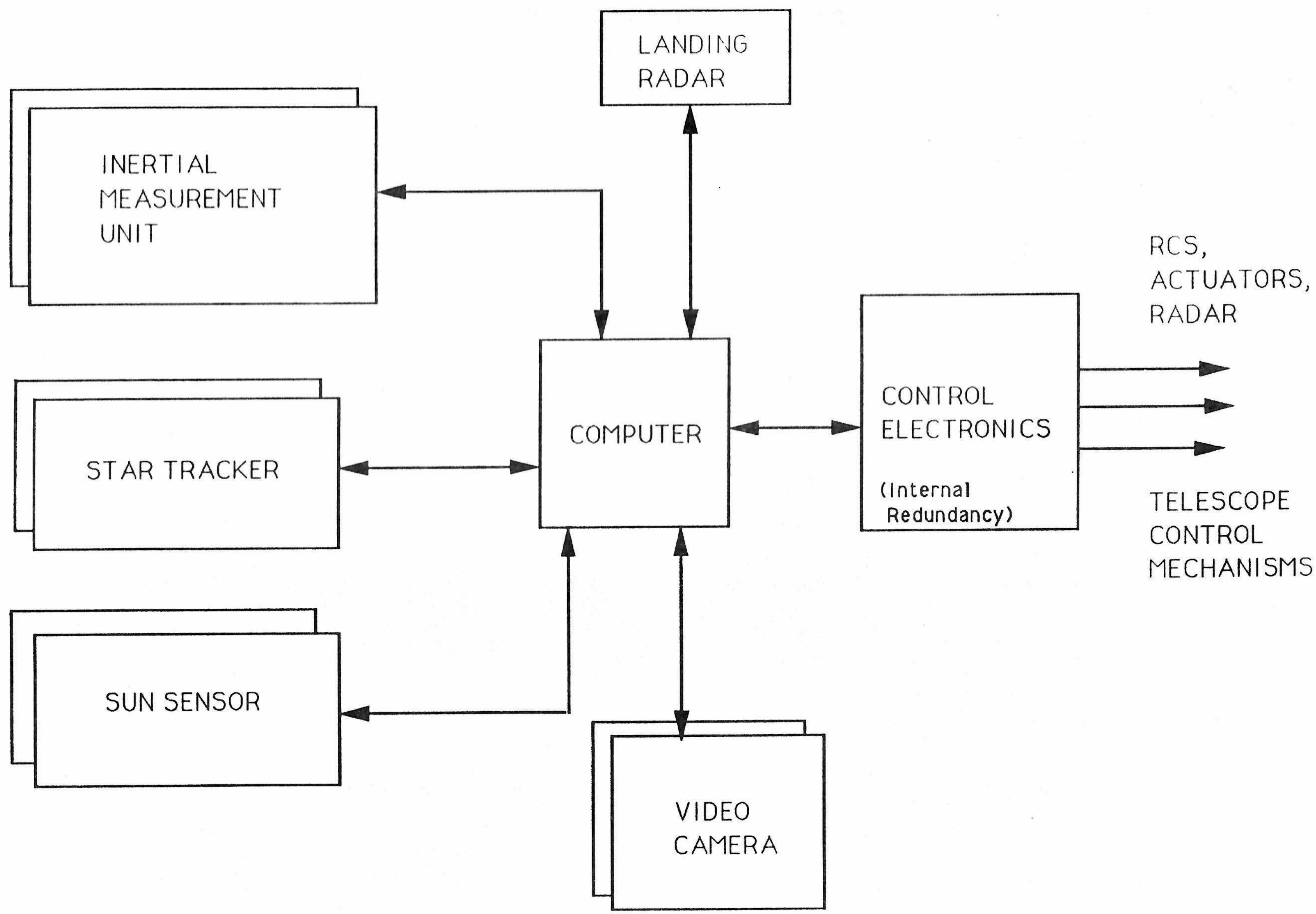
Recommend	Wt. each (Kg)	Total wt. (Kg)	Power reqmt (W) (each)
2 - IMU	3.8	7.6	27/40
2 - Star tracker and sun shield	4.3	8.6	10
2 - Sun sensor and electronics	1.4	2.8	3
1 - Landing radar	38.6	38.6	123
1 - Control electronics	40.0	40.0	70
2 - Video camera	7.0	14.0	10
Miscellaneous	5.0	5.0	
		Total wt. 116.6 Kg	

Requires: Throttleable main engine, RCS for maneuvers, coast attitude control & roll control during main engine burns, midcourse and landing

GN&C BLOCK DIAGRAM

Two inertial measurement units, star trackers, and sun sensors provide attitude information for stabilization, maneuvering, descent, and landing. Two video cameras and the landing radar provide site selection and navigational information for landing. Control electronics provides an interface to the RCS, actuators, and the RL10 for maneuvers and descent, and to reorient the telescope boresight, detector array, and sunshade for telescope use after landing. Limited redundancy is incorporated to ensure that credible single point failures will not impair the mission.

GN&C BLOCK DIAGRAM

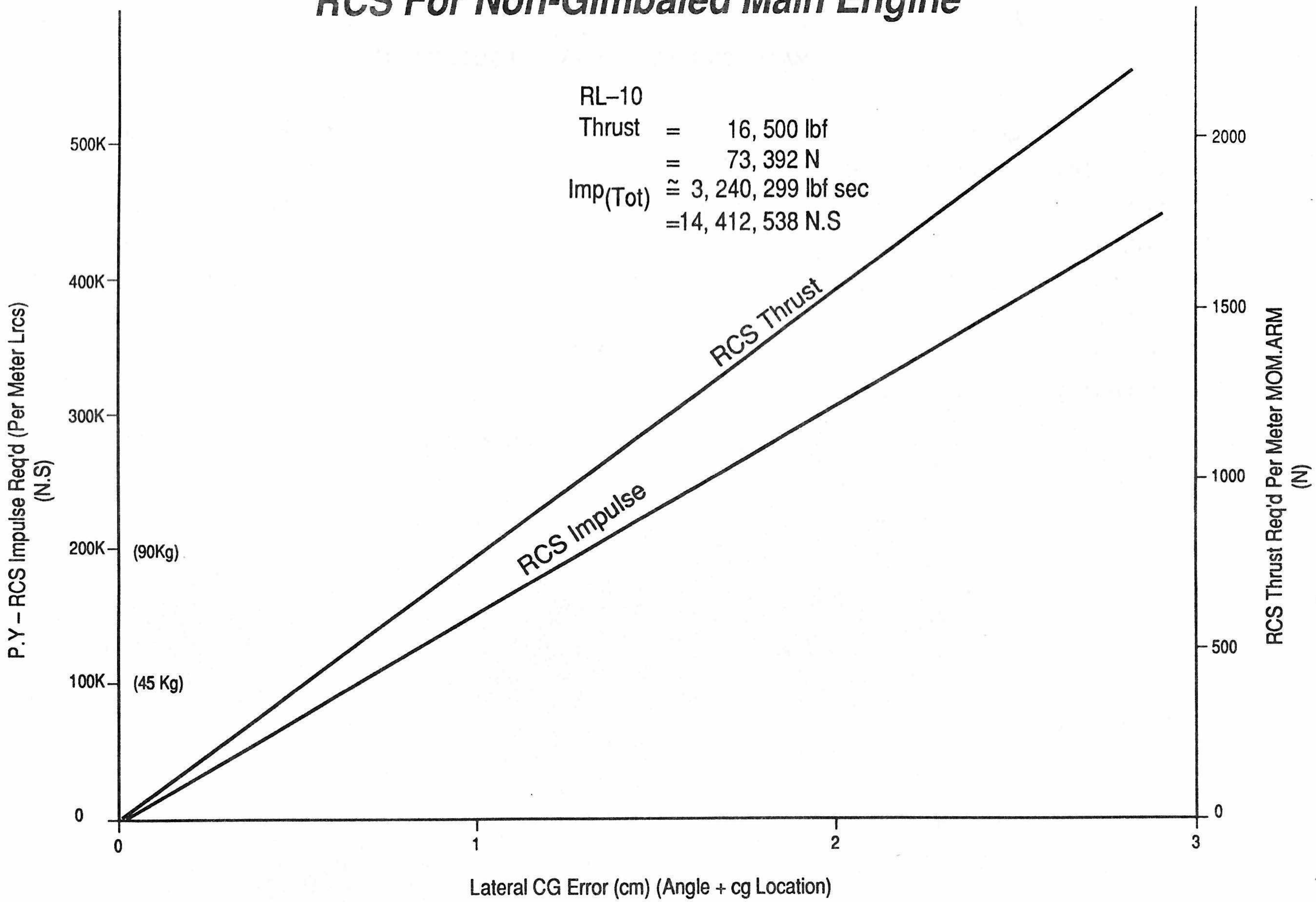


RCS FOR NON-GIMBALED MAIN ENGINE

To reduce the overall length of the LTT and to avoid engine clearance issues in the configuration, the use of a non-gimbaled, embedded main engine was discussed. Removing the gimbal capability of the main engine will produce torques on the lander that must be opposed by the RCS system to maintain attitude control. The facing chart shows the RCS impulse and thrust required as a function of lateral distance between the vehicle cg and the main engine thrust vector, given a one meter moment arm between the cg and the RCS thrusters. (Note that thrust levels for a particular thruster/cg configuration can be found by dividing these numbers by the actual moment arm.) Gimbaling the main engine will significantly reduce the thrust and impulse requirements for the reaction control system.

1-2648-1-61D

RCS For Non-Gimbaled Main Engine



LBB/PD12
25 Jul 91

MAIN ENGINE GIMBAL REQUIREMENT

The facing chart addresses the issue of embedding the engine in the vehicle. Embedding the engine moves the gimbal point closer to the vehicle cg, requiring larger gimbal angles and increasing the clearance necessary for engine motion. These gimbal angles also increase as the lateral cg error (e) increases. A six degree gimbal capability has been baselined that should accomodate configuration changes. It is recommended that the main engine not be embedded in the vehicle, to ensure a reasonable vertical distance between gimbal point and vehicle cg.

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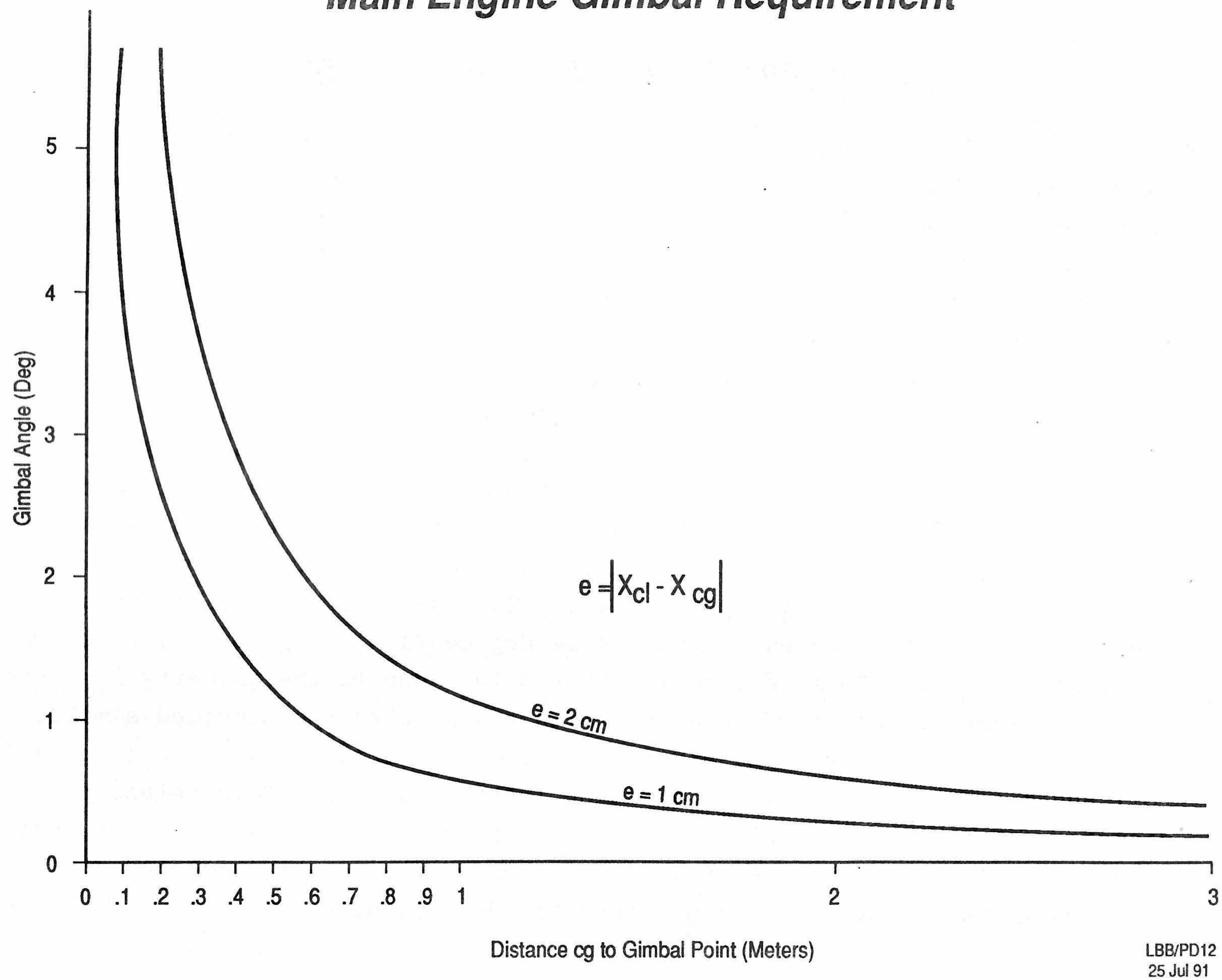
Main Engine Gimbal Requirement



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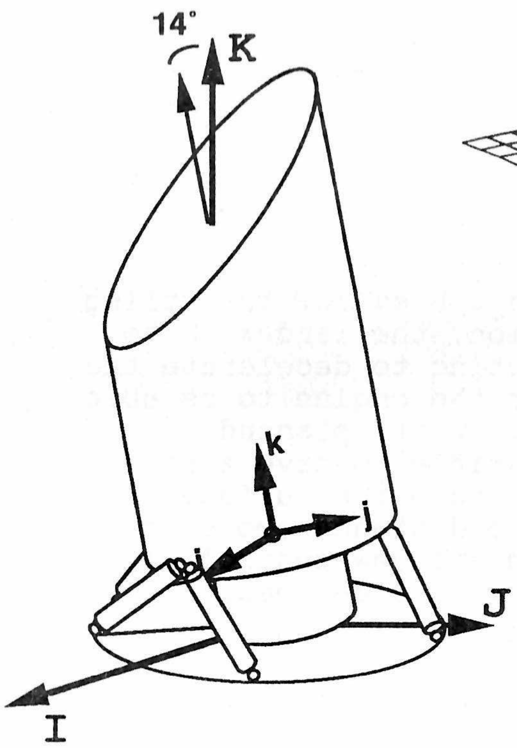
25 Jul 91

HEXAPOD ROLL ANGLES FOR 2M TELESCOPE

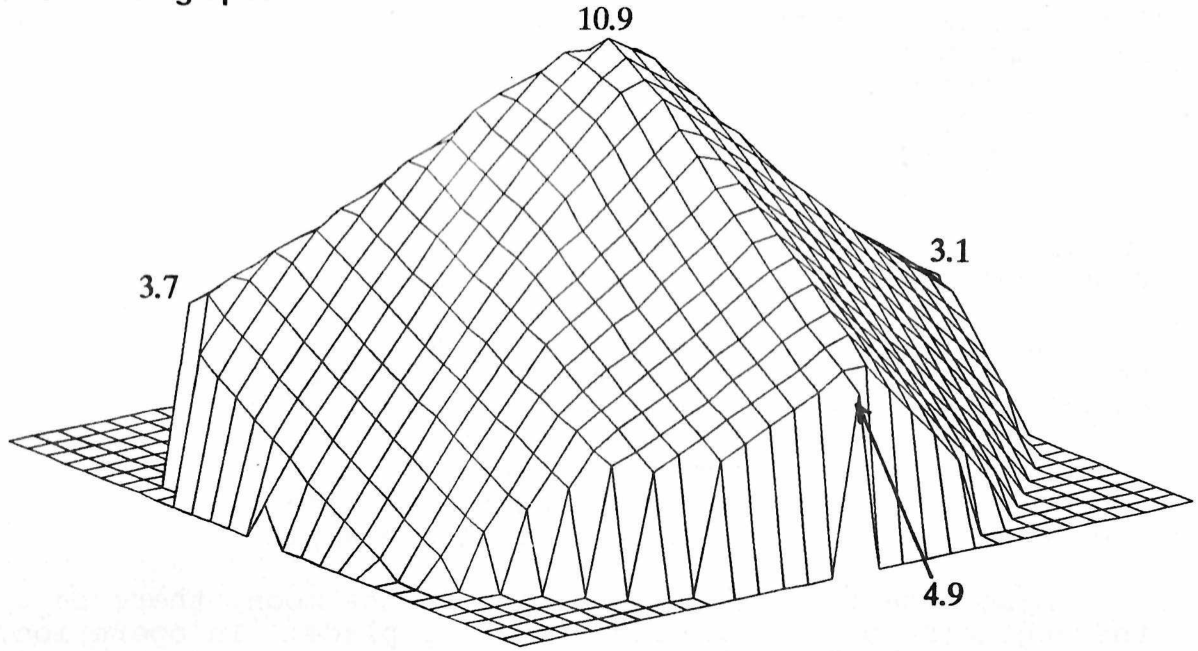
Telescope orientation errors occur not only due to attitude errors immediately before touchdown, but also due to lunar surface variations and final landing gear configurations after crushing. Preliminary assessments indicate that both azimuth and elevation corrections will be necessary to align the boresight to zenith, and roll correction will be required for sunshade alignment and gross detector plane rotation. Options for pointing mechanisms include various azimuth-elevation gimbal mounts, either secondary or primary strut adjustments in the lander legs, or hexapod mounts such as those considered on a 4 m telescope previously studied. Kinematic analysis of hexapod mounts for a 2 m telescope using the AXAF mirror assembly linear actuators provides a 14 degree half-angle viewing space in azimuth-elevation, and roll angles across that circular viewing space that are maximum at 10.9 degrees along the telescope optical axis. The maximum roll angles decrease as the optical axis is rotated out to the 14 degree viewing space rim, as shown by the surface plot. (This surface is "triangular" due to the geometry from three pairs of linear actuators.) Analysis with a 4 m telescope provided similar limits on azimuth, elevation, and roll, indicating that the viewing space and roll angle is more sensitive to actuator length and stroke than telescope diameter. Hence limited roll motion can be provided by a hexapod mount, but assessment of these and other pointing mechanisms cannot be completed until the roll requirements due to landing errors are determined.

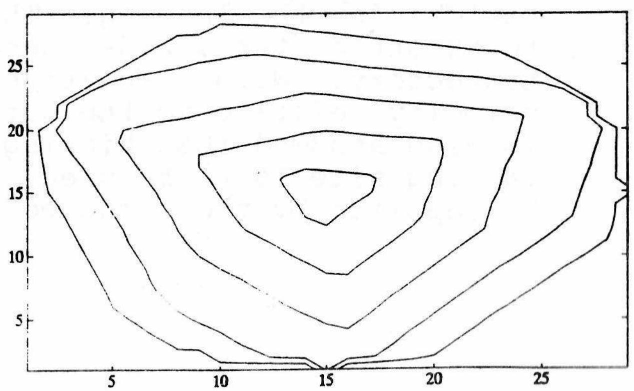
ROLL ANGLES FOR 2M TELESCOPE

- 14° half-angle viewing space provided by hexapods
- Roll angle is maximum at center of viewing space



Roll Angles in Degrees





4.2 PROPULSION/RCS

LTT PROPULSION SYSTEM

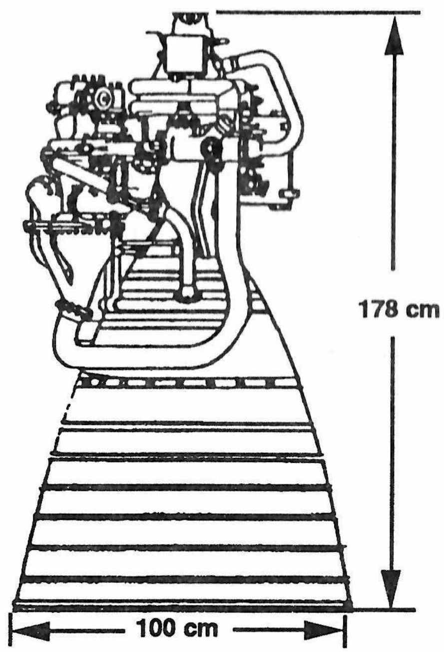
The desire to maximize any telescope on the surface of the moon means that the subsystem weights associated with its delivery must be minimized. To carry out that mandate, the propulsion system wet weight must be at a minimum. Intuitively, this means use of a LOX/LH_2 cryogenic system with its high performance capability. A cryogenic system also minimizes propellant loading and has the capability of reducing the pressurization system weight further by using autogenous pressurization. A trade with earth storable propellant contrasted with the cryogenic propellants is presented for comparison.

To limit the system mass, the assumption of no redundancy in the design was assumed. This is an acceptable assumption as the telescope is an unmanned asset and is also flown on an expendable launch vehicle.

As with any experiment or payload, there is a desire to expend funds on the payload rather than developing a new spacecraft system. Since there is currently no existing lander being flown or under development, a clean sheet approach was assumed for this design. This also gives a better idea of the maximum cost of that system. However, this cost can be mitigated somewhat by employing previously developed components and filling in with those items that are readily available: tank sets, for example.

Since the telescope is to land on the moon, there is a need for throttling the engine(s) as the landing is taking place. In operation, the lander is to come in on a predetermined trajectory, continuously thrusting to decelerate the payload for the landing. The guidance requirement is for the engine to be able to throttle over a wide range to adjust the flight to follow its planned trajectory. With a relatively constant thrust, it is possible to have zero velocity relative to the surface of the moon as the legs touch the surface. It is also assumed that biasing the engine cutoff some finite distance above the landing site is to be used. It is assumed that the pitch and yaw control will be supplied by the gimbaled main engine, on the order of 4 to 6 degrees.

RL10A-3-3A Performance Characteristics

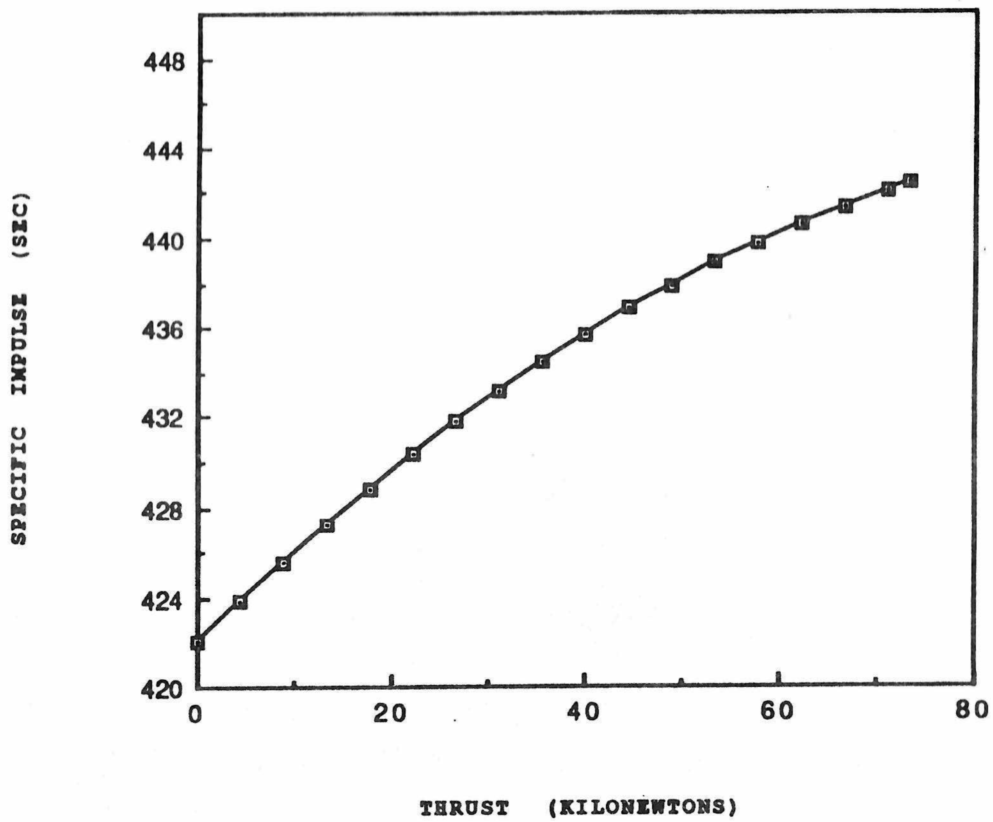


Vacuum Thrust (N)	73,500
Nominal Vacuum Specific Impulse (sec)	444
Expansion Ratio	61
Restarts	
Low NPSP (kPA) (ox./fuel)	200/296
Autogenous Tank Pressurization	Fuel Only

LTT PROPULSION SYSTEM (Cont'd)
EFFICIENCY LOSS DUE TO THROTTLING

The engine first assumed for the main propulsion system of the lander design was the RL10A-3-3A, a LOX/LH_2 engine manufactured by Pratt & Whitney. This engine has been operational since the early 1960's and had the approximate thrust level required for the lander application. It has also been demonstrated as being capable of 10 to 1 continuous throttling over the range of interest, and currently had been selected to demonstrate a 4 or 5 to 1 throttle capability for the Single Stage To Orbit vehicle. The requirements for thw LTT are within the 4:1 range. This engine also supports autogenous tank pressurization, but for the fuel side only. That is, the hot hydrogen gas is routed back to the fuel tank for pressurization. However, it was recognized that the technology used on the SSME could have been applied here for total autogenous pressurization. This was considered and rejected as being an expense that was not required. General characteristics were taken from literature and were catalogued. It should be noted that the nominal vacuum specific impulse listed is for the rated vacuum thrust. When the engine was throttled, there was a corresponding drop in specific impulse. This was assumed to follow the curve as shown in the accompanying figure. Gimbal movement capability of the engine was listed as +/- 4 degrees in a square pattern.

EFFICIENCY LOSS DUE TO THROTTLING
RL10 PERFORMANCE



PROPELLANT BUDGET BREAKOUT

A propellant loading of 3,231 kg and a 5 percent ullage fraction were used to size the tank dimensions. Estimated losses due to boiloff and residuals reduced the usable propellant to 3,071 kg, including 46 kg of reserve. Although the engine mixture ratio is 5 to 1, the effective loading mixture ratio, due to boiloff on a one for one ratio for LOX/LH_2 respectively, is 4.66 to 1. No attempt was made at this point to include engine start/stop losses in the analysis.

PROPELLANT BUDGET BREAKOUT
LTT LANDER: TITAN IV-CENTAUR

MAIN PROPELLANTS:	TOTAL	LH2	LOX	
USABLE	3024	504	2520	
BOILOFF	97	48	48	
RESERVE	46	8	38	
RESIDUALS	64	11	54	
TOTAL	3231	571	2660	KG

EFFECTIVE MIXTURE RATIO = 4.66

GBK/PD13
11/08/91

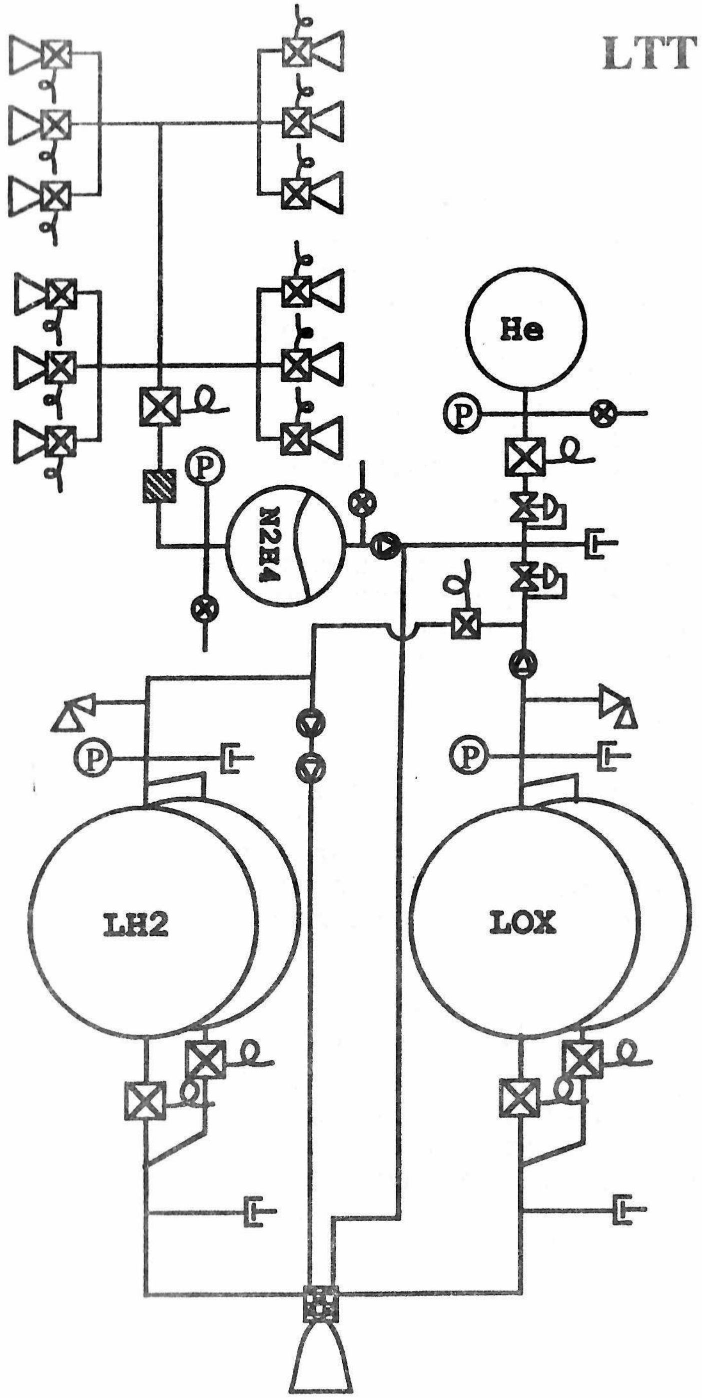
LTT LANDER PROPULSION SYSTEM

With the engine characteristics in hand, the basic schematic of the lander was developed. Packaging dictated that four tanks, two fuel and two oxidizer, be employed. The diameter of the tanks was held constant allowing the oxygen tank to be spherical while the fuel tank was elongated by the addition of a cylindrical section. It was assumed that the manufacturing of the spherical bulkheads would be cheaper than using elliptical tanks, and that with equal diameter, a cost savings might also be realized. The diameter for the equal diameter tanks was approximately 132 cm and the fuel tank overall length was 349 cm. If the fuel tank diameter was increased to approximately 140 cm, the total height was reduced correspondingly 28 cm.

Tank isolation valves allowed loading each propellant tank separately through quick disconnect fill and drain valves. The isolation valves also are needed during operation to prevent pressurization gases from being ingested through the engine. This condition can occur if one tank is depleted before the other due to imbalance in the flow paths or in the pressurization of the tanks. Sensors in the tank are required for this operation.

It was assumed that the loading pressure was as low as possible allowing the boiloff to pressurize to 0.34 MN/m^2 operating conditions during the transit to the moon. This is a valid assumption as the first firing is scheduled for the midcourse correction which takes place some 3 days into the mission. However, to ensure that the tank was at the proper pressure, a pre-press capability was provided. To guard against over pressurization during boiloff, and subsequent tank rupture, relief valves are placed in the system. It was recognized that a liquid/vapor separator may be necessary for relief operation or that the propellants may need to be settled prior and during venting: this issue was not addressed beyond its recognition during the study. Since the pressure build-up in the fuel tank could influence the engine initial operation, check valves were used to isolate the two. This still permitted the autogenous pressurization of the tanks by the engine, once any over pressure drops below operating pressures.

LTT LANDER PROPULSION SYSTEM



	MASS STATEMENT (KG)			
	MAIN PROP		RCS	
EQUIPMENT	LIQUID	GAS		TOTAL
TANK	159.8	30.5	6.4	
FILL & DRAIN VALVES	18.2	0.9	0.9	
ISOLATION VALVE	9.0	1.4	2.8	
PRESSURE TRANSDUCERS	0.9	0.5	0.5	
PRESSURE REGULATOR		2.7	2.7	
RELIEF DEVICE		6.4		
CHECK VALVES		4.6		
PROPELLANT FILTER			0.5	
ENGINES/THRUSTERS	165.9		27.3	
TVC	26.4			
MISCELLANEOUS	56.8	4.5	4.1	
TOTAL DRY WEIGHT	437.0	51.5	45.2	533.7
PROPELLANT: USABLE	3070.9		37.3	3108.2
RESIDUAL/BOILOFF	160.9		1.4	162.3
PRESSURANT		4.5		4.5
SUBSYSTEM WEIGHT				3808.7

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11/20/91

LTT LANDER PROPULSION SYSTEM (Cont'd)

The oxidizer pressurization and hydrogen pre-press was supplied from a high pressure helium bottle, initially charged to approximately 27.6 MN/m^2. Since it also supplied the pressurization for the regulated monopropellant hydrazine RCS system and the main engine pneumatic system, a two stage regulation scheme was employed. The first supplied pressure to the RCS and main engine at 3.1 MN/m^2, while the second supplied the oxygen tank 0.34 MN/m^2 helium. A valve isolated the helium in the high pressure tank until orbit has been achieved. Residual pressure in the 60 cm outside diameter helium tank was based on a final pressure of 3.8 MN/m^2. Room temperature was assumed for the first approximation mass calculations.

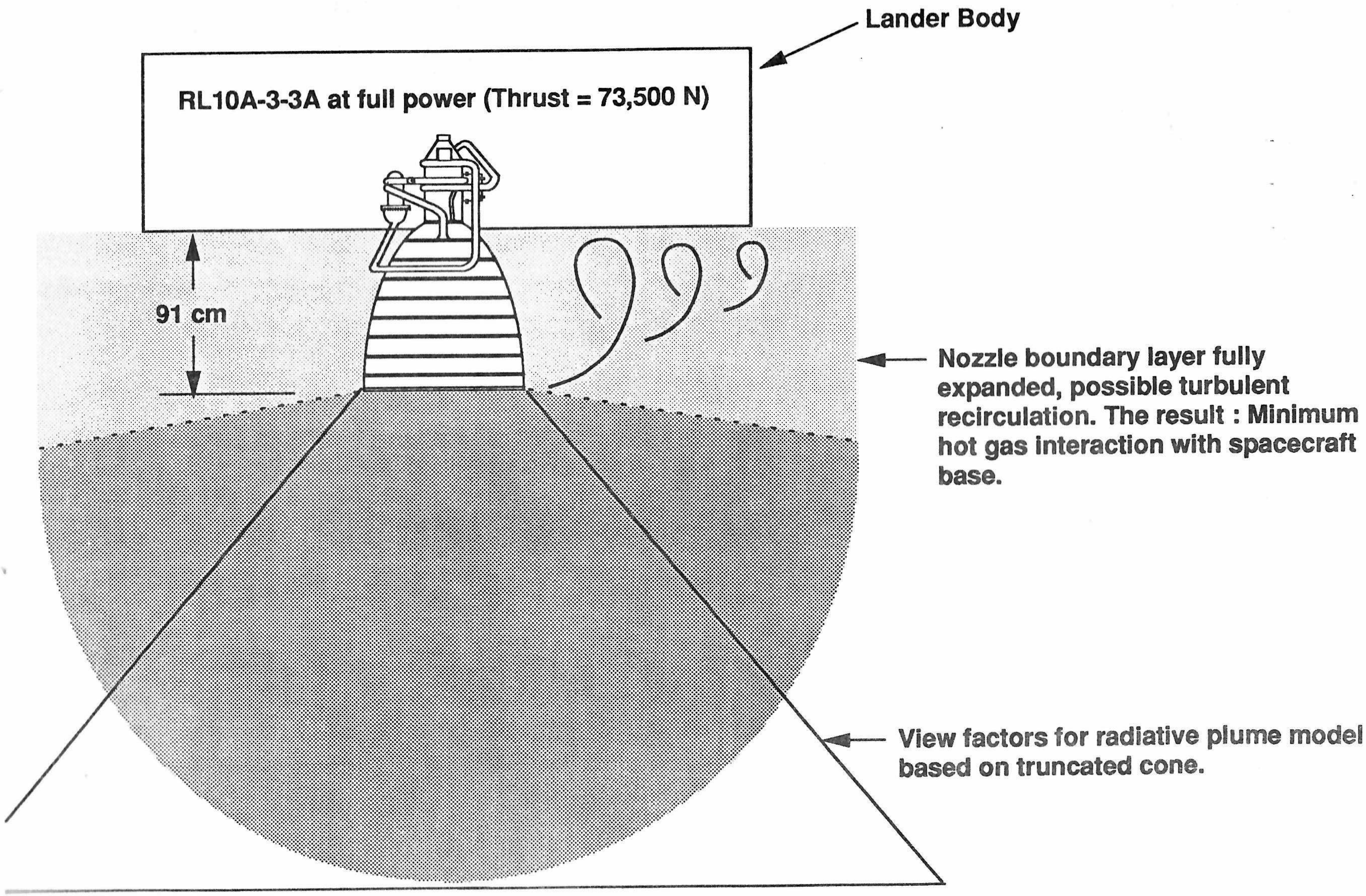
Fill and drain valves bracket the hydrazine tank allowing the system to be loaded at the hazardous handling facility. The tank, as configured for the EXOSAT program, had a propellant acquisition system consisting of an elastomeric diaphragm. This tank was assumed for the RCS propellant tank in this design. The tank was a sphere, 49 cm in outside diameter designed with a burst pressure of 5.2 MN/m^2.

The RCS thruster system is required to provide the roll control during main engine burn and trim in the pitch/yaw directions after separation and at the end of the burns. There is also some positioning required as the landing is effected. The thrust level for the single seated thruster is unknown at this time, but is assumed that it will be in the 111 Newton class. The mass statement reflects using this class of thruster.

The one thing needing additional definition is the loading of the cryogenics and the design of the accompanying plumbing system through the shroud. It should be included in future definition studies. Base heating is also an area of concern. The facing chart indicates that heat from the engine exhaust may impinge upon the lander body, depending upon the nozzel distance from the lander structure and the thrust level of the engine. The gimbal angle may also be a factor in the amount of heating.

Lunar Transit Telescope

Base Heating : First Order Approximation



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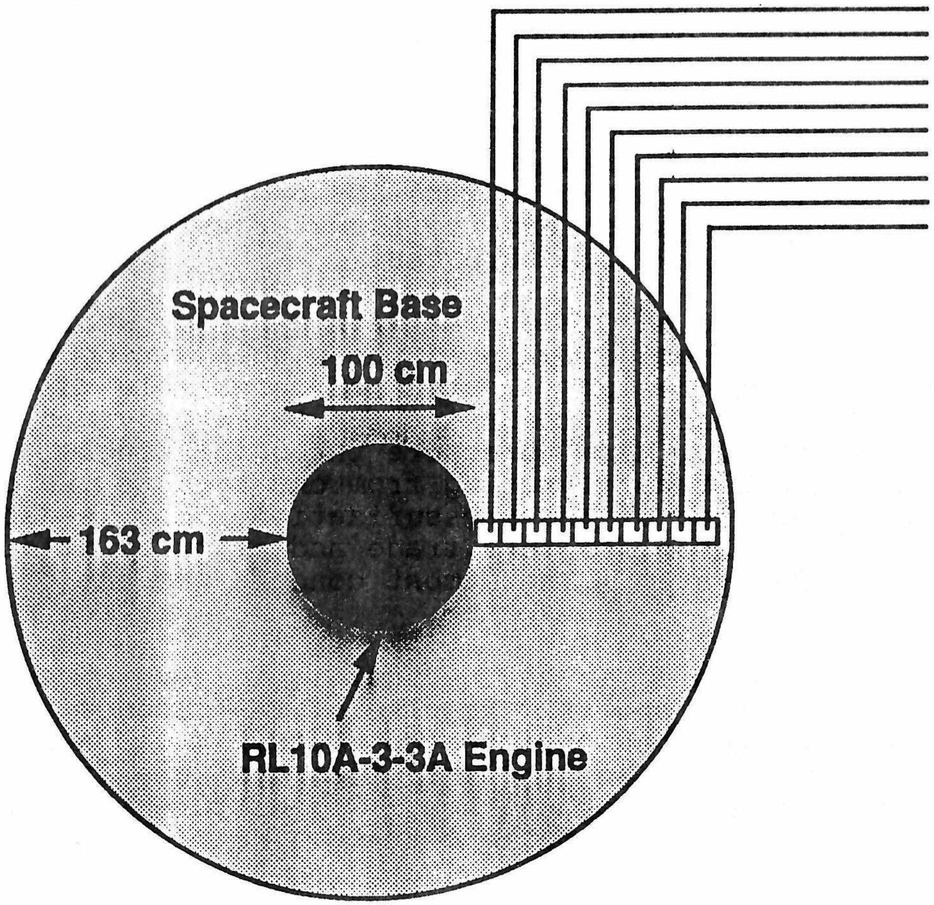
LTT BASE HEATING

A rough first order approximation calculation was made to determine the magnitude of the base heating from the main engine exhaust for input to the base thermal control design. The geometry was determined from the layout drawings and integrated into a computer program. The results of the modeled system indicated that the bulk of the exhaust products mass was within a cone with a half angle of 75 degrees. The recirculated mass in the base region was determined to be negligible, and therefore ignored. The greatest heat was due to plume radiation from the plume cone. Calculations for two conditions were made, full thrust and 25 percent thrust, to understand the magnitude of the flux as the engine was throttled. The data was presented as a function of the radial distance along the base of the vehicle.

Lunar Transit Telescope

Base Heating



Thermal Heating due to Radiation *

Full Thrust	25% Thrust
-6355	-4627
-6292	-4583
-6229	-4536
-6166	-4489
-6100	-4442
-6034	-4395
-5968	-4347
-5901	-4300
-5835	-4249
-5769	-4202

units: $\frac{\text{watts}}{\text{m}^2}$

Ref:
* JANNAF equations for plume radiation heat transfer

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PROPULSION SYSTEM COMPARISONS

During the conduct of the design, a trade study was made to determine whether a cryogenic or earth storable propulsion system should be used. The assumptions used were as follows. The total impulse of the LOX/LH_2 system that was required to perform the mission, including reserve, was calculated with an average specific impulse. An equivalent propellant weight of the NTO/MMH system was then calculated based on the earth storable engine specific impulse. The RCS total impulse was also calculated and the resulting propellant weight added to the main engine requirements. With this total, the earth storable concepts total propellants were estimated. Two systems were compared with the cryogenic baseline. One used an engine system that had a simple, but relatively heavy, blowdown pressurization while the other assumed that a pump fed engine was available. Both had the RCS propellant in common tankage with the main propulsion engine(s) and the thruster assumed for the RCS function was the Shuttle vernier 111 Newton thruster.

The pressure fed system assumed an OMS engine with a steady state specific impulse of 313 seconds. The basic design followed the single string philosophy and the schematic resembled the OMS design. Instead of resealable valves, pyrotechnic valves were used. A pair of pressurization tanks provided the equal volume propellant tanks the 2.1 MN/m^2 pressure through a single regulator. To keep the propellant from migrating from one tank to the other, check valves were located in each leg of the pressurization lines leading to the tanks. Packaging was not an issue for this trade and as such, it did not enter into the refinement of the system. The utmost concern was one of total subsystem weight minimization.

The pump fed engine system assumed an engine that is in development and was predicted to have a steady state specific impulse of 340 seconds. Since the thrust level is not as large as the other two concepts, it was assumed that two would be required. This could also be thought of as a single large engine weighing an equivalent amount as a pair of engines. The mixture ratio used was 2 to 1 and will require different size propellant tanks. As

PROPULSION SYSTEM COMPARISONS:

		EARTH STORABLE (NTO/MMH)	
	LOX/LH2	PRESSURE	PUMP FED
DRY WEIGHT (KG)	448	854	334
USABLE PROPELLAT (KG)	3256	4612	4162
TOTAL SUBSYSTEM WEIGHT (KG)	3880	5589	4586
SPECIFIC IMPULSE (SEC)	440	313	340
MIXTURE RATIO	5:1	1.65:1	2:1

LTT PROPULSION SYSTEM COMPARISONS (Cont'd)

with the pressure fed system, the basic schematic resembles the OMS. The major difference between the two is the pressures that the propellant tanks operate. The pump fed system assumed that the tanks were at 0.34 MN/m^2. This translated into a smaller quantity of helium needed with a resulting smaller helium tank. It also made the propellant tanks lighter.

A comparison of the three systems shows that the pump fed storable system is the lightest while the total system weight favors the cryogenic system. This is essentially the same results found in "Integrated Hydrogen/Oxygen Technology Applied to Auxiliary Propulsion Systems, " September 1990, under NASA Lewis Research Center contract number NAS3-25643, performed by Rockwell International. There are some additional considerations besides weight that are of interest. While the storable is heavier, there is no boiloff to consider with its associated problems. The bulk density is greater and therefore the propellant can be packaged more efficiently. The storable is a simpler system which should, but not necessarily, translates into a more economical system. One of the major cost drivers is the development of an engine with throttle capability.

The cryogenic system on the other hand has higher performance which produces a lower total system weight. The thermal control lends itself to a passive application technique that requires no energy expenditure. An autogenous pressurization system will minimize the pressurization required and the corresponding system tank weight. The engine system has already demonstrated throttle capability. But there is a question of the operating range of the corresponding pump fed earth storable engine. This is not to imply that a pump fed storable system cannot be made to work, it simply indicates that the engines being developed are not being done with throttling in mind. And finally, the propellants are non toxic.

PROPULSION SYSTEM COMPARISON

CRYOGENIC

- HIGHER PERFORMANCE = LOWER PROPULSION SYSTEM WEIGHT (30 % LESS)
- NO HEATER POWER REQUIRED FOR MAIN PROPELLANT CONDITIONING
- AUTOGENOUS PRESSURIZATION REDUCES PRESSURIZATION REQUIRED
- PROPELLANT TANK PRESSURES ARE LOWER, THEREFORE TANK WEIGHT LESS
- EXISTING ENGINE WITH DEMONSTRATED THROTTLE CAPABILITY
- PROPELLANTS ARE NON TOXIC: HANDLING MORE SIMPLE

STORABLE

- BOILOFF NOT INCURRED
- BULK DENSITY GREATER; THEREFORE, PROPELLANT VOLUME SMALLER (2.5:1)
- MORE SIMPLE AND, THEREFORE, CHEAPER MAIN ENGINE

GBK/PD13
08/23/91

4.3 COMMUNICATIONS AND DATA HANDLING

LTT CONSIDERATIONS FOR LUNAR COMMUNICATIONS

Early in the LTT study it was decided that one set of subsystems could be designed to serve the LTT during transit to the moon, while soft landing on the moon and support telescope operations for 5 to 10 years after landing on the moon. During transit to the moon the data rates are expected to be low, a few kbps for monitoring and housekeeping. During actual landing on the moon the computer operations and data rates could be rather high depending on the systems used, but they are not expected to exceed the telescope operating requirements.

The LTT operating requirements after landing on the moon drive the design of the communications and data handling system. Location on the lunar surface will determine whether the communications are direct line-of-sight or require some sort of relay satellite. Fortunately, most of the sites suggested would permit direct communication with the Earth. A typical raw data rate is 31 Mbs which could be compressed by a 10:1 ratio. This translates to a 3.1 Mbs data rate, which can be transmitted continuously or stored at the telescope site and transmitted when needed. Several frequency bands are available, each with its own advantages and disadvantages. The control mode is another criteria to be considered, obviously the most desirable being a completely automated, remotely controlled system requiring no human support except possibly for repairs.

CONSIDERATIONS FOR LUNAR COMMUNICATIONS

- LOCATION
  - Near Side- Direct Communication
  - Far Side- Requires Relay Satellite
  - All Suggested Sites on Near Side

- DATA RATE
  - Average Raw Data Rate of 31 MBS before compression
  - Assume 10:1 compession for 3.1 MBS transmission rate
  - Continuous transmission vs. Intermittent
- FREQUENCY OF TRANSMISSION
  - S band, X band, or K band
- CONTROL
  - Completely automated,remotely controlled
  - No human intervention except for repairs

C&DH SYSTEM REQUIREMENTS/ASSUMPTIONS

The C&DH system will be operational from lift off, transit to the moon, during landing and for long term lunar operations. This chart lists the requirements and assumptions that were made in the design of the C&DH system for the LTT during operations from the moon while taking data. The maximum transmitted data rate was assumed to be 3.0 Mbs, based on a typical pixel exposure per unit time. The location of the telescope was assumed to be on the side of the moon nearest the Earth, providing direct communication access. During transit and landing low gain antennae will be used. The telescope high gain antennae will automatically deploy upon landing. After deployment the telescope will be controlled by the Earth Ground Station.

Lunar Transit Telescope
C&DH System Requirements/Assumptions

o Data Rate--- Assume 3.0 Megabits/second Maximum Rate.

o Location--- Assume Near Side- Direct Communication with Earth.

o Power Source--- Internal- Approx. 180 watts in full operation.

o Deployment--- Automatic on Landing- No EVA Required.

o Control/Operation---From Earth Ground Station.

DEEP SPACE NETWORK

The chart on the facing page describes some of the salient features of the Deep Space Network, NASA's scientific telecommunications network operated by the Jet Propulsion Laboratory. The DSN consists of Stations at 3 locations around the earth; in California, Spain and Australia. Each location features a wide variety of telecommunications equipment manned by experienced personnel on a continuous basis.

Continuous communications coverage with the LTT would require at least three ground stations. If the DSN were not available then three new ground stations with some of the capabilities of the DSN would be required to support the LTT.

DEEP SPACE NETWORK

- DSN is NASA's largest and most sensitive scientific telecommunications and radio navigation network.

- DSN is a facility of the NASA Office of Space Operations and is operated by the Jet Propulsion Laboratory.

- DSN consists of 12 Deep Space Stations at 3 locations:

  (1) Goldstone, CA
  (2) Madrid, Spain
  (3) Canberra, Australia

- Each complex consists of four Deep Space Stations with large parabolic dish antennas:

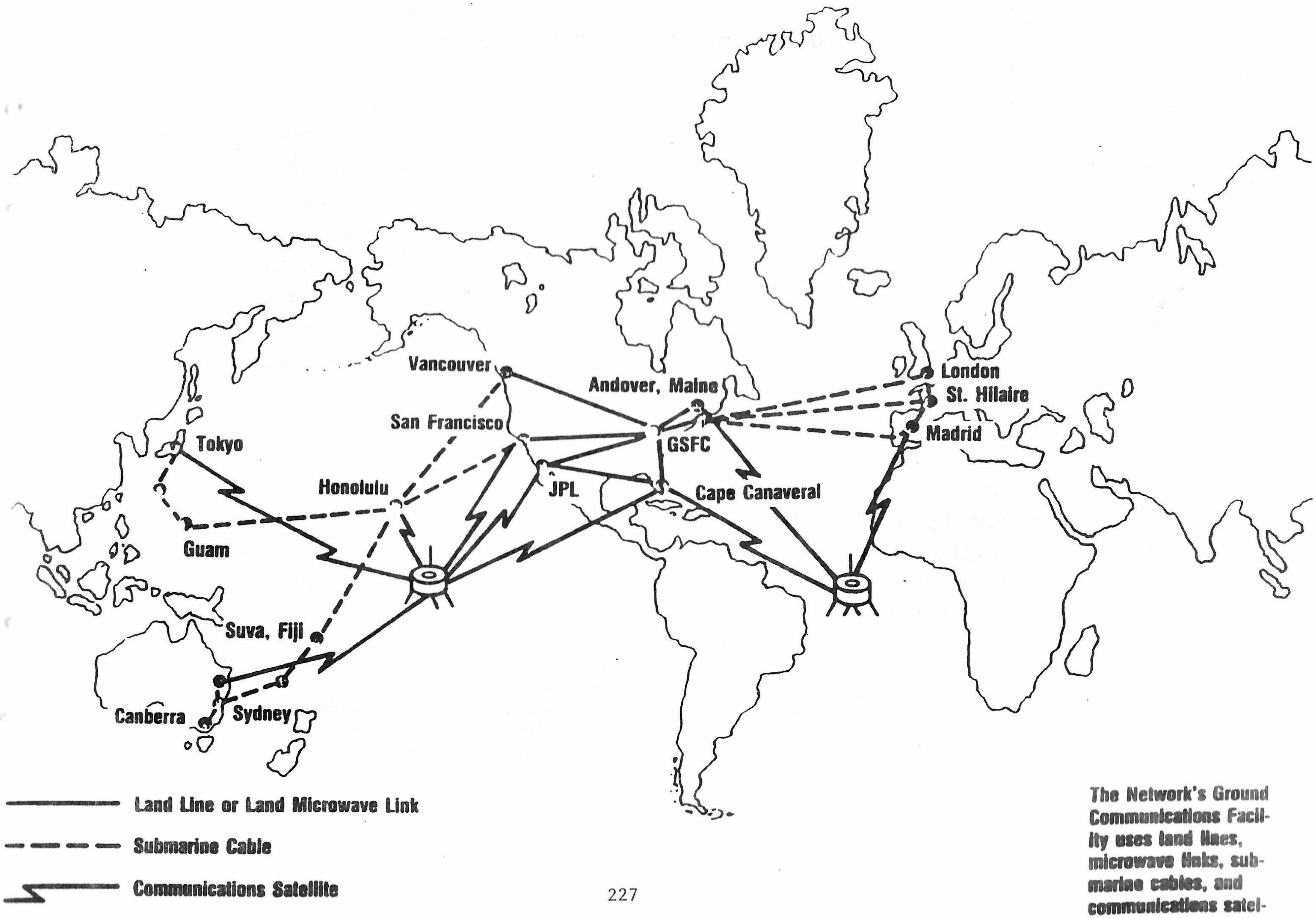
  (1) One 26 meter (85 feet) diameter antenna.
  (2) Two 34 meter (111 feet) diameter antennas.
  (3) One 70 meter (230 feet) diameter antenna.

- Each complex has a data processing center manned by a six-person crew and is operated on a 24 hour-7 day basis. Signals from three spacecraft can be received and processed simultaneously.

DEEP SPACE NETWORK FACILITIES MAP

This map shows the location of the facilities of the Deep Space Network, and the type of links that connect them together. These include land lines, microwave links, submarine cables and communications satellites.

The LTT would probably have a central location as a science operations center and to distribute the data to various investigators. There has been some discussion about beaming the real time data into class rooms for educational purposes and to stimulate interest in astronomy.



The Network's Ground Communications Facility uses land lines, microwave links, submarine cables, and communications satel-

SAMPLE LINK CALCULATION

This chart provides a sample calculation of the communications link needed to provide LTT-to-earth communication. This calculation assumes the use of the 34 meter dish of the Deep Space Network on the Ka-band. The calculations show that a 30.6 dB Power-Gain Product is needed to support a 3.0 Mbps data rate at a minimum 10 degree antenna elevation and allowing for 10% bad weather effect. A 30 cm (one foot) diameter dish antenna on the lunar surface will provide 37.5 dB with a beam-width of 2.1 degrees at the transmission frequency. Since the Earth subtends an angle of only 1.8 degrees, a one foot dish would cover the entire earth at the lunar distance.

If the LTT's position and orientation on the moon were accurately known and if everything were inertially fixed then the LTT could have fixed antennae. However, analyses have shown that when viewed from the moon the Earth appears to move over a half cone angle of several degrees during one year. Also landing dynamics can cause several degrees of rotation in the LTT's attitude relative to the Earth's position. These considerations require that the HGA have two degree of freedom gimbals to assure correct Earth pointing.

12/16/91:HB

SIMPLE LINK CALCULATION

* ASSUME Ka-BAND COMMUNICATIONS USING DSN 34 M DISH

* 30.6 dB PGP NEEDED FOR 3.0 Mbs @ 10 DEG ELEVATION AND 90% WEATHER

* A 30 CM DISH WILL PROVIDE 37.5 dB GAIN

* ONE FOOT DISH HAS A BEAMWIDTH OF 2.1 DEGREES AT 33 GHz (Ka-BAND)

* EARTH SUBTENDS AN ANGLE OF 1.8 DEGREES AT LUNAR DISTANCE

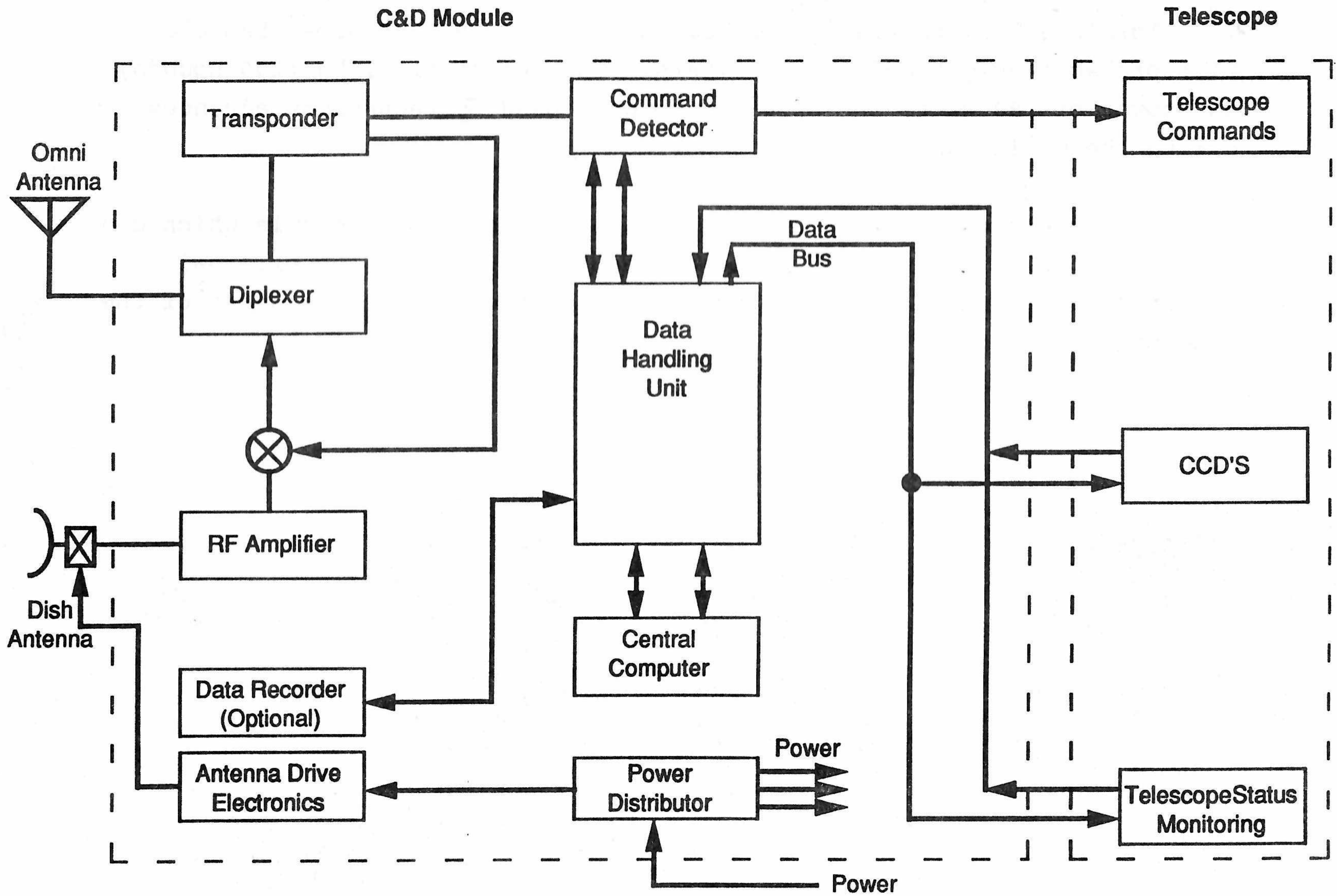
C&DH SIMPLIFIED BLOCK DIAGRAM

This diagram depicts the basic elements of the Communications and Data Handling System for the LTT. Commands are received from the Earth Station and processed through the command detector and passed on to the telescope. Data from the telescope and instruments are fed through a data bus to a data handling unit, processed and transmitted to the ground station.

The C&DH has been designed to satisfy the long term science operations of the LTT on the moon. However, the short term needs, 3-4 days transit and minutes while landing, may also drive the computer needs. Landing dynamics require rapid computations, and automatic landing requires special software and sensors. The landing needs can be satisfied by a specially designed and programmed card in the computer. One C&DH system can satisfy all the LTT's requirements and be much more reliable and lighter weight than separate systems for the lander and telescope.

1-1845-1-37D

Lunar Transit Telescope
C&DH Simplified Block Diagram



C&DH SUBSYSTEMS EQUIPMENT LIST

This chart lists the typical components of the C&DH subsystem along with probable weight and power numbers. The weight of 91 kg (200 pounds) and power of 180 watts can probably be decreased by technology advances during the design and development period.

The module components can be packaged into one small module which can be located at any convenient place on the LTT. However, the external components, redundant antennae, should be located on opposite sides of the telescope/lander.

Lunar Transit Telescope
C&DH Subsystem Equipment List

Module Components	Quantity	Weight (Lbs)	Avg. Power (W)	Unit Size (cm)
Transponder	1	15	18	36 x 11 x 25.2
Command Detector	1	3	10	3 x 10 x 15
Diplexer	1	2	-	5 x 15 x 3
Central Data Unit	1	10	40	26.7 x 21 x 16.5
RF Amplifier	1	10	12	8 x 7 x 2
Central Computer	1	20	40	22.7 x 17.8 x 20.3
Data Recorder	1	15	18	17.3 x 20 x 28
Ant. Drive Elect.	2	20	15	3 x 4 x 1
Power Dist.	1	5	7	5 x 10 x 15
Structure & Cabling	1	50	-	———
Subtotal		150 (lbs.)	160	
External Components				
Omni Antennas	2	2	-	3 x 3 x 30
Hi-Gain Antennas	2	10	-	TBD
Antenna Mounts	2	18	-	-
Antenna Drives	2	20	20	10D x 30
Subtotal		50 (lbs)	20	
Total		200 (lbs)	180 Watts	

4.4 ELECTRICAL POWER

A COMPARISON OF POWER SYSTEMS

Three types of power systems, solar array/battery, solar array/regenerative fuel cell, and nuclear, were considered for the LTT. Although two separate power systems could be used on this project an attempt to use one common system was pursued to minimize vehicle weight. Since the surface system (i.e. telescope requirements) had the highest energy storage demand it was the driver in determining the type of system to be utilized.

The accompanying chart shows a comparison of the mass of the systems required to fulfill the surface requirement. Note the comparison is based on a 1 kW average requirement. Power system requirements were not available at the times of the study but the results are still applicable. Actual weights are defined later in the report. As can be seen here the length of the lunar night drives battery and fuel cell masses for energy storage to excessive values. The nuclear systems (both using isotope heat sources) have much lower masses and offer long mission times. The radioisotope thermoelectric generator (RTG) was chosen over the dynamic isotope power system (DIPS) for reasons of development status. The major disadvantages of the selection are the environmental and political concerns.

Development status of the four options is shown in the lower half of the chart.

A Comparison of Power Systems Assuming 1 kW Continuous Operation

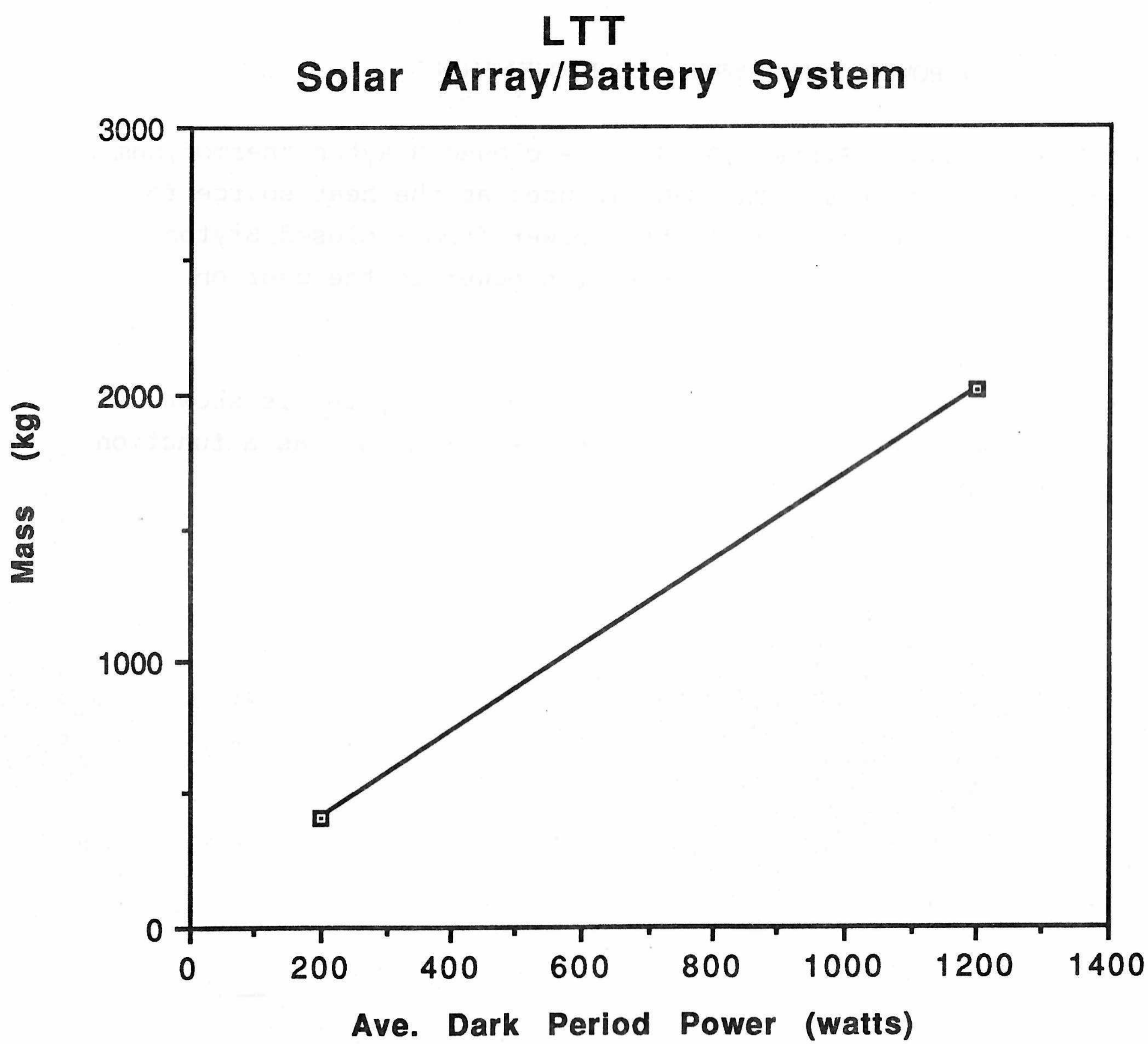
	Weight (kg)	Disadvantage
RTG	184*	High thermal load, environmental and political concerns, launch pad cooling required
DIPS	191*	High thermal load, environmental and political concerns, launch pad cooling required
Solar Array/ Battery	1700**	Excessive weight (batteries)
Regenerative Fuel Cell	464**	Excessive volume (hydrogen tanks)

* Weight of radiation shield and power cables are neglected
** Solar array is assumed to be fully oriented. Cable weights are neglected

	State of Development of the Technology
RTG	Ready to fly
DIPS	Proven technology but requires considerable DDT&E
Solar Array/ Battery	Battery subsystem requires some development
Regenerative Fuel Cell	Mature technology but would required some DDT&E

SOLAR ARRAY BATTERY SYSTEM

System mass as a function of average dark period power is shown here. The period of solar occultation is assumed to be a 14 earth day period. Current state of the art solar array and battery technology was assumed. Two degree of freedom array orientation was assumed to achieve minimum size and mass. No distribution system or cables masses are included on this plot.

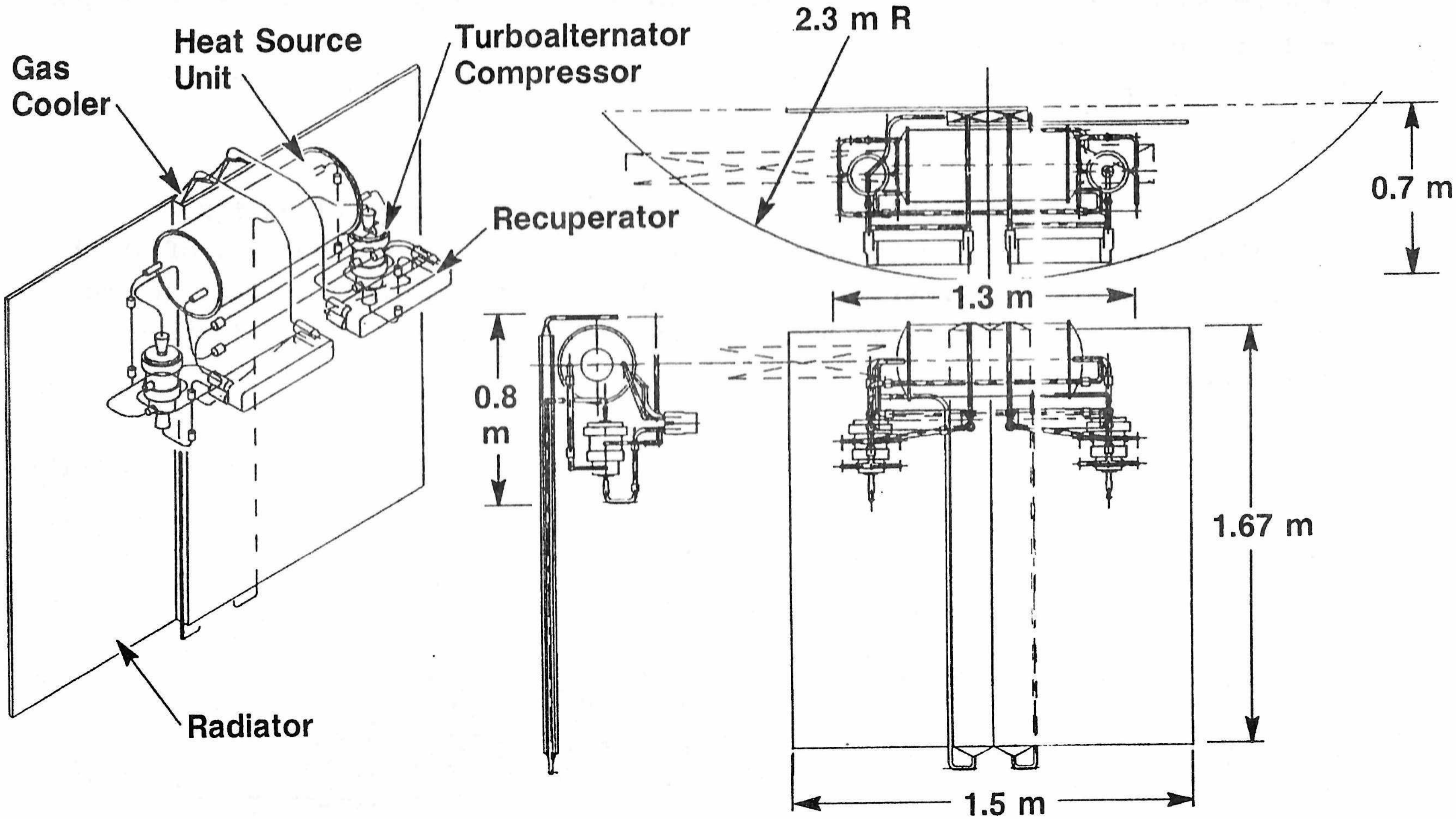

LTT
Solar Array/Battery System
Mass (kg)
3000
2000
1000
0
0
200
400
600
800
1000
1200
1400
Ave. Dark Period Power (watts)

LOW POWER DIPS SYSTEM CONFIGURATION

The Dynamic Isotope Power System (DIPS) is a closed Brayton thermodynamic cycle used to generate electricty. The GPHS is used as the heat source for the thermodynamic cycle. Conditioned electric power from a closed Bryton cycle turboalternator compressor provides electric power to the user on demand.

A drawing showing overall system dimension of a DIPS system is shown here. Typically its power level may range up to 2 kW or higher as a function of its operating temperature and radiator area.

Low Power DIPS
System Configuration

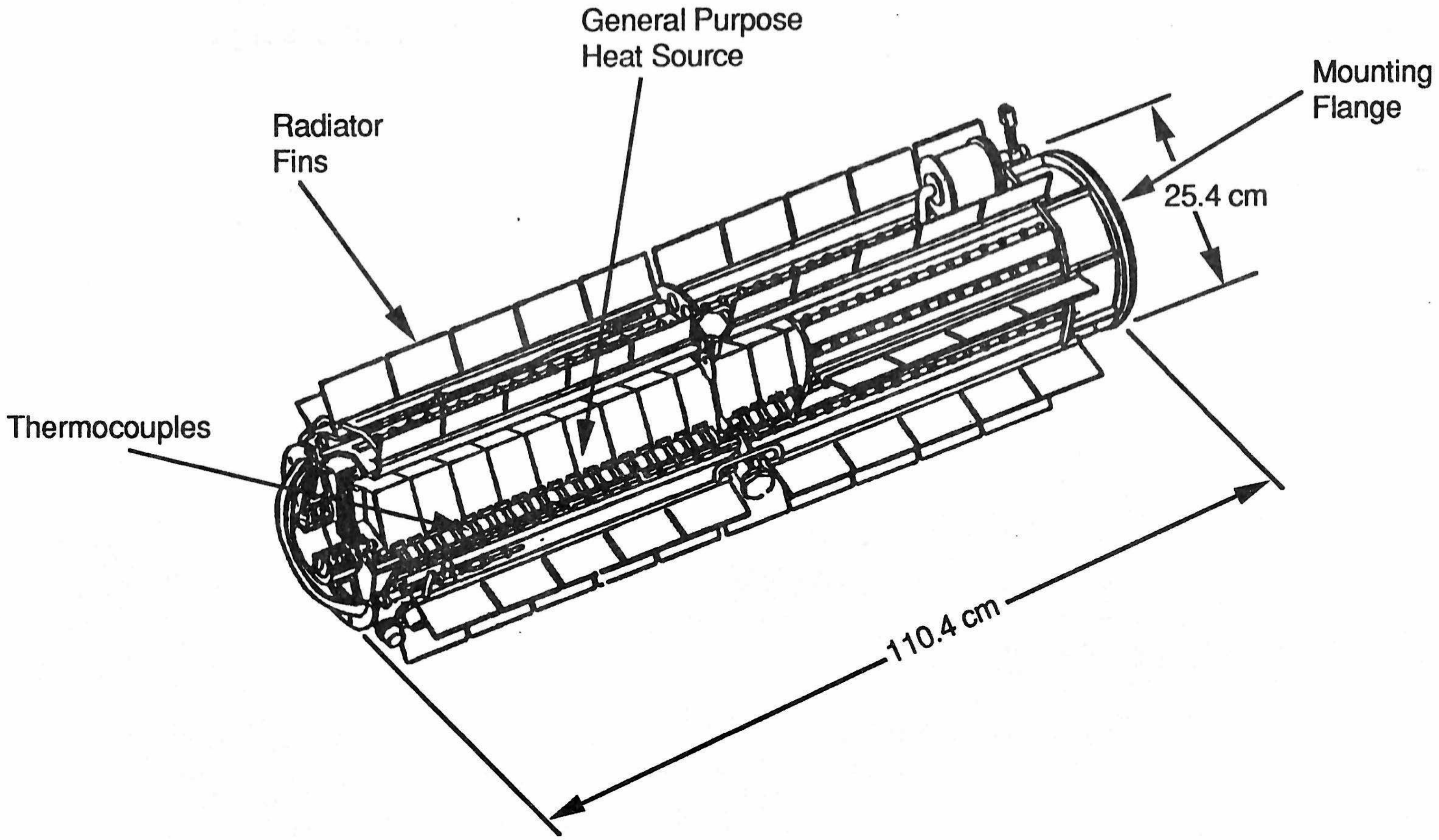


GPHS-RTG DESCRIPTION

A picture of the GPHS-RTG shown in cutaway reveals the fuel blocks and the thermoelectric converters on the interior. Fins cover the exterior of the module and provide adequate cooling for the module when not obstructed. Mounting is accomplished through a flange on the inboard end (non-domed end) of the module.

The RTG contains eighteen 250 W (thermal) heat sources along its longitudal axis. The GPHS generates approximately 4,500 W (thermal) of which approximately 315 W (electric) is produced by the thermoelectric unit around the perimeter. The remaining heat is radiated by the fins at a rejection temperature of 580 K.

General Purpose Heat Source – RTG



RTG/DIPS FUEL

The DIPS and the RTG use the same fuel element, Pu-238, as a general purpose heat source (GPHS). The characteristics of the fuel element are shown in the acompaning chart. These elements are supplied by DoE and incorporated into the energy conversion system manufactured by the DIPS or RTG supplier.

1-2288-1-23Ta

RTG/DIPS Fuel

Radioisotope	Pu-238
Class of Emitter	α
Half life	87.7 yr.
BOL/EOL Ratio	1.09 (11 yr.)
Compound Form	PuO_2
Melting Temperature	4352° F
Density	10 gm/cc
Watts/gm	0.39
Curies/watt	30
*Pb Shielding	.25 cm

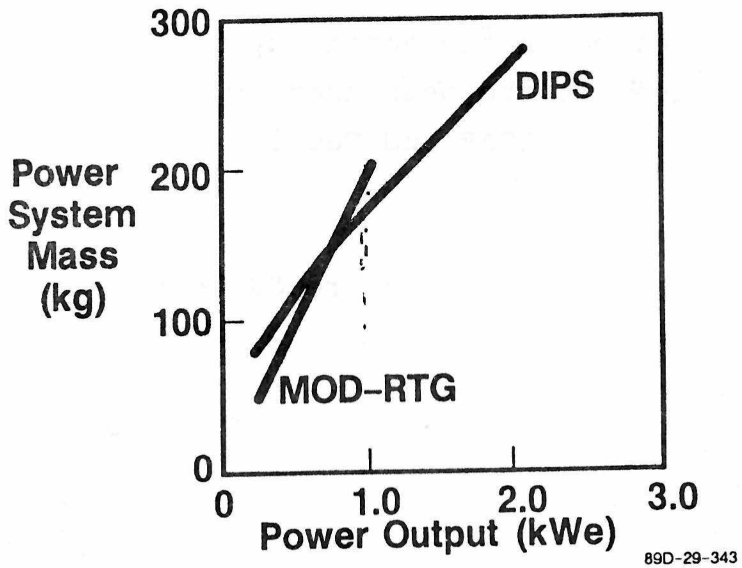
*Lead shielding for 1-kWt heat source for 10 mrem/h at 1 meter

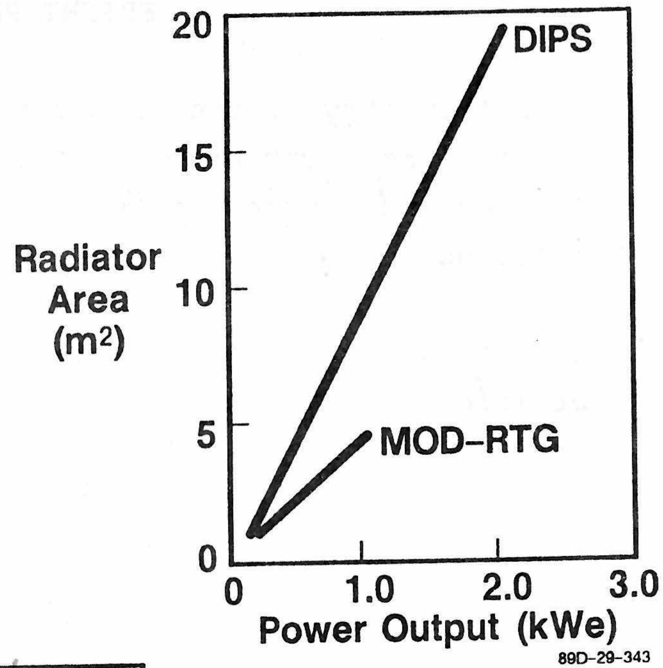
LOW POWER DIPS SYSTEM MASS

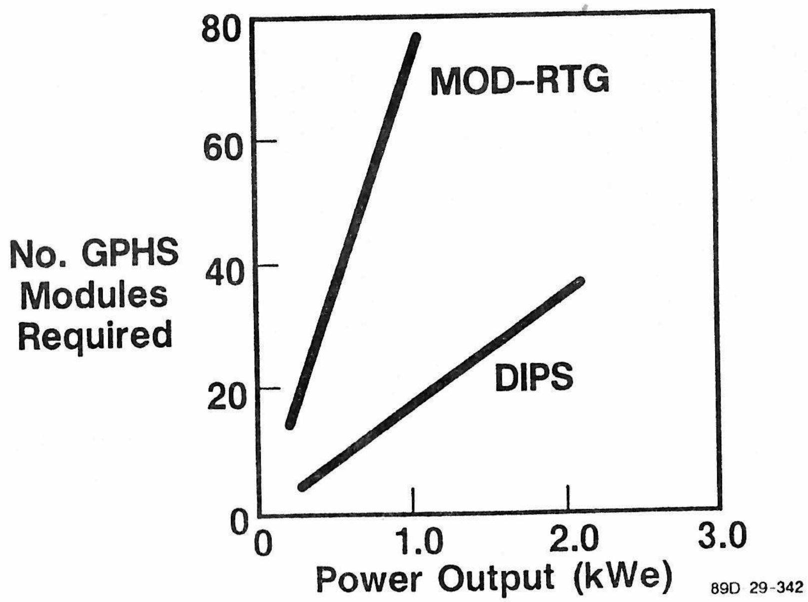
A Dynamic Isotope Power System (DIPS) was considered for the power source. A comparison of the DIPS with an RTG in terms of system mass, radiator area, and fuel quantity is shown on this chart by Rockwell International. As can be seen here, had the requirement remained at or above 1 kW the DIPS would have become a contender for the LTT power system.

Mass, state of development, time of availability, reliability and cost made the RTG the choice for this study.

Low Power DIPS System Mass and Area Characteristics and Fuel Requirements Compared to Advanced MOD–RTG







Rockwell International
Rocketdyne Division

90D–29–79
883

11/22/91:HHH

FLIGHT PROVEN ISOTOPE HEAT SOURCE

A summary of the flight history of RTG's is given in the accompanying chart. The units proposed for the LTT are identical to the ones used on Galileo and Ulysses spacecrafts and proposed for use on CRAF and Cassini missions.

As was shown on an earlier chart the RTG is a flight proven, ready to fly article.

Flight Proven Isotope Heat Source

- 28 years of safe flight experience
- Radioisotope thermoelectric generators (RTGs')
  - Over past 26 years, 6 units flown on 20 DoD and NASA space missions
    - 8 earth orbit
    - 5 lunar surface
    - 7 planetary exploration

Type Proposed For LTT

- 3 advanced RTG's, using the general purpose heat source (GPHS) modules, were shuttle-launched -
  - 2 in 1989 on Galileo missions
  - 1 in 1990 on Ulysses mission
- 32 RTG's delivered to DoD for special terrestrial applications

LTT POWER REQUIREMENTS

The power requirements for the LTT are identified here by mission phase. The transit and surface requirements are those of the landing vehicle while the surface requirements are those of the telescope. Operation times for each of these phases were obtained from the mission timeline (see section 5.2). Applying these to the power requirements resulted in the energy requirements of the three mission phases. Only the lunar surface energy required for the lunar night is recorded here, the duration being the driver on the power system storage sizing.

LTT Power Requirements

Subsystem	Mission Phase		
	Transit (Watts)	Landing (Watts)	Surface (Watts)
GN & C	233	356	30
C & DH	100	100	180
Thermal	90	90	200
Propulsion	120	120	nil
Experiment	nil	nil	100
Contingency	107	133	90
Totals (Watts)	650	800	600

Energy Requirements

Transit 100 hours x 650 W = 65 kWh

Landing 1 hours x 800 W = 0.8 kWh

Surface 336 hours x 600 W = 202 kWh (lunar night requirements)

RTG RADIATION EXPOSURE, END AND SIDE READINGS

The accompanying charts addresses the radiation from the RTG unit. The major portion of the radiation is alpha particles and are of little concern, to the spacecraft and its subsystems. The neutron and gamma radiation along the longitudial axis of the RTG and along a line perpendicular to the mid point of the longitudial axis as a function of distance are tabulated. This level is not believed to be of sufficient magnitude at the distance involved to cause a problem. Avionics and science instruments can be shielded if further study shows a requirement.

The numbers indicate that at a given distance from the RTG, more radiation is experienced from the broad side than from the end. These radiation levels are measured on a test article and are representative of those that could be expected on a flight article.

A study that should be performed in the next study phase of the LTT design is an evaluation of the radiation effects on the avionics and science instruments. Preliminary investigations indicate there are no show stoppers and shieldinig can be incorporated if required. Another factor in favor of CCDs is their low operating temperature. Tests have shown that neutron damage to CCD's is greatly reduced if operated below 160 K.

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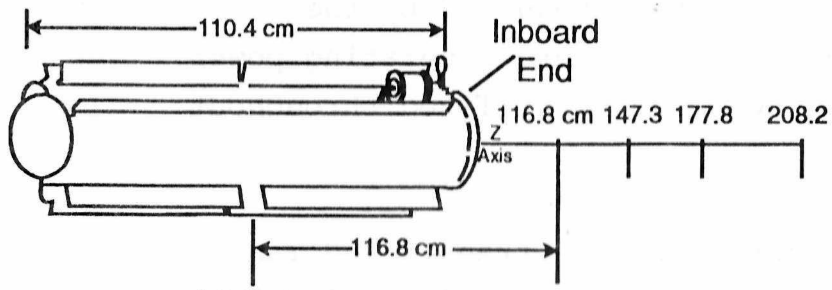
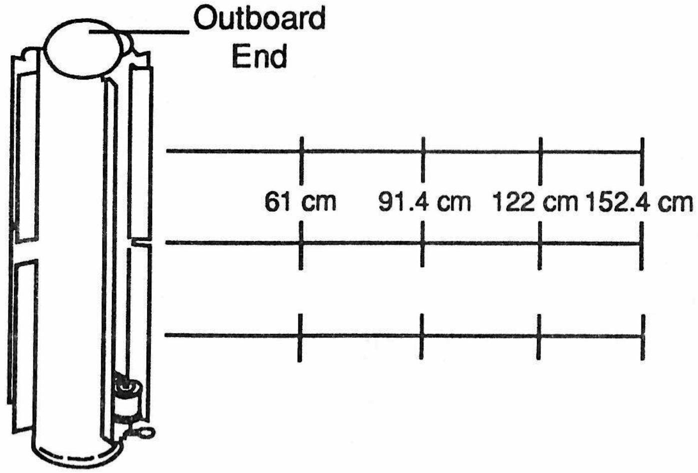
RTG Radiation Exposure

On Center Axis Reading



Dist. (cm.)	Neutron m rem/hr	Gamma m rem/hr	Total m rem/hr
116.8	17.7	3.2	20.9
147.3	11.5	1.9	13.4
177.8	8.3	1.2	9.5
208.2	6.9	1.0	7.9

Mid-Span Readings



Dist. (cm.)	Neutron m rem/hr	Gamma m rem/hr	Total m rem/hr
61	86.9	29	115.9
91.4	47.9	14	61.9
122	30.7	8	38.7
152.4	21.9	5.5	27.4

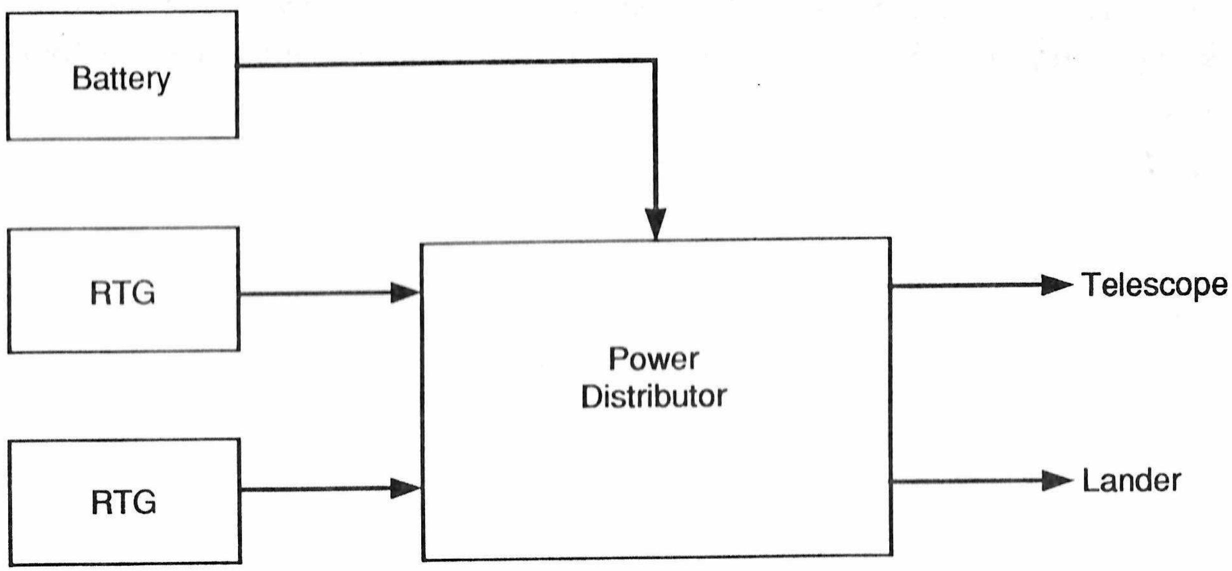
LTT POWER SYSTEM BLOCK DIAGRAM

The power system chosen for the LTT included 2 RTG's and primary batteries. The 2 RTG's were chosen to supply the power required by the surface operating systems of the LTT. Since these units are generating power from the time the GPHS's are installed their power can be utilized throughout the mission.

As noted in the power requirements shown earlier, the transit and landing phase power levels are greater than the surface requirements. These levels are in excess of the RTG's capability and must be supplemented by another source. Since these are one time requirements a primary battery was chosen to supply the energy not available from the RTG's.

A simplified block diagram of the proposed system is illustrated along with power requirements by mission phase.

LTT Power System Block Diagram



Power Budget

Transit	650 W
Landing	800 W
Surface	600 W

LTT POWER SYSTEM MASS

The power system mass is approximately 199 kg. Since the RTG's are an existing design, their mass is a well established quantity. The remaining quantities may vary slightly depending on final requirements and technologies implemented.

LTT Power System

2 RTG's @ 55.9 kg each	112 kg
1 Battery	55
1 Distribution system	32
*Total	199 kg

* Cable weights not included

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SECTION 5

5. MISSION ANALYSIS

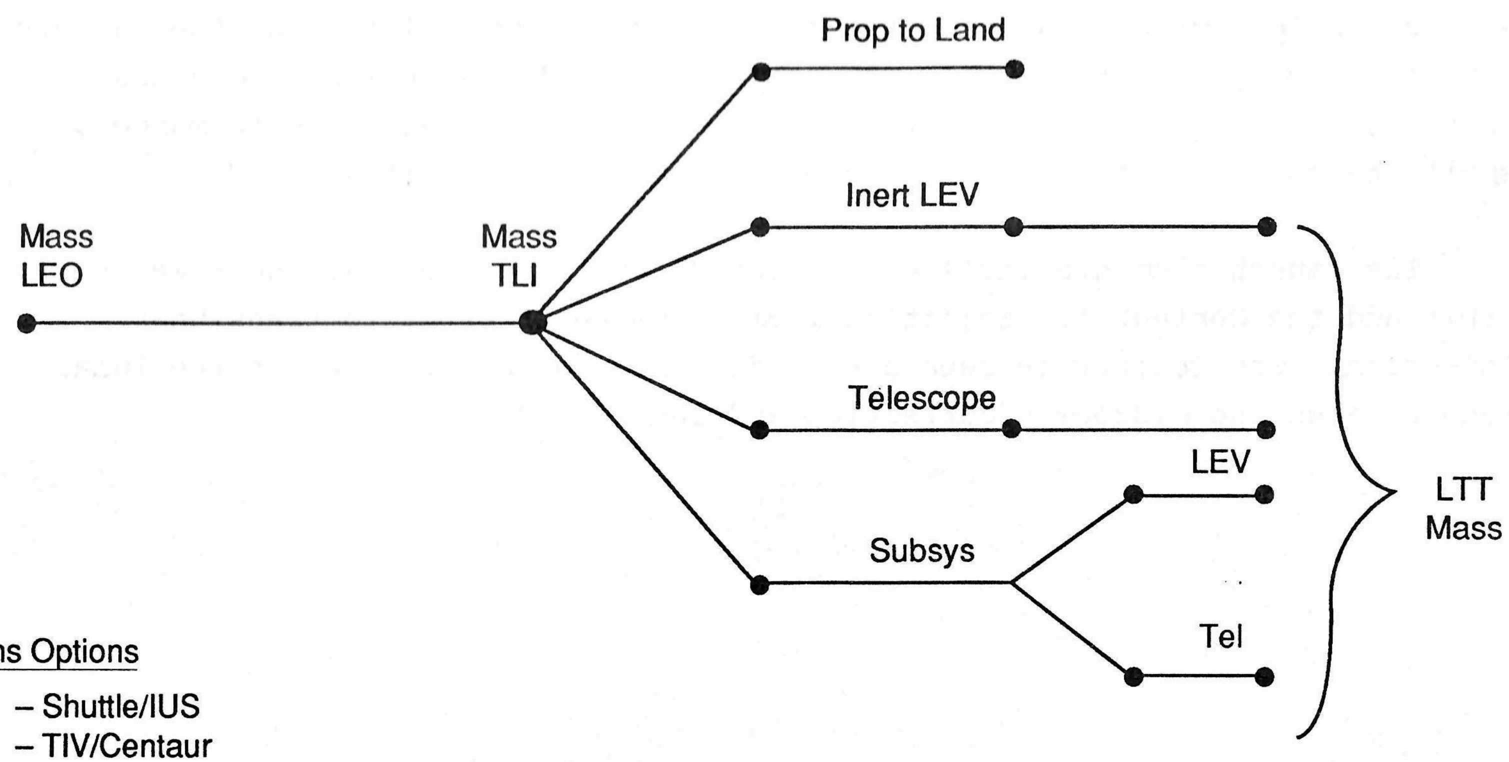
5.1 TRANSPORTATION

CANDIDATE LAUNCH VEHICLES

One of the guidelines for this study is to use an existing launch vehicle for placing the LTT into a translunar injection orbit, another is to make the primary mirror as large as possible subject to the launch and transportation vehicle constraints. A quick review of existing, or expected to exist, launch vehicles was made. Most of the data on existing launch vehicles was taken from "International Reference Guide to Space Launch Systems", AIAA 1991 edition, by Steven J. Isakowitz. However, the performance data was not in the form needed such as the translunar injection (TLI) mass, and the ratios between the TLI and propellant mass needed to land on the moon. Moreover, the weights of the inert lander and the useful payload that is landed on the moon are needed. These parameters were obtained by using scaling equations based on trajectories run. It is assumed that the launch vehicle with its upper stage placed the LTT with lander into TLI, after which the upper stage is separated from the LTT/lander. The lander must have enough propellant for midcourse corrections and to soft land on the moon. About one half of the TLI mass is propellant and the other is inert lander and telescope.

Since the Titan IV/Centaur should be operational in time for the LTT and its performance is greater than the other candidates, hence permitting a larger telescope diameter, it was selected for the LTT. The Titan IV/Centaur performance can be improved by structural modifications to both the Centaur and Titan Stage 2, but the modification cost is not known.

Lunar Transit Telescope is The Integrated Telescope and Lander With Their Subsystems



Trans Options

- Shuttle/IUS
- TIV/Centaur
- TIV + USRM/Modified Centaur
- New NLS With New Upper Stage

LTT DELIVERY PROFILE

A typical LTT delivery profile is shown on the facing page. This is neither an optimum nor worst case. When a firm launch date is given, a launch window study should be undertaken. This profile is based on a launch date of 16 September 2002. The profile starts with a TLI burn from a 28.5 degree orbit at 100 nautical miles. A midcourse burn occurs 72 hours later and then a soft landing at 96 hours after TLI. The projected landing site was 40 degrees N latitude and 70 degrees W longitude. This is not critical to performance as small changes at midcourse can be used to land at other sites.

The launch elements involved are the Titan IV for Earth to near Earth orbit and the Centaur for injection from a low earth orbit to translunar injection. The Centaur is separated before mid-course after which the lunar lander makes the midcourse correction and lunar landing.

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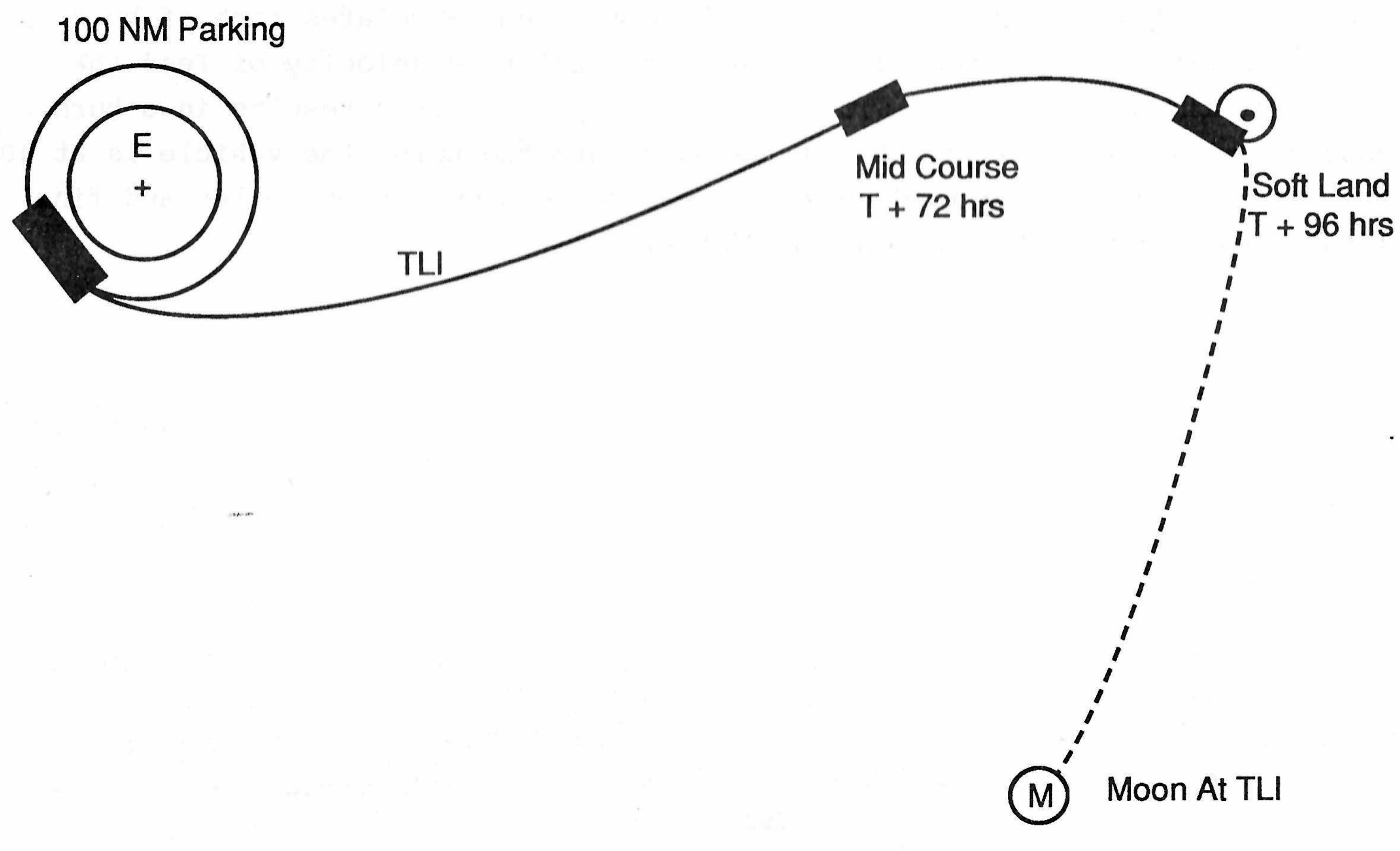
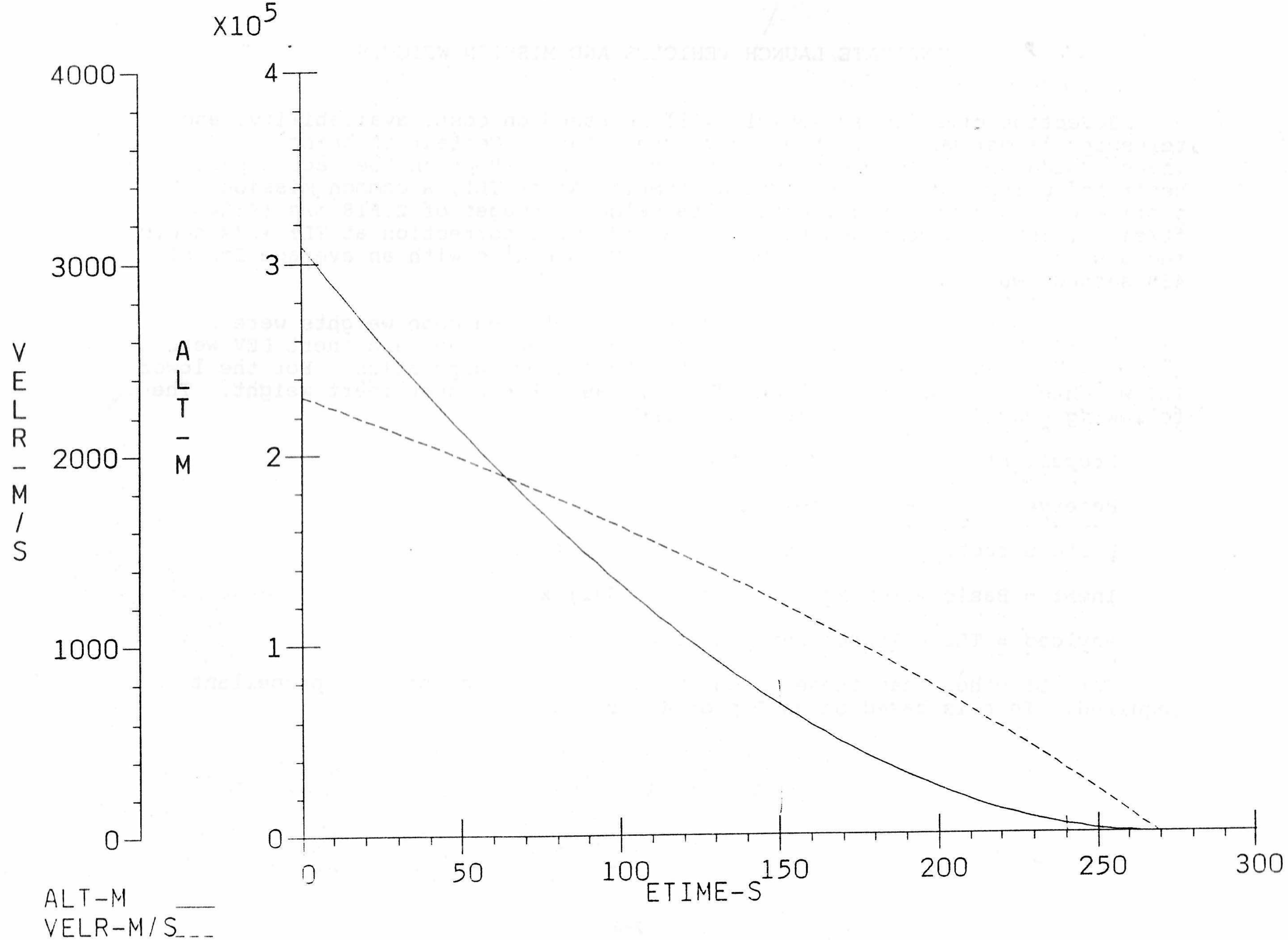
LTT Delivery Profile



SOFT LANDING

An empirical algorithm for a soft landing using a constant thrust level was used to generate the profile on the facing page. The midcourse correction burn targeted to a point 600 meter above the desired landing point. On reaching an incoming altitude of approximately 308 km, the engine starts throttling to 10,000 pounds thrust. The algorithm calculates time of burn and guide angles such that touchdown occurs at a relative velocity of less than 1 m/s. For the TLI weight of 6,349 kg (14,000 pounds) this results in a burn of approximately 270 seconds. At 180 seconds into the burn, the vehicle is at 40 km. At this point, the landing site could be scanned for obstacles and final corrections made to the landing trajectory.

SOFT LANDING: EMPIRICAL ALGORITHM 20 NOV 1991



CANDIDATE LAUNCH VEHICLES AND MISSION WEIGHTS

Selection of a launch vehicle will be based on cost, availability, and telescope/lander weights. Presently, the Titan IV Centaur is being investigated for use. The propellant used values shown on the facing page were generated using the computer program "IMP." After TLI, a common mission profile was used that required a delta velocity budget of 2,818 m/s (9,245 ft/s). This was a combined total for a midcourse correction at TLI + 72 hours and a soft landing at TLI + 96 hours. An RL10 engine with an average Isp of 438 seconds was used.

Data other than propellant used and the total on moon weights were calculated using scaling equations. Initially the lander and inert LEV were estimated using an equation supplied by the Analex Corporation. For the lower TLI weights, the equation estimated an unreasonable higher inert weight. The following equations were devised and used.

Propellant = TLI - On Moon (from IMP)

Reserve = 0.056 x Propellant

Basic Structure (Avionics, etc) = 391 kg (860#)

Inert = Basic + (On Moon - Reserve - 391) x 0.37

Payload = TLI - Propellant - Reserve - Inert

For TLI other than those given, use 0.49 x TLI to estimate propellant required. This is based on an Isp of 438 seconds.

9/24/91

CANDIDATE LAUNCH VEHICLES

VEHICLE	TLI	PROP USED	INERT LANDER	PAYLOAD	EST COST
* DELTA 7925	2,900 LB	1,445	1,030	325	50 M
* ATLAS IIAS/CENTAUR	5,500 LB	2,675	1,530	1,145	120 M
* SHUTTLE/IUS	5,500 LB	2,675	1,210	1,540	245 M+
* ARIANE 5 (EST)	7,070 LB	3,425	1,820	1,635	110 M+
*** TITAN IV/CENTAUR**	**14,000 LB**	**6,835**	**3,050**	**3,730**	**227 M**
* MOD TITAN/CENTAUR	21,000 LB	10,430	4,235	5,750	? M

NOTES:

1. TITAN IV/CENTAUR SELECTED FOR 2 M LUNAR TRANSIT TELESCOPE, BECAUSE OF AVAILABILITY AND PAYLOAD CAPABILITY. TRY ATLAS IIAS FOR 1 M LTT.

2. FOR ANY CANDIDATE LAUNCH VEHICLE A NEW LUNAR LANDER IS REQUIRED

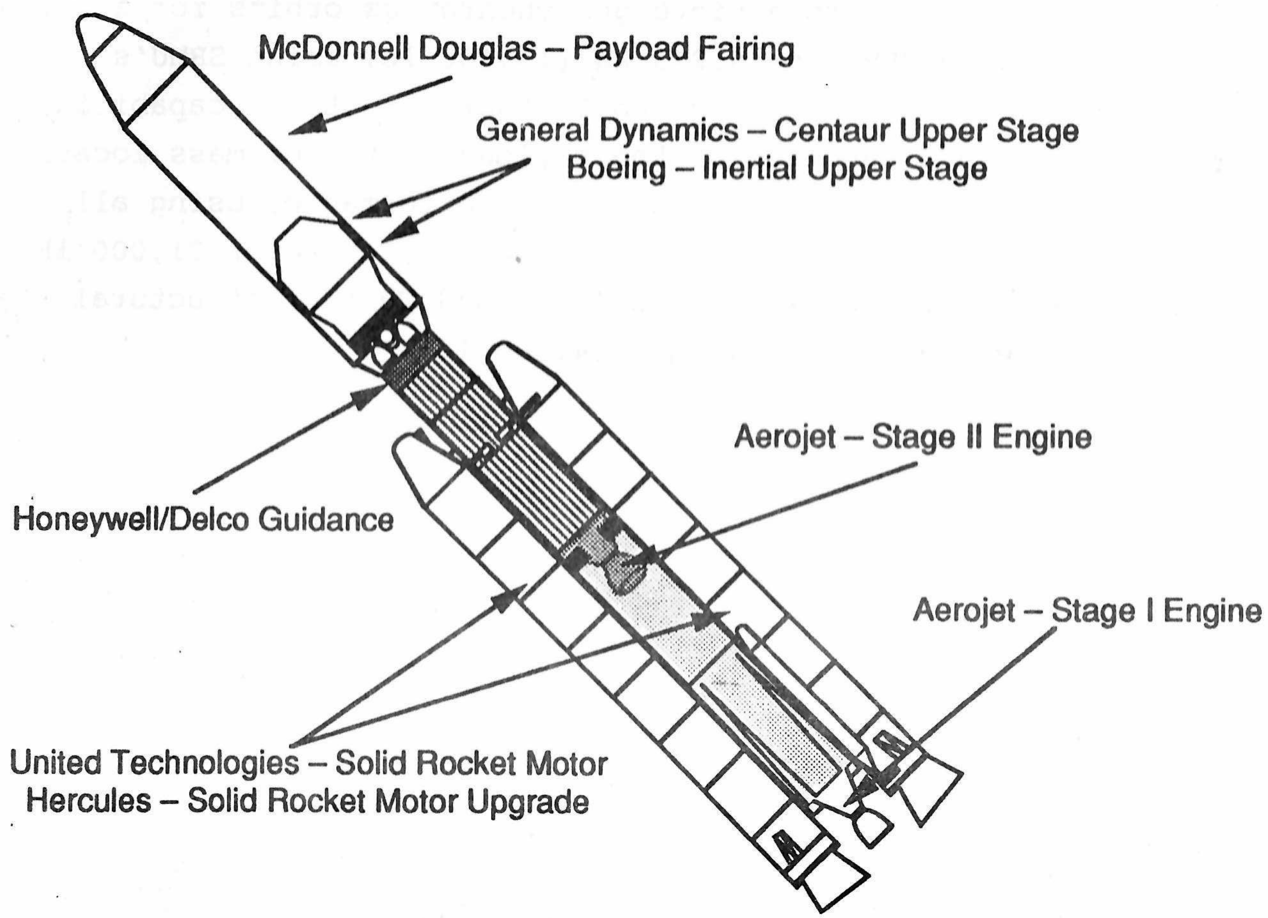
TITAN IV TEAM

The contractors responsible for the major elements of the Titan IV are shown on the facing chart. Martin Marietta is the prime contractor and General Dynamics is responsible for the Centaur upper stage. For the LTT the Titan IV/Centaur with the solid rocket motor upgrade (SRMU) has been selected as the preferred launch vehicle.

Data for the Titan IV/Centaur was obtained from "Titan IV Centaur, Program Overview", September 15, 1989, by R. Ernst and J. Moehlonpak.

Titan IV Team

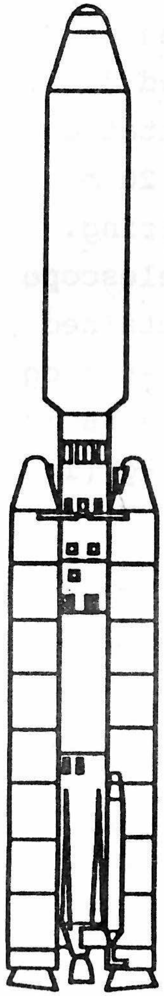
Martin Marietta - Prime Contractor



TITAN IV 401 CONFIGURATION

The Titan IV 401 is designed for launch from the eastern test range. It utilizes the Centaur upper stage to achieve geosynchronous orbits for a 4,773 kg (10,500 lb) payload using SRM's or 6,273 kg (13,800 lb) using SRMU's. However, the payload weight is limited by the Centaur structural capability of 5,215 kg (11,500 lb) and is a function of the payload center of mass location relative to the attachment plane. The theoretical performance, using all available propellants, is considerably greater (about 9,545 kg or 21,000 lb TLI) than the performance dictated when both Titan and Centaur structural limitations are considered (about 6,364 kg = 14,000 lb).

Titan IV 401 Configuration



401
CCAFS

Configuration

- 7-segment SRMs or SRMU
- 200-inch diameter PLF
  - — Length options: 66,76 & 86 ft.
- Centaur G' (modified) Upper Stage

Reliability

- Titan/Centaur
  - — 98% goal
  - — 97.91% predicted w/Centaur

Launch Availability

- 85% (Pitch Bias & Load Relief)

Launch Processing

- CCAFS/ESMC
  - — Vertical Integration Bldg.
  - — Solid Motor Assy Bldg
  - — Launch Complex 41
  - — Launch Control: Bldg. 27-200

Development Status

- All design reviews complete
- SRMU integration initiated
- Initial Launch Capability 1st qtr 1991

Performance with 86' PLF (Centaur Structural Limit 11,500 lbs.)

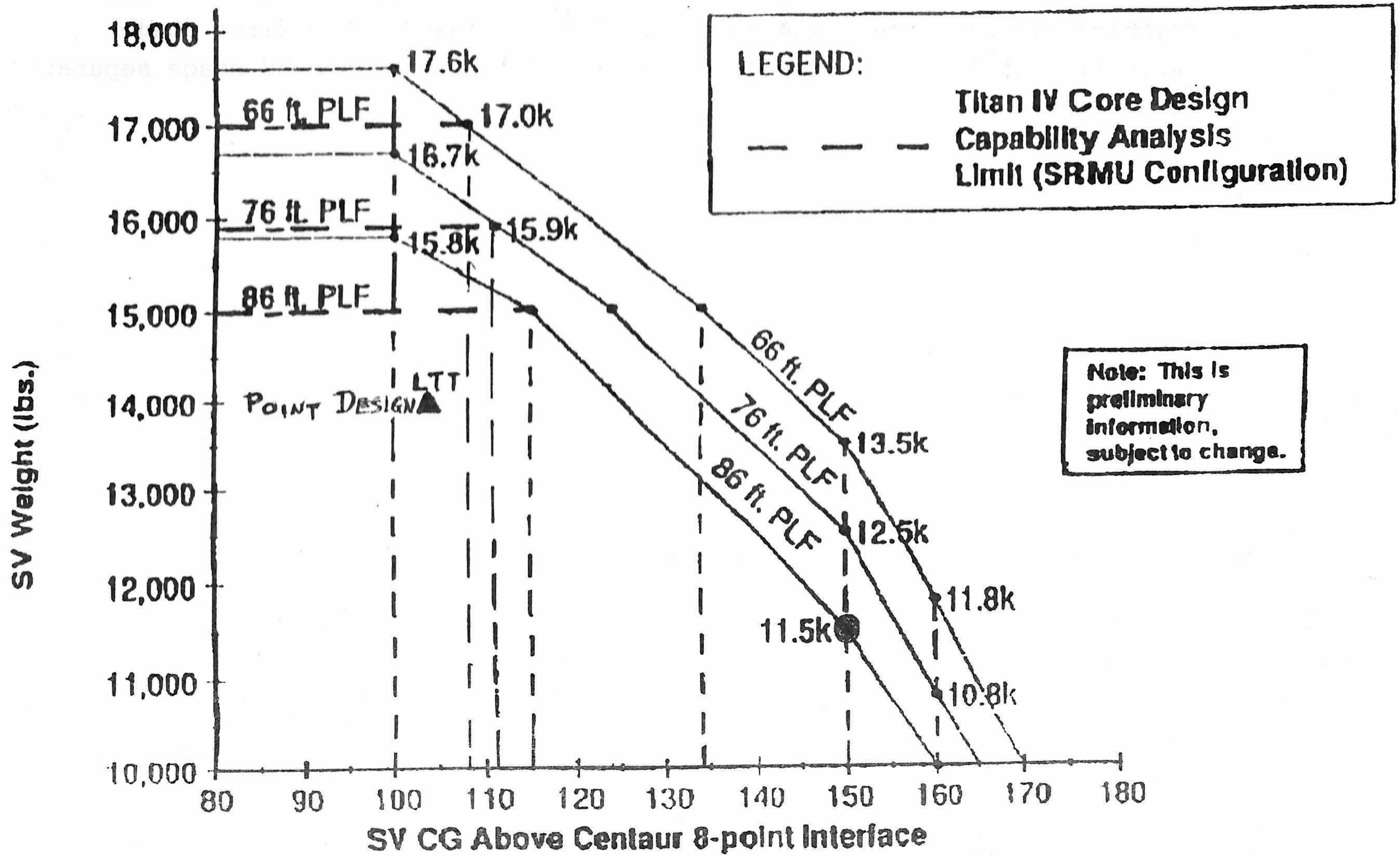
Mission	7-Segment SRMs		SRMU	
	Spec	Min Vehicle	Spec	Min Vehicle
GSO	10,000 lbs	10,200 lbs	12,700 lbs 13,500 lbs goal	13,400 lbs
12-hr	11,500 lbs	14,500 lbs	TBD	18,600 lbs
24-hr	N/A	10,500 lbs	N/A	13,800 lbs

57-BHW/AS-882492-98

TITAN IV/CENTAUR PAYLOAD WEIGHT VERSUS CG LIMITS

The total LTT system weight is limited by the Titan IV/Centaur performance capability, the payload fairing, the Titan IV structural limitations, and the Centaur structural limitations. The performance capability of the Titan IV/Centaur to a translunar injection orbit is over 9,070 kg (20 klb) and does not play a major role for this mission as the Titan IV core design limitations are lower. The core design limits vary from 7,710 kg (17 klb) for the 20 m (66-foot) payload fairing to 6,803 kg (15 klb) for the 26 m (86 ft) fairing. These limits are further modified by the cg location of the combined telescope and lander. The facing page shows the pertinent curves. These were obtained from Lewis Research Center. The present LTT design, 6,349 kg (14 klb) at a cg of 267 cm (105 in), is well within limits for any fairing. Currently, a 56 m (66 ft) fairing is being used with an arbitrary payload limit of 6,349 kg (14 klb) to allow for future growth.

Titan IV(SRMU)/Centaur Estimated Capability (Airloads)

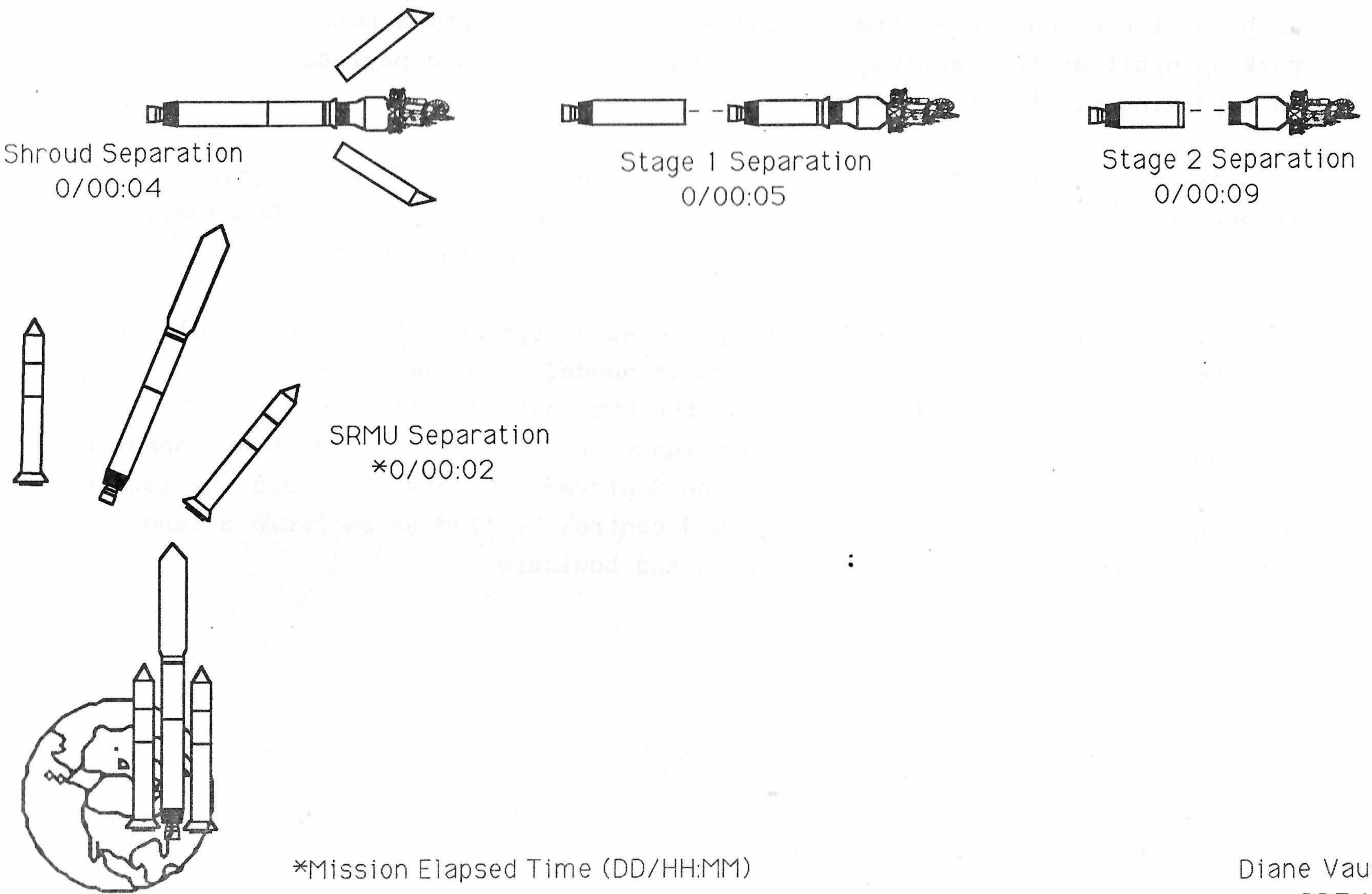


5.2 TIMELINES AND OPERATIONS

LUNAR TRANSIT TELESCOPE (LTT) TRANSPORTATION TIMELINE

The LTT will be launched by the Titan IV with a Centaur upper stage. The solid rocket motors will separate at burnout. At approximately 4 minutes MET (mission elapsed time), the shroud will be released. The first stage separation will follow shortly afterwards followed by second stage separation at 9 minutes MET.

Lunar Transit Telescope (LTT) Transportation



*Mission Elapsed Time (DD/HH:MM)

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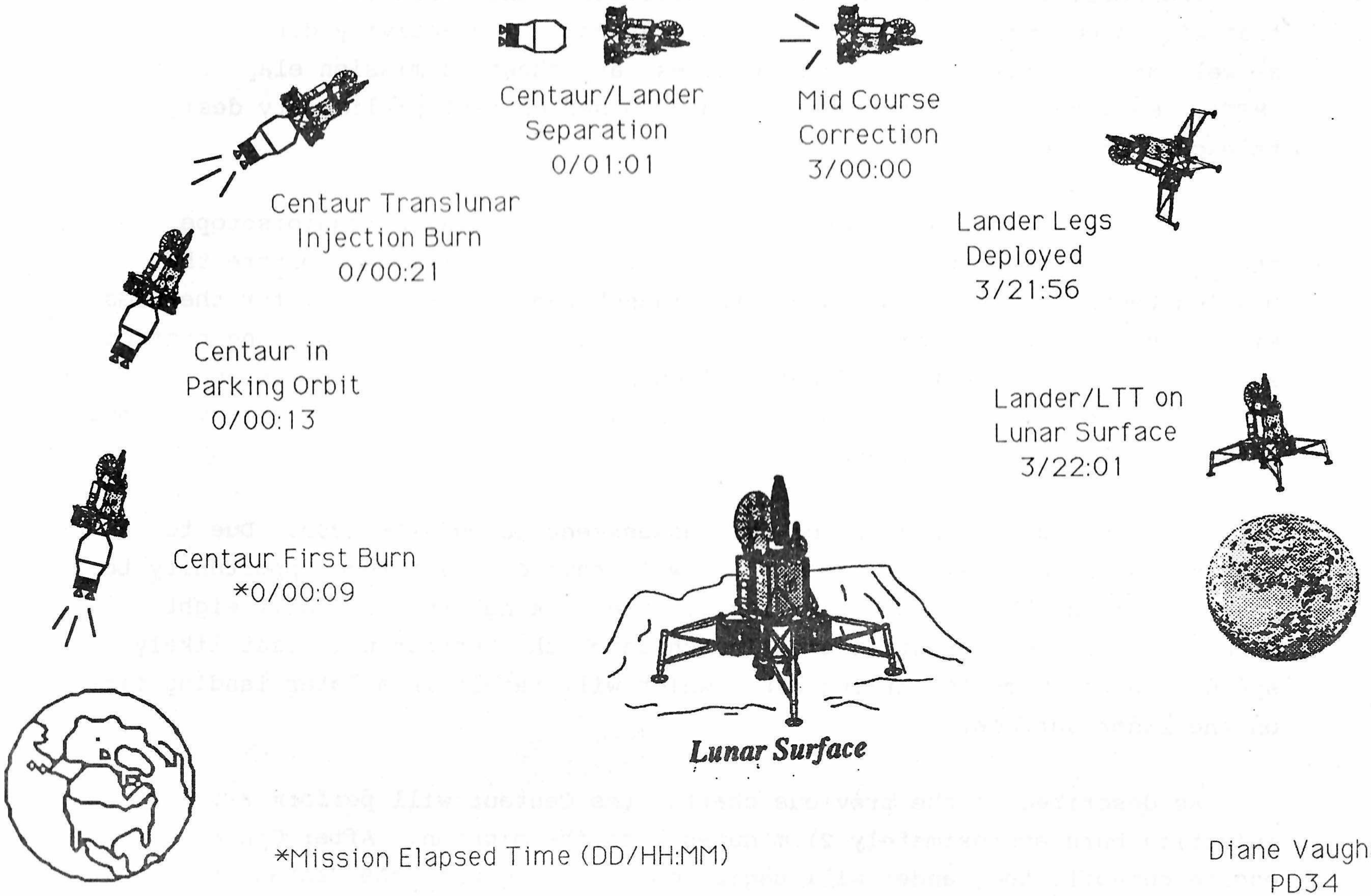
LTT TRANSPORTATION TIMELINE (continued)

As soon as the Titan IV second stage is adequately separated, the Centaur, with the lander and lunar transit telescope, will perform a burn to achieve a parking orbit at 100 nautical miles. The upper stage and payload will remain in a parking orbit for 8 minutes.

At approximately 21 minutes MET, the Centaur will perform a translunar injection burn. The Centaur will stay attached to the LTT/lander for about 40 minutes and will then separate and remain in a translunar orbit.

Mid-course correction will take place at 3 days MET. (The legs could be deployed prior to mid-course correction if needed to minimize heat flux on them.) Following the mid-course burn, the LTT payload will coast until utilization of the retro burn for soft lunar landing. The lander will maneuver to landing position and the legs will be deployed. Cameras on-board the lander will communicate with Earth-based ground control to find as suitable a landing site as possible, free from large rocks and boulders.

Lunar Transit Telescope (LTT) Transportation



*Mission Elapsed Time (DD/HH:MM)

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LTT ACTIVATION TIMELINE (Titan IV/Centaur)

The following pages describe the activation and verification activities that will occur prior to LTT science operations. The activity duration times, as well as the beginning and ending times, are shown in mission elapsed time (MET). Many duration times are estimated based on past preliminary design telescope studies.

The power system used for the LTT/lander will be RTGs (radioisotope thermoelectric generator) which will be activated several days before the mission begins. Closer to launch, the launch pad cooling system for the RTGs will be disconnected from the vehicle. The guidance, navigation, and control system will also be activated before launch. Outgassing will begin and continue throughout the mission. Housekeeping data will be available via omni antennas after shroud release.

The assumed launch date for this assessment is on 9/16/2002. Due to launch window constraints, the Centaur will have one near-term opportunity to reach the moon; therefore, it will stay in a parking orbit for only eight minutes. For a different assumed launch date, the Centaur will most likely spend a longer time in parking orbit which will result in a later landing time on the lunar surface.

As described on the previous charts, the Centaur will perform a translunar injection burn approximately 21 minutes into the mission. After Centaur main engine cut-off, the lander will begin communicating with the Centaur to obtain an attitude reference.

LTT TIMELINE ACTIVITIES	DURATION	MET DD/HR:MIN START	END
* Activate G&NC subsystems on launch pad		0/00:00	
* Begin outgassing		0/00:00	
* ASRM separation		0/00:02	
* Payload Fairing separation		0/00:04	
* Initiate housekeeping data stream		0/00:05	
* Titan IV Stage 1 separation		0/00:05	
* Titan IV Stage 2 separation		0/00:09	
* Centaur first burn		0/00:09	
* Centaur MECO-1		0/00:13	
* Centaur/Lander in parking orbit	0/00:08	0/00:13	0/00:21
* Command Lander from Centaur telemetry	0/00:05	0/00:13	0/00:18
* Activate sun sensors and star trackers	0/00:02	0/00:18	0/00:20
* Centaur Translunar injection burn	0/00:07	0/00:21	0/00:28
* Centaur MECO-2		0/00:28	
* Lander RCS on standby	0/00:01	0/00:28	0/00:29
* Establish separation attitude	0/00:02	0/00:29	0/00:31
* Centaur RCS inhibit	0/00:01	0/00:31	0/00:32
* Lander attitude reference from Centaur	0/00:03	0/00:32	0/00:35

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LTT ACTIVATION TIMELINE (Centaur separation)

In preparation for separation, the Centaur will transfer command control to the LTT/lander and the lander on-board computer will be activated. The LTT/lander will separate from the Centaur at approximately one and a half hours after lift-off. The Centaur will perform a small burn in order to separate from the LTT/lander. When there is an adequate distance between the LTT/lander and Centaur (45 minutes is a best guess), the Centaur will perform a disposal burn to an unknown destination out of the path of the LTT/lander.

Following separation, tip-off rates will be removed using the inertial measurement unit outputs and the reaction control system to stabilize the LTT/lander spacecraft. A series of maneuvers will then be required to update the inertial measurement unit and possibly to a preferred thermal attitude. The inertial measurement unit will be updated once again and a state vector navigation update from the ground will be received before the LTT/lander maneuvers to an attitude suitable for a mid-course correction. Mid-course correction takes place approximately 3 days MET (mission elapsed time). Following the mid-course correction burn, the lander will have the opportunity to maneuver to a preferred thermal attitude. At a later time, the LTT/lander will maneuver to acquire stars for an inertial measurement unit update and will receive a lander state vector update to prepare for landing.

LTT TIMELINE ACTIVITIES	DURATION	MET DD/HR:MIN START	END
* Transfer command control to Lander from Centaur	0/00:26	0/00:35	0/01:01
* On-board Lander computer activation	0/00:21	0/01:01	0/01:22
* Centaur/Lander separation	0/00:10	0/01:22	0/01:32
* Enable Centaur RCS	0/00:01	0/01:32	0/01:33
* Centaur separation burn		0/01:33	
* Centaur disposal burn		0/02:18	
* Centaur Destination?			
* Activate Lander computer confidence test	0/00:21	0/01:32	0/01:53
* Activate Lander thrusters	0/00:02	0/01:53	0/01:55
* Remove tip-off rates	0/00:02	0/01:55	0/01:57
* Maneuver to acquire stars for IMU update	0/00:20	0/01:57	0/02:17
* Maneuver to preferred thermal attitude?	0/00:30	0/02:17	0/02:47
* Maneuver to acquire stars for IMU update	0/00:20	2/23:10	2/23:30
* State Vector Navigation update (from ground)	0/00:00	2/23:30	2/23:30
* Maneuver for burn attitude	0/00:30	2/23:30	3/00:00
* Mid-course correction		3/00:00	
* Maneuver to preferred thermal attitude?	0/00:30	3/00:01	3/00:31
* Maneuver to acquire stars for IMU update	0/00:20	3/20:16	3/20:36
* Lander state vector update	0/00:00	3/20:36	3/20:36

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LTT ACTIVATION TIMELINE (Lunar landing)

Prior to landing, the lander radar will be activated and checked out. The spacecraft will maneuver to a burn attitude and perform an additional maneuver for antennae orientation. Cameras will be present on the lander to provide pictures for ground control interface as the spacecraft descends. The LTT/lander will make a direct descent to a previously chosen site that will be adjusted upon descent as necessary. After a braking burn for a soft landing, the LTT/lander will land on the lunar surface hopefully at a site with minimal slope, free of rocks and boulders.

Immediately after landing, the LTT will acquire the Sun to determine the proper location of the LTT sunshade. The sunshade will then slowly rotate to the required location. The high gain antennae will be deployed and oriented as necessary for Earth communication. The pointing control system hardware and software will be activated to point the telescope line of sight to zenith and to compensate for landing on a slope, etc.

The science instrument command and data handling system is activated beginning at about 3 days and 22 hours MET. Pyrotechnic fasteners will fire allowing the aperture cover to retract. The focal plane data stream is initiated to obtain the required data to be used in telescope alignment and focusing activity. The focal plane is then rotated to the correct position to allow the stars to track across the CCD array properly. The telescope is then aligned and focused for this position. Mirror focusing and adjustment is expected to last about 7 days. The focal plane will be rotated again once the proper focusing is obtained if necessary.

LTT TIMELINE ACTIVITIES	DURATION	START	END
		MET DD/HR:MIN	
* Activate/checkout lander radar	0/00:10	3/20:36	3/20:46
* Maneuver to burn attitude	0/00:30	3/20:46	3/21:16
* Maneuver Lander/LTT for antennae orientation	0/00:30	3/21:16	3/21:46
* Braking burn for soft landing	0/00:15	3/21:46	3/22:01
* Lander/LTT on Lunar surface		3/22:01	
* Sun Acquisition	0/01:00	3/22:01	3/23:01
* Deploy/Orient High Gain Antenna	0/00:30	3/22:01	3/22:31
* Pointing ctrl system h/w & s/w activated - rqmt	0/00:02	3/23:01	3/23:03
* Science instrument C&DH system activated	0/04:00	3/22:31	4/02:31
* Rotate sunshade/telescope	0/01:00	3/23:03	4/00:03
* Open aperture cover	0/00:20	4/00:03	4/00:23
* Initiate focal plane data stream	0/00:30	4/00:23	4/00:53
* Rotate focal plane	0/01:00	4/00:53	4/01:53
* Telescope alignment activity	7/00:00	4/01:53	11/01:53
* Primary mirror focus & fine adjustment	7/00:00	4/01:53	11/01:53
* Secondary mirror focus & fine adjustment	7/00:00	4/01:53	11/01:53
* Tertiary mirror focus & fine adjustment	7/00:00	4/01:53	11/01:53
* Rotate focal plane	0/01:00	11/01:53	11/01:5

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LTT ACTIVATION TIMELINE (Science operations)

Following the mirror focusing and adjustment activity, science instrument activation will begin and data acquisition will be initiated. It is expected that the science instrument verification will require about 90 days as was planned for the Hubble Space Telescope. Lunar Transit Telescope operations are planned to begin about 113 days after lift-off.

LTT TIMELINE ACTIVITIES	DURATION	MET DD/HR:MIN START	END
* Begin science instrument activation	11/15:00	11/02:53	22/17:53
* Initiate data acquisition	0/02:00	22/17:53	22/19:53
* Science instrument verification	90/00:00	22/19:53	112/19:53
* Begin LTT operations		112/19:53	

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5.3 CELESTIAL MECHANICS AND STAR TRACKS

LUNAR TRANSIT TELESCOPE SITE SELECTION CRITERIA

This chart lists some requirements that were assumed in establishing an acceptable lunar location for the LTT (reference Section 2.4). The LTT light shade is to be designed to keep both sunlight and earthshine out of the LTT aperture. Subsequent charts will present an analysis of the light shade with respect to its shape, size, and orientation.

The probability of avoiding large terrain slopes, boulders, and craters would be maximized by locating the LTT in one of the Mare (pronounced mah-ray) regions of the Moon as far from craters and boulders as possible.

For continuous line-of-site communications with Earth, the LTT site must be chosen such that the Earth remains completely above the lunar horizon. The motion of the Earth as viewed from the LTT site is caused by the optical librations. This phenomenon is explained more fully in subsequent charts.

The dense, bright starlight of the Galactic plane would interfere with the LTT astronomy. Therefore, opportunities for viewing near the Galactic pole should be provided. A lunar latitude of about 30 degrees provides Galactic North Pole viewing during one portion of the lunation cycle.

An LTT site near the trailing edge of the Moon (eastern limb) would result in somewhat lower micrometeoroid velocities at the telescope. However, other than the Mare Crisium area, there are not many candidate locations near the eastern limb that are not surrounded by rough terrain and craters.

The lunar sunrise and sunset relative to light shade design are discussed in detail in this section. The star tracks on the CCD detector array were assessed and the pointing accuracy requirements for the telescope were determined. At 40 degrees latitude the telescope should be zenith pointed to approximately 5 arcminutes in the meridian plane and 5-10 degrees perpendicular to the meredian plane. The detector array should be aligned E-W to an accuracy of approximately 7 arcseconds.

LUNAR TRANSIT TELESCOPE SITE SELECTION

SITE SELECTION CONSIDERATIONS

- AVOID EARTH SHINE IN THE TELESCOPE APERTURE
- AVOID SUNLIGHT IN THE TELESCOPE APERTURE
- MINIMIZE SUNSHADE HEIGHT/WEIGHT
- MINIMIZE MICROMETEROID FLUX
- CONTINUOUS COMMUNICATION WITH EARTH
- PROVIDE PERIODS OF VIEWING NEAR PERPENDICULAR TO GALACTIC PLANE
- MINIMUM TERRAIN SLOPE, ROCKS, AND CRATERS
- MINIMIZE DUST BY LOCATING FAR FROM MANNED ACTIVITIES

AZIMUTH OF LUNAR SUNRISE AND SUNSET

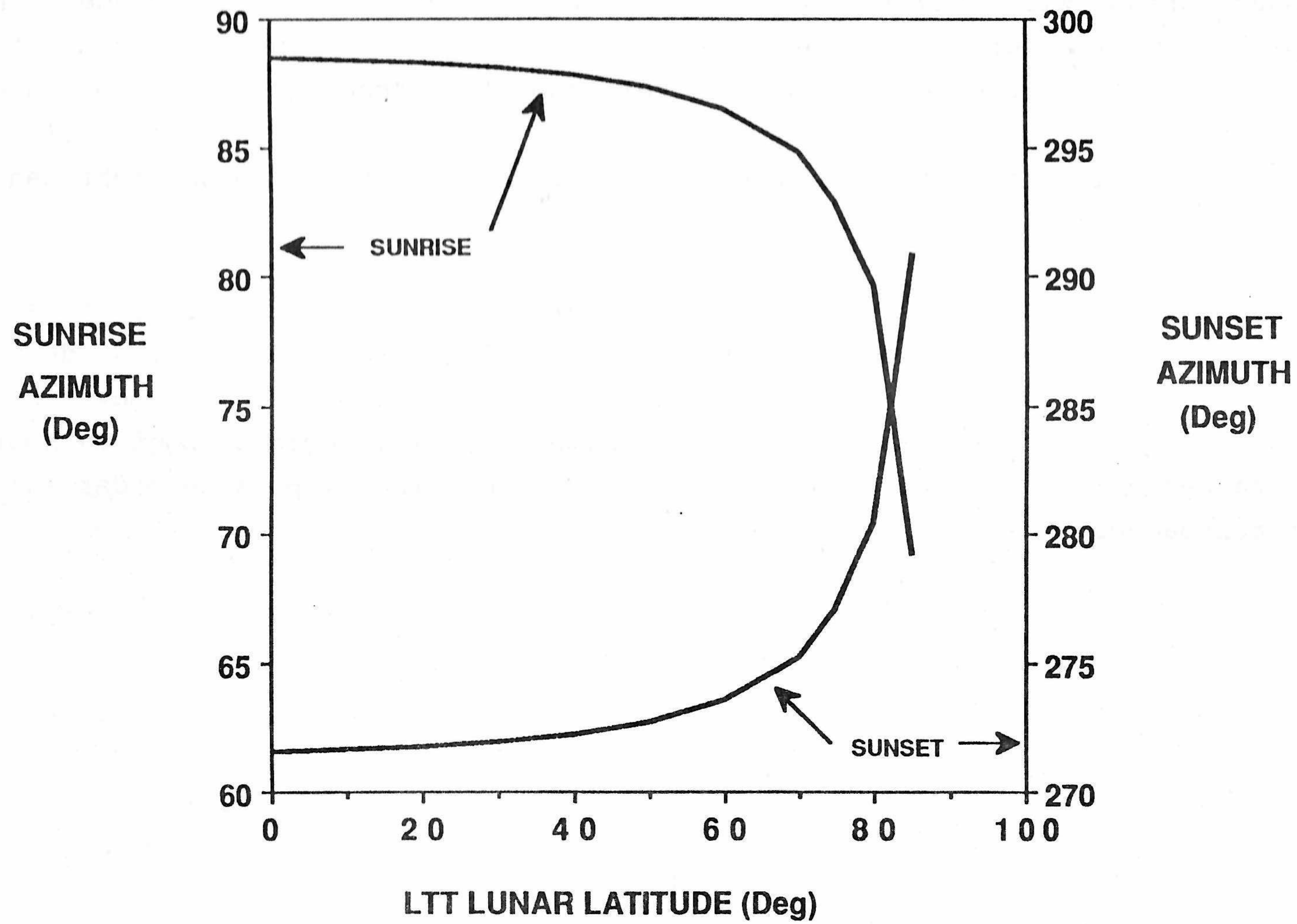
For sun shading, the optimum orientation is to point the high side of the shade due south. As the Moon rotates on its axis, the Sun, as viewed from the telescope site, will rise in the east (approximately), move across the lunar sky (at about 12 degrees/day), then set in the west. If the altitude of the Sun causes the edge of the Sun to peek above the plane of the shade opening, then the inside of the shade will be illuminated. This will occur for slightly more than half of the year when the Sun's limb has a positive declination (north of the equatorial plane). During this period, for northern lunar latitudes, the Sun will begin to rise above the lunar horizon a few degrees north of east (azimuth less than 90 degrees) and the Sun will finish setting a few degrees north of west (azimuth greater than 270 degrees). This happens because the Moon's axis is tilted slightly (about 1.5 degrees) from the ecliptic pole, producing lunar "summer" and "winter" as the Earth-Moon system revolves around the Sun.

The accompanying figure shows sunrise and sunset azimuths for northern latitudes when the Sun is at its maximum declination (worst case) of 1.5 degrees (lunar midsummer).

In generating these data, the radius of the Sun was not neglected. Therefore, the term "sunrise" refers to the instant that the first ray of light from the upper edge of the Sun is visible on the horizon. "sunset" refers to the instant that the last ray of light from the upper edge of the Sun disappears below the horizon.

AZIMUTH OF LUNAR SUNRISE AND SUNSET

FOR VARIOUS LTT SITE LATITUDES



SUNRISE RELATIVE TO THE LTT LIGHT SHADE

During lunar summer, the direction of the rising Sun is a few degrees north of east. The direction of the setting Sun is a few degrees north of west. Therefore, during lunar summer, the first rays of light from the Sun as it begins to rise above the lunar horizon will unavoidably illuminate the inside of the light shade. The same problem will occur near sunset. Some type of light baffling interior to the light shade will be required to keep the telescope from "seeing" these "hot" areas. Eventually, as the Sun continues to rise, it will move to an azimuth that will place it beneath the plane of the light shade opening. At this time, no sunlight can enter the shade opening.

This chart illustrates a rising Sun as viewed edgewise to the LTT light shade for an LTT viewing declination of 40 degrees and a light shade angle of 62 degrees. For this case, the Sun rises at an azimuth of 87.8 degrees and finally moves beneath the plane of the light shade opening at an azimuth of 94.3 degrees about 19 hours after the beginning of sunrise. The same geometry occurs in opposite order as sunset approaches.

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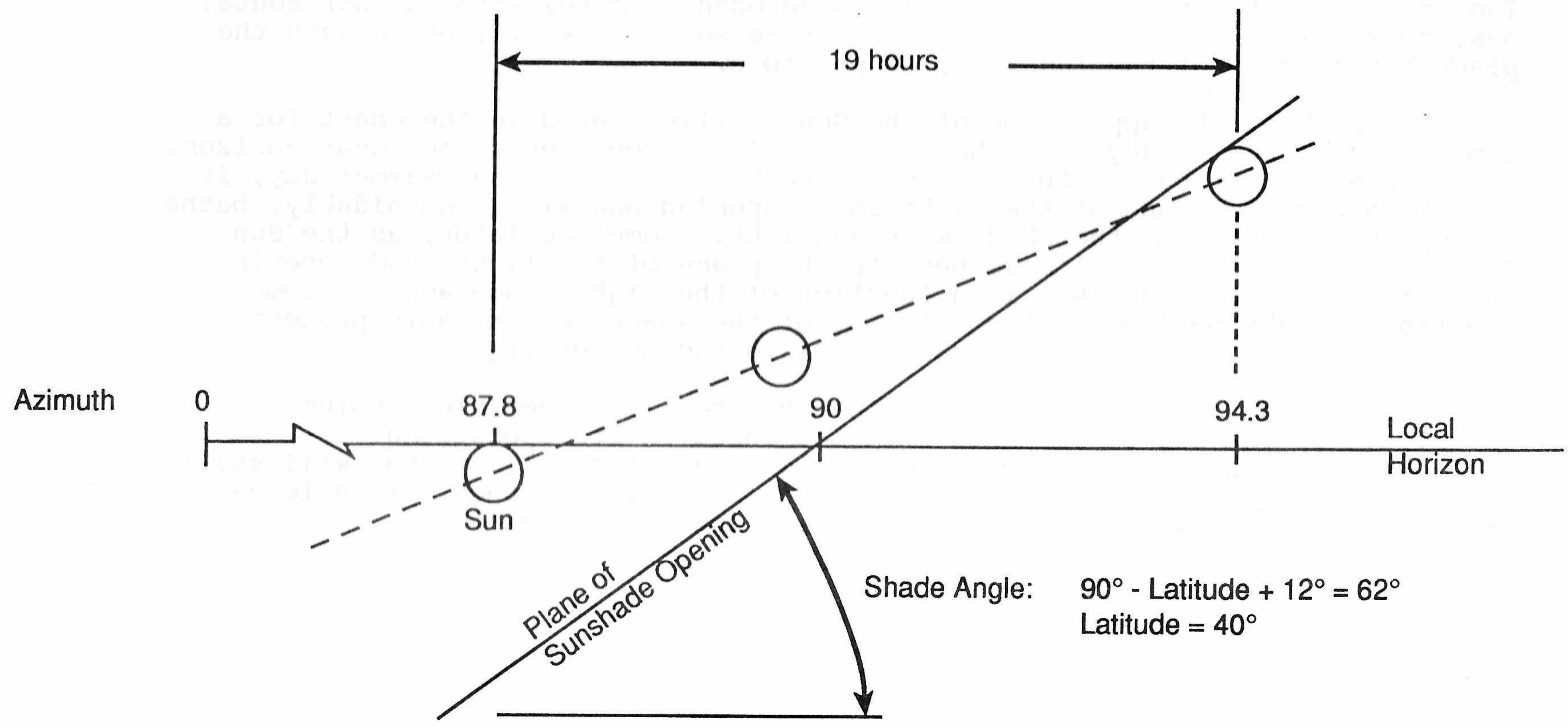
Sunrise Relative To the LTT Light Shade



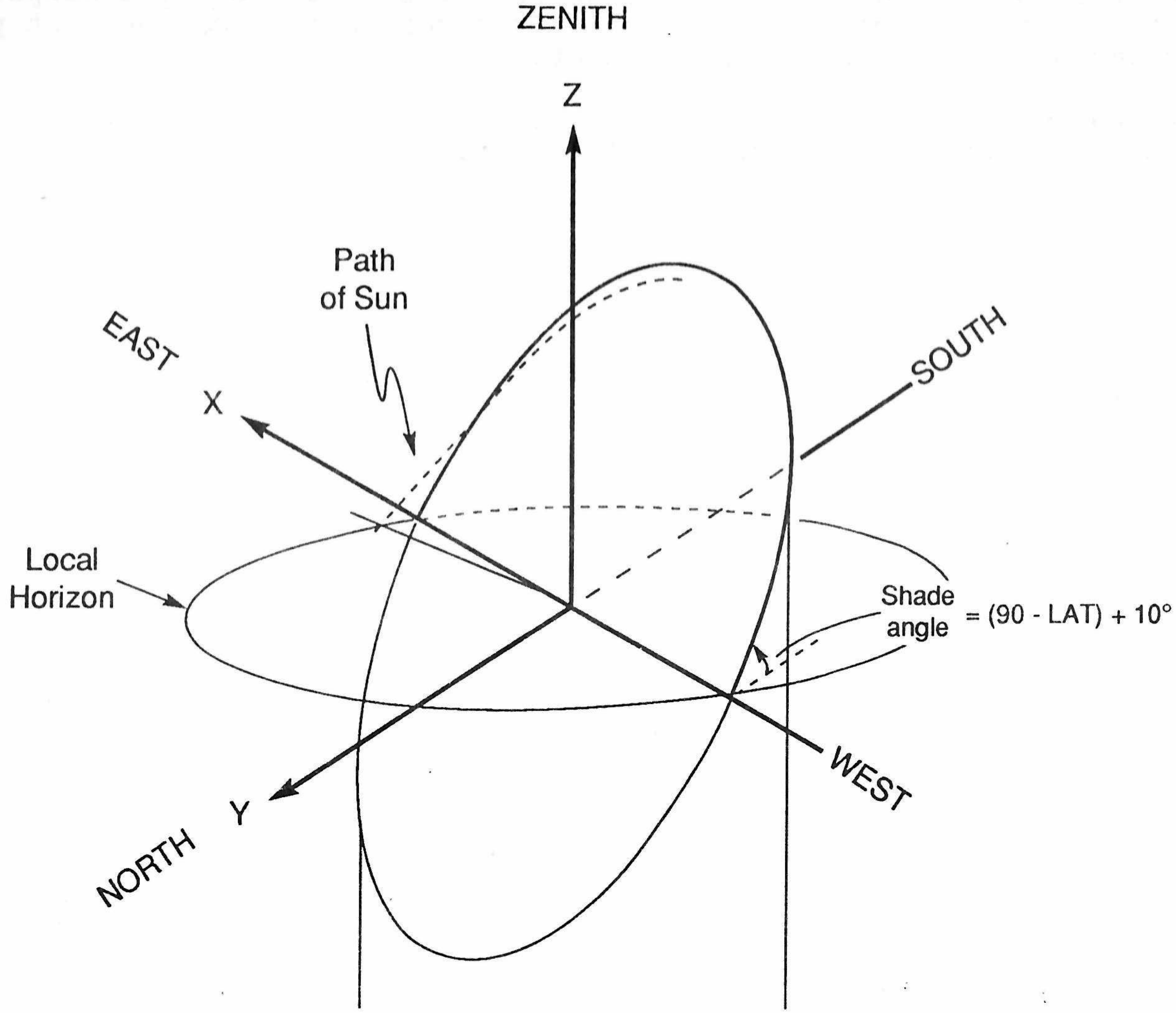
LTT LIGHT SHADE ORIENTATION FOR SUNLIGHT PROTECTION

This chart illustrates a shade oriented high-side-south. This is the optimum orientation of the light shade for providing maximum protection against sunlight entering the telescope aperture. As the Moon turns on its axis, the Sun will appear on the eastern horizon, rise in the sky to a maximum altitude determined by the telescope site latitude, and then set on the western horizon. The angle that the plane of the light shade opening makes with the horizontal must be sufficiently large so that the entire solar disk will be beneath the plane for as much of the lunar day as possible.

The path of the upper limb of the Sun is illustrated in the chart for a typical lunar summer day. The horizontal ellipse represents the lunar horizon. When the Sun first begins to rise above the horizon on a lunar summer day, it will be above the plane of the light shade opening and will, unavoidably, bathe the opposite inside of the shade with sunlight. Sometime later, as the Sun continues to rise, it will move beneath the plane of the light shade opening. The time it takes to do this is a function of the light shade angle. Some additional light baffles on the interior of the shade can probably protect sufficiently against this unavoidable Sunrise and Sunset light.

Light shade angle may be constrained by the launch vehicle's volume envelope. If the LTT is launched with the shade in its operational configuration, then a 60 degree shade angle is about the limit that will still allow the LTT to fit within the launch vehicle's payload space. If a longer payload shroud is available, this constraint could be relaxed.

LTT Light Shade Orientation



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LTT LIGHT SHADE BAFFLES

This chart illustrates how additional baffles, internal to the light shade, could effectively "hide" the illumination of the sunshade from the telescope. This illumination on the inside surface of the light shade occurs during lunar summer at sunrise and sunset and is unavoidable. This inside illumination lasts only for a portion of the daylight period until the Sun completely disappears below the plane of the light shade opening. It occurs again for the last portion of the lunar day as the Sun approaches sunset.

This diagram is only a schematic and is intended to illustrate the principal only. It does not represent an actual baffle design.

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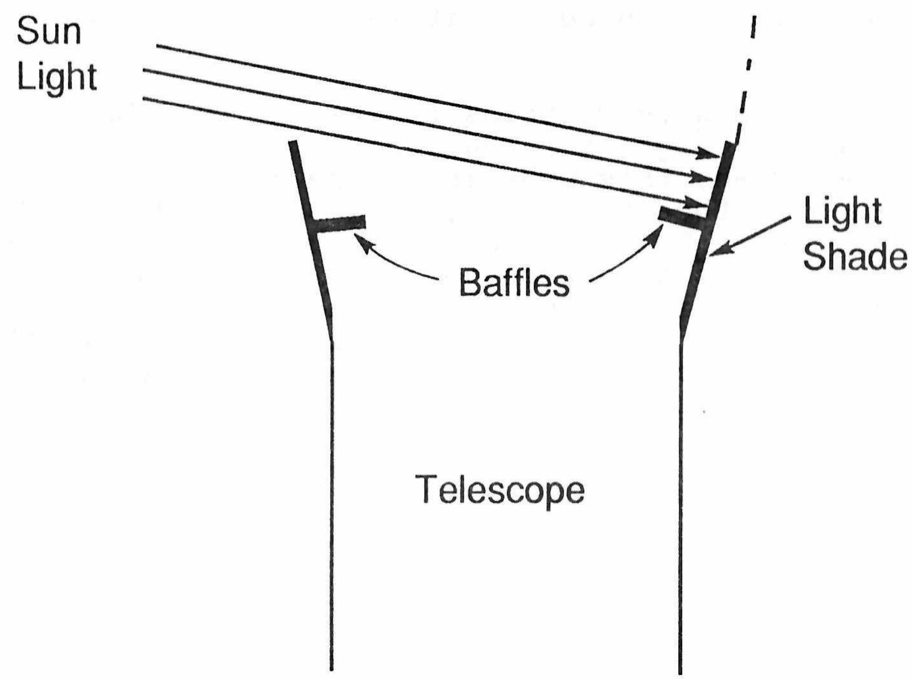
LTT Light Shade Baffles



During lunar summer sunlight will enter the light shade at sunrise and sunset. Additional baffling, as illustrated, will be necessary.

LTT LIGHT SHADE ANGLE FOR EARTH SHINE AVOIDANCE

The LTT light shade can also serve to shade the telescope from earthshine. The same side of the Moon always faces the Earth. Hence the Earth, as observed from a lunar site, would always appear in the same part of the Sky. Actually, during part of each month, the Moon exposes more of its north polar region of the Earth and two weeks later exposes more of its south polar region to the Earth. Also during part of each month, the Moon exposes more of its left side to the Earth and two weeks later exposes more of its right side to the Earth. These apparent "nodding" motions of the Moon are called optical librations in latitude and longitude. The Earth, therefore, appears to remain over the central region of the lunar surface moving between $\pm$ 6.7 degrees latitude and $\pm$ 8 degrees longitude.

The region of the lunar sky that always contains the Earth could be described by a cone of approximately 11 degrees. When the Earth is observed from an equatorial site at the prime meridian, this cone is in the direction of zenith. When it is observed from a northeastern site, this cone is directed toward the southwest above the horizon. This is the scenario that is illustrated in the accompanying chart. Also shown is a representation of the plane of the light shade opening. Remember that the high side of the shade must face south to provide maximum protection from sunlight. Adjusting the shade angle until the plane of the shade opening is just tangent to the cone surface ensures that light from the Earth can never enter the telescope tube.

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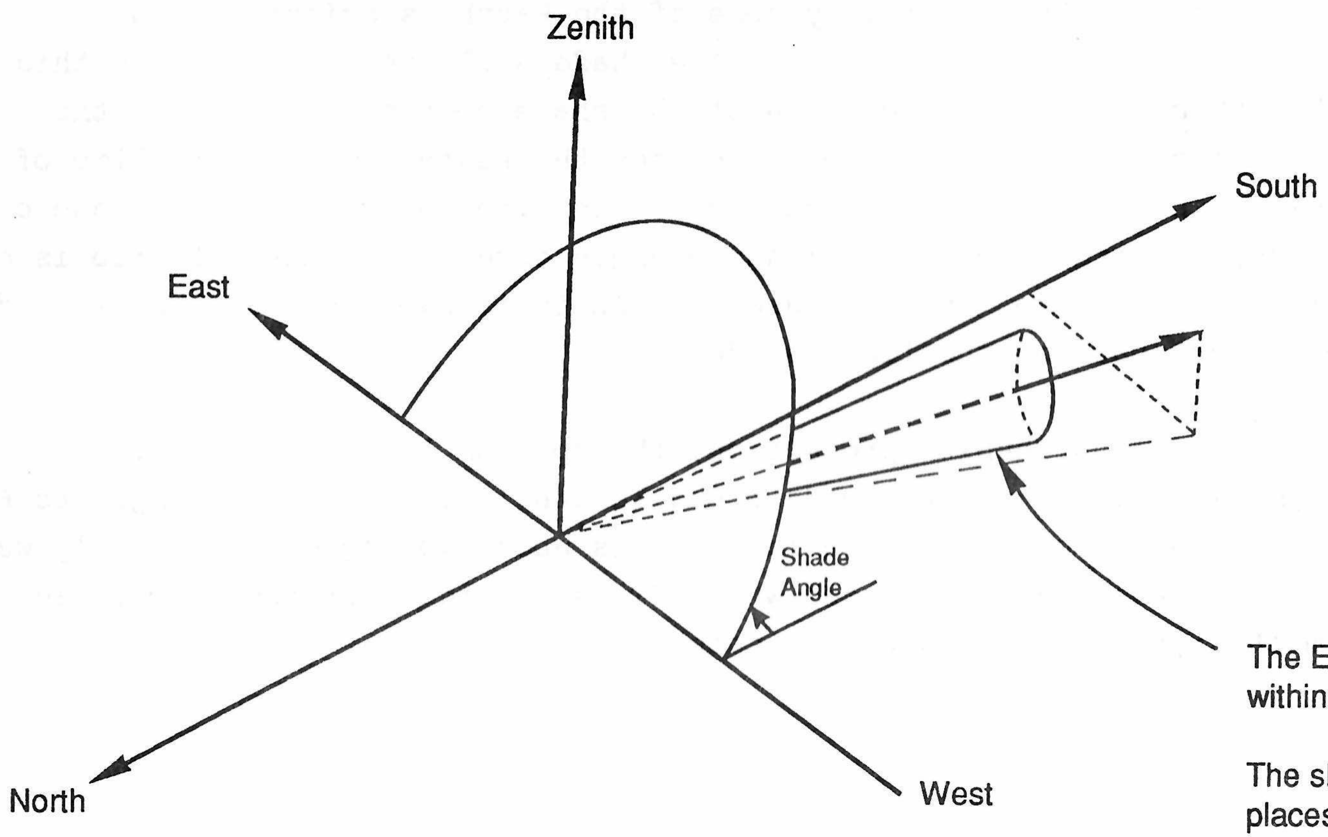
LTT Light Shade Angle For Earth Shine Avoidance



The Earth is always within this 10° cone

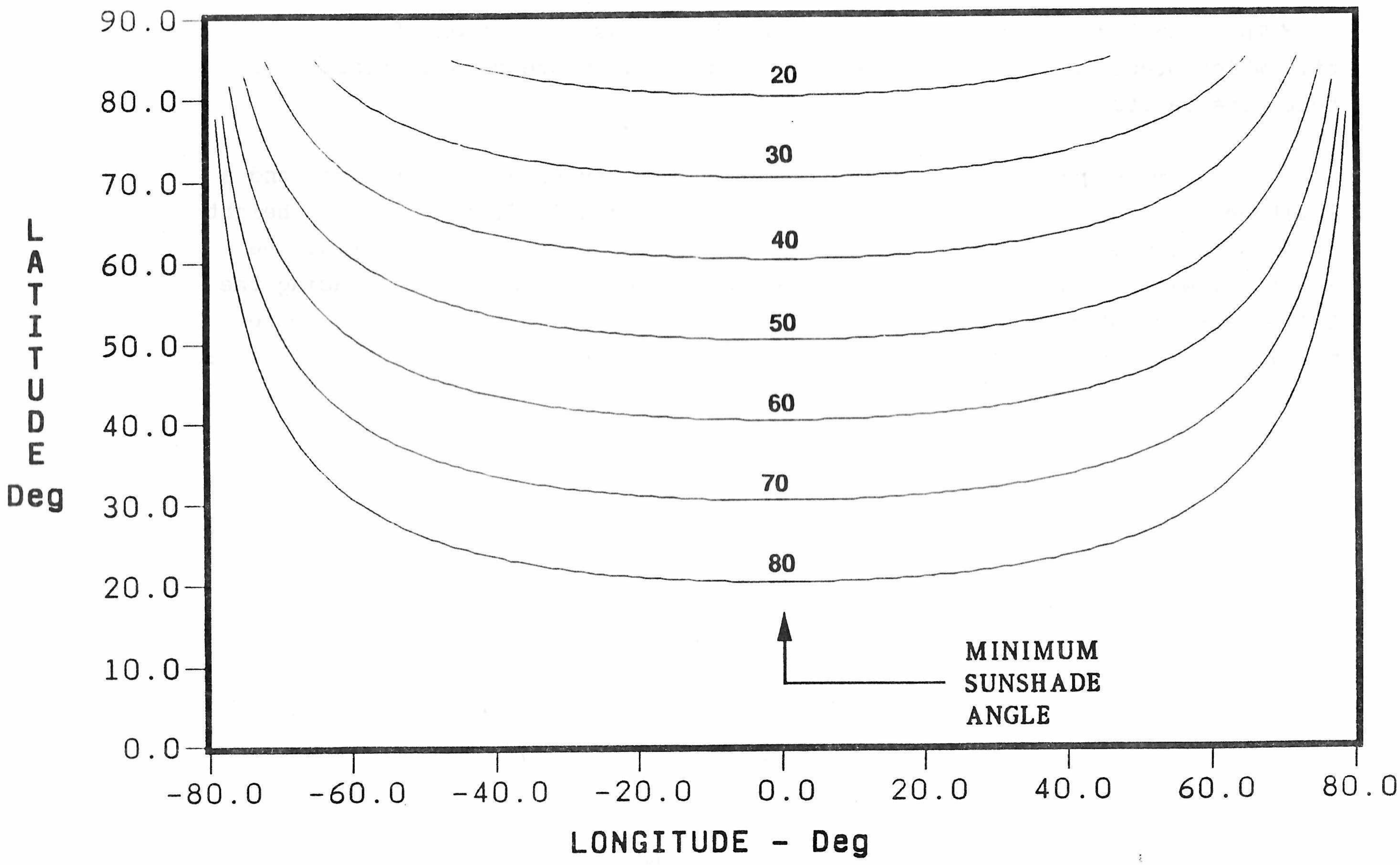
The shade angle which places the plane of the light shade opening tangent to the 10° cone will preclude earth shine

TRANSIT TELESCOPE LIGHT SHADE ANGLES REQUIRED TO AVOID EARTHSHINE

The high side of the light shade must face south to provide maximum protection against sunlight. If the LTT site is located northeast or northwest of the central region, the visibility cone of the Earth is oriented in a southwestern or southeastern direction. The shade angle required to keep this earth visibility cone beneath the plane of the shade opening is shown in the accompanying chart. As the LTT site approaches the eastern or western limb of the Moon, higher shade angles are required to keep the Earth below the plane of the light shade opening. The minimum shade angle occurs when the LTT site is on the prime meridian because, in this case, the Earth is due south of the LTT and behind the tallest part of the light shade.

For purposes of earthshine protection, the optimum location for LTT is on the central meridian. If the payload shroud length limits the shade angle to 60 degrees, then the minimum acceptable latitude is about 40 degrees. The only way to keep the shade angle at 60 degrees when LTT is located near the limb is to locate at much higher latitudes as indicated.

TRANSIT TELESCOPE SUNSHADE ANGLES
REQUIRED TO AVOID EARTH SHINE



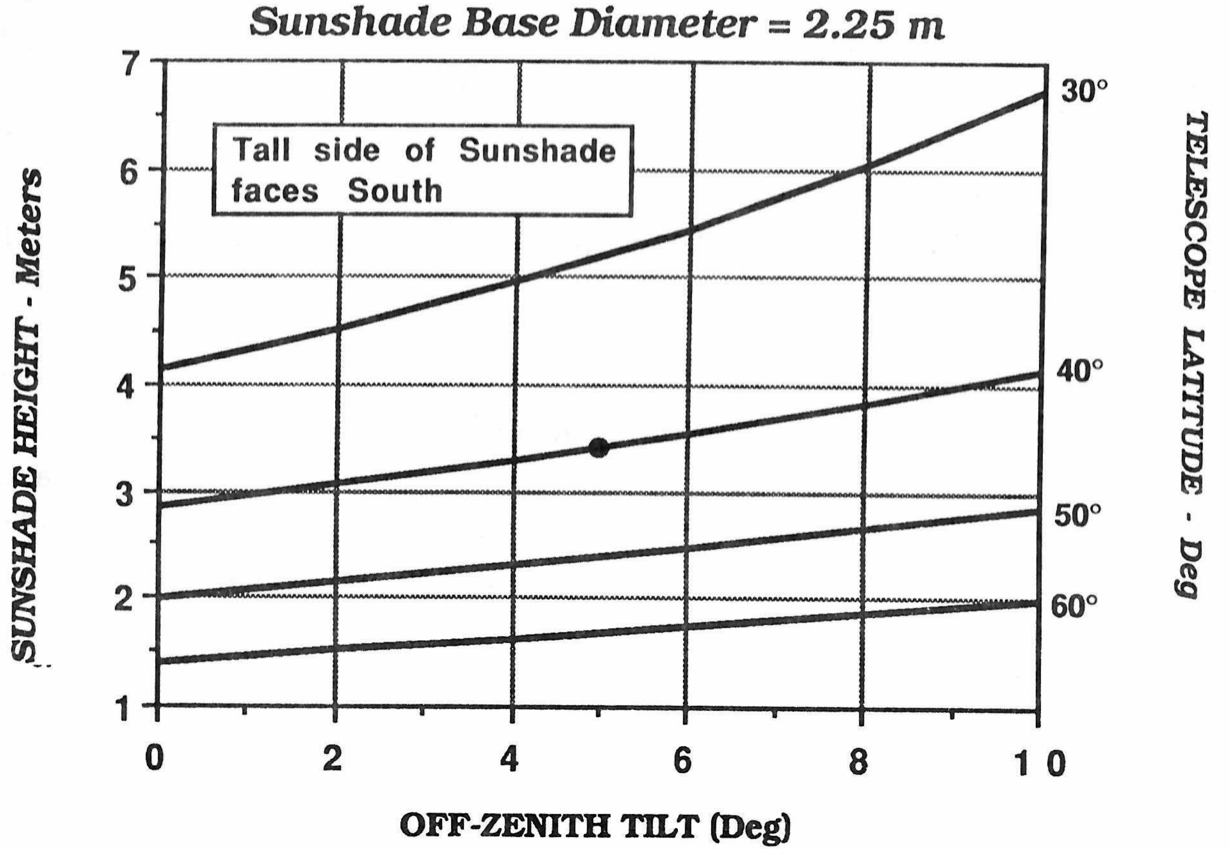
LTT SUNSHADE HEIGHT

This chart shows minimum LTT sunshade height as a function of off zenith tilt which precludes sun interference. These results are parameterized over the telescope latitudes.

The example shown on the chart is for an LTT having a tilt of 5 deg and a latitude of 40 degrees which results in a truncated cylindrical sunshade height of about 3.5 meters. At this latitude, viewing can be continuous during the lunar day without fear of violating the sun incidence constraint. Placing the LTT at a high latitude to avoid direct sunlight may be in conflict with other mission requirements. For example, viewing the center of the galactic plane can only be done from latitudes near the equatorial plane for small field of view telescopes.

LUNAR TRANSIT TELESCOPE SUNSHADE HEIGHT WHICH ENSURES NO SUNLIGHT ENTRY

(TRUNCATED CYLINDER SUNSHADE)



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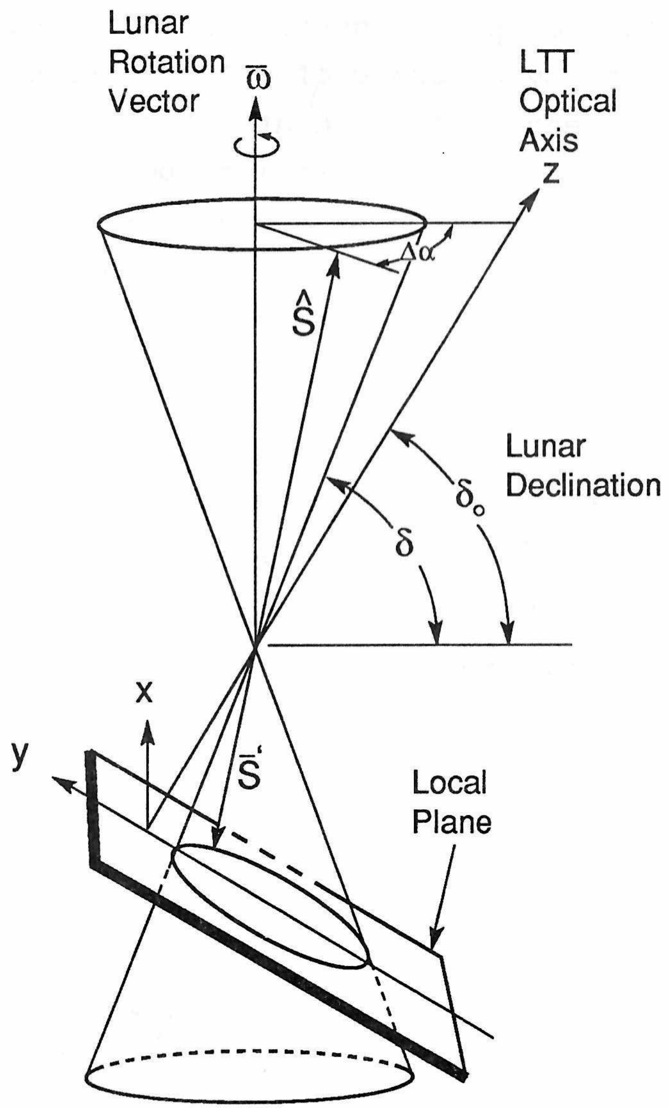
STAR IMAGE TRACK ON THE LTT FOCAL PLANE

A convenient way of visualizing how a star image moves in the focal plane of the LTT is to consider the telescope fixed, with the stars in the lunar sky moving. This apparent motion of the stars is caused by the rotation of the Moon on its axis. The stars will appear to move in concentric circles centered about the Moon's polar axis. As the Moon rotates, a vector, S, from the telescope to a star will remain on the surface of a cone as depicted in the figure. Using the "Pin-hole Camera" analogy, a ray of light from the star will pass through the pin-hole and intersect the telescope focal plane as shown by vector S'. This vector also traces out a surface of a cone.

The path of the image on the focal plane, therefore, is defined by the intersection of a plane and a cone, which yields a conic section. For the case illustrated in the figure, the resulting image path is an ellipse. Parabolic paths are obtained when the LTT viewing declination (and the star's declination) is near 45 degrees. Hyperbolic paths result when the LTT viewing declination is less than 45 degrees.

Star Image Track on the LTT Focal Plane



12/9/91:jm

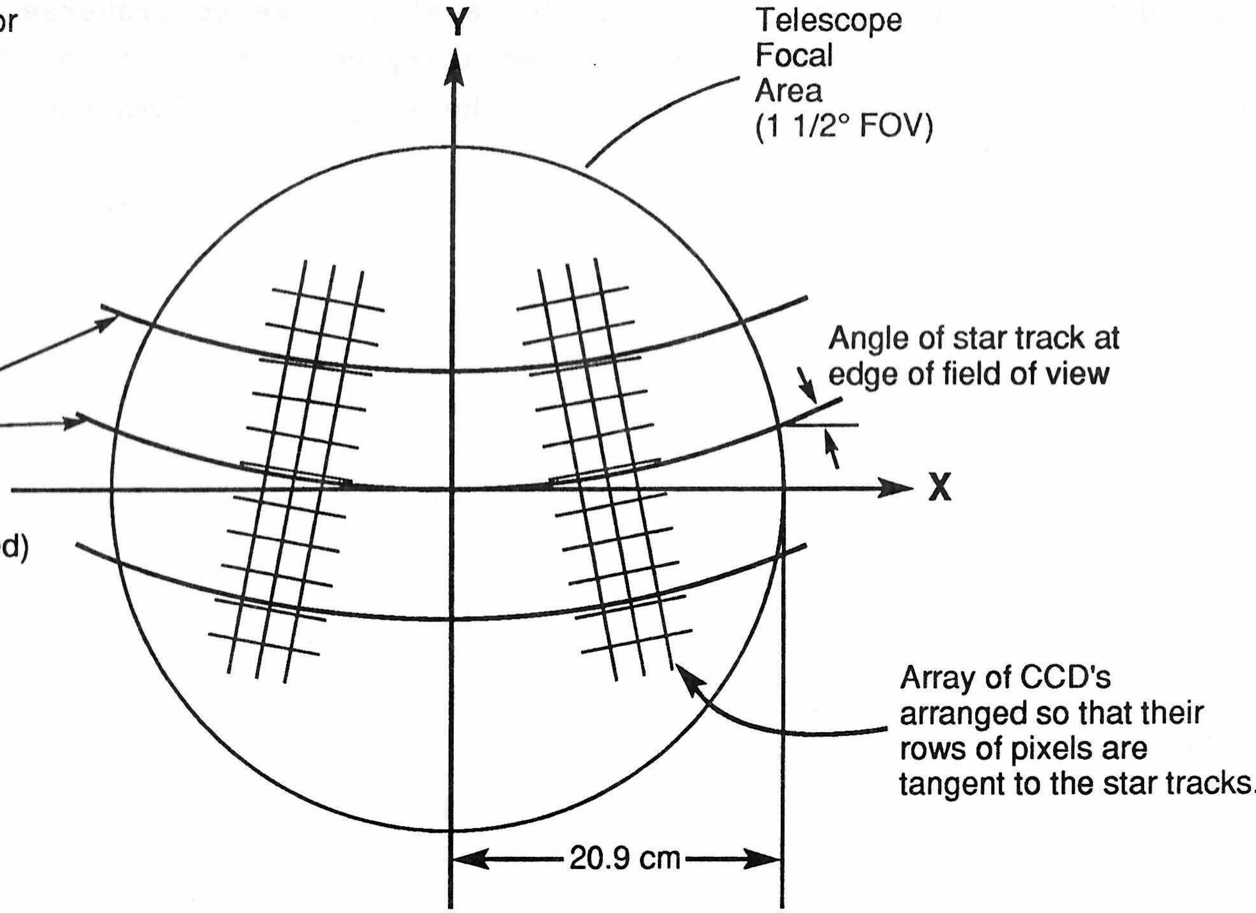
CHARACTERISTICS OF THE LTT FOCAL PLANE SENSOR ARRAY

The LTT sensor, on which the optical system projects the star images, will consist of a flat plate covered by a mosaic of CCD's (charge coupled devices). For the baselined effective focal length of 800 centimeters and 1.5 degrees field of view, the LTT focal area will be a circular region about 21 centimeters in radius. As the Moon rotates on its axis, an image of a star, as it passes through the LTT field of view, will traverse across the plate of CCD's on a slightly curved path. The curvature of these star image paths depend on the pointing declination of the LTT and the declination of the star.

Each CCD is made up of rows of pixels. The size of these pixels may be as small as 5 microns. These rows may be on the order of 3000 pixels in length. The pixel rows could be slightly curved to match the curvature of the star image tracks. The CCD's are arranged on the sensor plane so that their rows of pixels are tangent to the expected star image tracks.

Characteristics of the LTT Focal Plane Sensor Array

If conventional integrating CCD's are used, the focal plane array must be precisely designed for the planned pointing declination



Assumes 800 cm effective focal length and 1 1/2° total field of view

CCD ALIGNMENT ON THE FOCAL PLANE ARRAY

After deciding on exactly what lunar declination the LTT is to point, the CCD sensors must be arranged so that each row of pixels on the CCD is tangent to the curved star image tracks. This enables a star image to traverse a single pixel row as the Moon rotates. This is necessary so that a single CCD can accurately accumulate the pixel charges as the image moves down the row.

If the LTT is slightly misaligned, the curvature of the image track will be different from what was planned. In this case, the pixel rows (especially at the edge of the array) will no longer be tangent to the tracks and the track of a star image will cross a CCD at a small angle to the pixel rows. It is assumed that a star image, as it moves across a CCD, should not "smear" across a row of pixels more than 0.1 pixel. This means that the angle between the pixel row and the star image track cannot exceed 6.87 arcseconds (assuming 3,000 pixels comprise a row).

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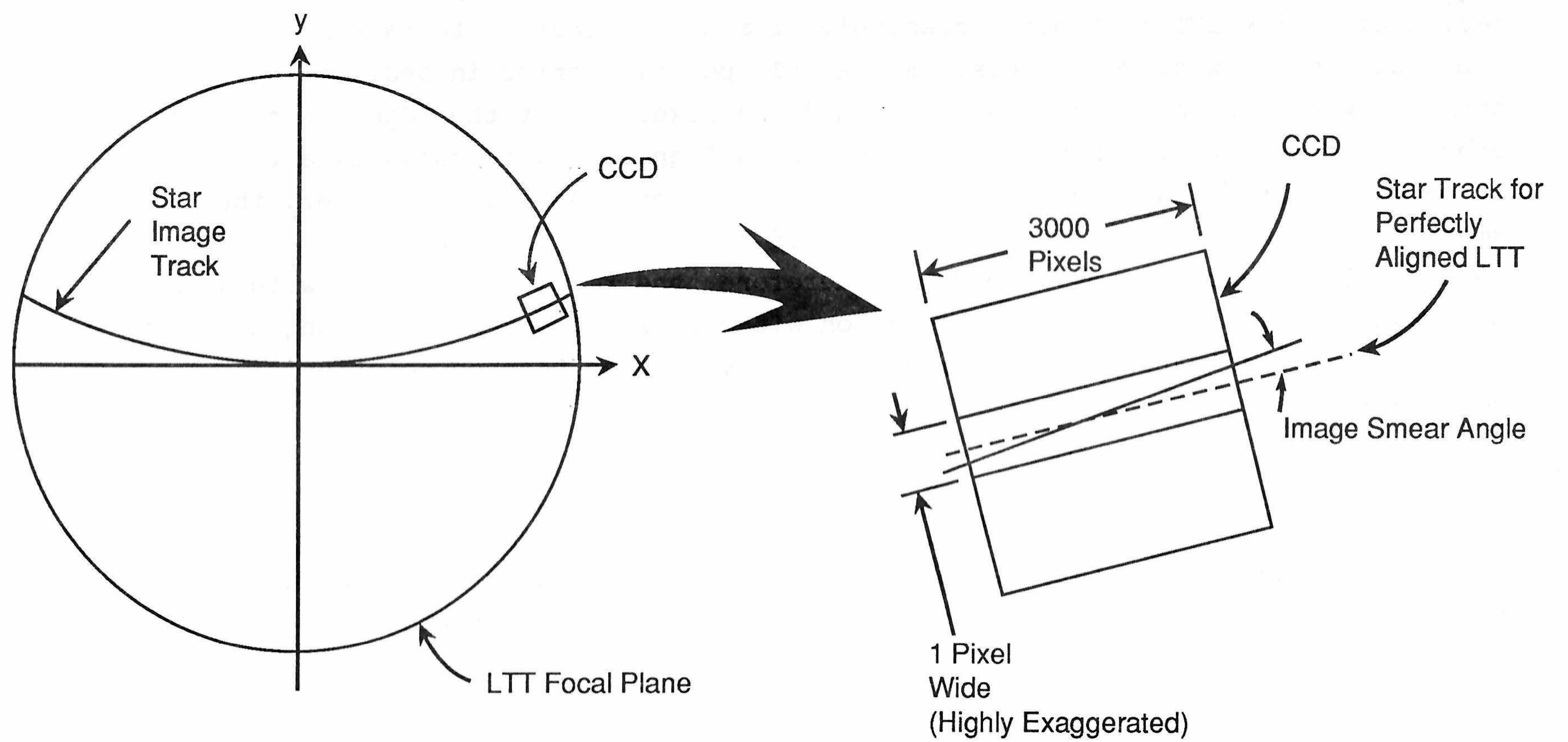
CCD Alignment On The Focal Plane Array



The image smear should not exceed .1 pixels on a single CCD.
The maximum allowed smear angle

$$= \tan^{-1}\left(\frac{.1}{3000}\right)$$

$$= 6.87 \text{ arcseconds}$$

STAR TRACK "SMEAR" RESULTING FROM LTT POINTING ERROR

If a focal plane array is constructed for an assumed LTT pointing declination, the LTT must point precisely at that declination to have no image smear across a row of CCD pixels. For an LTT pointing error in declination, the angle between the resulting star track and the pixel row at the edge of the focal area can be computed. This angle is the "smear" angle and is shown as a function of pointing error and "planned" pointing declination. As stated earlier, the smear angle cannot exceed 6.87 arcseconds if the maximum allowable smear is .1 pixel during transit across one CCD. Therefore, the required accuracy with which LTT must point to the planned declination without exceeding the smear angle constraint is indicated as 5 arcminutes for a 40 degree planned pointing declination.

Remember that this analysis is predicated on the use of the conventional integrating CCD's that are currently in use. These stringent pointing requirements could be relaxed significantly by using a sensor that does not require custom fabrication to match an expected star track curvature.

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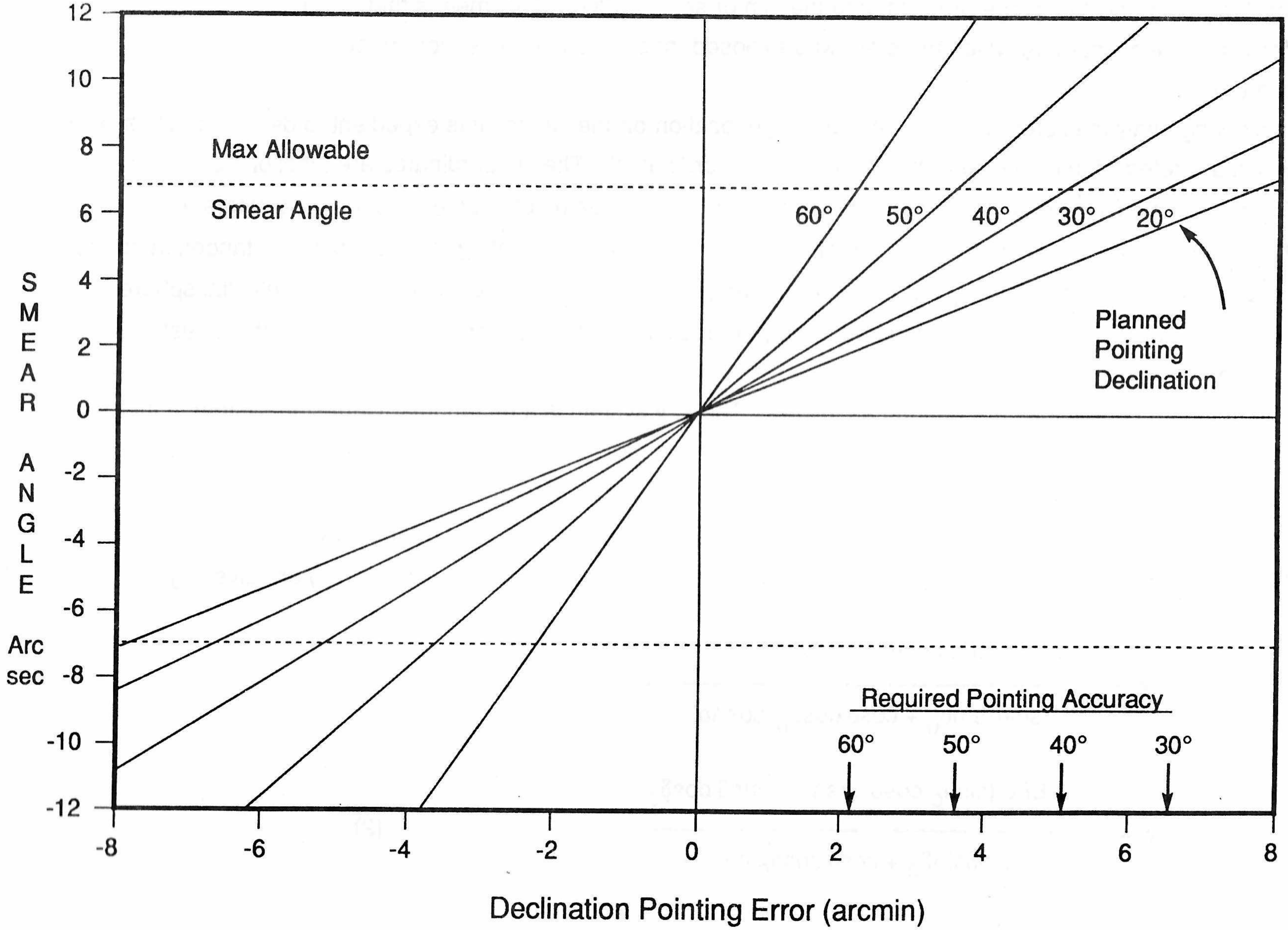
Star Track Smear Resulting From LTT Pointing Error



Star Tracks on the LTT Focal Plane Array

In photographic Astrometry, the geometry of the "gnomonic" projection has been used as the basis of the model for describing how the image projects onto the film or sensor plane. This means that it is assumed that the camera (or telescope), at which the plate was exposed, has the same projection properties as a "pinhole" camera.

In deriving analytic expressions for the star image location on the sensor, it is expedient to define variables (x,y), which are referred to in the literature as "standard coordinates". These coordinates are functions of the star's Right Ascension, α and declination, δ (referenced to Lunar equator, of course) and the parameters α and δ_0.

The standard coordinates are the coordinates at which a plane representing the focal plane is tangential to the celestial sphere. The point (α_0, δ_0) is called the tangential point and is the location on the celestial sphere where the optical axis (which is assumed to be perpendicular to the sensor plane) intersects the celestial sphere.

The standard coordinates are defined as follows: The x, y system is a rectangular Cartesian system in the plane of the focal plane sensor. For LTT, the positive y axis points toward the North and the positive x axis points Eastward. Now, if LTT is not aligned exactly in the plane of the local meridian, x must point in the direction of $\bar{\omega} x \bar{z}$, where $\bar{z}$ is a vector representing the telescope optical axis and $\bar{\omega}$ is the Lunar spin vector.

The standard coordinates of an object at (α, δ) with respect to the tangential point (α_0, δ_0) are given by

$$x = \frac{EFL \cos\delta \quad \sin\Delta\alpha}{\sin\delta \ \sin\delta_0 + \cos\delta \cos\delta_0 \ \cos\Delta\alpha} \qquad (1)$$

$$y = \frac{EFL\ (\sin\delta_0 \cos\delta \cos\Delta\alpha \ - \sin\delta \cos\delta_0)}{\sin\delta\sin\delta_0 + \cos\delta \cos\delta_0 \cos\Delta\alpha} \qquad (2)$$

The quantity, EFL, in equations (1) and (2) is the Effective Focal Length of the Telescope. Using the pinhole camera analogy, EFL is the distance from the pinhole to the film plane. Equations (1) and (2) are derived as follows:

First, the components of a vector to a star at a right ascension and declination of (α, δ) are determined in the local telescope reference frame. In the figure on Page 314, XYZ is an inertial cartesian frame where the XY plane is the moon's equatorial plane and X is in the star's meridian plane. In the $x_L y_L z_L$ Telescope reference system the z_L axis is the telescope optical axis and is pointing at some declination, δ_0, the $x_L y_L$ plane is parallel to the LTT focal plane, and the x_L axis points generally eastward. As the moon rotates on it's axis, the telescope reference system rotates about the moon's spin axis (z).

Referring to page 314, a unit vector to the star is

$$\hat{S} = \begin{matrix} \cos\delta \\ 0 \\ \sin\delta \end{matrix} \qquad \text{in the inertial XYZ frame.}$$

This vector can be transformed to the LTT $x_L y_L z_L$ system by carrying out the following matrix operations:

$$\begin{matrix} 1 & 0 & 0 \\ 0 & \cos(90-\delta_0) & \sin(90-\delta_0) \\ 0 & -\sin(90-\delta_0) & \cos(90-\delta_0) \end{matrix} \quad \begin{matrix} \cos(\Delta\alpha+90) & \sin(\Delta\alpha+90) & 0 \\ -\sin(\Delta\alpha+90) & \cos(\Delta\alpha+90) & 0 \\ 0 & 0 & 1 \end{matrix} \begin{pmatrix} \cos\delta \\ 0 \\ \sin\delta \end{pmatrix}$$

Making the substitutions:

$$\cos(\Delta\alpha + 90) = -\sin\Delta\alpha$$
$$\sin(\Delta\alpha + 90) = \cos\Delta\alpha$$
$$\cos(90 - \delta_o) = \cos\delta_o$$
$$\sin(90 - \delta_o) = \cos\delta_o$$

and carrying out the matrix multiplication:

$$\begin{pmatrix} -\sin\Delta\alpha & \cos\Delta\alpha & 0 \\ -\sin\delta_o \cos\Delta\alpha & -\sin\delta_o \sin\Delta\alpha & \cos\delta_o \\ \cos\delta_o \cos\Delta\alpha & \cos\delta_o \sin\Delta\alpha & \sin\delta_o \end{pmatrix} \cdot \begin{pmatrix} \cos\delta \\ 0 \\ \sin\delta \end{pmatrix}$$

$$S = \begin{pmatrix} -\cos\delta \sin\Delta\alpha \\ \sin\delta \cos\delta_o - \cos\delta \sin\delta_o \cos\Delta\alpha \\ \sin\delta \sin\delta_o + \cos\delta \cos\delta_o \cos\Delta\alpha \end{pmatrix} \qquad (3)$$

Equation (3) above gives the components of a unit vector which points to some star of interest. This vector is expressed in a frame of reference located at the LTT where the z_L axis is the optical axis and the $x_L y_L$ plane is perpendicular to the optical axis. The x_L axis points approximately eastward. It points exactly in the direction of $w \times z_L$. The y_L axis points exactly in the direction of $z \times (w \times z_L)$.

As illustrated in the figure on Page 315, the LTT focal plane is parallel to the $x_L y_L$ plane and is displaced from it by the Effective Focal Length (EFL). So the $x_L y_L$ components expressed by equation (3) are not the components of the star's image on the focal plane. The star's image on the focal plane is formed by vector S as shown in the figure. The components of S can be found from the similar triangles formed by the two vectors and their z components. The z component of S is clearly -EFL, therefore,

$$\frac{\underline{S}}{S} = \frac{-EFL}{S_z}$$

and $$\vec{S} = \frac{-EFL}{S_z} \hat{S}$$

The x and y components of S above are the coordinates of the star image on the LTT focal plane and are:

$$x = \frac{EFL \cos\delta \ \sin\Delta\alpha}{\sin\delta \ \sin\delta_o + \cos\delta \ \cos\delta_o \ \cos\Delta\alpha} \tag{4}$$

$$y = \frac{EFL\,(\sin\delta_o \ \cos\delta \ \cos\Delta\alpha - \sin\delta \ \cos\delta_o)}{\sin\delta \ \sin\delta_o + \cos\delta \ \cos\delta_o \ \cos\Delta\alpha} \tag{5}$$

Equations (4) and (5) can be combined by eliminating $\Delta\alpha$. This results in a conic equation in x and y.

First, solve equation (5) for $\cos\Delta\alpha$.

$$\cos\Delta\alpha = \frac{\tan\delta\,(EFL \ \cos\delta_o + y \sin\delta_o)}{y \cos\delta_o - EFL \sin\delta_o} \tag{6}$$

Next, substitute (6) into (4) and solve for $\sin\Delta\alpha$

$$\sin\Delta\alpha = \frac{x \tan\delta}{EFL \sin\delta_o - y \cos\delta_o} \tag{7}$$

combining (6) and (7) with $\sin^2\Delta\alpha + \cos^2\Delta\alpha = 1$

yields

$$x^2 + (\sin^2\delta_o - \cos^2\delta_o \cot^2\delta) y^2 + 2\,EFL\,(\sin\delta_o \cos\delta_o + \sin\delta_o \cos\delta_o \cot^2\delta) y$$
$$+ EFL^2 (\cos^2\delta_o - \sin^2\delta_o \cot^2\delta) = 0 \qquad (8)$$

or

$$Ax^2 + By^2 + Cy + D = 0 \qquad (8a)$$

where $A = 1$

$B = \sin^2\delta_o - \cos^2\delta_o \cot^2\delta$

$C = 2\,EFL\,(\sin\delta_o \cos\delta_o + \sin\delta_o \cos\delta_o \cot^2\delta)$

and $D = EFL^2 (\cos\delta_o - \sin^2\delta_o \cot^2\delta)$

Consider the special case when the LTT is pointed at a lunar declination of 45 degrees and we wish to view a star whose lunar declination is 45 degrees.

For this case, $B = 0$, $C = 2EFL$, and $D = 0$

and equation (8) becomes

$$y = \frac{x^2}{2\,EFL}, \quad \text{a parabola.}$$

If the pointing declination is greater than 45 degrees, then B is positive and equation (8a) becomes

$$Ax^2 + By^2 + Cy + D = 0 , \qquad \text{an ellipse}$$

If the pointing declination is less than 45 degrees, then B is negative and equation (8a) becomes

$$Ax^2 - By^2 + Cy + D = 0 , \qquad \text{an hyperbola.}$$

Equation (8a) can be solved for y for any value of x by applying the quadratic formula:

Repeating equation (8a): $x^2 + By^2 + Cy + D = 0$

$$y = \frac{-C \pm \sqrt{C^2 - 4B\,(D + x^2)}}{2B} \tag{9}$$

and

$$\frac{dy}{dx} = \frac{-2X}{\pm\sqrt{C^2 - 4B\,(D + x^2)}}$$

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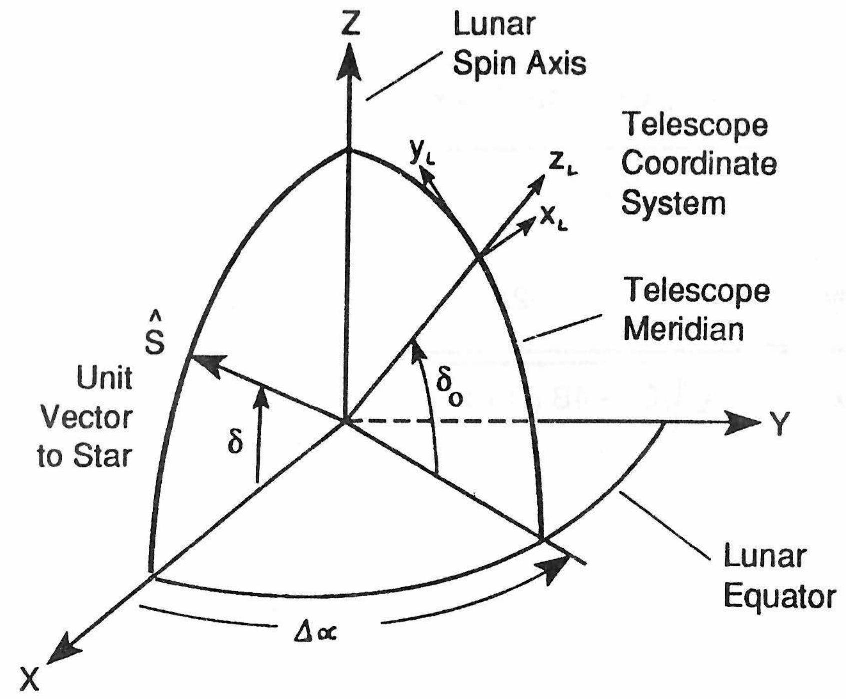
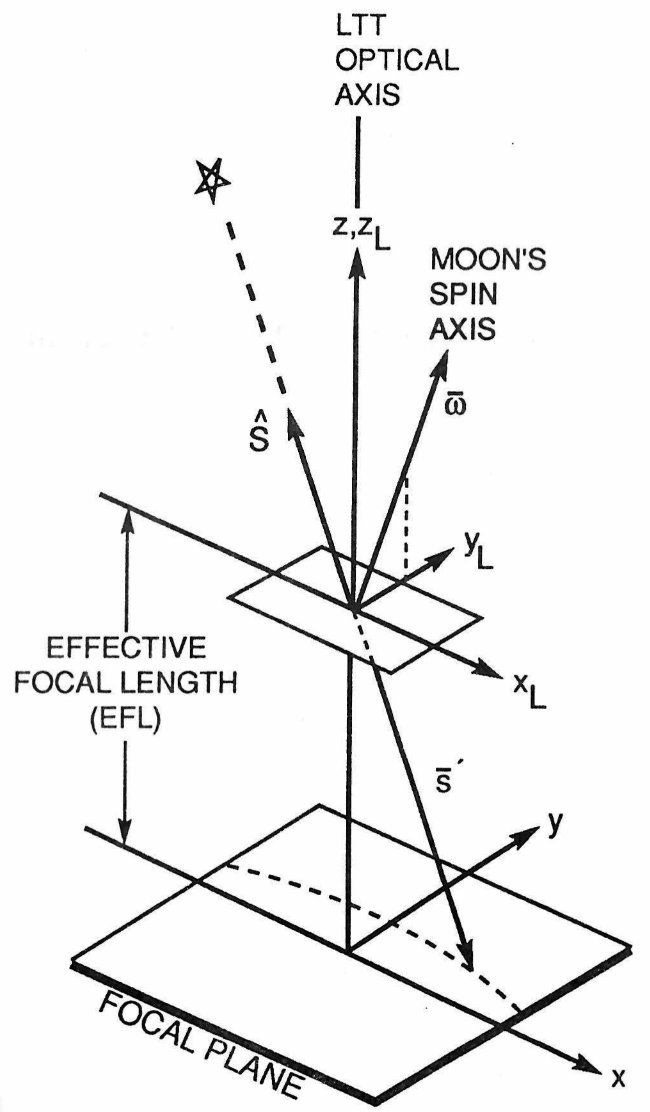
Model for Derivation of Star Image Path on the LTT Focal Plane Array



The LTT Focal Plane Coordinate System



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SECTION 6

6. CONFIGURATION TRADES AND OPTIONS **PAGE**

6.1 ATLAS IIAS TELESCOPE (1M)

LLT GUIDELINES USING THE ATLAS IIAS

At the request of NASA Headquarters, MSFC was asked what type telescope could be launched to the moon using a lower cost vehicle than the Titan IV. Either the Titan II or the Atlas IIAS was suggested as alternatives, the study team selected the Atlas IIAS/Centaur. A one meter telescope was suggested as a possibility. A top level study was conducted to determine what diameter telescope could be launched with the Atlas, and determine how the scientific benefits are impacted.

The science instruments and optics were scaled from the 2 m LTT and each discipline engineer was asked to estimate weights for his subsystem without doing detailed design analysis. At first it was thought that the Surveyor which was launched years ago with an Atlas/Centaur might be a candidate lunar lander for a small telescope. The Surveyor, Viking and Luna 16 were assessed as candidate carriers. However, as will be discussed, these did not prove fruitful, so a new lander was designed for the Atlas IIAS. But even with a new down sized lander, it could not support a 1 m telescope. The study team concluded that the telesope size was not the major design driver, and that many of the systems and components required to land on the moon do not depend on the size of the telescope or the lander.

LTT GUIDELINES USING THE ATLAS IIAS

1. ASSUME THE CENTAUR UPPER STAGE WITH THE ATLAS IIAS

2. ASSUME THE ATLAS CAN PUT 5,500 LB INTO TRANS LUNAR INJECTION

3. ASSUME HALF PROPELLANT AND HALF INERT LANDER AND TELESCOPE

4. SCALE THE 2 M LTT DOWN TO ABOUT 1 M. WHAT CAN BE LAUNCHED?

5. A THREE OR FOUR MIRROR TELESCOPE IS REQUIRED FOR A 2 DEG FOV

6. ESTIMATE SUBSYSTEMS AND SYSTEMS MASS TO SUPPORT A 1M LTT

7. ASSESS THE SURVEYOR LEV AS A CANDIDATE FOR A SMALL LTT

8. DESIGN A NEW LUNAR LANDER FOR THE ATLAS.

9. KEEP BOTH THE TELESCOPE AND LANDER DESIGNS SIMPLE AND LOW COST

10. EVALUATE THE SCIENCE IMPACT OF A SMALLER TELESCOPE

TELESCOPE SIZE AND SCIENCE RETURN

What science is lost if the telescope shrinks just a little? The nominal 2 m mission goal is to see down to a V magnitude of 27. This allows observaation of a normal spiral galaxy out to a redshift of 1 and a giant elliptical to a redshift of over 4, allowing detailed evolutionary studies. The volume of space sampled by a 1 m LTT will be more than six times smaller than a 2 m LTT; the spiral galaxy will be visible only to a redshift of 0.5, and the giant elliptical to a redshift of less than 2. Much of the evolution of normal galaxies occurs at a redshift between 0.5 and 1; the evolution of quasars, many of which seem to reside in giant elliptical galaxies, seems to occur most rapidly at redshifts around 3. The reduction of LTT capability would therefore significantly reduce its sensitivity to the most interesting cosmological investigations. To amplify this point: not only does the telescope see less far, but the particular distance range that is lost is exactly that range most critical for these important cosmological questions. Contrariwise, increasing the diameter a bit gives investigators more leverage on these questions but not dramatically so; it is in the regime between d = 1 m and d = 2.5 m that the ability of LTT to study galaxy evolution or not is determined. Naturally, a larger diameter telescope would allow a whole new set of questions to be addressed, but the 2 m telescope is well designed to address the galaxy evolution question which is one of the key cosmological issues.

The various galactic (Milky Way) studies suggested for the LTT are also reduced in scope by a smaller aperture. The smaller volume sampled means that only half as many objects are available for study, and so in a sense only half as much science is realized. However, the difference for these investigations seems to be quantitative, in contrast to the qualitative difference made to the cosmological studies. The most important effect is the loss of angular resolution in the survey of the diffuse infrared background, as accurate maps of this will be important when studying infrared point sources with ground based and SIRTF-like telescopes of lower angular resolution.

TELESCOPE SIZE AND SCIENCE RETURN

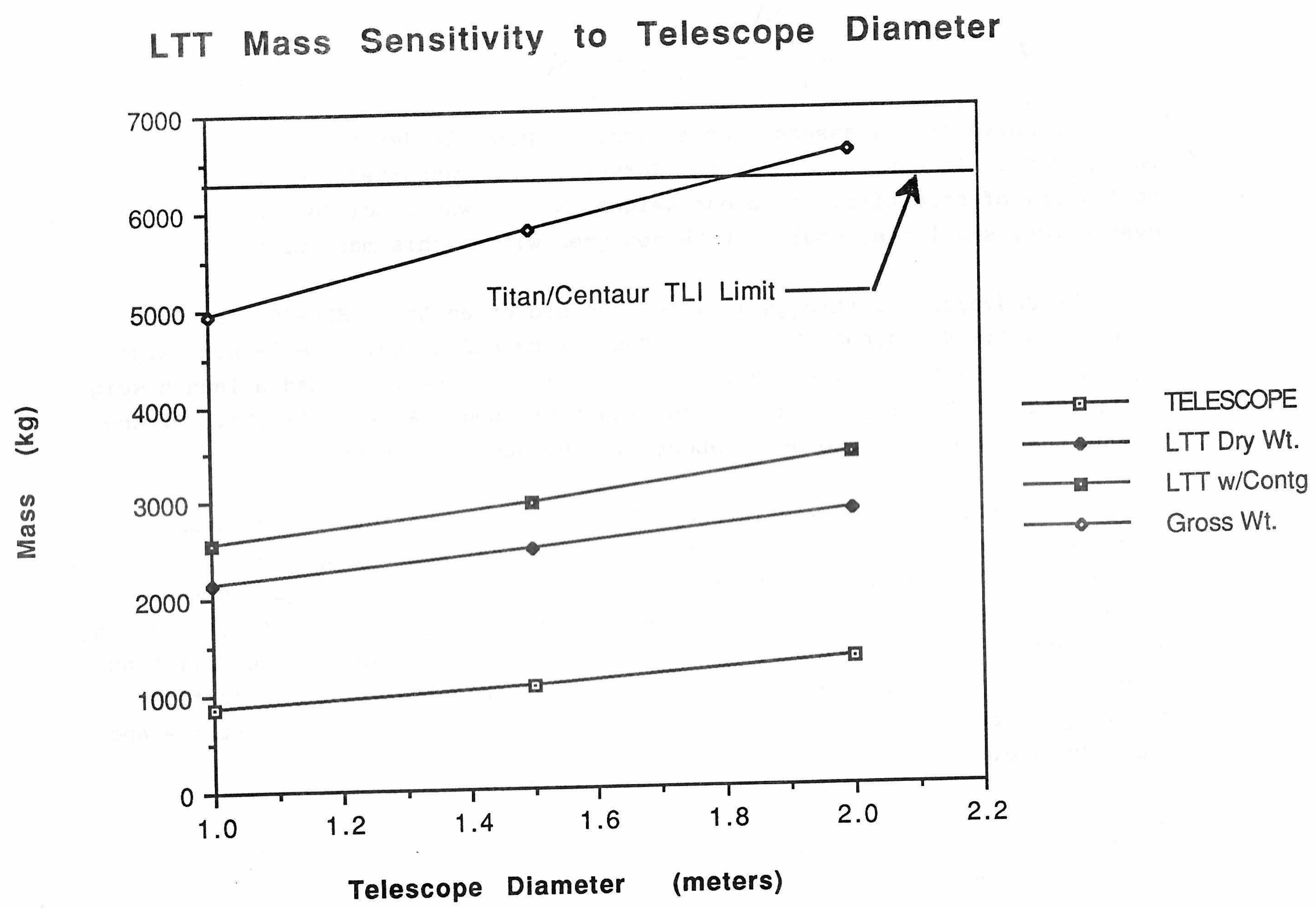
1. NUMBER OF PIXELS REDUCED TO FILL NEW FOCAL PLANE
2. SAME NUMBER OF FILTERS
3. DATA RATE APPROXIMATELY SCALES LINEARLY WITH FOCAL PLANE DIAMETER
4. PHYSICAL SIZE SCALES WITH FOCAL PLANE
5. MASS SCALES ROUGHLY WITH FOCAL PLANE AREA
6. THERMAL REQUIREMENTS UNCHANGED
7. ALIGNMENT REQUIREMENTS UNCHANGED

LTT MASS SENSITIVITY TO TELESCOPE DIAMETER

A study was performed to evaluate the impact of telescope diameter on vehicle mass. The results show a 33% weight reduction in telescope mass would result from halving the telescope diameter. An 18% accompanying reduction would be realized in the lander with a total vehicle weight reduction of approximately 24%.

These reductions, in addition to mirror mass, are in structure and propellant. The subsystem to support the one meter telescope and lander are the same as for the two meter design. This total reduction of approximately 1,590 kg does not seem to justify the losses in achievable science experienced through telescope diameter reduction (See science returns on previous page).

Values used in this analysis were based on the technologies and concepts used in the LTT integral design described in this report at a particular point in the design and may vary from the final design numbers. Future design may vary from these values but should be reflected proportionally throughout the system with mirror diameter.


LTT Mass Sensitivity to Telescope Diameter
7000
6000
5000
4000
3000
2000
1000
0
Mass (kg)
Titan/Centaur TLI Limit
1.0
1.2
1.4
1.6
1.8
2.0
2.2
Telescope Diameter (meters)
TELESCOPE
LTT Dry Wt.
LTT w/Contg
Gross Wt.

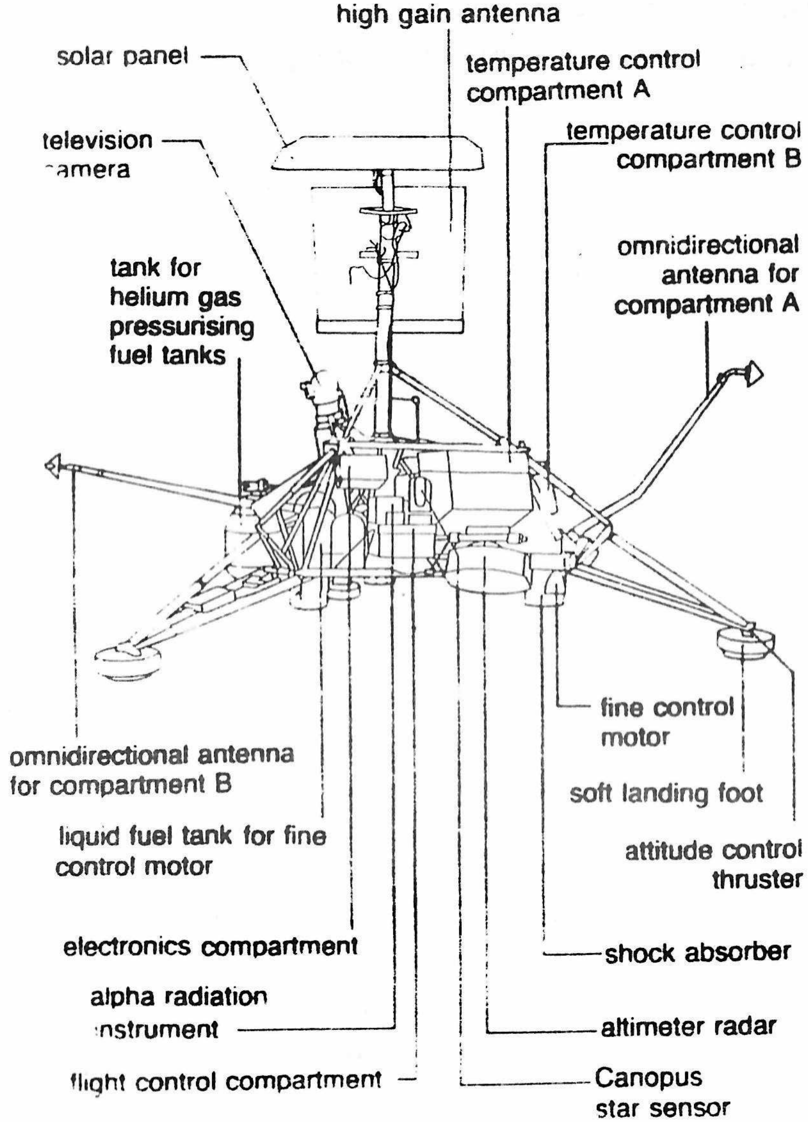
SURVEYOR

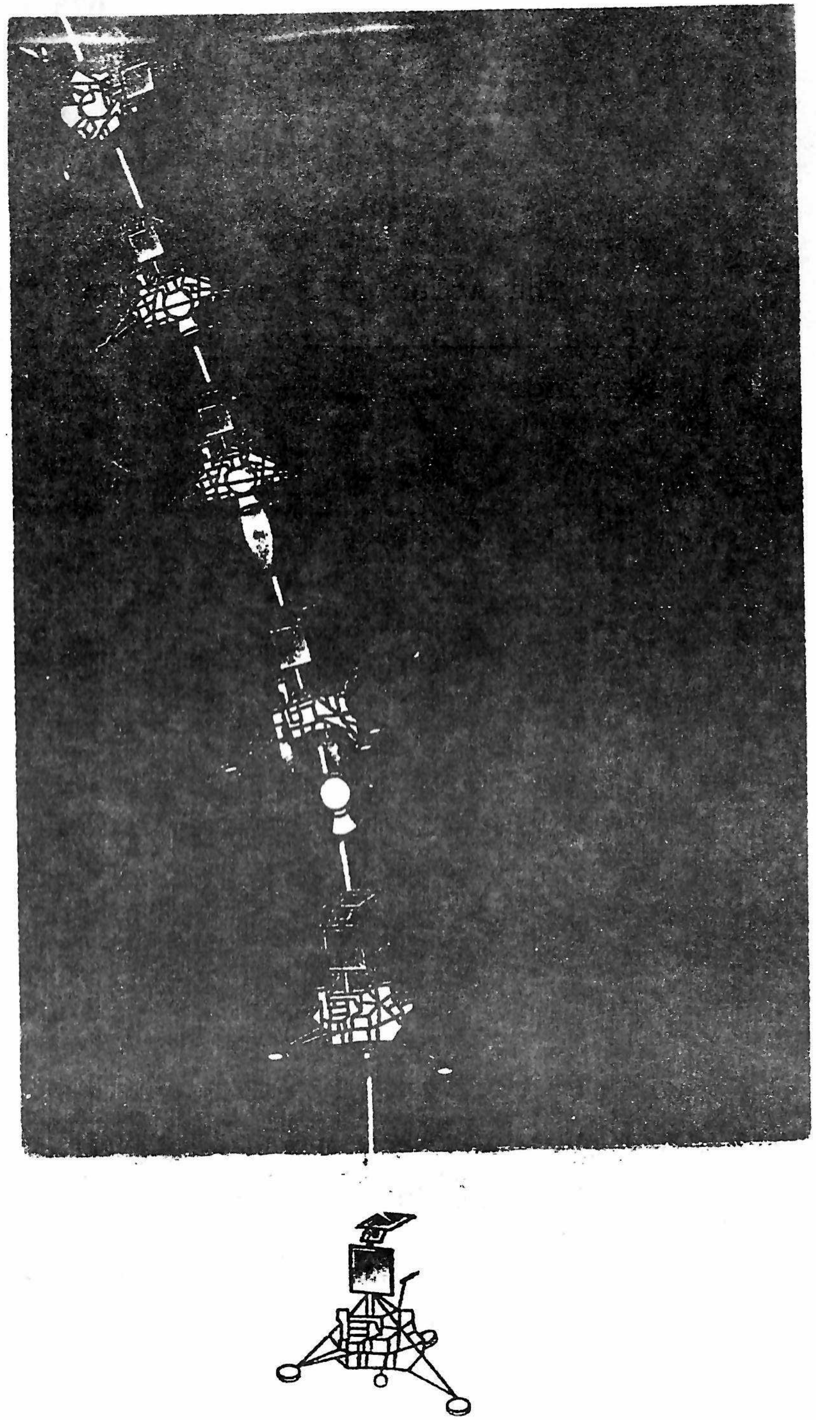
The Surveyor was assessed as a possible lunar lander for a small telescope on the moon. It was found that the Surveyor only supported from 27 to 52 kg (60 to 114 lb) of scientific equipment weight, and it was concluded that a telescope, even a very small one, could not be designed within this mass constraint.

The Surveyors, 1 through 7, were launched on an Atlas SLV-3C first stage with a 133,446 N (30,000 lbf) thrust Centaur second stage. The launch weight ranged from 998 to 1,134 kg (2,200 to 2,500 lb). Surveyor 7 had a launch weight of 1,038 kg (2,288 lb) and a landing weight of about 283 kg (625 lb). Of the weight that landed on the moon, about 114 lbs was science equipment.

The Surveyor lander was intended to set a large variety of payloads gently down on the moon. It used a jettisonable spherical solid propellant retro-rocket to cancel the velocity prior to landing. Three throttleable liquid vernier engines, of 75 N (16.8 lbf) each, were used for the final stage of soft landing. Solar cells and silver-zinc batteries were used as the power source, limiting operations to the lunar day. It was successful in gathering lunar surface data, taking pictures and testing to see if the lunar surface would support the Apollo Lunar Module.

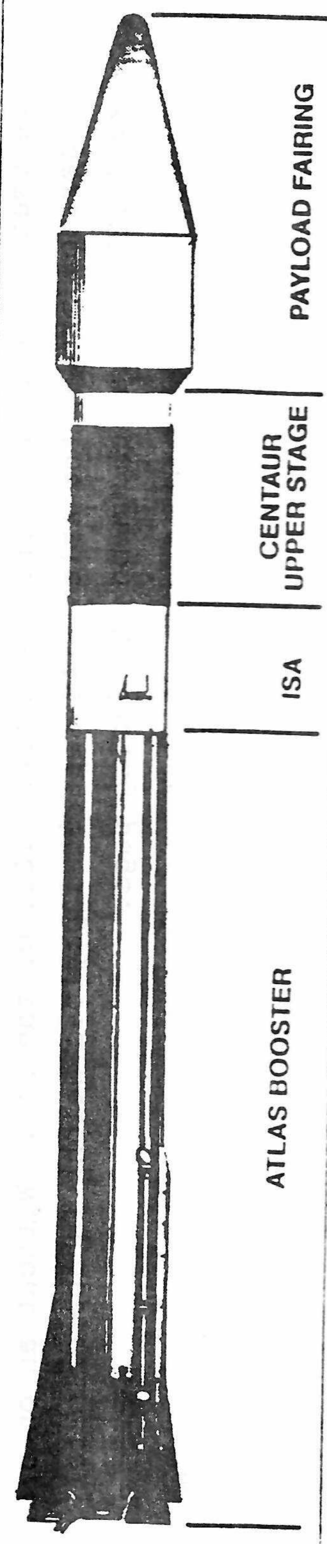
Surveyor: Configuration and Landing Profile





ATLAS IIAS

The Atlas IIAS and its major elements are described on the facing page. Of particular interest to the LTT are the dimensions of the two payload fairings.



PAYLOAD FAIRING

FEATURES:
- SIZE: 13.75 FT DIA x 40.1 FT LONG (4.19M x 12.22M) OR 10.8 FT DIA x 34.0 FT LONG (3.30M x 10.36M)
- WEIGHT: 4,524 LB (2052 KG) (LARGE) 3,084 LB (1399 KG) (MEDIUM)

SUBSYSTEMS:
- FAIRING: ALUMINUM SKIN/STRINGER & FRAME CLAMSHELL
- SEPARATION: PYRO BOLTS & SPRING THRUSTERS

INTERSTAGE ADAPTER (ISA)

FEATURES:
- SIZE: 10.0 FT DIA x 13.0 FT LONG (3.05M x 3.96M)
- WEIGHT: 1,149 LB (521 KG)

SUBSYSTEMS:
- STRUCTURE: ALUMINUM SKIN/STRINGER & FRAME
- SEPARATION: PYROTECHNIC (FLEXIBLE LINEAR-SHAPED CHARGE)
- ROLL CONTROL: ROLL CONTROL MODULE INSTALLATION

ATLAS BOOSTER

FEATURES:
- SIZE: 10.0 FT DIA x 81.7 FT LONG (3.05M x 24.9M)
- PROPELLANT: 344.8 KLB LO_2 & RP-1 (156.4 METRIC TONS)
- GUIDANCE: FROM UPPER STAGE

SUBSYSTEMS:
- STRUCTURE: PRESSURE-STABILIZED STAINLESS STEEL TANKS
- SEPARATION: BOOSTER ENGINE PACKAGE
  - PNEUMATICALLY ACTUATED SEPARATION LATCHES (10)

  SUSTAINER SECTION
  - RETRO ROCKETS (8)
- PNEUMATICS: HELIUM (PRESSURE TANKS)
  - HYDRAULICS & LUBRICATION SYSTEM
- PROPULSION: ROCKETDYNE MA-5A (SEA-LEVEL CHARACTERISTICS)

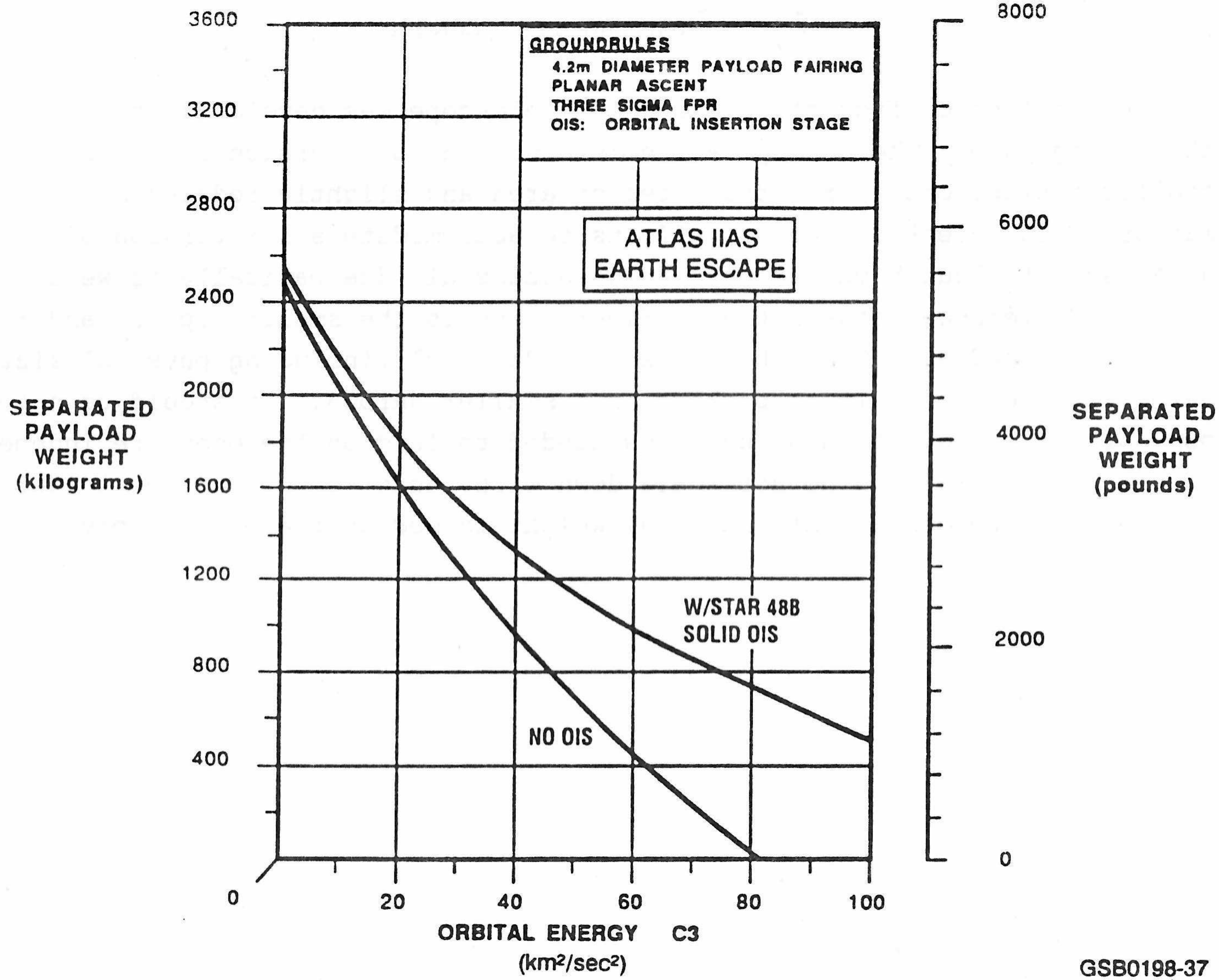
  BOOSTER ENGINE PACKAGE
  - THRUST: 414.0 KLB (LIFTOFF) (1841 KN)
  - Isp: 264.0 SEC

  SUSTAINER ENGINE
  - THRUST: 60.5 KLB (269 KN)
  - Isp: 220.0 SEC
- HYDRAULICS: FLUID – PROVIDES 2 SEPARATE SYSTEMS FOR GIMBALLING MAIN ENGINES
- AVIONICS: FLIGHT CONTROL, TELEMETRY, RANGE SAFETY COMMAND, ELECTRICAL POWER, COMPUTER-CONTROLLED ATLAS PRESSURIZATION SYSTEM, RATE GYRO UNIT

Atlas II subsystems

ATLAS IIAS EARTH ESCAPE PERFORMANCE

An Earth escape trajectory with an approximate C3 of 2 (km^2/sec^2) will be used for the translunar injection (TLI) of the LTT. Without an on orbital insertion stage, the Atlas IIAS separated payload weight is approximately 2,400 kg (5,292,lb) as shown on the facing page.



Atlas IIAS Earth escape performance.

1 M ATLAS IIAS CONCEPT

A baseline configuration for a 1 m telescope was developed and is shown in the facing page. The configuration is a scaled down version of the 2 m LTT with smaller optics, one-fourth the detector area and slightly reduced subsystems support requirements. However, efforts to accommodate a 1 m version of the LTT on an Atlas IIAS launch vehicle proved unsuccessful, due basically to weight. The primary difference between the configurations is the smaller optics and reduced propellant loading. The subsystems are identical (including physical size) which makes it more difficult to package in a smaller shroud. It should be noted that many of the components and functions needed to land on the moon are independent of the lander size and do not scale down with lander size. As the lander becomes smaller less percentage of the total weight landed on the moon is payload.

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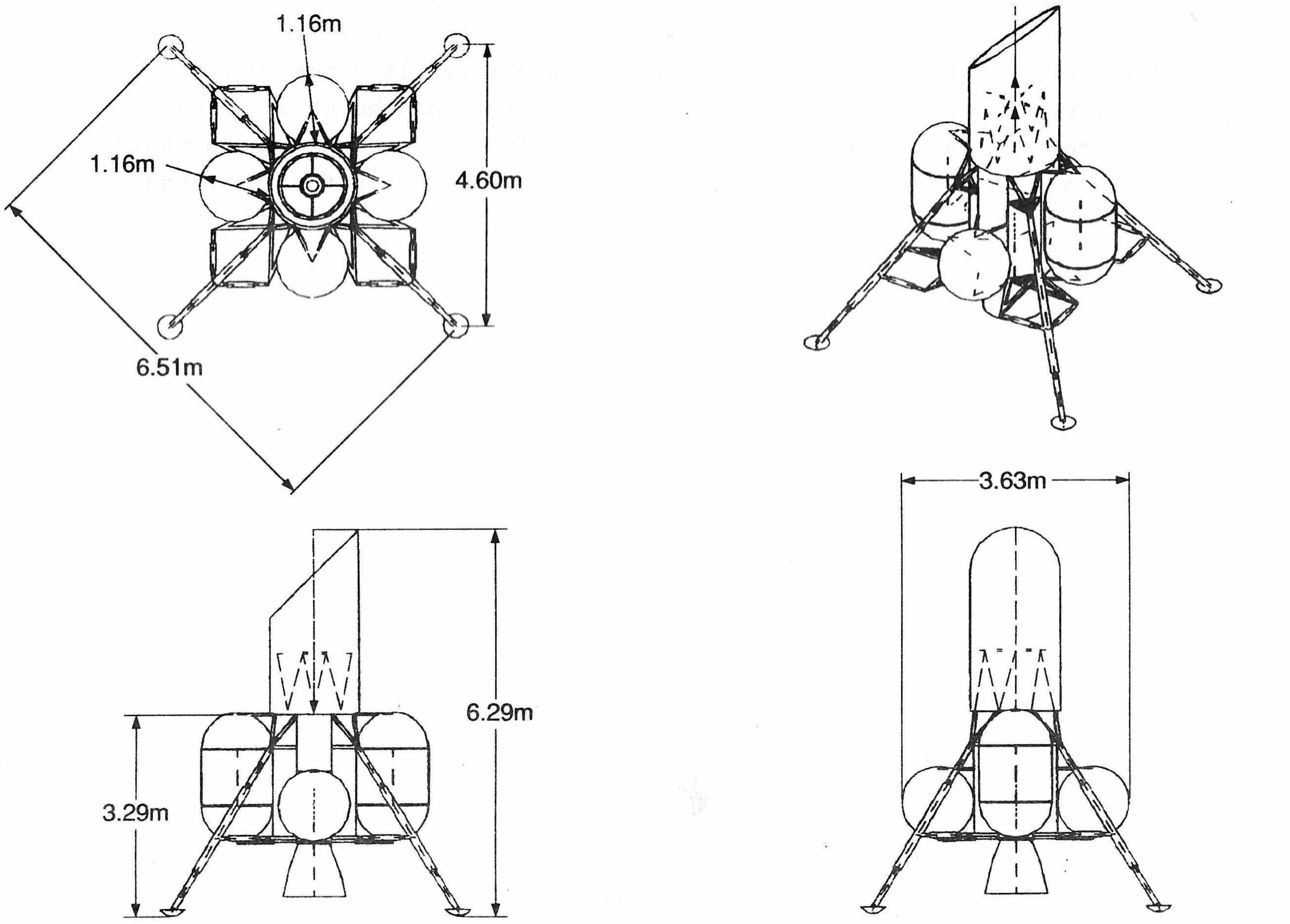
Lunar Transit Telescope
1m Atlas IIAS Concept



1 M LTT ATLAS IIAS CONCEPT
LAUNCH CONFIGURATION

The Atlas shroud is considerably smaller than the Titan both in length and diameter, but the smaller telescope and lander fit it rather well. The facing chart shows the 1 m LTT inside the launch shroud of the Atlas. There is adequate room for the lander legs to fold under the LTT between the launch vehicle interface structure and the shroud, and room at the top for additional length.

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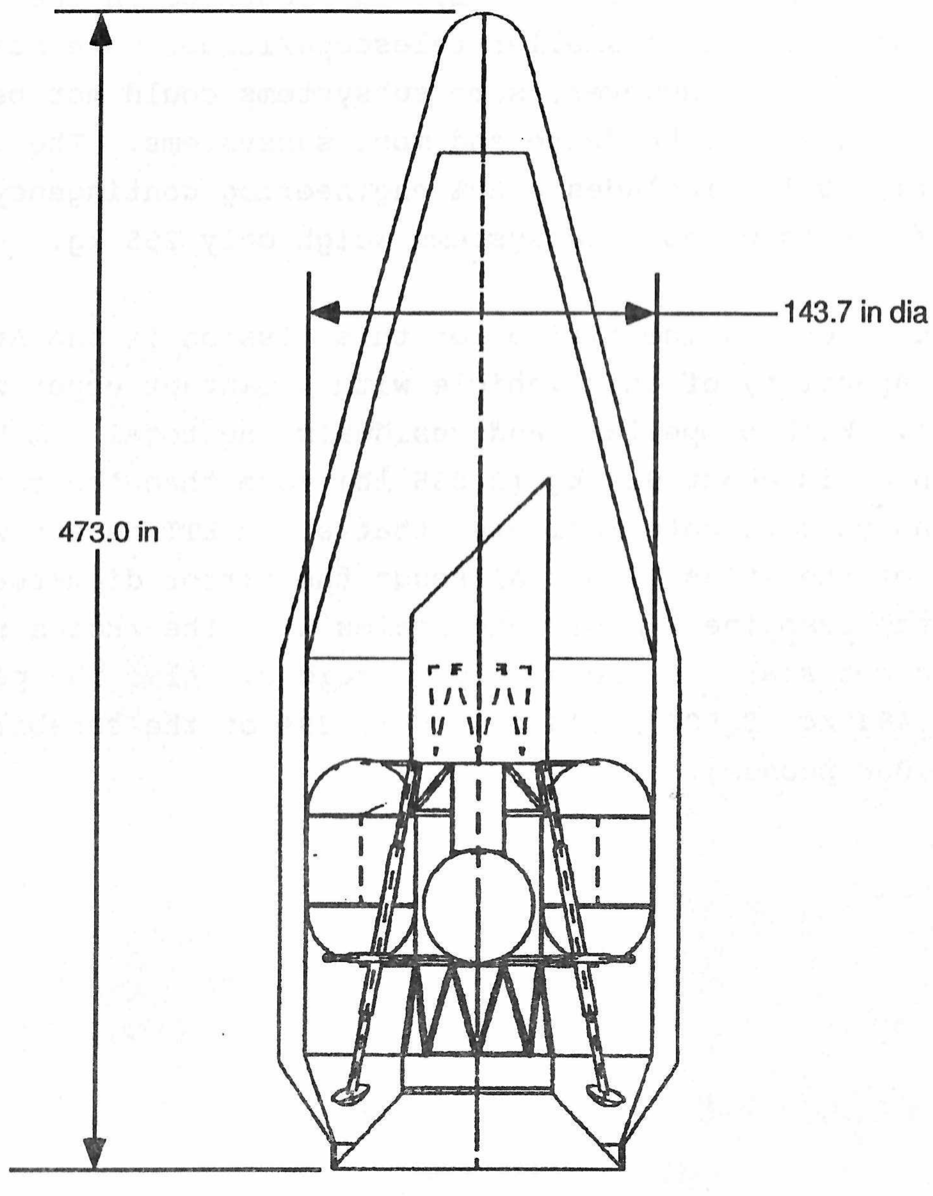
Lunar Transit Telescope
1m Atlas IIAS Concept



1M LTT MASS STATEMENT

The mass statement for the 1 m LTT is presented in the same format as the 2 m LTT. The masses for this smaller telescope/lander were scaled where possible from the 2 m baseline. However, some subsystems could not be scaled down, such as the propulsion system hardware and most subsystems. The total system dry mass of 2,027 kg (4,470 lb) includes a 20% engineering contingency. Note that the telescope components without subsystems weigh only 255 kg.

The launch vehicle identified for this mission is the Atlas IIAS. The performance capability of this vehicle with a Centaur upper stage to TLI is 2,494 kg (5,500 lb). With propellant and residuals the total 1 m LTT mass is 3,417 kg (7,535 lb) which is about 923 kg (2,035 lb) more than the capability of the Atlas IIAS. The analysis to date indicates that a 1 m LTT is not within the capabilities of the Atlas IIAS. Although the mirror diameter is only one half the size of the baseline 2 m LTT and scales with the radius squared, the subsystems do not scale in the same proportions. Also the performance of the Atlas IIAS 2,494 kg (5,500 pounds) is only 39% of the capability of the Titan IV 6,349 kg (14,000 pounds).

LUNAR TRANSIT TELESCOPE

TRANSIT TELESCOPE LAUNCHED ON ATLAS IIAS	(LBS)	(KG)
OPTICS (1m primary/4 Mirror Sys.)	97	44
SCIENCE INSTRUMENT	88	40
SENSORS/ELECTRONIC MONITORING	110	50
MIRROR SUPPORT STRUCTURES	97	44
LIGHT SHADE,SHELL,APERTURE COVER	82	37
METERING STRUCTURE,POINTING MECHANISMS	88	40
THERMAL CONTROL SYSTEM	308	140
ELECTRICAL POWER SYSTEM	437	198
INTEGRATION,ELECTRICAL	276	125
COMMUNICATIONS & DATA MANAGEMENT SYS.	201	91
GUIDANCE, NAVIGATION & CONTROL SYS.	257	117
PROPULSION SYSTEM	805	365
REACTION CONTROL SYSTEM	97	44
CORE/THRUST STRUCTURES	221	100
TANK SUPPORT,MM SHEILDING,& MISC.MECH.	110	50
LANDING GEAR (INCL.DEPLOYMENT MECH.)	232	105
CENTAUR INTERFACE STRUCTURE	221	100
CONTINGENCY (20%)	745	338
TOTAL DRY MASS	4470	2027
PROPELLANT (USABLE,RESIDUALS,BOILOFF,RCS)	3065	1390
TOTAL LTT SYSTEM MASS	7535	3417
ATLAS II PERFORMANCE LIMIT	5500	2494
PAYLOAD AVAILABLE	-2035	-923
		1/15/92

1 M LTT STRAWMAN SCIENCE INSTRUMENT

The instrument for the 1 m LTT has 1/4 the number of CCDs contained in the 2 m LTT. The 2 m case had a 35 x 35 array of CCDs which is too large for the 1 m case, the array size blocks some of the 1 m telescope's aperture. An 18 x 18 CCD (324 CCDs) array has been assumed for the 1 m LTT and the physical characteristics scaled downward from the 2 m instrument.

The physical characteristics for a CCD array sized for the 1 m LTT is 0.3 x 0.3 x 0.3 m with a 40 kg mass. The readout rate is 16 Mbps, and assuming a 10 to 1 compression ratio the average data rate returned to Earth is 1.6 Mbps. The average power needed during operations is 65 W of which 50 W is for cooling. The estimated weight does not contain a mechanical cooler or radiators. However, an instrument radiator is included in the thermal control mass estimate.

Instrumentation to monitor the lunar environment and telescope house keeping functions are the same as for the 2 m LTT. The estimated mass is 50 kg and the average power and data rate are 50 W and 8 kbps respectively.

The optical system weights are found in Section 2.2. An areal density of 45 kg/m^2 was used for the primary and 60 kg/m^2 for the other mirrors. The total glass weight is only 44 kg.

1 M LTT
STRAWMAN SCIENCE INSTRUMENT

ASSUMPTIONS:

1. MOSAIC ARRAY OF OPTICAL/ULTRAVIOLET CCD'S

2. 36,864 N-S X 36,864 E-W PIXELS (MAY BE SMALLER IN N-S)

3. 324 CCD'S, EACH WITH 2,048 X 2,048 PIXELS (1.4 BILLION PIXELS)

4. 5 N-S READOUTS OF 88 BIT/S/PIXEL GIVING 16 MBPS

CHARACTERISTICS:

1. SIZE, 0.3 S-N X 0.3 E-W X 0.3 METERS

2. MASS ESTIMATE 40 KG
   - CCD'S AND PACKAGING 5 KG
   - ELECTRONICS 10 KG
   - COOLING EST. 25 KG

3. PASSIVE COOLING OF DETECTOR IF POSSIBLE
   - 170 K FOR UV CCD'S
   - 65 K FOR IR CCD'S
   - LESS THAN 10 K VARIATION DURING OPERATION
   - CAN BE WARM DURING LAUNCH AND COAST PHASES, AVOID LARGE VARIATIONS

SUPPORT REQUIREMENTS:

1. AVERAGE DATA RATE OF 16 MBPS BEFORE COMPRESSION

2. AVERAGE POWER: 65 W: 15 W ELECTRONICS + 50 W COOLING

3. FOV SWATH SHOULD BE PERPENDICULAR TO THE GALACTIC PLANE.

4. DETECTOR ARRAY MUST BE ALIGNED WITH E-W LUNAR ROTATION (3 ARCSEC)

STRUCTURES MASS SUMMARY FOR 1 M LTT

As a part of the LTT study, a mass estimate of the structural components for a one meter diameter LTT to be launched on an Atlas IIAS was compiled. The structural masses were estimated by scaling from the 2 meter LTT values. The mirror support structure was scaled based on the square of the diameter of the primary mirror. Surface area was used for the sunshade and shell estimates, and the metering structure value was based on engineering judgement.

Lunar Transit Telescope
Structures Mass Summary
1 Meter Diameter Telescope

Part	Material	Mass (kg)
Metering Structure	Invar 36 and Aluminum	25.0
Primary Mirror Support	Invar 36 and Aluminum	44.0
Telescope/Lander Interface	Weldalite™ 049	N/A
Sunshade	Aluminum 2219	13.48
Shell	Aluminum 2219	15.57
Tanks	Weldalite™ 049	56.2
Core	Aluminum 2219	100.0
LTT/Centaur Interface	Aluminum 2219	100.0
Landing Gear	Beryllium-Aluminum	105.0

- Structural details and secondary structure design will affect these mass estimates

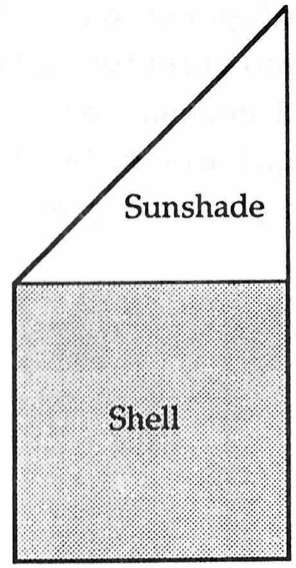
THERMAL ANALYSIS & DESIGN

MASS ESTIMATE FOR 1 M DIAMETER LTT

The preliminary thermal control system mass estimate includes foam and MLI for two LH_2 tanks, MLI for two LO_2 tanks, MLI for small pressurant and propellant tanks, radiators, subsystem insulation, shell and sunshade MLI covering, reflective shields near the RTGs, base heat shield, and thermal surface treatments (paint, foil, etc.). These masses were scaled from those calculated for the 2 m diameter LTT using the appropriate factor. In most cases, mass of the thermal control components is proportional to surface area and a ratio of the 1 m to the 2 m area was used as a scaling factor. As is shown in the table, an attempt has been made to separate the lander components from the telescope components. In some cases, such as tank insulation, the correct classification was obvious. The proper division for subsystem insulation and avionics radiators was more subjective and a portion of the mass has been allotted to each system. The individual estimates are shown in the table, with the TCS total of 139.7 kg. It should be noted that, for this estimate, the sunshade is the angled portion of the LTT 'tube' and the shell is the cylindrical portion.

Thermal Control System
Weight Estimate for 1 m diameter LTT

Lander Components	Weight Estimate (kg.)
Hydrogen tank MLI	14.8
Hydrogen tank foam	9.6
Oxygen tank MLI	8.2
Pressurant tank insulation	0.6
Propellant tank insulation	0.6
Insulation for subsystems	4.5
Avionics radiator(s)	15.1
Base heat shield	41.2
Surface treatments	6.8
Misc. TCS	9.5
Lander Total =	110.9
Telescope Components	
Instrument radiator	2.3
Sunshade fabric	2.1
Shell insulation	3.7
RTG shields	6.0
Insulation for subsystems	2.3
Avionics radiator(s)	7.6
Misc. TCS	4.8
Telescope Total =	28.8
LTT Total	**139.7**



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GN&C FOR 1M LUNAR TELESCOPE

The guidance, navigation, and control equipment for the 1 m diameter telescope must perform exactly the same functions as those required for the 2 m configuration, with the same accuracy and redundancy requirements. Initial stabilization after tipoff, attitude control during coasts, burns, and maneuvers, and descent burn and touchdown must still be controlled to meet the same mission requirements as those of a larger telescope. Hence the GN&C equipment list, with weights and power requirements, would not vary as the telescope diameter changes.

GN&C for Lunar Lander for Lunar Telescope

Requirement

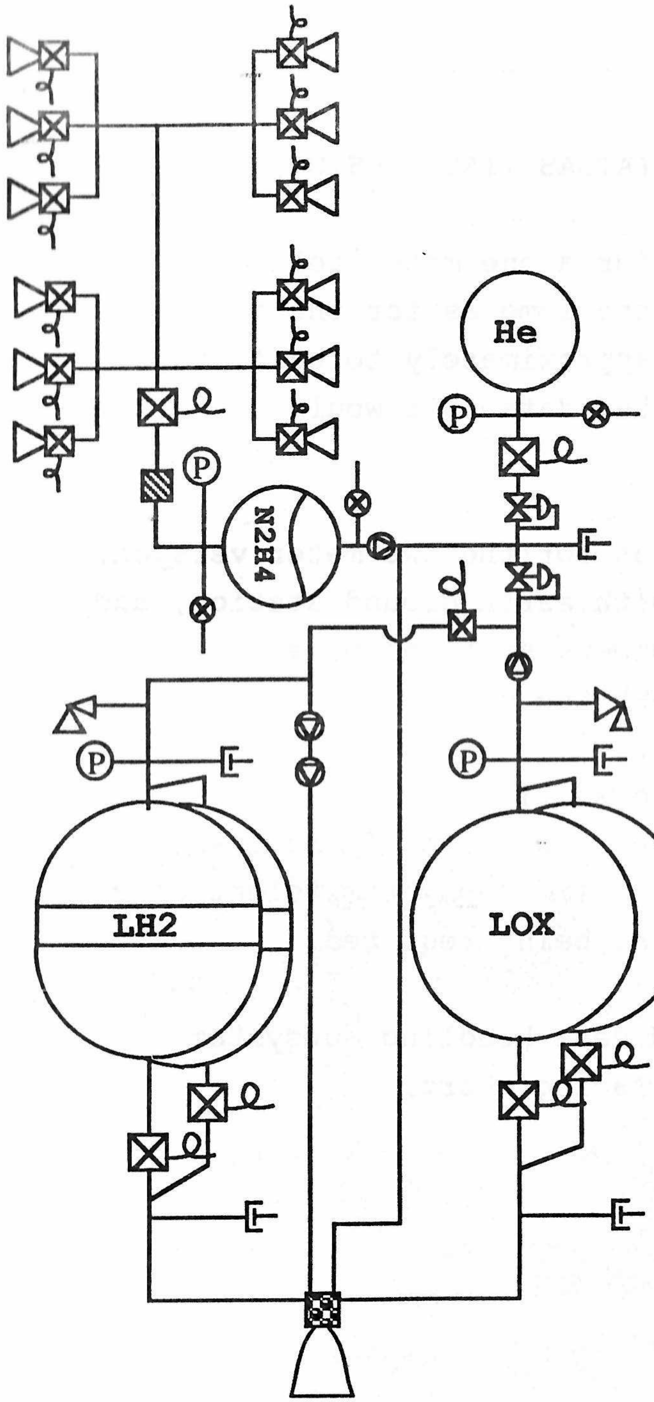
- o Provide GN&C from TLI to lunar orbit
- o Provide GN&C for deorbit/braking & landing on unprepared surface
- o Assumes State Vector Updates from ground

Recommend	Wt. each (Kg)	Total wt. (Kg)	Power reqmt (W) (each)
2 - IMU	3.8	7.6	27/40
2 - Star tracker and sun shield	4.3	8.6	10
2 - Sun sensor and electronics	1.4	2.8	3
1 - Landing radar	38.6	38.6	123
1 - Control electronics	40.0	40.0	70
2 - Video camera	7.0	14.0	10
Miscellaneous	5.0	5.0	
		Total wt. 116.6 Kg	

Requires: Throttleable main engine, RCS for maneuvers, coast attitude control & roll control during main engine burns, midcourse and landing

ATLAS IIAS PROPULSION

The basic design of the propulsion system for the lander will not change as the launch vehicle changes from the Titan IV/Centaur to the Atlas IIAS/Centaur. The same compliment of hardware is required for the smaller lift capacity vehicle; however, the amount of propellant required is less as the telescope and total payload is downsized. This translates into smaller sized main propellant tanks, but does not appreciably change the total dry weight. Since there is not another developed/operational cryogenic engine available, the RL10 has been retained. There was also no attempt to refine the RCS propellant loading from the previous estimate. Operation of the system is identical to the large lander.



ATLAS II-AS
LTT LANDER

MASS STATEMENT (KG)

EQUIPMENT	MAIN PROP LIQUID	MAIN PROP GAS	RCS	TOTAL
TANK	56.2	25.0	6.4	
FILL & DRAIN VALVES	18.2	0.9	0.9	
ISOLATION VALVE	9.1	2.8	1.4	
PRESSURE TRANSDUCERS	0.9	0.5	0.5	
PRESSURE REGULATOR		2.7	2.7	
RELIEF DEVICE		6.4		
CHECK VALVES		4.6		
PROPELLANT FILTER			0.5	
ENGINES/THRUSTERS	165.9		27.3	
TVC	26.4			
MISCELLANEOUS	41.5	4.3	4.0	
TOTAL DRY WEIGHT	318	47	44	409
PROPELLANT: USABLE	1250		37	1287
RESIDUAL/BOILOFF	98		2	100
PRESSURANT		3		3
SUBSYSTEM WEIGHT				1799

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12/04/91

COMMUNICATIONS AND DATA HANDLING SUBSYSTEM (ATLAS IIAS VERSION)

The communications and data handling subsystem for a one meter telescope to be flown on the Atlas IIAS would remain essentially the same as for the two meter telescope. Although the data rate would be reduced approximately to half, data compression would still be required and the transmitted data rate would be in the same design range as before.

All the same design considerations would apply as for the two meter version; lunar near-side location, continuous communication with earth ground station, and complete automation of operation. The Deep Space Network stations or a comparable dedicated network would receive and process the data.

The weight and power requirements would remain essentially the same. The power requirement of 180 watts might be reduced slightly because of the lower data rate, but the decrease would not be significant. The subsystem weight of 91 kg (200 lb) would not change due to the same equipment being required.

A block diagram of a typical communications and data handling subsystem for a one meter Lunar Transit Telescope is shown on the facing chart.

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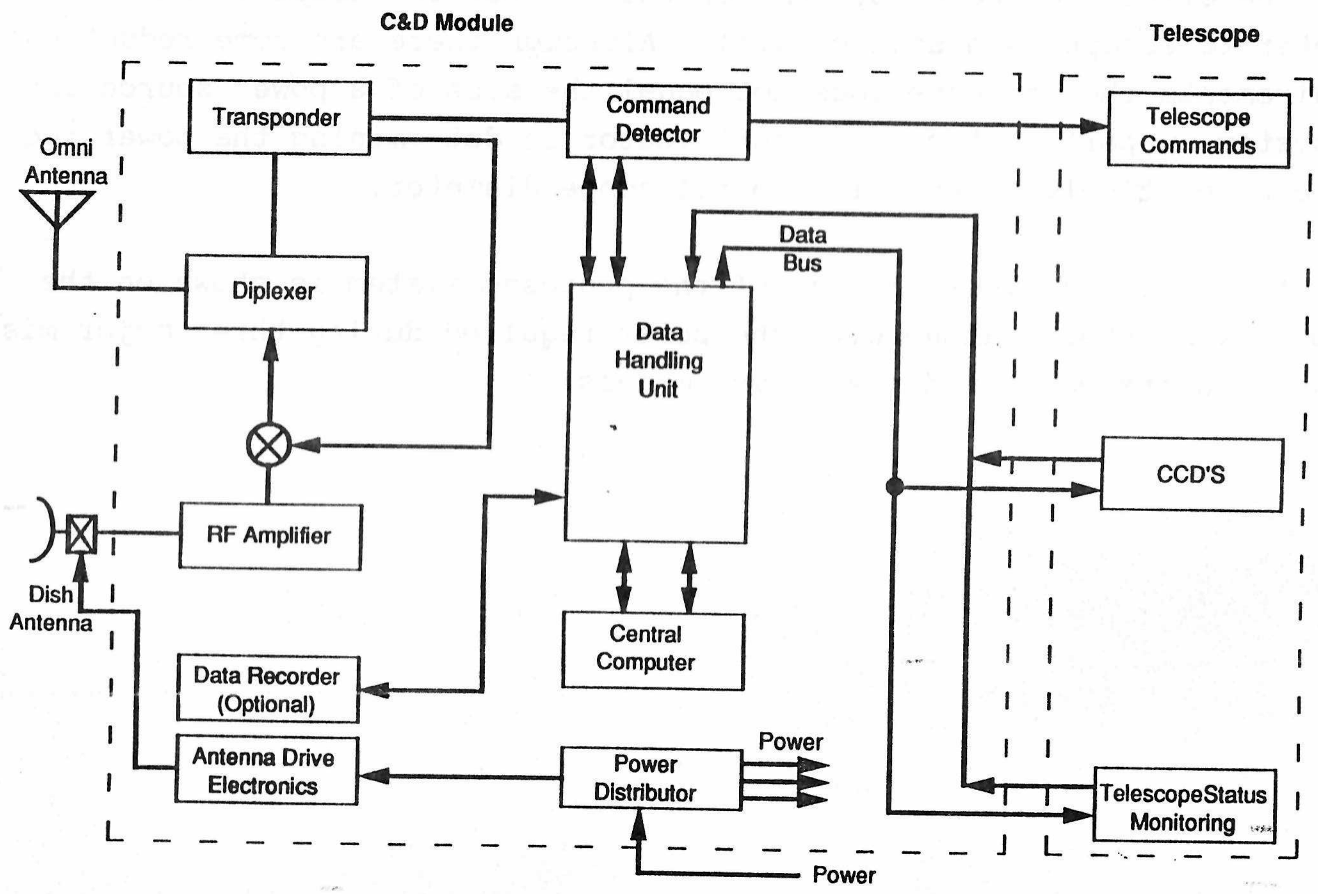
Lunar Transit Telescope
C&DH Simplified Block Diagram



ONE METER TELESCOPE POWER SYSTEM

The electrical power system for the 1 meter telescope is the same as for the 2 meter telescope (see section 4.4). Although there are some reductions in power requirements the decrease does not equal the size of a power source increment. Subsystem support is the major contributor in determining the power requirements and are relatively insensitive to telescope diameter.

A simplified block diagram of the proposed system is shown on the accompanying chart, along with the power required during three major mission phases and the estimated power system mass.

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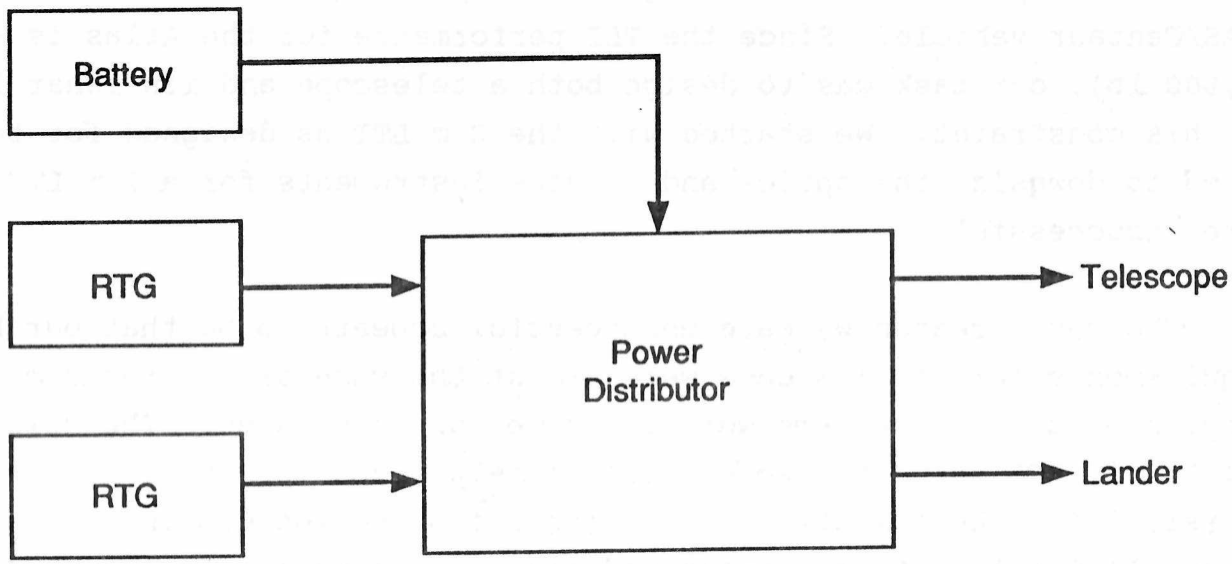
One Meter Telescope Power System



Power Budget

Transit	650 W
Landing	800 W
Surface	500 W

System Mass

RTG's	112 kg
Batteries	54 kg
Pwr. Distr.	32 kg
	198 kg

ATLAS IIAS TELESCOPE SUMMARY

An attempt was made to design a 1 m aperture LTT for launch with an Atlas IIAS/Centaur vehicle. Since the TLI performance for the Atlas is about 2,494 kg (5,500 lb), our task was to design both a telescope and its lunar lander subject to this constraint. We started with the 2 m LTT as designed for the Titan IV and tried to downsize the optics and science instruments for a 1 m LTT. However, we were unsuccessful.

The basic reason we were unsuccessful appears to be that our basic support 0equirements for the 1 m case were almost the same as for the 2 m case, and the propulsion and RCS systems were the same for both cases. The RL10 engine and 111 N (25 lbf) RCS thrusters were appropriately sized for the 2 m Titan case, but are oversized for the 1 m Atlas case. The use of common components designed for 6,349 (14,000 lb) TLI are oversized for a 2,494 kg (5,500 lb) TLI vehicle. Other viable approaches were not studied. Other options and different engines should be assessed for a lunar lander designed specifically for the Atlas. Also, the basic support requirements should be scrubbed and reduced, especially electrical power requirements.

ATLAS IIAS TELESCOPE SUMMARY

- AN ATTEMPT WAS MADE TO DESIGN A 1 M APERTURE TELESCOPE WITH ITS LUNAR LANDER FOR LAUNCH WITH AN ATLAS IIAS/CENTAUR.

- WE WERE NOT SUCCESSFUL. THE ENSUING DESIGN WAS TOO HEAVY.

- THE ATLAS TLI MASS IS 2,494 KG BUT THE TOTAL LTT SYSTEM MASS WAS 3,376 KG.

- THE TELESCOPE MASS (234 KG) SCALED DOWNWARD FROM THE 2 M CASE, BUT THE SUBSYSTEMS (1,066 KG) DID NOT.

- OBSERVATIONS
  - THE PROPELLANT AND RCS WAS OVERSIZED FOR THE ATLAS
  - THE SUBSYSTEMS SUPPORT REQUIREMENTS WERE NOT REDUCED
  - A CERTAIN AMMOUNT OF SUBSYSTEMS ARE REQUIRED REGARDLESS LTT SIZE
  - STUDYING OTHER VIABLE OPTIONS MIGHT LEAD TO SUCCESS

6.2 LANDING GEAR OPTIONS

GENERAL CLASSES OF LANDING GEAR

NASA has a successful history of building manned and unmanned surface landers: Apollo Lunar Excursion Module (LEM), Surveyor, and Viking. The current design may utilize some of the same concepts, design techniques and ideas to build on the success of the previous work and knowledge base. The Russian space agency has designed unmanned lunar landers which have also been researched for ideas applicable to the LTT effort. Most were designed in the early to mid-sixties making much of the technology out dated. The early programs, both NASA and Soviet, have a common bond in that the landers were built for a series of missions with a common delivery method (lander) but varying scientific payloads and objectives with each new mission. Similarly the LTT may be of some use for other scientific instruments that are desired on the lunar surface early in the next decade. Early NASA landers served as a starting point for ideas from which the LTT lander design could be derived. The driving requirement was that the LTT lander be of minimum mass and size to land an optical telescope upon the lunar surface. The Surveyor and Viking landers are characterized by a relatively small ratio of payload mass to lander and propellant mass. Both used solid propellants which are major contributors to the large lander percentages.

The landing gear on the LTT was limited both in mass and available packaging space during launch. The landing gear will be utilized only once, since the LTT is an unmanned vehicle. While designing the lander, it was discovered that if the propellant was loaded as low as possible the landing cg was also lowered. This lower cg increases the static stability of the vehicle. Thus the landing gear in most concepts was integrated around the fuel tanks and mounted on the thrust structure.

General Classes of Landing Gear

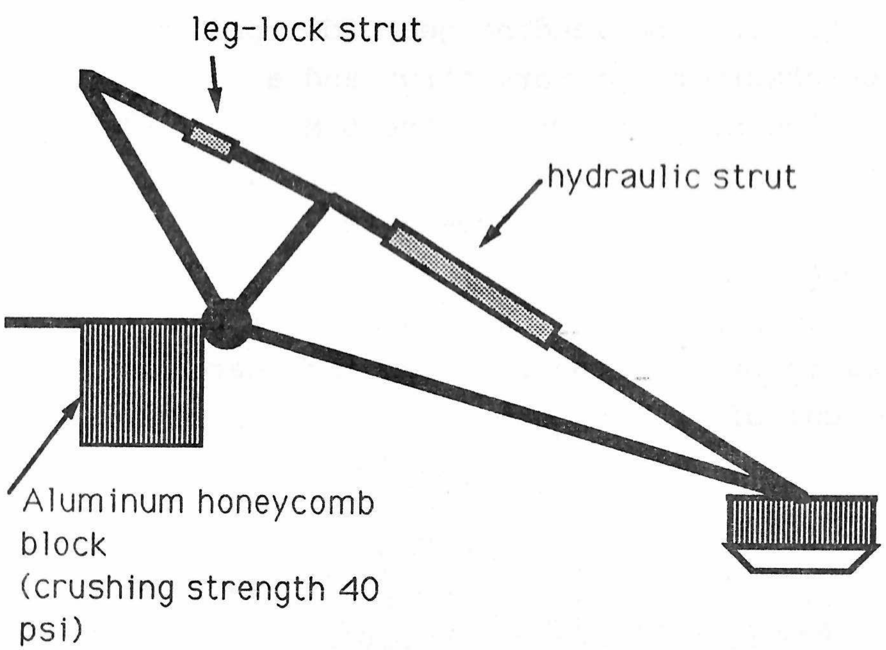
1. Surveyor Style
2. Apollo Style
3. Modified Apollo
4. Fixed Gear
5. Crushable Landing Gear

I. SURVEYOR STYLE LANDING GEAR

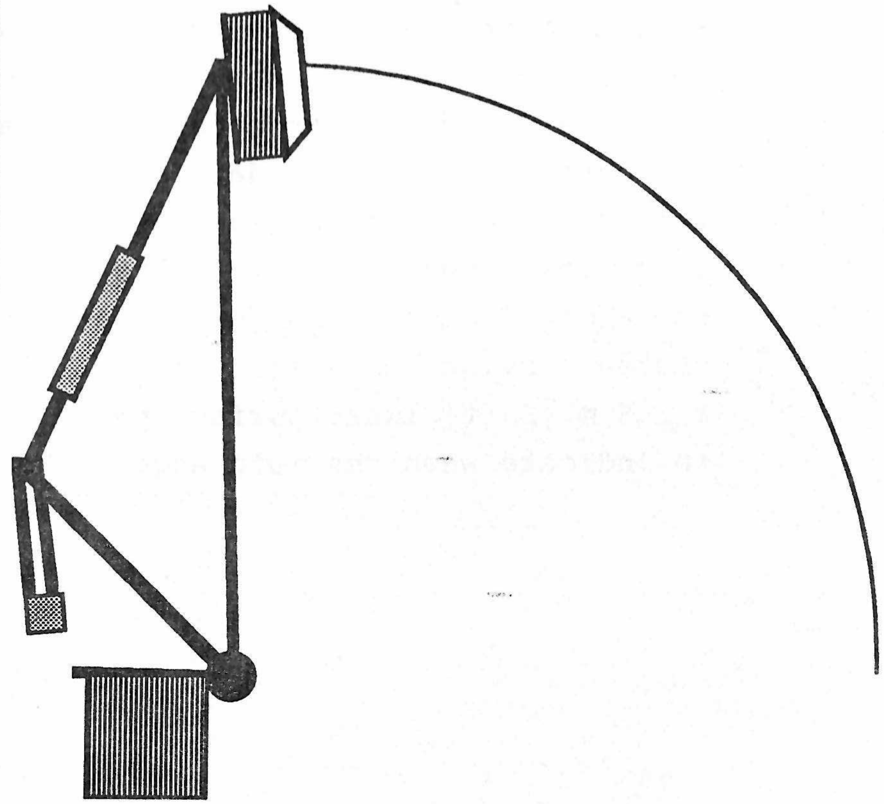
The Surveyor unmanned lunar probe used a three legged landing gear arrangement. The landing gear was stowed upright during launch on an Atlas. Just before touch-down several explosive bolts holding the landing gear in place were released, allowing the landing legs to deploy under the force of a spring loaded lock strut. The landing leg/gear absorbed the landing shock by using crushable aluminum honeycomb on the landing pad and under the landers frame. The primary landing strut also incorporated a hydraulic strut to counteract horizontal forces on the lander. Of the six Surveyors that landed on the lunar surface, none had any critical failures of the landing gear.

I. Surveyor Style

A. Original Surveyor Configuration–
Deployed Mode



B. Original Surveyor Configuration–
Stowed Mode

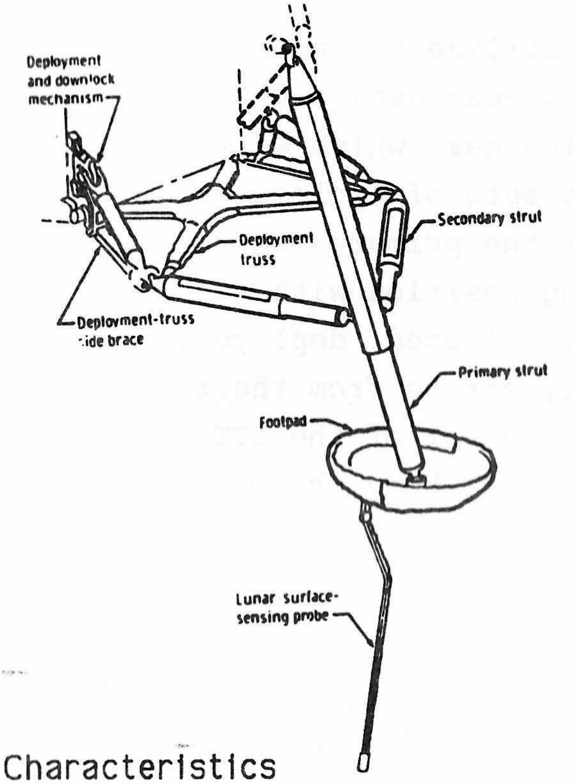


II. APOLLO LM STYLE LANDING GEAR

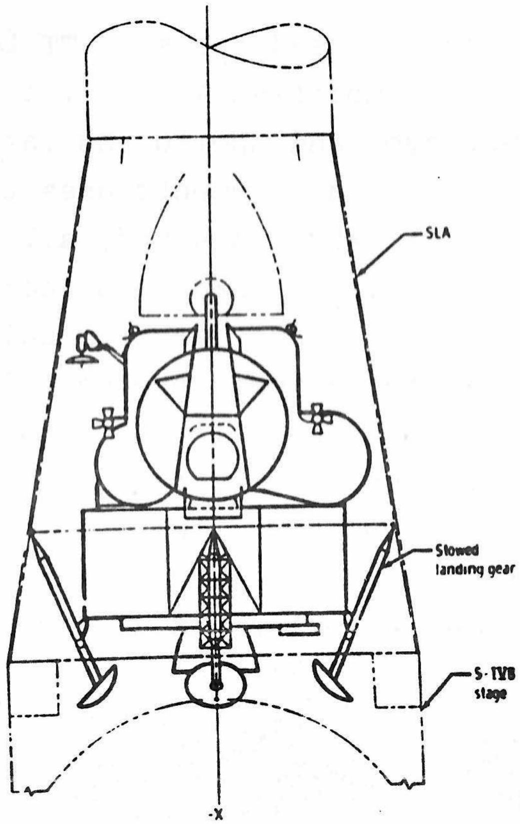
The Apollo Lunar Excursion Module (LEM) utilized a four legged landing gear arrangement. The landing gear was stowed under the LEM during launch and trans-lunar flight. Each leg is an inverted tripod type landing gear, which consisted of a primary strut and two secondary struts joined near the foot-pad. The LEM was required to retract for stowage due to the large landing gear tread. The tread radius is defined as the distance from the vehicle longitudinal axis to the circle containing the landing gear foot pad. The final landing gear consisted of a shock absorbing primary strut and a deployable set of secondary struts. Unlike the Surveyor craft, the LEM utilized four landing gear instead of three for further stability. Before touch-down, two explosive bolts were used to activate a deployment and down-lock device. The deployment device unfolded the secondary struts, thereby moving the primary strut into a locked position. The landing gear had a 1.5 m (5 ft) lunar surface probe attached to each of the four landing gear to indicate when the main engine should be cut off.

II. Apollo Style Landing Gear

A. LM Landing Gear



B. LM in Payload Envelope



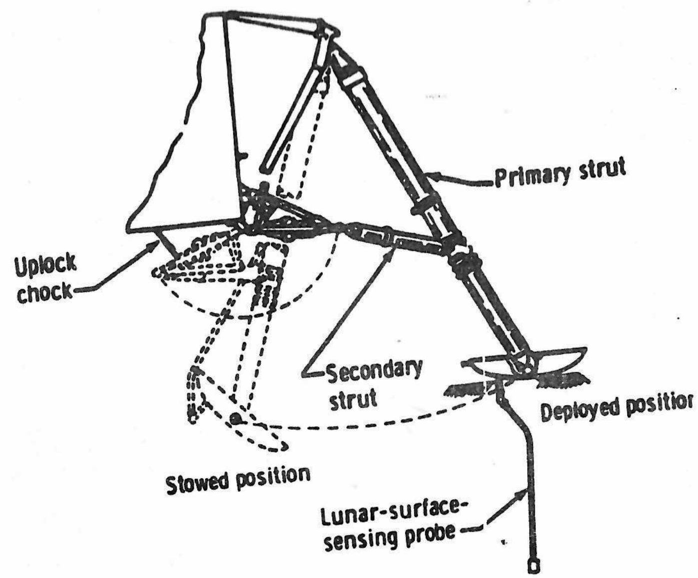
Characteristics

Utilizes both horizontal area
and vertical area under the lander

Locked in place by titanium strap while in flight/
explosive uplock and spring action into deployed position

Design Limits of: 10 fps vertical and 4 fps horizontal velocity

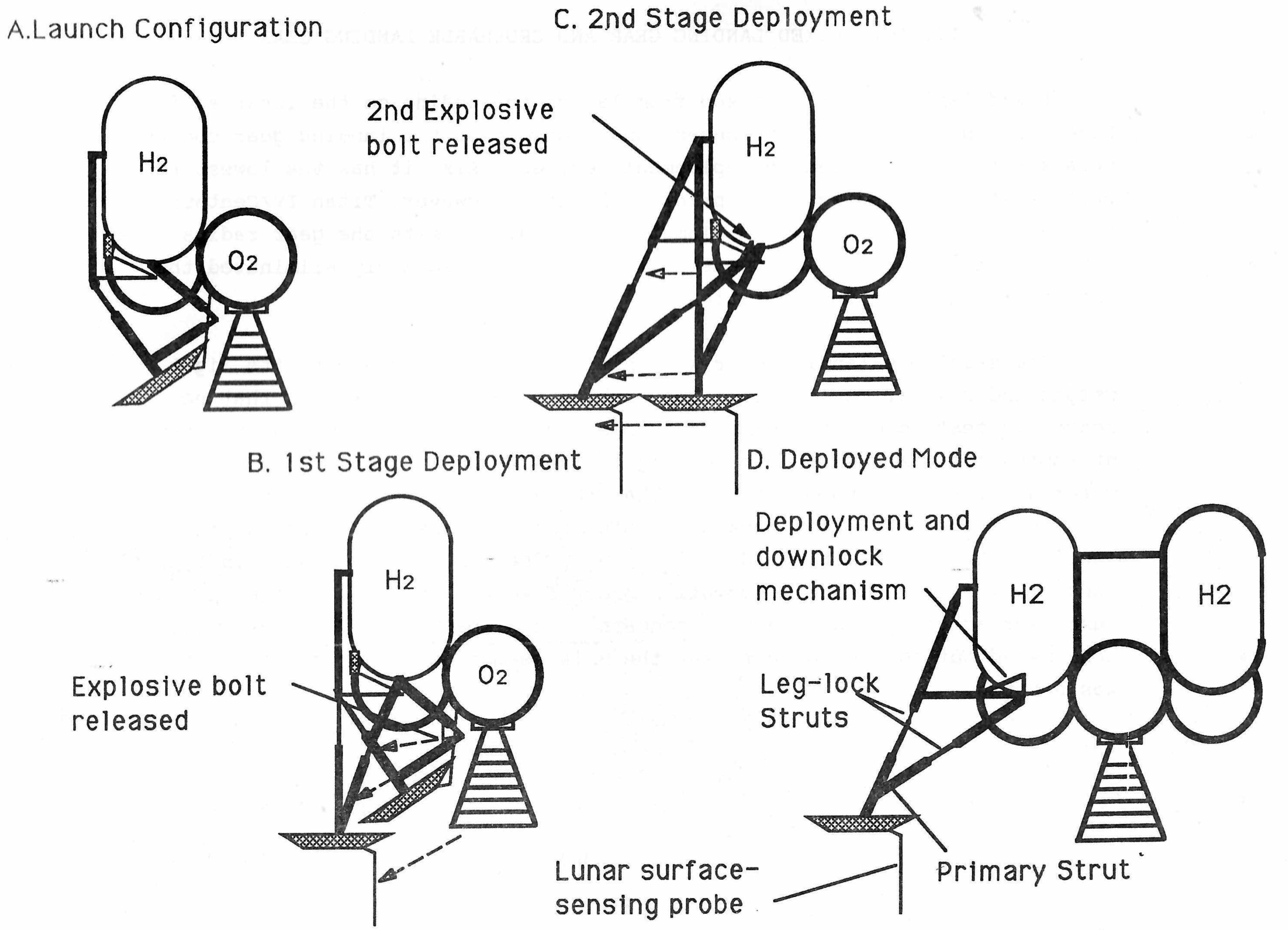
C. Stowed and Deployed Positions of the Landing Gear



III. MODIFIED APOLLO LM STYLE LANDING GEAR

This design concept arose out of need for the LTT to utilize the space under the bottom of the craft. The modified Apollo landing gear uses a two stage deployment, unlike the Surveyor and Apollo LEM landing gear which were one stage deployments. The two stage deployments uses two sets of explosive bolts on each gear. The first bolt, when released, allows the primary landing strut to swing from under the propellant tanks to a locking position with the upper primary strut. The second bolt, when electronically released, deploys a down-lock device (spring loaded) that extends the secondary struts from their stowage and locked positions under the propellant tanks. Several of the LTT concepts had the 1.78 m (70") RL10 engine mounted below the LTT body which produced a high center of gravity (cg). To lower the cg, propellant tankage should be placed as low as possible. The landing gear was designed to fit under the propellant tankage, but above the end of the RL10 nozzle.

III. Modified Apollo Style



III/IV. FIXED LANDING GEAR AND CRUSHABLE LANDING GEAR

Fixed landing gear is fixed from launch to landing on the lunar surface. Fixed landing gear is the strongest and lightest of the landing gear concepts because it does not require deployment devices. Also it has the lowest risk because its geometry is fixed prior to launch. However, Titan IV/Centaur payload fairing is limited in diameter which also limits the gear radius to an unacceptable value for landing stability. This immediately eliminated the option from further consideration.

The crushable landing gear was initially considered due to its light weight and ease of deployment. The problem with this concept is that no concerted testing on utilizing crushable materials in landing gear has been attempted by NASA. Crushable landing gear would be either a honeycomb material or an inflatable airbag. The crushable material could be mounted just below the propellant tanks and main frame to take the impact of the landing. No deployment would be required. The only drawback on this type of landing gear is that the deformation might make the lander unstable upon the lunar surface; tipping is also a concern. An airbag type system was also considered but the deployment near the RL10 seemed dangerous and the concept was dropped.

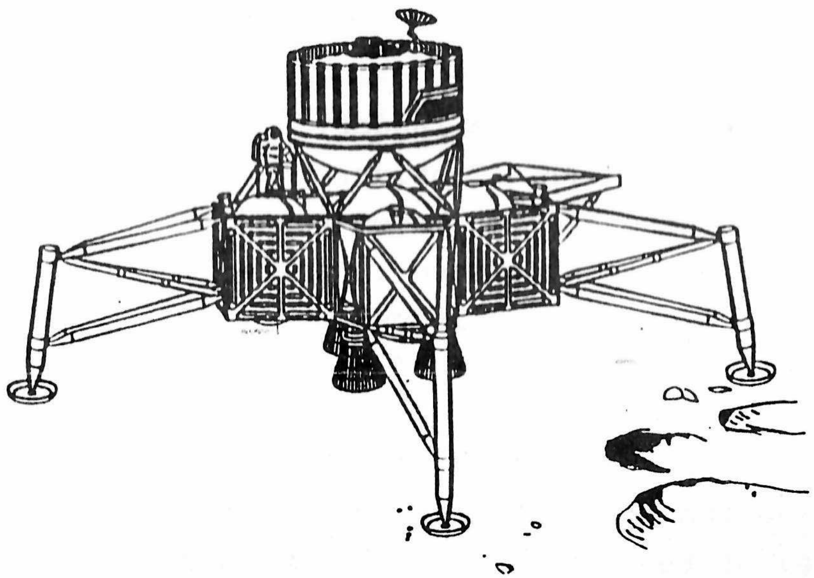
IV./V. Fixed Landing Gear/Crushable Landing Gear

Considerations

Requires large payload widths

Guaranteed deployment

Very predictable landing characteristics

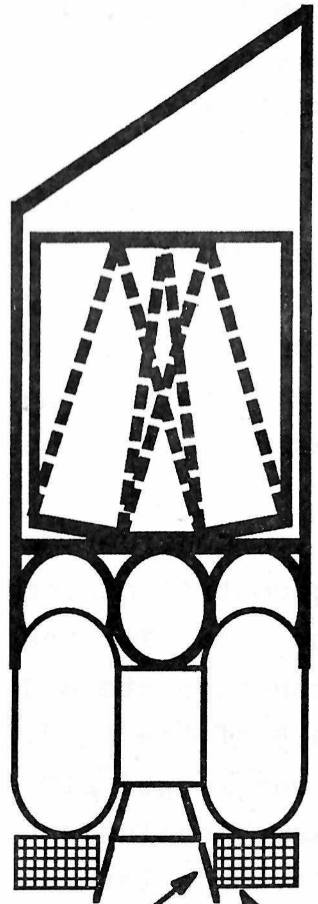


Considerations

Never been used as a landing pad

Tipping of lander may be of concern

Thermal distortion due to proximity of engine placement may be a problem



STATIC STABILITY ANGLE FOR 2 M LTT
THREE VS FOUR LEGS

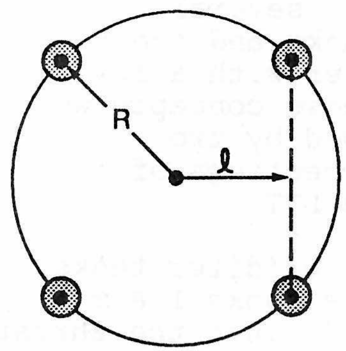
On an uneven, hard surface a three legged stool or table always sits stable with each leg bearing weight. However, under the same conditions, a four legged stool or table always sits unstable and the mass is unequally distributed between its legs: it is possible for one leg to be compleyely off the surface. Assuming the LTT legs will be flexible and spring loaded, then all of its legs, even the four legged case will bear weight on an uneven surface.

A brief assessment was made of three vs four legs for the lander, subsequently a four legged lander was selected for the LTT. Consider the static stability of the LTT sitting on a slope. There are two cases to consider: 1) one leg downhill and 2) two legs downhill. In case 1, there is no difference between the static stability of a three vs four legged LTT, assuming that both have the same landing gear radius and center of mass.

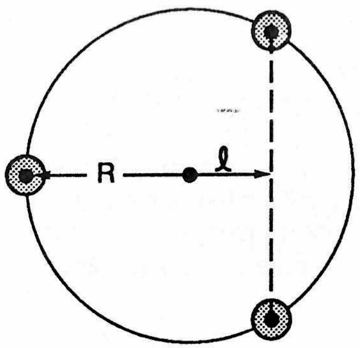
Consider case 2 with two legs downhill. The distance from the vehicle geometric center-line and the chord between the two downhill legs is defined by "l". For the four legged case "l" is 0.707 of the landing gear radius and can tolerate a 34 degree slope. In contrast, for the three legged case "l" is 0.5 of the landing gear radius and can only tolerate a 25 degree slope. The four legged LTT can land on a 36% greater slope than a three legged LTT. This same conclusion is expected to hold true for the dynamic analysis with the LTT having a lateral velocity just prior to touchdown.

Static Stability Angle for 2m LTT

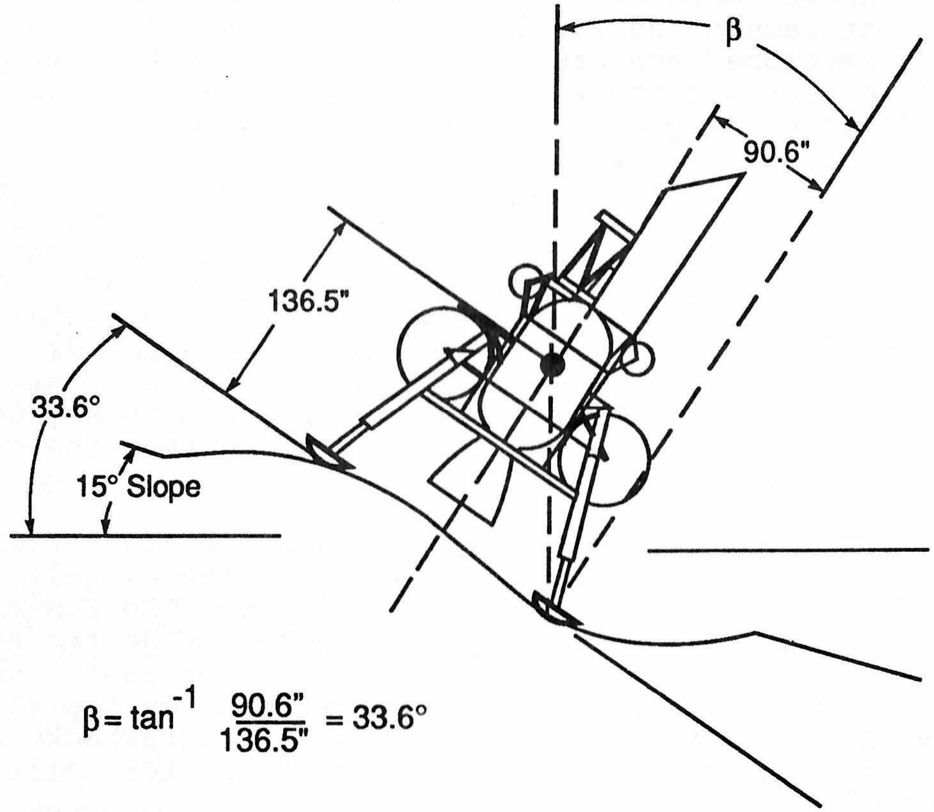
Three vs Four Legs



Four Legs
ℓ min = .707 radius
β = 34° slope



Three Legs
ℓ min = .500 radius
β = 25° slope



$$\beta = \tan^{-1} \frac{90.6''}{136.5''} = 33.6^\circ$$

6.3 TANKS AND LANDER DESIGN

LANDER CONCEPTS

After looking into preliminary designs of landing gears, several configuration concepts involving the number of propellant tanks and their number were developed. The objective was to have a LTT/lander with a low cg at launch and for landing on the lunar surface. Three of these concepts were developed and are illustrated on the following pages, followed by two comparison charts. The concepts evolved during the initial meetings of the LTT working group, helping toward selection of the reference LTT.

The first concept has two fuel and two long cylindrical oxidizer tanks (2 LO_2/2 LH_2) with the RL10 engine nozzle sticking out below the tanks 1.8 m (70"). Later in the study the engine was embedded 0.9 m (35") into the thrust frame to lower the launch and landing cg's. The RL10 is connected to an octagonal thrust frame with an outer diameter of 1.8 m (70"). Inside the thrust frame, the space above the engine is allocated for avionics equipment. The telescope sits on top of the thrust frame and above the long hydrogen tanks. The entire design with a 2 m telescope is 7.4 m (288") in height. It had a higher launch and landing cg than the other configurations, which is its major drawback.

The second telescope/lander configuration is characterized by eight propellant tanks (4 LO_2/4 LH_2). The overall configuration is shortened by using more tanks and embedding the RL10 1.5 m (60") into the thrust structure. Embedding the engine inside the thrust structure cuts down the amount of dead space under the propellant tanks to the bottom of the engine nozzle. With the telescope sitting on top of the tanks the configuration is 5.1 m (196") in height. Its major drawbacks are more complex fuel lines and increased surface area on the tanks for boil-off.

The third concept also uses eight propellant tank (4 LO_2/4 LH_2). The four LO_2 tanks are clustered around the RL10 and the telescope sits on top of the RL10. The four long, small diameter LH_2 tanks surround the telescope. This configuration has the lowest cg of all configurations examined, and the tanks may provide some protection for the telescope. Its major drawbacks are that the embedded telescope is limited to a 2 m or smaller primary and the complexity of the propellant feed lines.

LANDER CONCEPTS

- CONCEPT #1: FOUR TANK OPTIONS
- CONCEPT #2: EIGHT TANKS, TOP MOUNTED TELESCOPE
- CONCEPT #3: EIGHT TANKS, EMBEDDED TELESCOPE
- STATIC STABILITY OF LANDING CONFIGURATION
- CONFIGURATION COMPARISON

Lunar Transit Telescope Concept Drawing #1

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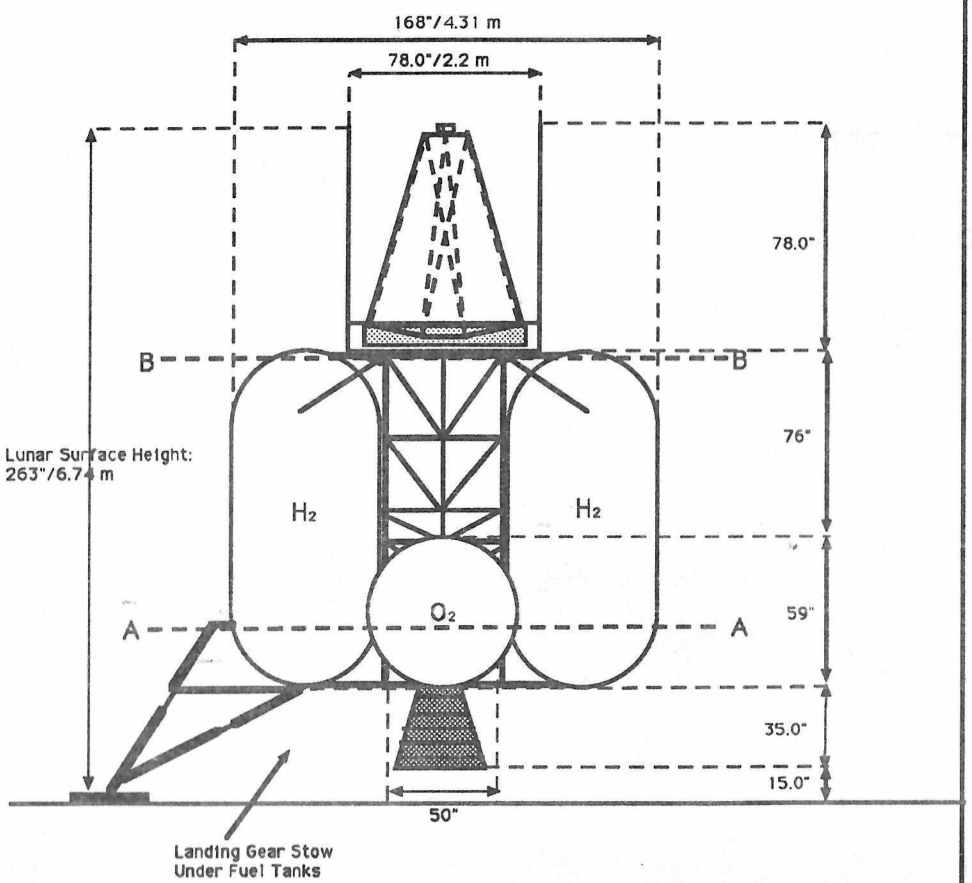
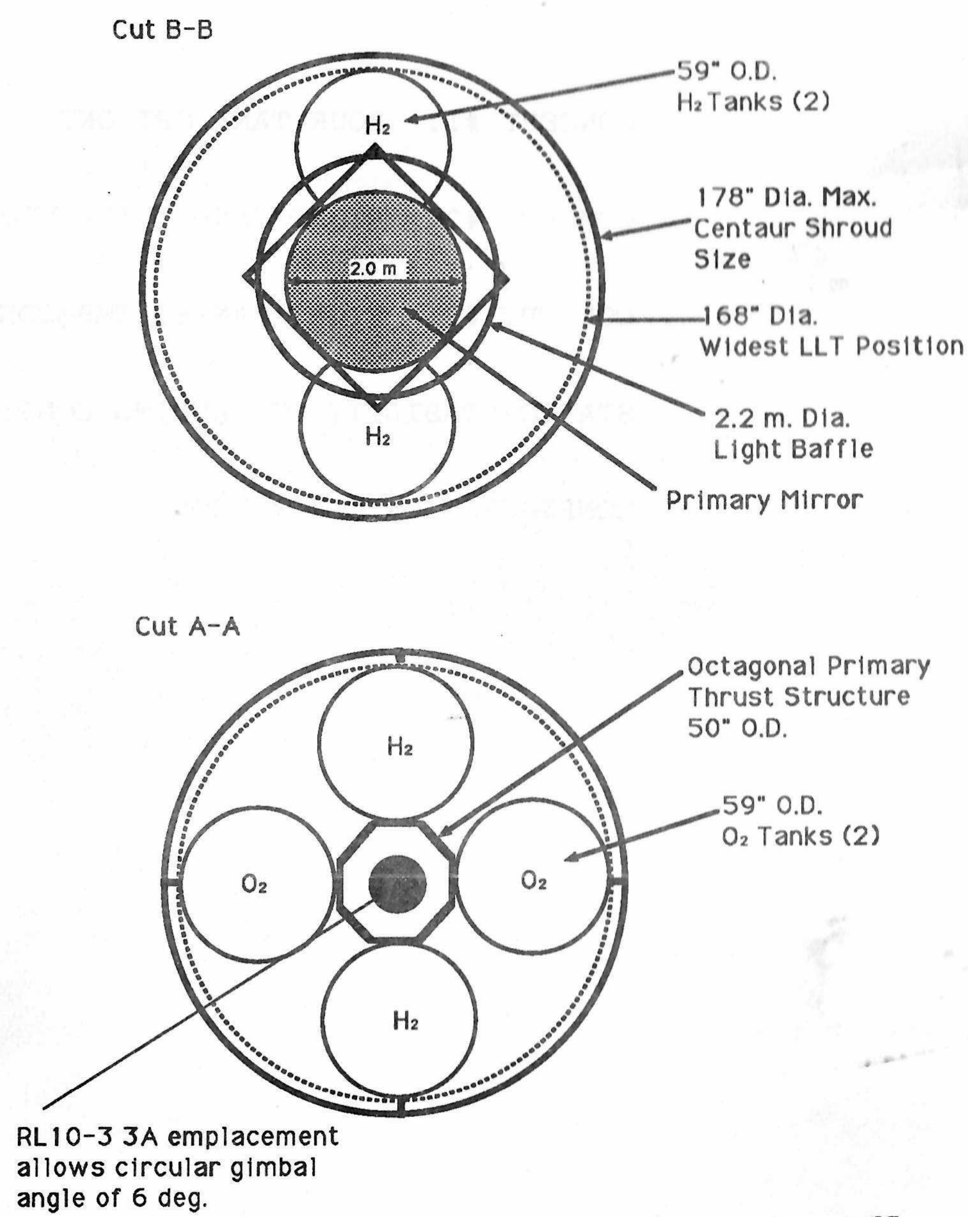
PD 34

Side View



Launch Pad/Payload Center of Gravity = 81.9"
Lunar Landing Center of Gravity = 108.42"

Top Views



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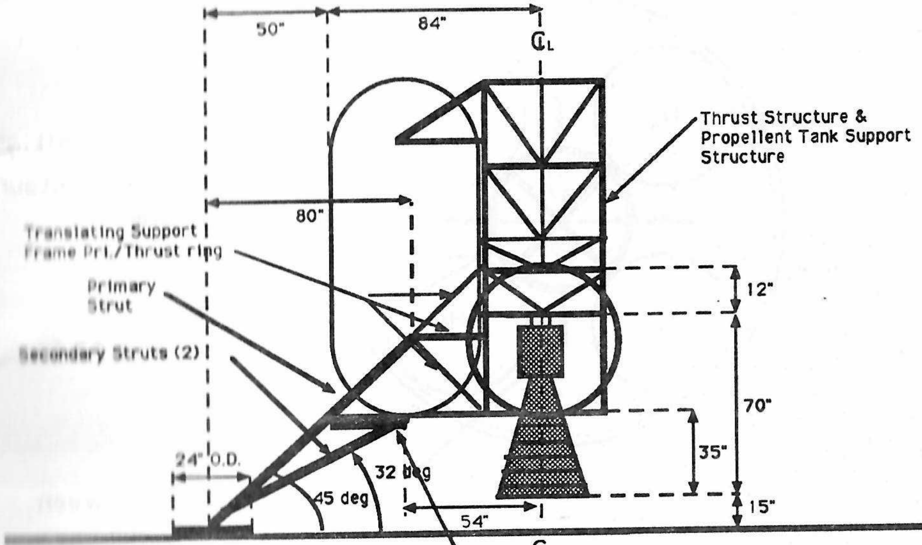
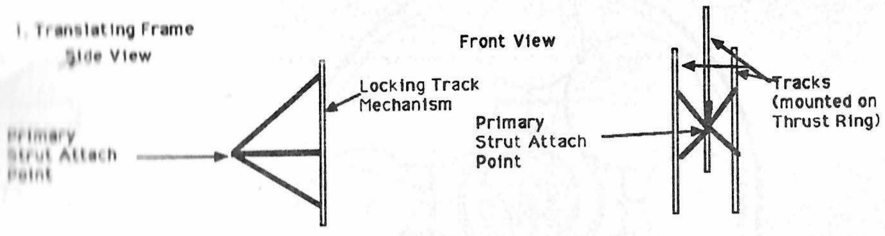
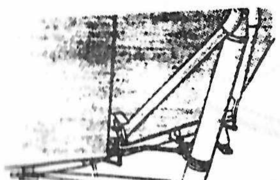
LTT Landing Gear for Concept # 1

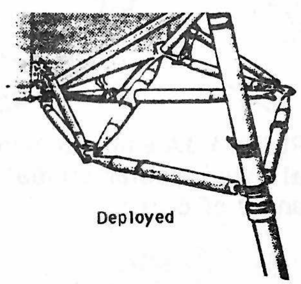
Side View
Landing Gear Dimensions







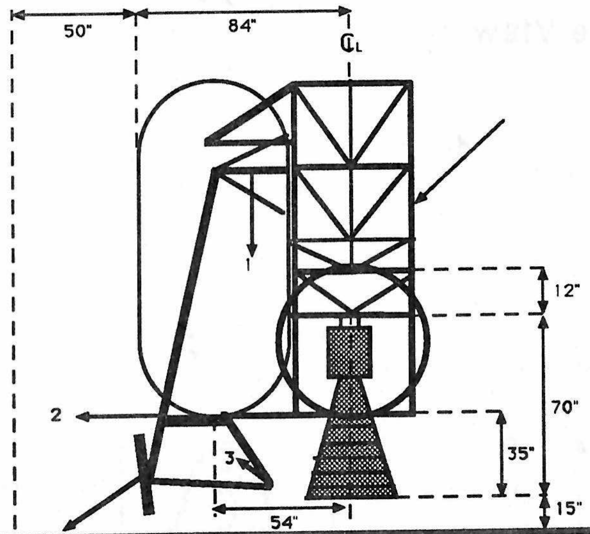
Stowed



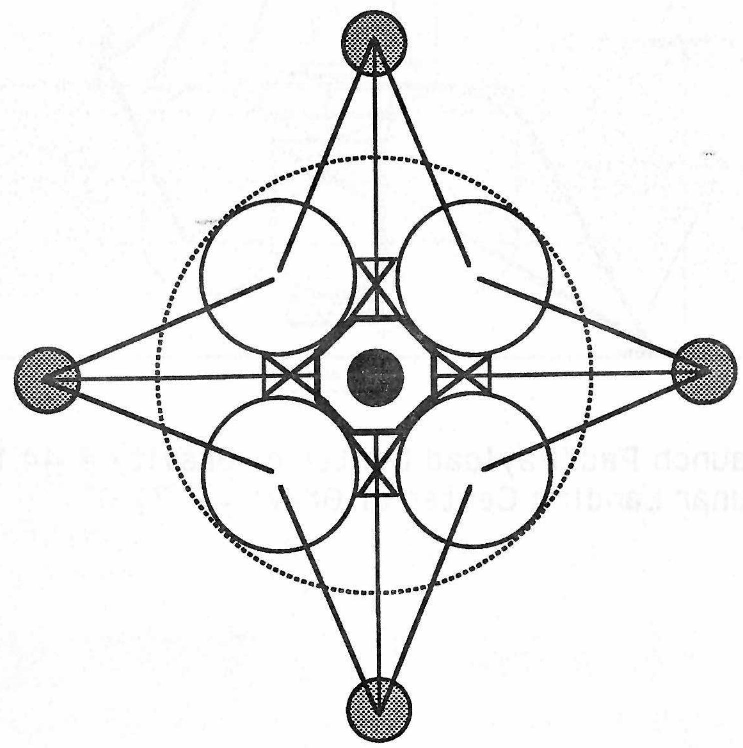
Deployed

Side View

Lunar Lander is Statically Stable toa lunar Slope less than 15 degrees

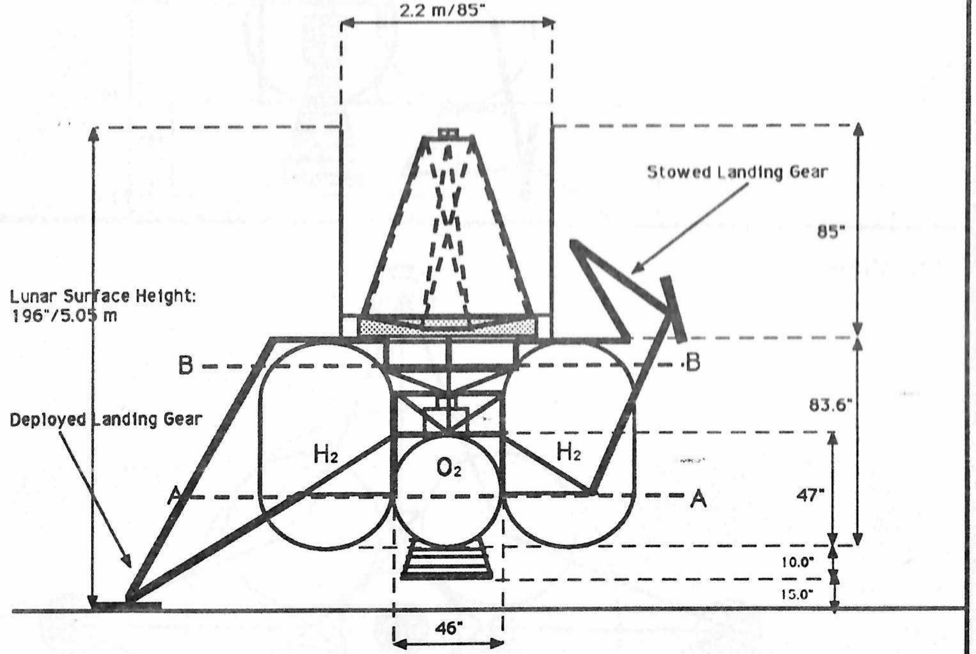


Top View



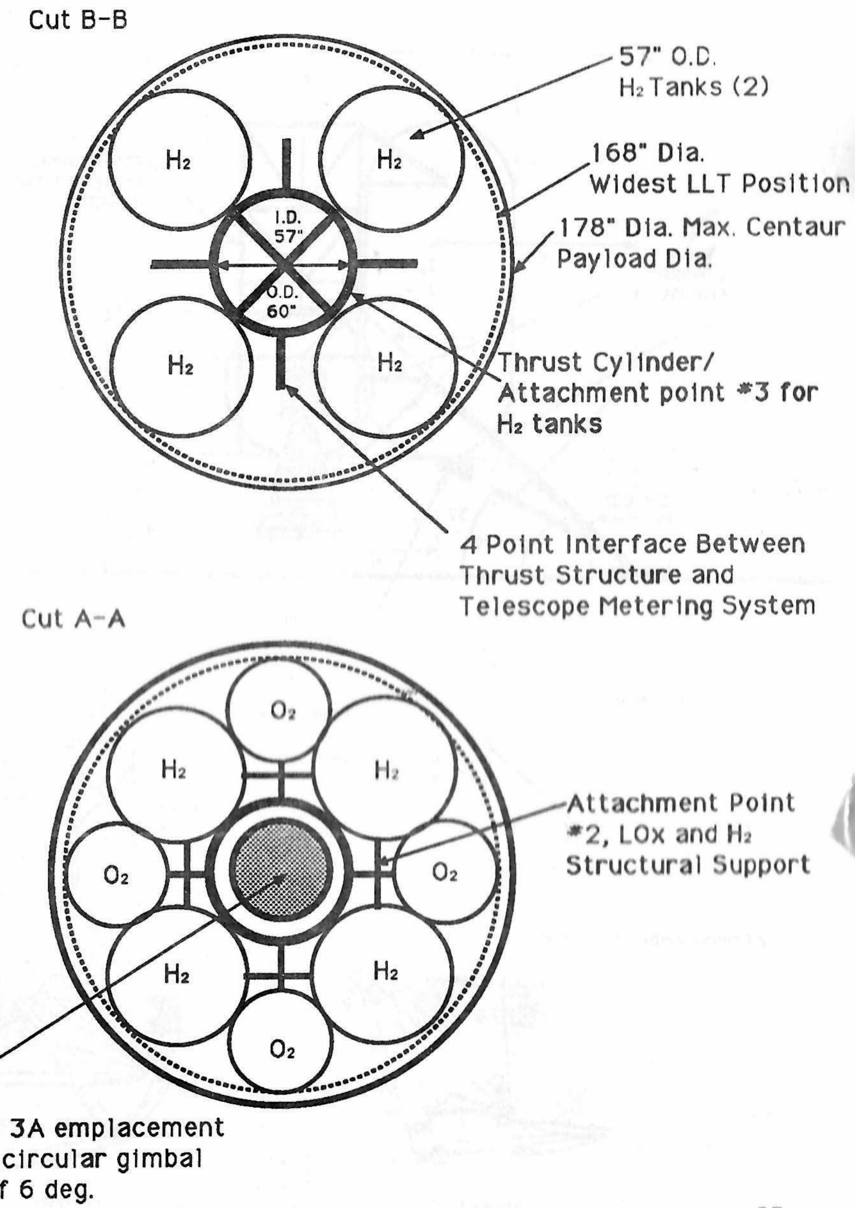
Lunar Transit Telescope Concept Drawing #2

Side View



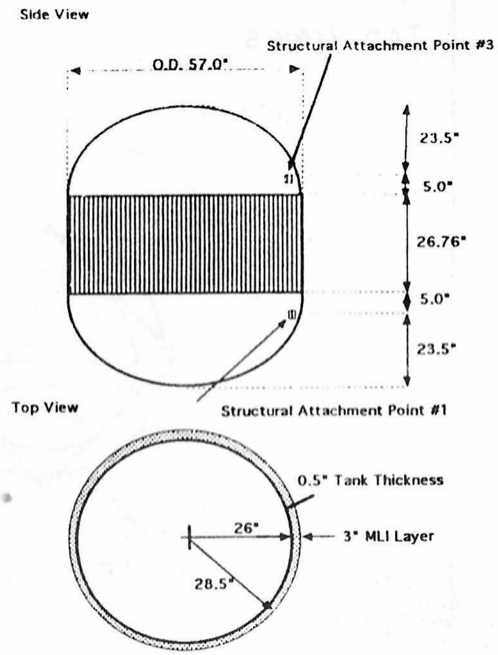
Launch Pad/Payload Center of Gravity = 44.1"
Lunar Landing Center of Gravity = 77.0"

Top Views



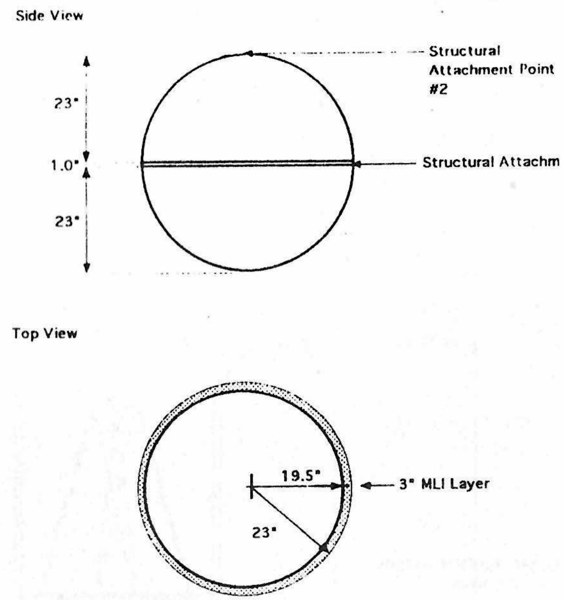
Hydrogen Fuel Tank Design

H_2: Load 302.0 cu. ftDescription 1. Spherical End Caps 2. Cylindrical Mid-sectionsNu. of Tanks: 40.D.: 57"I.D.:52"TPS: 3" thickness (MLI)Wall Thickness: 0.5"Cyl. Height: 26.8"Sph. O.D. Radius: 28.5"Height: 83.8" (6.98 ft)



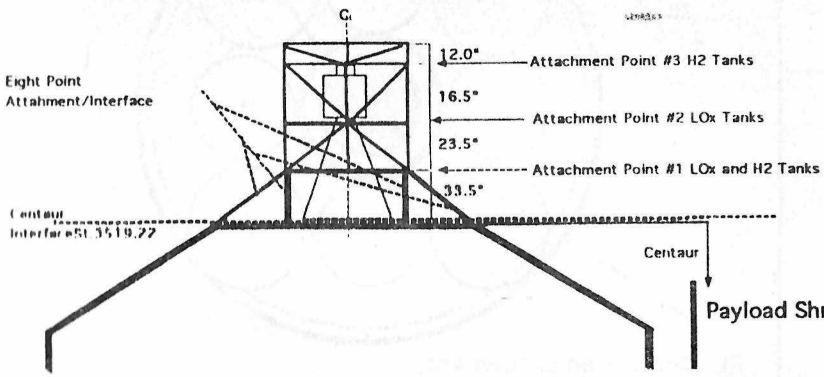
LOx Fuel Tank Design

O2: Load 92.0 cu. ft
Description
1. Spherical ends
2. Cylindrical Section
Nu. of Tanks: 4
OD.: 46"
I.D.:39"
TPS: 3" thickness
Wall Thickness: 0.5"
Cyl. Height: 1.0"
Sph. O.D. Radius:23"
Overall Height: 47"



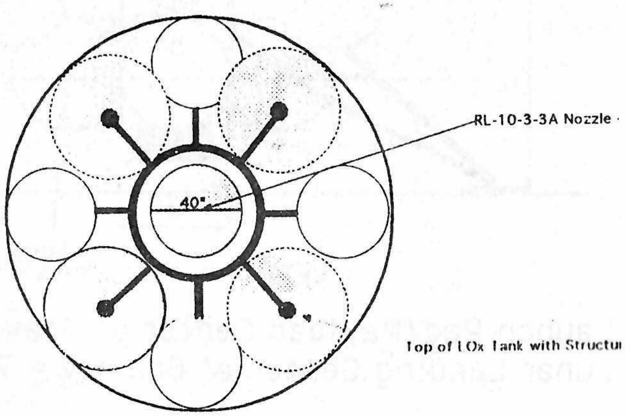
Thrust Structure/Support Structure

A. Side View, Primary Structure & Centaur Interface



B. Top View, Interface #2

View Elevation 57"Above Engine Nozzle



Lunar Transit Telescope Concept Drawing #3

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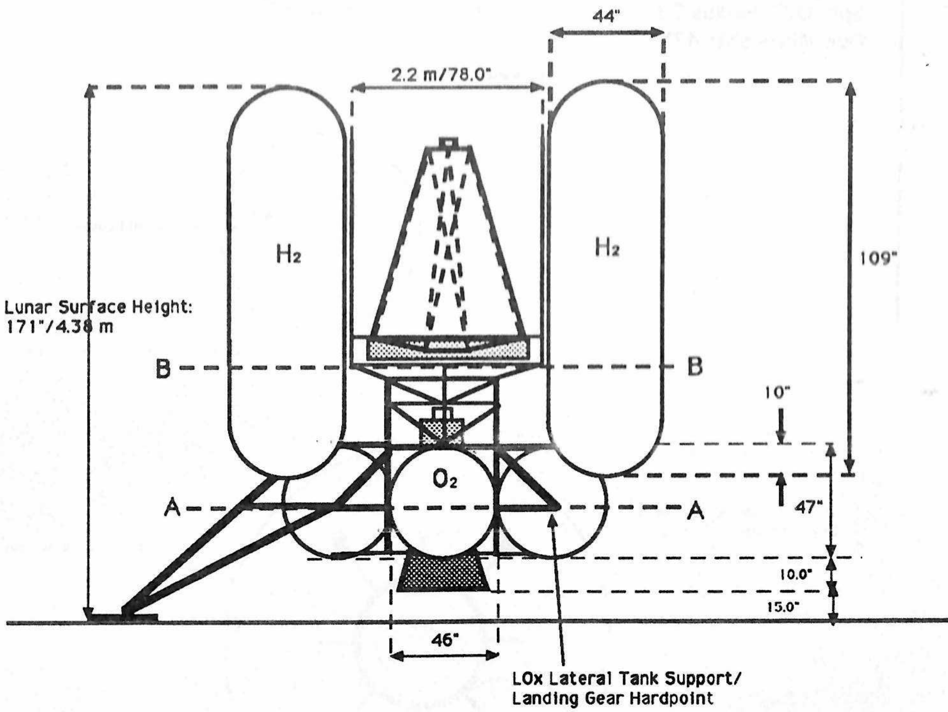
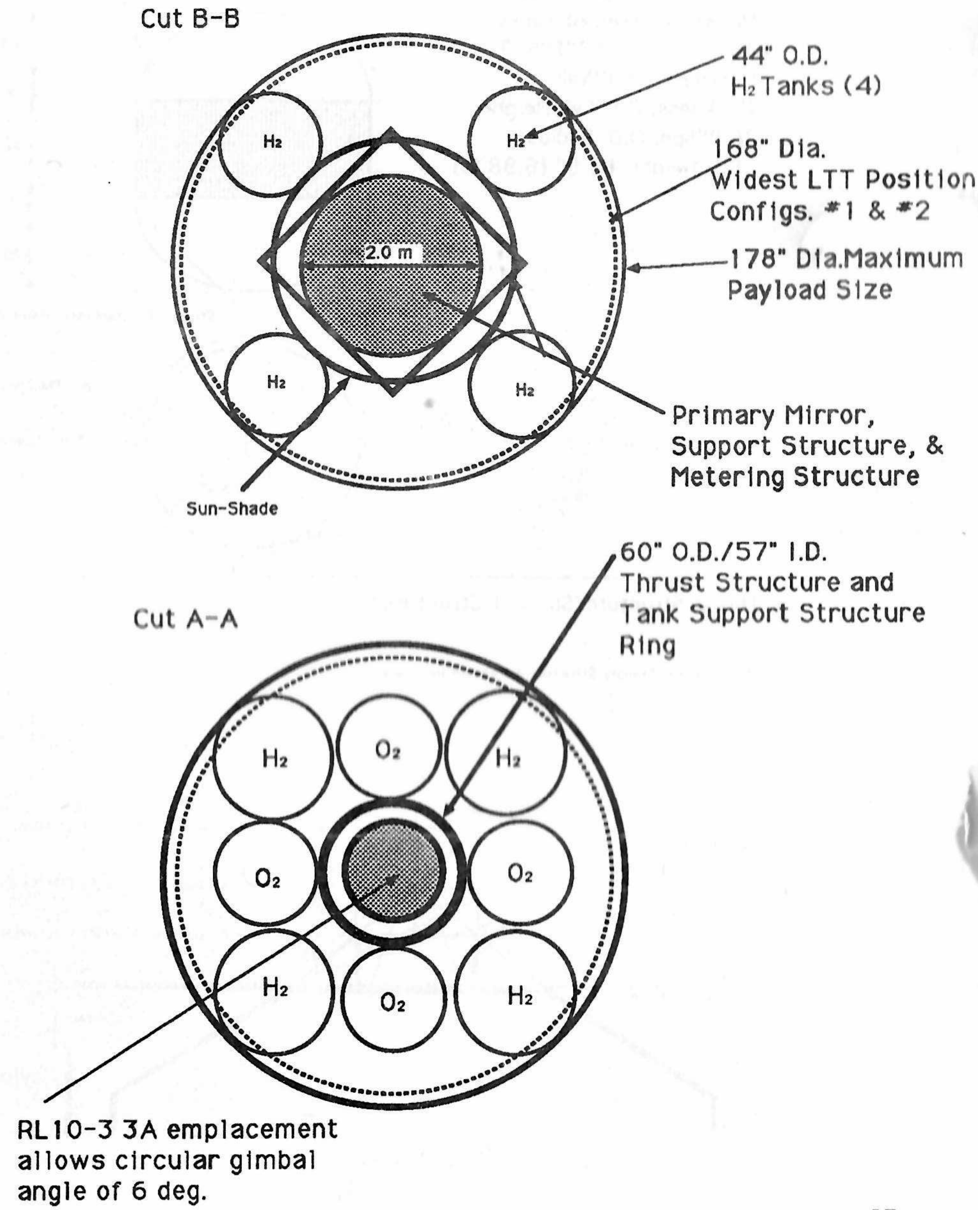
PD 34

Side View



Launch Pad/Payload Center of Gravity = 50.1"
Lunar Landing Center of Gravity = 70.6"

Top Views



LLT Landing Gear for Concept #3

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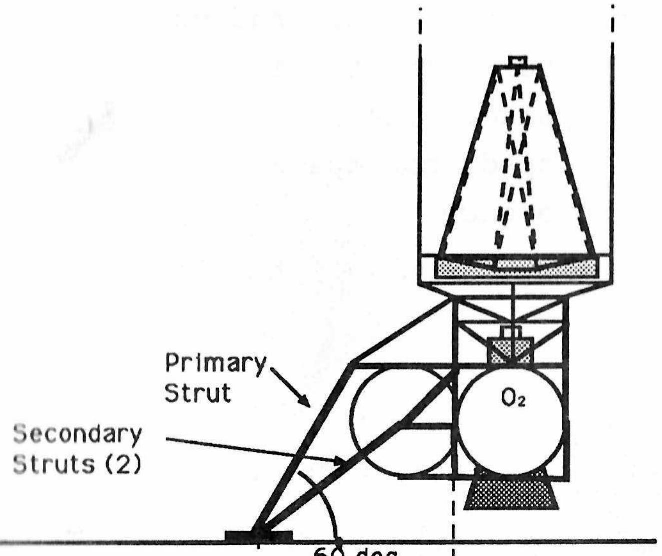
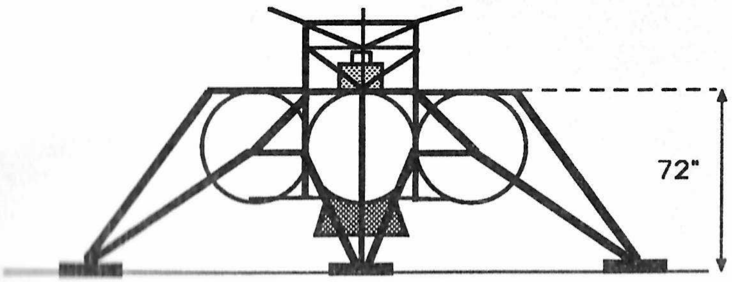
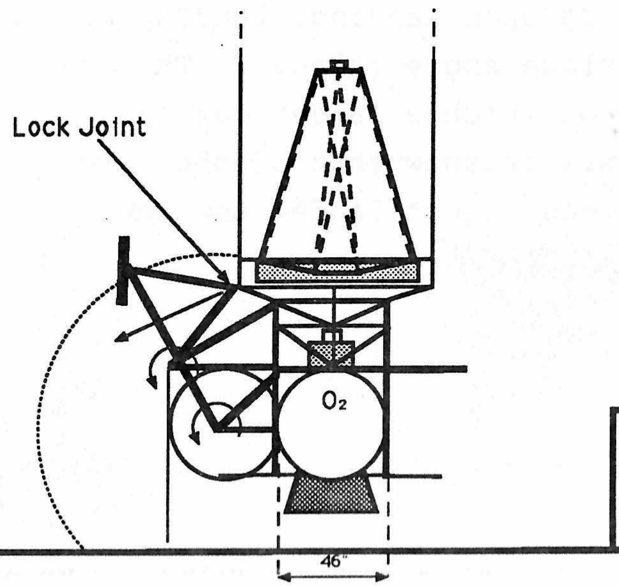
Side View



Side View



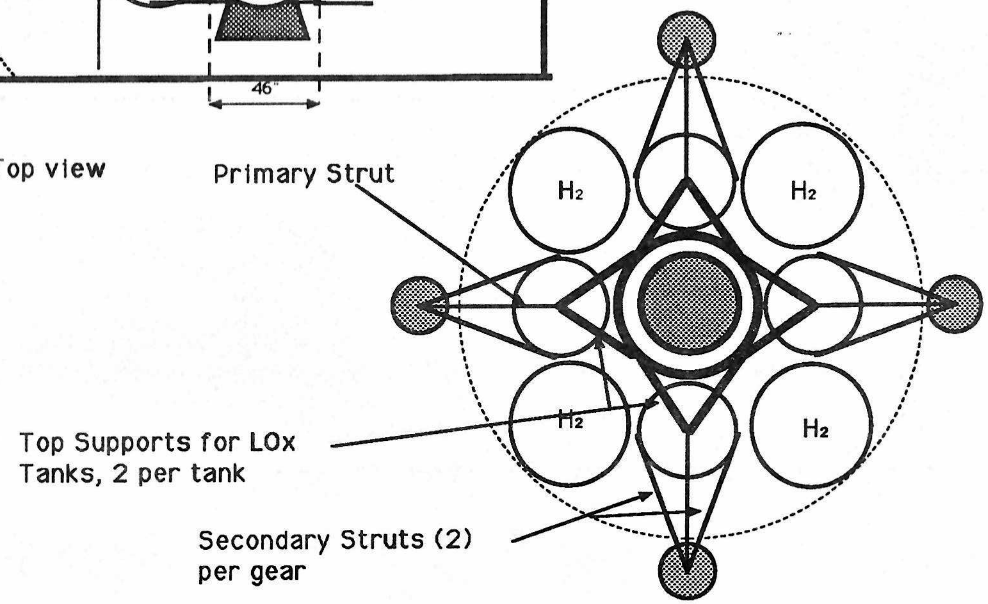
Side View Landing Gear Stowed



Since the propellent load is positioned low (10") above the engine nozzle, the landing gear must be stowed upright

Landing Gear is statically stable to Lunar Slope angles ~~greater~~ less than 35 degrees.

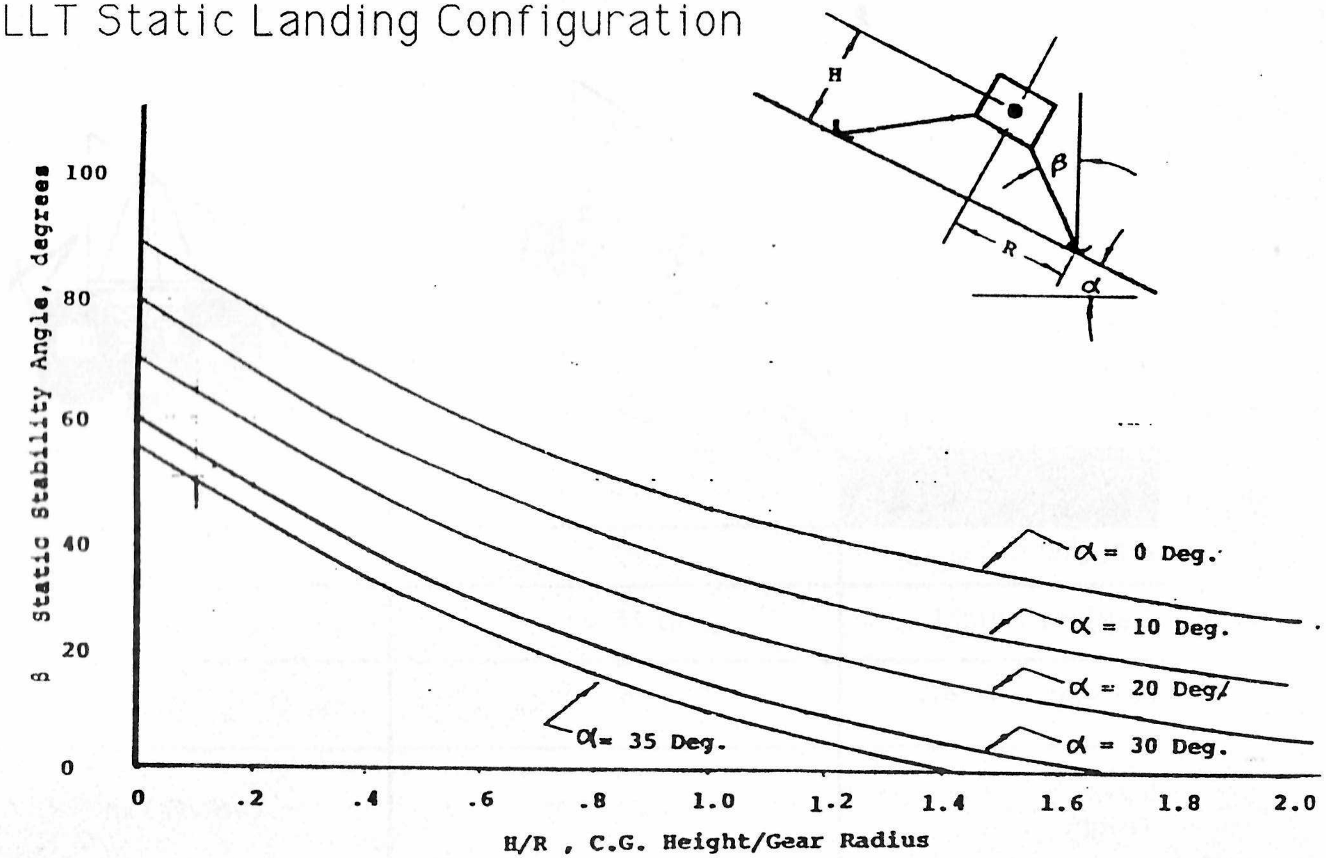
Top view



LLT STATIC LANDING CONFIGURATION

The static stability of the three configuration concepts were assessed and results are shown. The four characteristics determining stability are the height of the cg upon landing, landing gear angle (beta), landing gear radius (R), and the slope angle (alpha). The static stability characteristic is the maximum slope on which a lander can land upon without tipping. The preliminary results show that of the three concepts developed, concept 3 had the shortest, most squat lander and the largest slope tolerence.

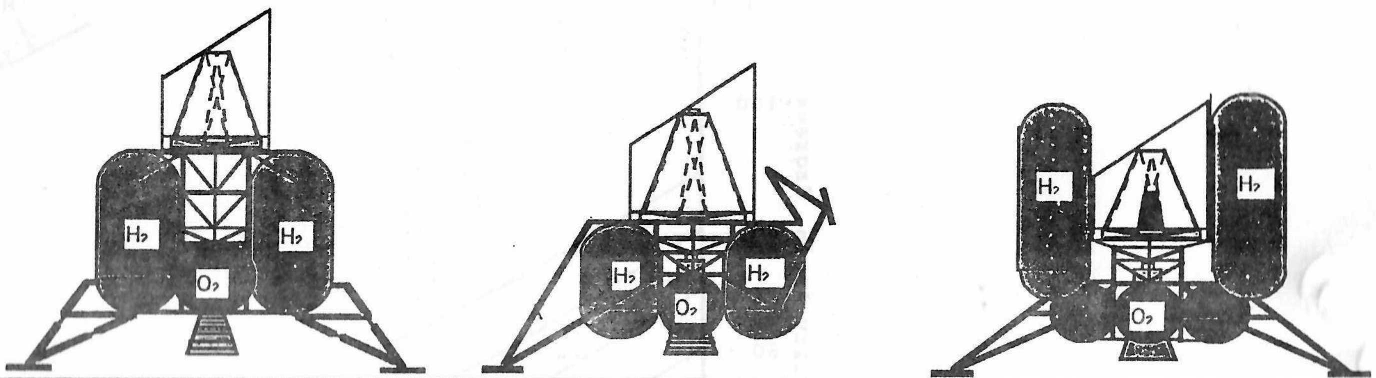
LLT Static Landing Configuration



Variation of the Static Stability Angle with C.G. Height to Landing Gear Radius Ratio

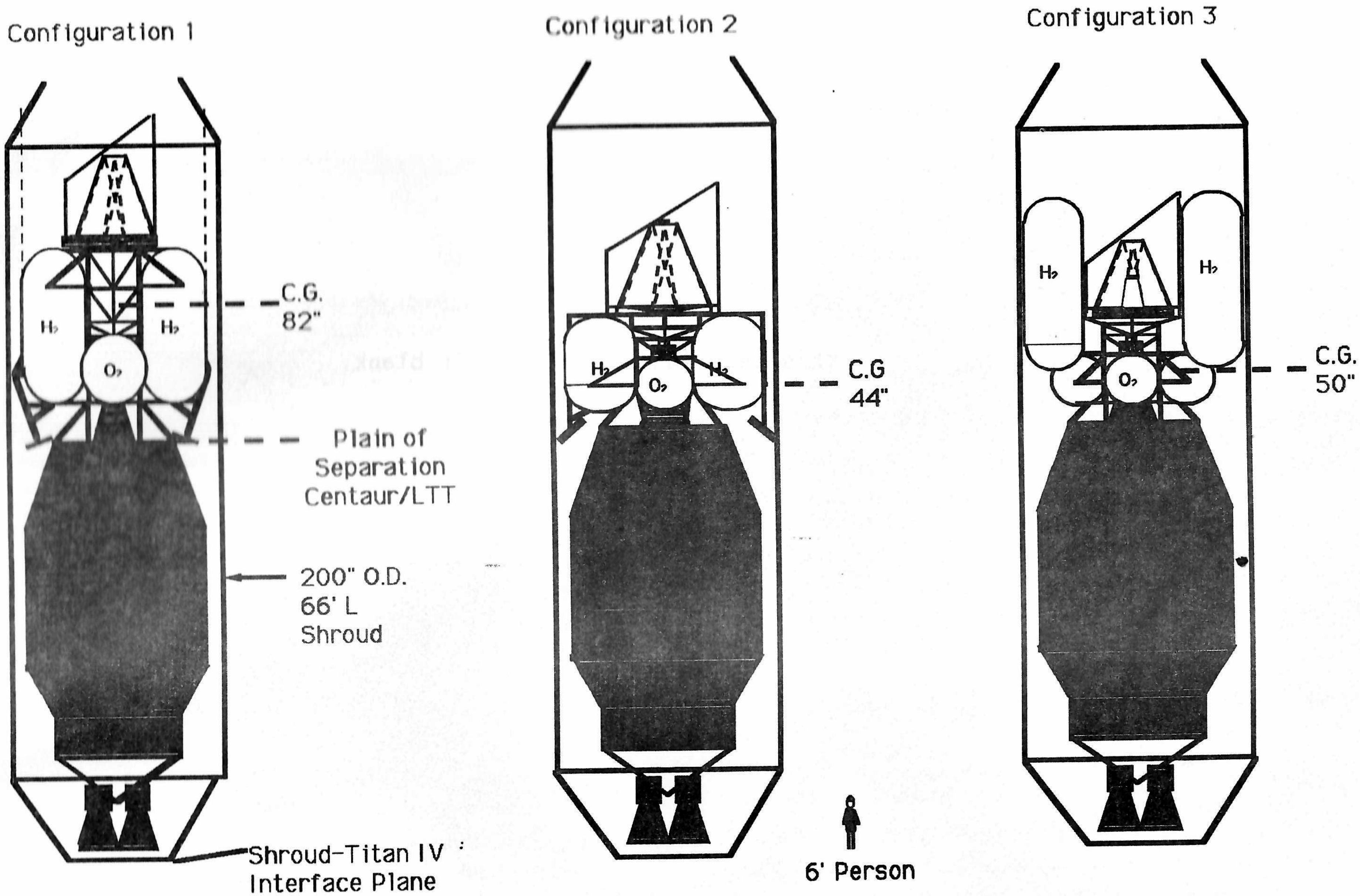
Configuration	Landing C.G.	Landing Gear Spread	Maximum Alpha
Concept #1	109"	134"	15 deg
Concept #2	77"	125"	25 deg.
Concept #3	71"	118"	+30 deg.

Lunar Transit Telescope Configuration Comparison



	Configuration #1	Configuration #2	Configuration #3
# Fuel/LO_x Tanks	2/2	4/4	4/4
Payload Height	248"/6.4 m	181"/4.6 m	156"/4.0 m
Payload Diameter	178"	178"	178"
Lunar Surface Height	263"/6.7 m	196"/5.1 m	171"/4.4 m
Weight (Dry) kg	3,115 →		TBD for ind. Config.'s
Landing C.G.	109"	77"	71"
Landing Gear Spread	134"	125"	115"
Max Static Stability Angle	15 deg.	25 deg	35+ deg

Lunar Transit Telescope LLT/Centaur Sizing Comparison



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SECTION 7

7.1 LTT SCHEDULE

LUNAR TRANSIT TELESCOPE SCHEDULE

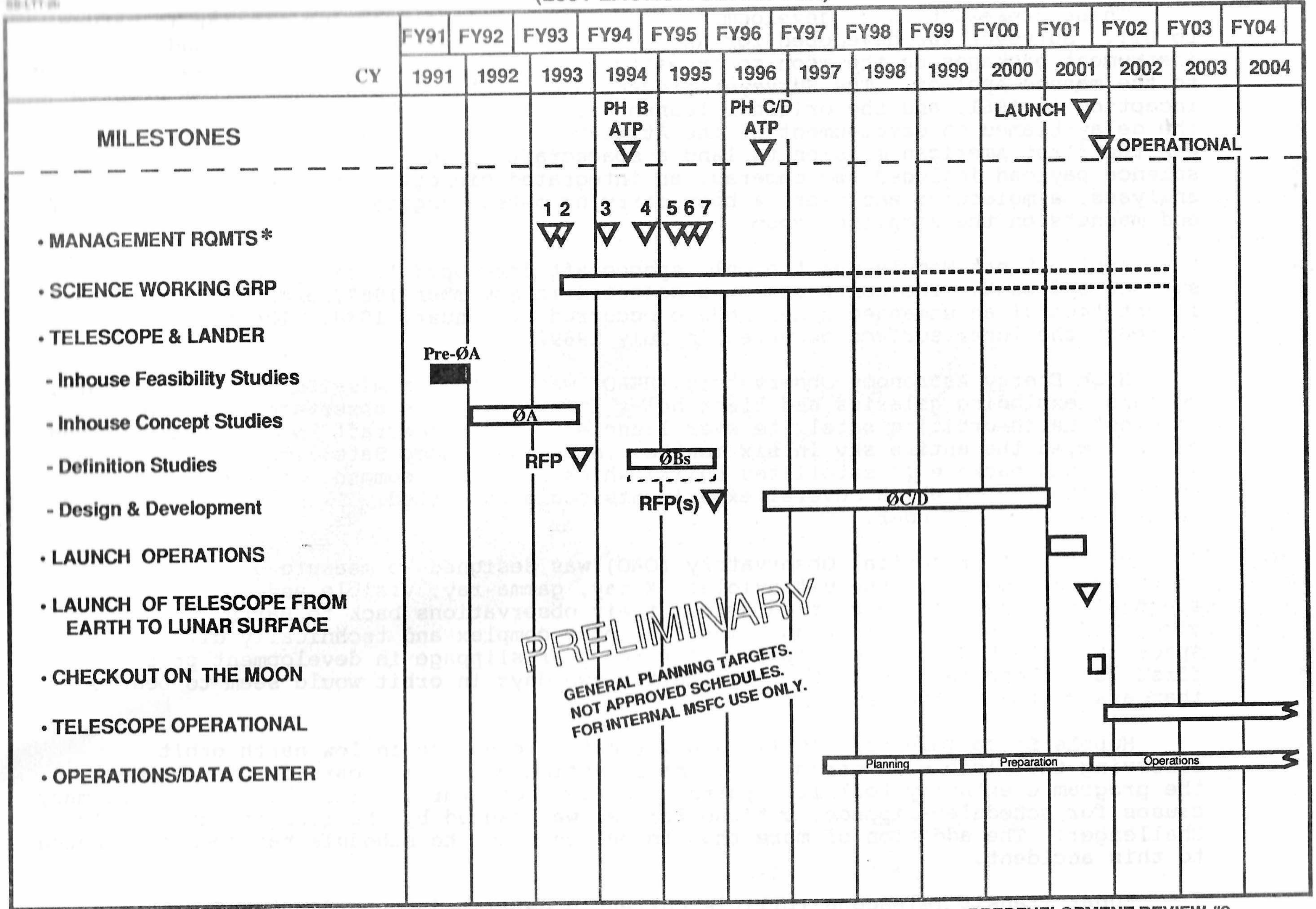
The LTT schedule is based on a 2001 launch goal. Assuming a five-year development, a Phase C/D new start is required in 1996 and Phase B should begin no later than 1994. An inhouse Phase A is planned to begin in January 1992, after the current Pre-Phase A studies are completed. Most of the Management Requirements shown are from OMB Circular A-109 and may be deleted if an A-109 waiver is obtained.

The five-year LTT development estimate is based on previous telescope project schedules. Possible long-lead items currently identified are the RTGs, CCD arrays, mirrors, and the launch vehicle. Detailed subsystem-level development schedules will be developed during the Phase B definition studies. Other long lead items and schedule drivers may be identified during these studies.

LUNAR TRANSIT TELESCOPE

(2001 LAUNCH SCHEDULE)

PS02/M. NEIN
PD11/B. DAVIS
PP02/ S. SPEARMAN
17 DEC 91



* 1 = MISSION NEEDS STATEMENT
2 = PREDEVELOPMENT REVIEW #1
3 = NEW START REVIEW
4 = MSFC SIGNED PIA
5 = DEFINITION REVIEW (NAR)
6 = SUBMIT PROJECT PLAN
7 = PREDEVELOPMENT REVIEW #2

NOTE: These are A-109 requirements. If A-109 waiver is obtained, some of these will not be required.

SELECTIVE HISTORY OF TELESCOPE & LANDER PROGRAMS

These are examples of development times for previous programs similar in nature to LTT. Surveyor was developed for NASA to soft-land television cameras and instrument packages on the moon to investigate the lunar environment and was a prelude to the manned Apollos. The Surveyor program encountered delays almost from program inception in 1961, and the original launch date stretched three years, with much of the delay blamed on development of the Atlas-Centaur launch vehicle. Viking Lander was the first American mission to land a spacecraft on another planet. Its surface science payload included two cameras, an integrated biology instrument for soil analyses, a molecular analyzer, a boom carrying meteorological sensors, a seismometer and magnets on the sampling scoop.

Apollo Lunar Module was the only spacecraft developed to carry man to the lunar surface and back. The contractor was selected in November 1962, and the Earth orbital flight test of an unmanned Lunar Module occurred in January 1968. The first mission to reach the lunar surface occurred in July 1969.

High Energy Astronomy Observatory (HEAO) was part of a mission to study pulsars, quasars, exploding galaxies and black holes in space. This observatory is the heaviest Earth-orbiting satellite ever launched. The spacecraft rotated end over end and surveyed the entire sky in six months. Small Astronomy Satellite (SAS-A) was a standardized package (3 satellites built) which contained command and control instrumentation to which several experiments could be fitted. This spacecraft was a spin-stabilized cylinder.

Orbiting Astronomical Observatory (OAO) was designed to measure the energy output of celestial bodies in the ultraviolet, X-ray, gamma-ray, visible and infrared spectral regions and transmit data from their observations back to Earth. At the time, OAO was termed by NASA as being "the most complex and technically difficult spacecraft the U.S. has developed." A three-year slippage in development prior to the first launch and failure of the OAO-1 after two days in orbit would seem to bear out that assessment of the program.

Hubble Space Telescope (HST) is a 2.4 meter telescope in low earth orbit observing the visible spectrum. Originally estimated at six years for development, the program eventually took 12.5 years from contract award to launch. There were many causes for schedule slippage, but the largest was caused by the loss of the shuttle Challenger. The addition of more than three years to the schedule has been attributed to this accident.

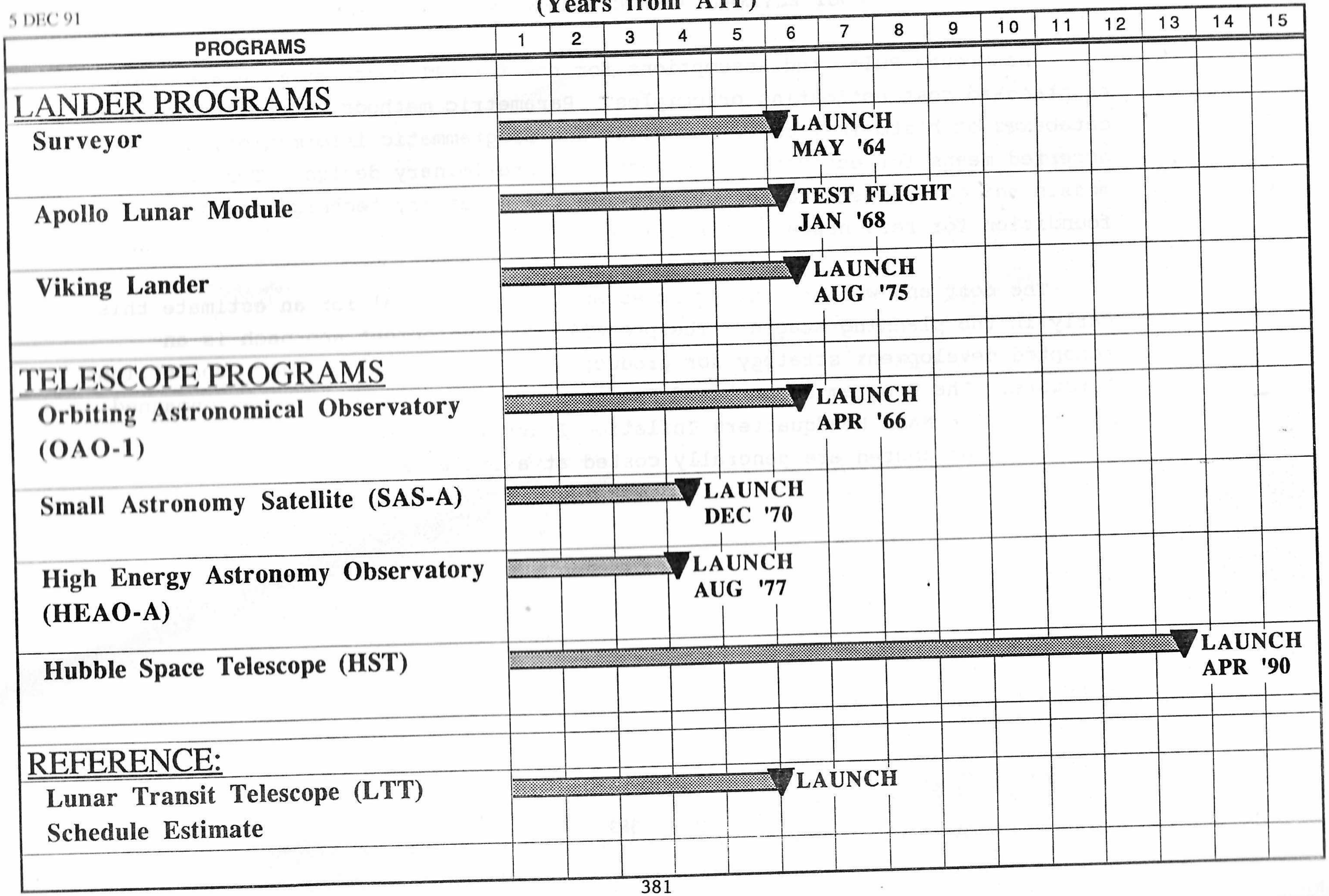
Lunar Transit Telescope (LTT) is currently estimated to be developed within a five year period.

Development History of Selected Telescope & Lander Programs

(Years from ATP)

5 DEC 91

MSFC/PP02-SPEARMAN



7.2 COST ESTIMATE

COST ESTIMATE GROUND RULES AND ASSUMPTIONS

The Ground Rules and Assumptions for the for LTT cost estimate are based on standard cost estimating principles. Parametric methods, supported by databases of historical cost, technical and programmatic information, are the accepted means for estimating the cost of a preliminary design. The cost models and databases used for LTT provide the necessary techniques and foundation for reasonable cost estimates.

The cost and weight contingencies applied are normal for an estimate this early in the planning stages. The protoflight development approach is an accepted development strategy for producing high quality, unmanned, spaceflight hardware. The Program Support and Fee percentages are normal for an unmanned mission. The NASA Headquarters Inflation Index is the recognized standard. The items not costed are generally costed at a later date.

Lunar Transit Telescope Cost Estimate Ground Rules and Assumptions

- o The Telescope and Science Instrument costs are estimated using the Scientific Instrument Cost Model, Version 4.0 and the Multivariable Instrument Cost Model.

- o The Lander/Spacecraft subsystem costs are estimated using the Apollo Lunar Module, Viking Lander, Galileo Orbiter, and Centaur G-Prime data from the NASA Cost Model (NASCOM). The structure and reaction control subsystem costs are estimated using the Unmanned Spacecraft Subsystems Cost Model. Complexity factors are applied to some subsystems to account for technology improvements. The system level costs are estimated using NASCOM developed factors.

- o Individual RTG cost of 28.6 million dollars provided by the DOE.

- o Costs for the RL-10-3 engine are taken from NASCOM. A nonrecurring cost equal to 10% of the recurring is added to acount for any modifications.

- o Integration and test costs are estimated using the Mission System Integration and Test (MSI&T) Cost Model, Version 2.0.

- o All spaceflight hardware is assumed to be procured using the protoflight approach.

- o All estimated costs are escalated to Fiscal Year (FY) 1991 dollars using the NASA Headquarters Inflation Index dated May 23, 1991.

- o Program Support costs are assumed to be 5% of the Flight Hardware costs.

- o A contingency cost of 30% of the Flight Hardware plus Program Support cost is included.

- o A fee of 10% on all costs is assumed.

- o All cost models are weight driven, a 20% contingency is added to all weights.

- o Items currently not costed are: Ground Systems Development; Ground Systems Operations (Data Analysis); Flight Operations; Launch Vehicle and Launch Operations; Phase A and B.

COST ESTIMATE SUMMARY

The LTT cost estimate is based on the traditional NASA procurement approach. The Telescope/Science Instrument is costed as an above average risk. The Science Instrument detector array requirements appear to be stringent and the detector chips have yet to be designed or built (new technology). The Lander/Spacecraft is costed at a slightly below average risk. The technology associated with landers and spacecraft is stable. Because of this, complexity factors are used that decrease the cost slightly (approximately 10%). The Integration and Test effort for LTT is costed at normal risk.

Lunar Transit Telescope Cost Summary

2m Telescope

	FY91$M
Telescope/Science Instrument	114.8
Lander/Spacecraft	887.3
Integration & Test	61.4
Total Flight Hardware	1063.5
Program Support (5%)	53.2
Total With Program Support	1116.7
Contingency (30%)	335.0
Total With Contingency	1451.7
Fee (10%)	145.2
Total LTT Flight System	1596.9

LTT ESTIMATE VERSION 2.4, 11/19/91

COST REDUCTION GROUND RULES AND ASSUMPTIONS

The cost reduction ground rules and assumptions are self explanatory. The main tenents are to define the requirements, freeze the cost, understand all risks, minimize technology development, minimize contractor oversight, and reward people and organizations for saving money.

Fully developing and understanding detailed requirements means that more time and effort is spent in Phase A/B determining what LTT will do, and be capable of doing, rather than how it will be done. A realistic cost target must be set and treated as a fixed design parameter. The cost must be obtainable both from budgetary and requiremental points of view. By understanding and managing risks, suprises that increase cost can be minimized. Technology development costs money. The less technology development associated with LTT the lower the costs. Minimizing contractor oversight frees the contractor to develop a lander that meets all of the requirements using the organization and methods that the contractor determines will reduce cost. Minimal oversight combined with financial rewards or penalties based on how well the requirements are met, allows NASA to act as a consumer, rather than a monopolistic purchaser. The establishment of incentive plans for saving money is important to encourage NASA employees to apply these new business practices. Contractors should enact similar plans; but, in the spirit of the previous tenet, this should not be specified by NASA.

LTT Development Cost Reduction Ground Rules and Assumptions

o Phase B Changes
- Establish strong, dedicated project team representing multiple disciplines
- Provide sufficient civil service manpower and financial resources to fully develop and understand the requirements
- Use Continuous Improvement (CI) and design-to-cost techniques to rigorously develop the requirements
- Treat cost as a design parameter equal to all other design parameters, freeze the cost at the system and subsystem level
- Understand all technical, management, and cost risks; develop contingency plans where necessary
- Use or evolve and hardware from existing designs, minimal technology development
- Breadboard as much hardware as possible, especially hardware determined to be high risk
- Develop detailed requirements, not a detailed design
- Freeze requirements at some point near the end of Phase B, allow no new requirements or changes to existing requirements
- Establish cost saving incentives for all Phase B participants

o Phase C/D Changes
- Use existing project team that was established in Phase B
- Minimal contractor oversight by NASA, hardware is built to satisfy a set of fixed requirements using the techniques preferred by the contractor
- Provide financial rewards for meeting requirements and penalties for failure to meet requirements
- Establish cost saving incentive plans for all NASA employees working LTT
- Early multi-year funding commitment from headquarters at agreed-upon levels

NEW APPROACH COST ESTIMATE SUMMARY

Applying the Phase B and C/D cost reduction methodologies reduces the total flight hardware cost by about 54%. The Lander/Spacecraft and Integration & Test cost reduction of 45% is based on Air Force experience in applying these methods. It is assumed that a full definition of the requirements and an understanding and management of the risks can eliminate the 30% cost contingency. Program Support and Fee decrease with the reduced hardware cost. At this time the risks associated with the Telescope/Science Instrument preclude cost reduction.

Lunar Transit Telescope Cost Summary

2m Telescope

FY91$M

	Traditional	New Approach	Savings
Telescope/Science Instrument	114.8	114.8	--
Lander/Spacecraft [1]	887.3	515.9	371.4
Integration & Test [1]	61.4	33.8	27.6
Total Flight Hardware	1063.5	664.5	399.0
Program Support (5%)	53.2	33.2	20.0
Total With Program Support	1116.7	697.7	419.0
Contingency [2]	335.0	--	335.0
Total With Contingency	1451.7	697.7	754.0
Fee (10%)	145.2	69.8	75.4
Total LTT Flight System	1596.9	767.5	829.4

[1] Cost reduced 45% based on Air Force Experience. Cost for RTG's and RL10 not reduced.

[2] Contingency elimination based on a full understanding and managing of all risks.

LTT ESTIMATE VERSION 2.4, 11/19/91

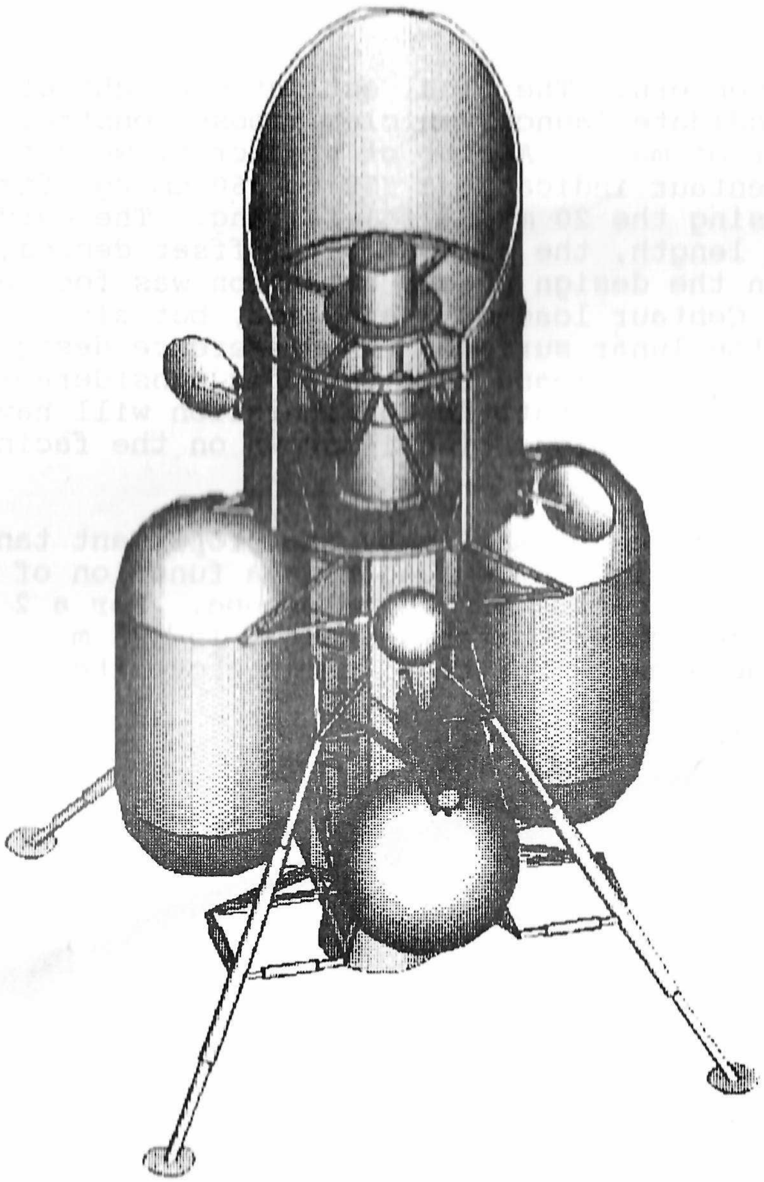
7.3 STUDY SUMMARY

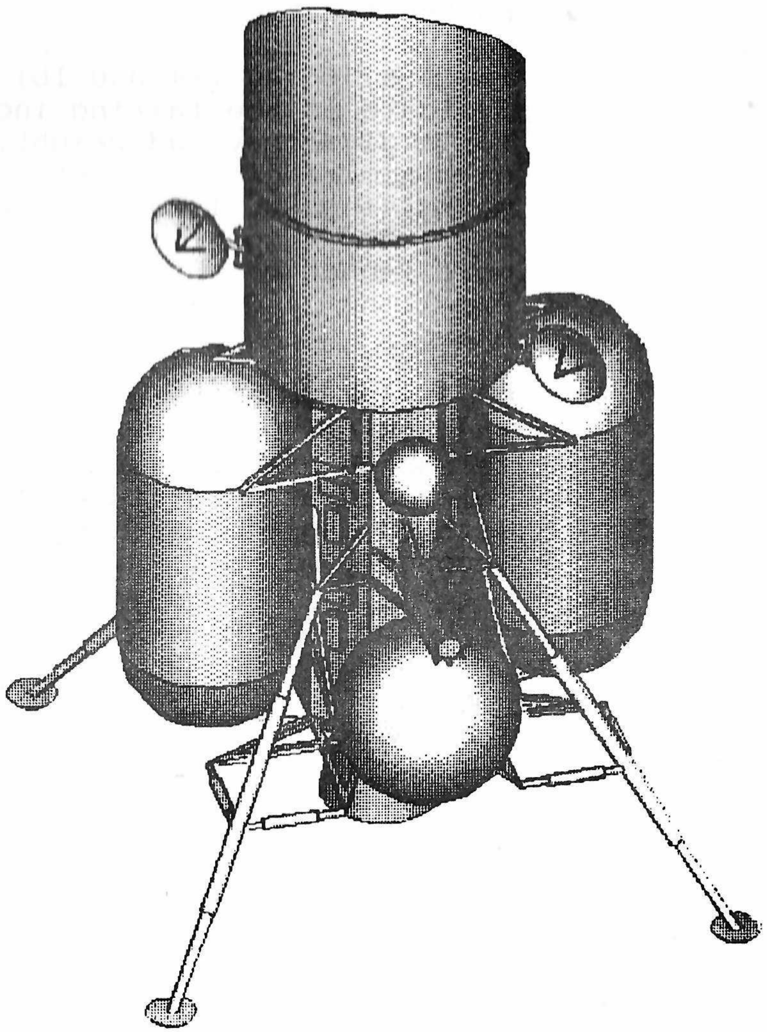
LTT DESIGN OVERVIEW

In the process of conceptualizing a Lunar Transit Telescope, a number of pertinent design considerations emerged for evaluation and discrimination. These included, but were not limited to, launch vehicle selection, the decision to use either cryogenic or storable propellants, and selection of the optical configuration. The scientific community desired a 2 m diameter diameter telescope, or as large a diameter as can be accommodated by the launch vehicle. These decisions have considerable effect on the performance and quality of the LTT design, and in some cases are so strongly influenced by the specific application, that final selection or deletion can only be made when evaluated for that singular case.

The desired early launch date (1996-2001 time period) dictated that only existing launch vehicles and technology be considered. Trades in the propulsion discipline indicated that it favored the cryogenic system for the lander. An Atlas IIAS and a Titan IV vehicle both using Centaur upper stages were studied. The Atlas offers a 3.6 m diameter payload envelope and 2,500 kg to translunar injection versus the Titan 4 m diameter shroud and 6,364 kg to TLI. The Titan shroud is also considerably longer than the Atlas. It was determined that for a 2 m class telescope and a lander utilizing cryogenic propellants that the Atlas was too volume and weight limited. The Titan was selected for the LTT, however, mass landed on the moon remains a concern for the LTT mission. The reference 2 m LTT is illustrated on the facing chart.

The propellant tank configuration has a major influence not only on the LTT's propulsion and thermal control system, but also on the LTT's structure. This is primarily due to the relatively large mass of propellant (about equal to the inert mass of the lander/telescope) and the physical dimensions of the RL10 engine. The design requirements for mounting a single large tank or a number of smaller ones, typically have a significant influence on the structural design of the spacecraft. Usually, the number of tanks and associated attachment hardware should be minimized to reduce heat leakage to a cryogenic system. The RL10 engine is approximately 178 cm in length. Unless the tank layout lends itself to embedding the engine, a relatively large system cg offset will occur. This can be overcome with a multiple or toroidal tank configuration with a recessed engine.





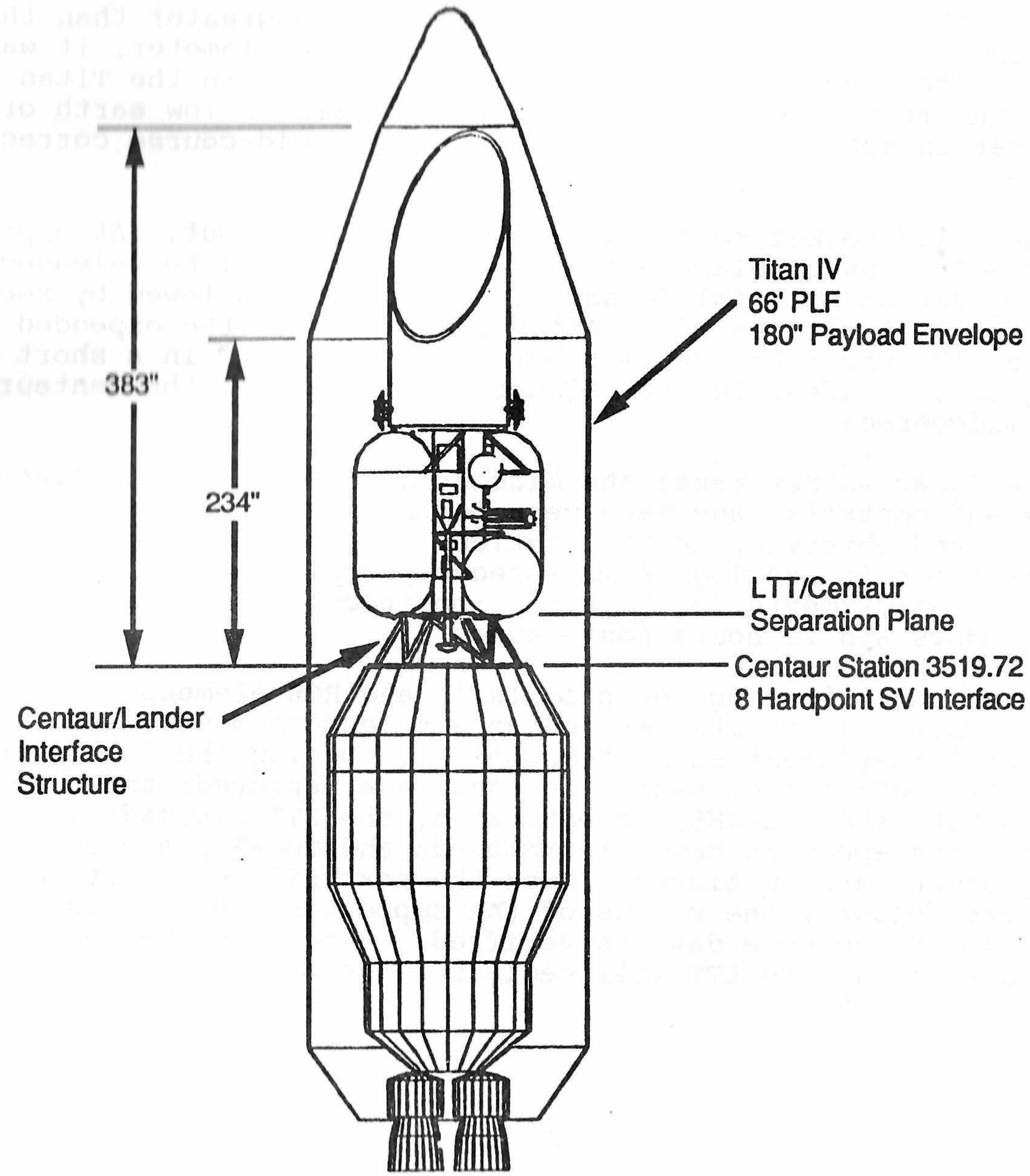
LTT LAUNCH CONFIGURATION

The LTT mass is of substantial concern. The final estimated weight of the LTT is 6,270 kg. Most of the candidate launch vehicles impose constraints on the location of the payload center of mass. A plot of spacecraft weight versus cg location for the current Centaur indicates a 355 to 360 cm cg offset for a 6,364 kg (14,000 lb) payload using the 20 m (66 ft) fairing. The curves indicate as the fairing increases in length, the allowable cg offset decreases for a given payload weight. Early in the design phase, attention was focused on minimizing the cg not only from a Centaur loading standpoint, but also for maximizing the landing stability on the lunar surface. The reference design 2 m LTT has a cg offset of only 267 cm, which means that there is considerably more launch weight margin than indicated and that the configuration will have excellent landing stability. The launch configuration is shown on the facing chart.

The overall length of the LTT is driven primarily by the propellant tank and optical configuration. The sunshade/lightshade length is a function of telescope diameter and desired latitude location of the telescope. For a 2 m telescope at a latitude of 40^{o}, the sunshade/lightshade height is 3.5 m with an angle of 62^{o} degrees, which determines the total heigh above the secondary mirror.

1-2613-1-59D

Lunar Transit Telescope Launch Configuration



TRANSPORTATION TIMELINE

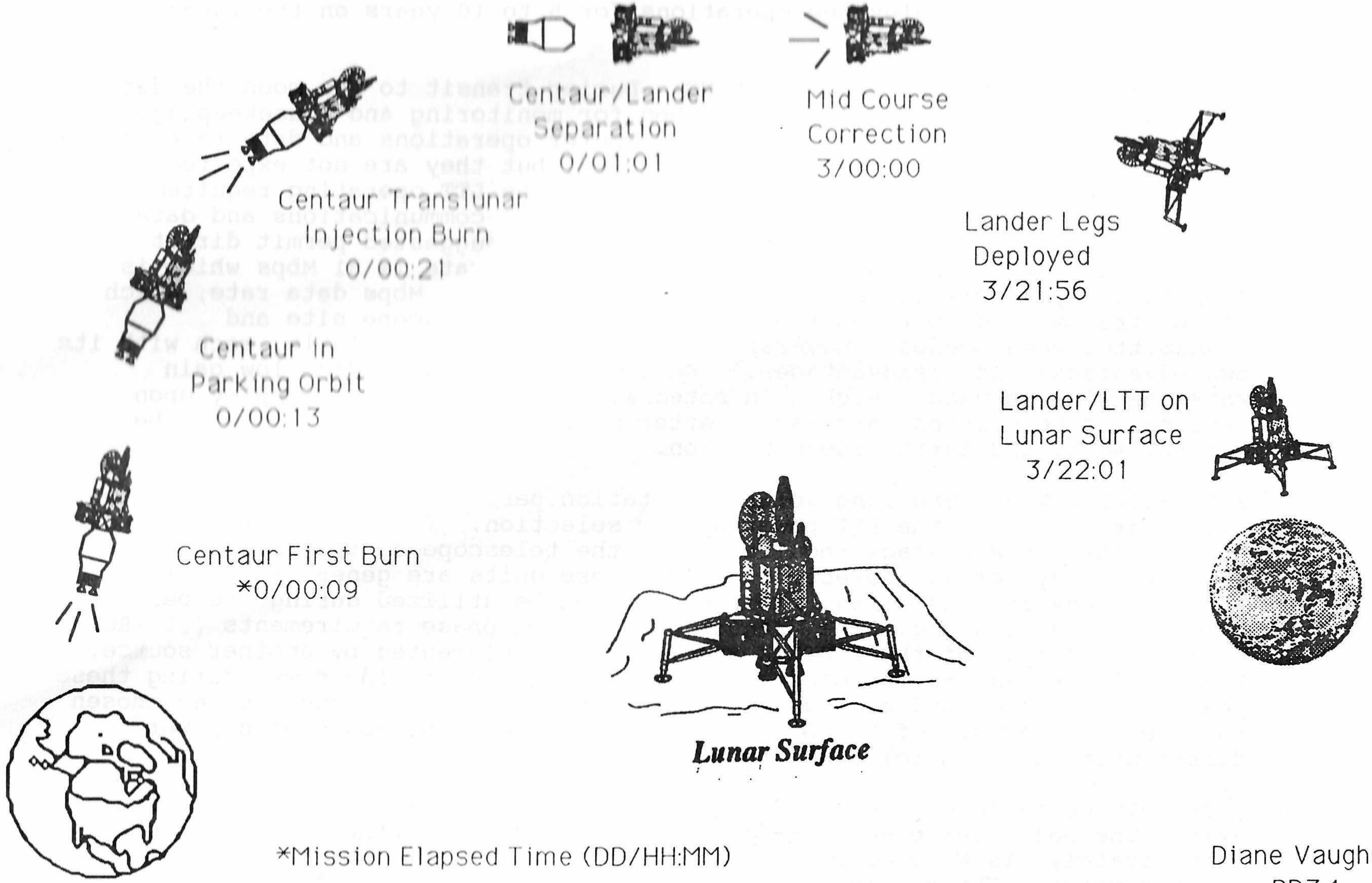
Since the Titan IV/Centaur is expected to be operational in the time frame for the LTT launch and its performance is greater than the other candidates, hence permitting a larger telescope diameter, it was selected for the LTT. Therefore the launch elements involved are the Titan IV for Earth to near Earth orbit, the Centaur for injection from a low earth orbit to translunar injection and the lunar lander for mid-course correction and lunar landing.

The solid rocket motors will separate at burnout. At approximately 4 minutes MET (mission elapsed time), the shroud will be released. The first stage separation will follow shortly afterwards followed by second stage separation at 9 minutes MET. After separation of the expended second stage of the Titan IV, the first Centaur burn places the LTT in a short duration parking orbit. After the translunar insertion burn the Centaur is separated before midcourse.

The lunar lander makes the midcourse correction ; activates its landing sensors and controls; and maneuvers for proper orientation of the LTT antenna, sunshade, and observing instrumetation. The system makes it final braking burn for the soft landing, simultaneously employing main and RCS thrusters to make final adjustments for telescope position and orientation. Soft touchdown occurs 3 days and 22 hours post-launch.

After lunar landing the propulsion and RCS elements are deactivated; the landing system begins its service as a telescope mount; and the activation of operational instrument subsystems begins. During the follow three hours the sun is acquired; the high-gain antennas are deployed; the Command and Data Handling Subsystem (C&DHS) is activated; the LTT sunshade is adjusted to final position; the aperture cover is removed; and focal plane instrument testing and adjustment are initiated. Over the next nineteen Earth-days the LTT optics are focused; the status of the supporting sensors is evaluated; and the acquisition of science data is verified. Some three Earth-months later, about three Lunar days, the LTT will begin science operations.

Lunar Transit Telescope (LTT) Transportation



*Mission Elapsed Time (DD/HH:MM)

Diane Vaughan
PD34

SUPPORT SYSTEM REQUIREMENTS

Early in the LTT study it was decided that one set of subsystems could be designed to serve the LTT during transit to the moon, while soft landing on the moon and during telescope operations for 5 to 10 years on the lunar surface.

COMMUNICATIONA and DATA HANDLING: During transit to the moon the data rates are expected to be low, a few kbps for monitoring and housekeeping. During actual landing on the moon the computer operations and data rates could be rather high depending on the systems used, but they are not expected to exceed the telescope operating requirements. The LTT operating requirements after landing on the moon drive the design of the communications and data handling system. Fortunately, most of the sites suggested permit direct communication with the Earth. A typical raw data rate is 31 Mbps which is compressed by a 10:1 ratio. This translates to a 3.1 Mbps data rate, which can be transmitted continuously or stored at the telescope site and transmitted when needed. Several frequency bands are available, each with its own advantages and disadvantages. During transit and landing low gain antennae will be used. High gain antennae will automatically deploy upon landing and be pointed earthward. After deployment the telescope will be controlled by the Earth Ground Station.

ELECTRICAL POWER: The long solar occultation period of the lunar night was the major driver in the LTT power system selection. To minimize weight and provide the 600 W average requirement of the telescope on the lunar surface an RTG powered system was selected. Since these units are generating power from the time they are installed, their output can be utilized during the period prior to lunar landing. The transit and landing phase requirements (650-800W) exceed the output of the 2 RTG's and must be supplemented by another source. Primary batteries are included in the EPS design to provide power during these phases when the demand exceeds the RTG's capability. The power system chosen for the LTT consists of 2 RTG's, primary batteries, and associated power distribution and control distributors.

The RTG contains eighteen 250 W (thermal) heat sources along its longitudal axis. The heat source generates approximately 4,500 W (thermal) of which approximately 315 W (electeic) is produced by the thermolectric unit around the perimeter. The remaining heat is radiated by the fins at a rejection temperature of 580 K.

SUPPORT SYSTEM REQUIREMENTS

* INSTRUMENTATION
  - FOUR MIRROR TELESCOPE WITH A 2 M DIAMETER PRIMARY
  - MOSAIC ARRAY OF 1,200 CCD'S AS TELESCOPE DETECTOR
  - INSTRUMENTS FOR MONITORING THE LUNAR ENVIRONMENT
  - INSTRUMENTATION FOR MONITORING THE TELESCOPE FUNCTIONS

* SUBSYSTEMS DESIGNED FOR THREE MISSION PHASES
  - SUPPORT TRANSLUNAR FLIGHT: MONITOR FUNCTIONS AND MIDCOURSE CORRECTION
  - TAKE OUT TERMINAL VELOCITY/LAND SOFTLY ON THE MOON
  - SUPPORT LTT SCIENTIFIC OPERATIONS OVER A 10 YEAR LIFE

* PROPULSION AND RCS SYSTEMS
  - THROTTLEABLE MAIN ENGINE (16 TO 4 KLBF)
  - CRYOGENIC MAIN ENGINE FOR HIGH Isp
  - HYDRAZINE REACTION CONTROL SYSTEM (TWELVE 25 LBF ENGINES)

* GUIDANCE, NAVIGATION AND CONTROL ISSUES
  - KNOWLEDGE OF TERRAIN, LUNAR LOCATION AND SLOPE TOLERANCES
  - GN&C ACCURACY, LANDING DYNAMICS AND ORIENTATION AT LANDING
  - CONTROL OF TELESCOPE LINE-OF-SIGHT AND E-W DETECTOR ALIGNMENT

* COMMUNICATIONS AND DATA HANDLING
  - UTILIZE DSN OR DEDICATED NETWORK
  - AVERAGE DATA RATE OF 31 MBPS ASSUMED BEFORE COMPRESSION
  - CONTROL AND OPERATION FROM GROUND STATION

* ELECTRICAL POWER SYSTEM
  - THREE PHASES: TRANSIT, LANDING AND GROUND OPERATIONS
  - 600 (OPS) TO 800 (LANDING) W POWER REQUIRED

ALIGNMENT REQUIREMENTS

The need for an alignment/pointing system was established late in the preliminary design phase. Subsequently several concepts were identified and briefly evaluated.

Various parts of the LTT have definite orientation and pointing accuracy requirements. The high gain antennae must point toward the Earth with a 1 degree accuracy, the telescope sunshade should be oriented toward the lunar equator (Southward for Northern latitude sites) with 1 degree accuracy, and the detector array should be aligned East-West with an accuracy in the second range. Ideally, the telescope line-of-sight should be aligned with local zenith. There has been considerable discussion about the effects of off zenith pointing, either in the meridian (North-South) or out of the meridian (East-West). The facing chart shows typical initial conditions that are expected just prior to landing, the slope tolerance that should be built into LTT, the alignment requirements and LTT needs. These are soft and subject to much debate and change as more assessments are made.

It was determined that when viewed from mid-latitudes on the moon the stars make a curved track on the science instrument detector at the focal plane of the LTT. If the CCD array, and possibly the pixels of individual CCDs are customized for a specific latitude, then the rows of pixels will be curved for the star track at that latitude. If because of GN&C errors, landing dynamics, or unknowns in the selected landing site, the LTT does not land with its telescope pointing to zenith, then its line of sight must be corrected by some type of mechanism. At approximately 40 degrees latitude, the telescope line of sight must point to zenith with an accuracy of 5 arcminutes in its meridian plane and several degrees perpendicular to the plane. Therefore, the pointing or leveling mechanisms should correct for a few degrees error and be accurate to within about 1 arcminute.

A top level assessment was made of four pointing options : a 2 DOF gimbal, screws on the lander legs, an arrangement of six linear actuators in a hexapod configuration, and linear joint actuators between each leg and the lander body. It was determined that a large range of movement would not be necessary. Because of its simplicity and low weight, leg screws are recommended for the LTT. With a four legged lander, level sensing and leveling is accomplished in two steps: 1) level across opposite diagonal legs and 2) then level across the diagonal of the two remaining legs.

ALIGNMENT REQUIREMENTS

* COMPENSATE FOR UNCERTAINTIES IN LUNAR TERAIN
  - KNOWN RESOLUTION OF LUNAR SURFACE
  - SIZE OF ROCKS AND BOULDERS
  - DEPTH OF SMALL CRATERS
  - SOFTNESS OF REGOLITH AND DEPTH OF DUST

* OVERCOME EFFECTS OF INITIAL CONDITIONS JUST PRIOR TO IMPACT
  - VERTICAL VELOCITY OF 3 M/SEC (10 FT/SEC)
  - HORIZONAL VELOCITY OF 1 M/SEC (3 FT/SEC)
  - UP TO 0.5 DEGREES ATTITUDE ERROR ON ALL AXES
  - POSITION ERROR OF 100 M (300 FT)

* OVERCOME FORCE AND MOMENTS IMPARTED TO LTT AT LUNAR IMPACT
  - DESIGN FOR SLOPE TOLERANCE UP TO 12 DEGREES
  - ROCKS AND BOULDERS UP TO 1-2 M IN DIAMETER; 0.5 M DEPTH
  - LANDING CAN IMPART ROLL ERROR TO LTT

* ALIGNMENT REQUIREMENTS
  - ORIENT THE SUNSHADE APEX SOUTHWARD
  - ALIGNE DETECTOR ARRAY E-W WITHIN 7 ARCSECOND
  - POINT THE TELESCOPE'S LINE OF SIGHT TO THE ZENITH
  - POINT HGS'S TOWARD EARTH

* LTT NEEDS
  - SUNSHADE ROLL MECHANISM FOR 1 DEG. ACCURACY, 10-15 DEG. RANGE
  - DETECTOR ALIGNMENT MECHANISMS FOR 1 ARC SEC ACURACY, 10-15 DEG RANGE
  - LINE-OF-SIGHT CORRECTION 1 MIN. ACCURACY, 12 DEG. RANGE
  - HGA GIMBALS FOR 1 DEG ACCURACY, 15 DEG RANGE

SUGGESTIONS FOR FUTURE WORK

The LTT study team has conducted a top level feasibility study for placing a 2 m transit telescope on the moon. The tasks included assessments of existing transportation systems, top level assessments of options for the telescope with its lunar lander, and the definition of a reference LTT. However, in all engineering and programmatic areas there were tasks identified that need additional study and assessments, some of these are listed on the facing page.

Although this study approaches being a Phase A rather than a feasibility study, the study team believes that additional definition, simulations and analysis are needed. There are options to be excercised, design concepts and structural analysis to be made, mechanisms to be defined, simulations to be programmed and evaluated, and requirements to be reassessed. In concert with our schedule, it is recommended that a Phase A study be initiated and conducted during CY 1992 to address the topics suggested for future work.

SUGGESTIONS FOR FUTURE WORK

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  - DESIGN CONCEPTS AND STRUCTURAL ANALYSIS
  - PRIMARY MIRROR MOUNTING OPTIONS
  - LANDER TO TELESCOPE INTERFACE STRUCTURE
  - DEFINITION OF RL-10 THRUST STRUCTURE AND FEED LINES
  - CLOSURES FOR SUNSHADE
  - TRADES OF SMALLER OPTICAL AND LAUNCH SYSTEMS

- DETAILS OF OPTICAL SYSTEM
  - DETECTOR DESIGN AND ALIGNMENT MECHANISMS
  - PRIMARY MIRROR MATERIALS AND DESIGN OPTIONS
  - MECHANISMS FOR ADJUSTABLE SECONDARY
  - ADAPTIVE/DEFORMABLE TERTIARY

- ASSESSMENTS
  - OPTIONS FOR LOS POINTING SYSTEMS
  - SEPARABLE LANDER AND TELESCOPE DESIGNS
  - MECHANISMS (LEG SCREWS, SUNSHADE, DETECTOR ROTATION, LOS POINTING)
  - THROTTLEABILITY OF RL10 ENGINE

- SIMULATIONS AND EVALUATION
  - LANDING DYNAMICS/SOFT LANDING SIMULATION
  - STRUCTURAL INTEGRITY AND LOADS
  - MORE DETAILED THERMAL MODEL
  - CONTROL, STRUCTURAL, THERMAL INTERACTION OF OPTICAL SYSTEMS

- REQUIREMENTS UPDATES
  - LAUNCH VEHICLE
  - SUBSYSTEMS SUPPORT
  - LANDER PROPULSION SYSTEMS
  - TELESCOPE ENVIRONMENT

12/27/91:BGD

RECENT REFERENCE TELESCOPE REPORTS
BY MSFC'S PROGRAM DEVELOPMENT OFFICE

1. LARGE LUNAR TELESCOPE (LLT), 16-M APERTURE UV/VIS/IR TELESCOPE, LLT-OO1, MARCH 1991

2. LUNAR CLUSTER TELESCOPE EXPERIMENT (CTE), 4-M TECHNOLOGY PRECURSOR TO LLT, LLT-002, MARCH 1991

3. HIGH EARTH ORBIT TELESCOPE (HEOT), 6-M UV/VIS/IR TELESCOPE, LLT-003, MARCH 1991

4. LUNAR TRANSIT TELESCOPE (LTT), 2-M APERTURE UV/VIS/IR TELESCOPE, LLT-004, JANUARY 1992