FINAL REPORT
ON PHASE I OF

A STUDY OF THE LUNAR ORBITER REGARDING ITS ADAPTABILITY TO EXPERIMENTS TO EVALUATE LUNAR SURFACE CHARACTERISTICS

L-6753

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FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
LANGLEY RESEARCH CENTER

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FOREWORD

This document is the final report for Phase I of Contract NAS1-6216. Its submission to NASA is required by Part IIIB of contract schedule dated May 3, 1966, as modified by letter from the Contracting Officer August 4, 1966 as follows:

"Volume I - ELF, Spherical Penetrometers, Low Density Impacter

"Volume II - Surface Experiments Utilizing a Fly-by and Earth Return Trajectory"

"Each Volume shall Contain"

"(a) Summary and conclusions section in the front of the document,

"(b) Graphs and figures accompanied by suitable descriptive information, and

"(c) An appendix containing pertinent technical working papers."

SUMMARY AND CONCLUSIONS

Payload Constraints

To carry additional experiments on the Lunar Orbiter requires assessment of the launch vehicle and spacecraft propulsion capability. The present Atlas/Agena can inject a 1000-pound spacecraft on a 90-hour translunar trajectory during any month of the year. Therefore, 1000 pounds was used for the weight limit for the penetrometer experiment studies. The propulsion capability of the spacecraft imposes a more restrictive limit on spacecraft weight which can be injected into the nominal Lunar Orbiter orbit. By accepting some restriction in launch opportunity, e.g., no launch in July or November and photography on either ascending or descending node, then a 950-pound upper limit on spacecraft weight may be used. The 950-pound limit applied to the flare and balloon experiments of the study.

The stripped weight of a Lunar Orbiter spacecraft for the flare and balloon experiments was 840 pounds. Since the camera could also be deleted for the penetrometer experiments, the stripped spacecraft would weigh about 686 pounds. The total weight allocation which could be made to the experiments was: 110 pounds for flare or balloon and about 314 pounds for the penetrometer.

Flare Experiment

Studies were conducted in a parametric manner so as to maximize the number of flares which could be carried.

The velocity increment required to deploy the flare to the lunar surface is greatest at the desired high impact angle. The minimum specified angle of 45° was chosen so as to permit carrying the largest number of flares. Also, the deployment velocity increment required increases with the desired high impact velocity, therefore the minimum specified 6000 ft/second condition was selected.

Studies of the instrumentation requirements were made to help size the flares. The features of the experiment sensors are as follows:

(1) Time to first peak detector. This instrument is simply a photomultiplier tube with appropriate optics selected, a high voltage power supply and a high response (220 KHz bandwidth) data transmission capability. Since the existing Lunar Orbiter video communication system has a band pass of 230 KHz, it was decided to use it to transmit the flash intensity in real time. To satisfy the desire for fidelity with high signal to noise ratio, a 5.08 cm, f/1 lens was selected. The field of view of this lens was 20°.

(2) Spectral data. The Lunar Orbiter camera could be modified to record the spectral data by changing the shutter to allow it to remain open two or three seconds, long enough to assure recording the event, and by the addition of a diffraction grating in front of the lens. Requirements for the size of flare were developed by consideration of the film sensitivity, of the light attenuation, of the lens and quartz grating, and of the resolution of spectral lines desired. Consideration of all these factors sets requirements on flare size and limits the range from which the flare can be viewed. The range of resolution considered was from 0.3 to 3.0 angstroms.

The rocket motor size to achieve the impact condition desired is related to both the velocity increment ($\triangle V$) and the mass to be delivered. With the objective to carry the greatest number of flares per mission, trade data were developed to permit optimizing the experiment within the total weight allocated to the flare packages.

Because of dispersions in the trajectory and time of the flare impact and the limited field of view of the camera in the downrange direction, there is less than a 3-sigma probability of recording the event at a range below 60 km. This might be resolved by turning the spacecraft 90° to the line of flight to provide the larger field of view in the same direction as the larger dispersions or by selecting a longer range condition or possibly by reducing dispersions.

(3) Temperature detector. The instrument required to record this data is very similar to the photomultiplier tube used for the "time to first peak" detector except that a filter is placed in front of the lens and the bandpass required is only 5 KHz. Since the spectrum between 3500 and 6500 angstroms is of interest, six filtered phototubes each covering 300 angstroms of the spectrum are required. This data can all be recorded on a small tape recorder (such as the Raymond Engineering 3-pound unit) and read out at a later time. It should be noted that this array of instrumentation accounts for about 20 pounds of added weight on the spacecraft.

Selection of a typical flare experiment.- Although all the problems were not solved in detail, it was shown that the Lunar Orbiter could be modified to carry three flares and record data related to chemical composition, hardness, and depth of the layer of surface material. The spacecraft would weigh 947 pounds at launch. Each flare would weigh 6.8 pounds and the delivery rocket would weigh an additional 12 pounds each. The impact conditions would be 45° and 6000 ft/second. The flare would be delivered from a nominal 46 km perilune-1850 km apolune orbit, but delivery would occur to the dark surface with impact 10° before perilune. The launcher tube would be located on the spacecraft with centerline parallel to the camera optical axis.

Balloon Experiment

The delivery of the balloon to the lunar surface from orbit is similar to delivery of the flare with the notable exception that the point of impact must be such as will permit high resolution photographs to be taken. Thus, impact must be near perilune and the orbit must be selected so as to provide oblique lighting on the imprint area. Since no additional sensors are required on the Orbiter, the weight which can be allocated to balloons and their delivery rockets is 90 pounds. With 20 pounds allocated to the launcher tube on the spacecraft, a total of four balloons could be carried. Each balloon weighed eight pounds with two pounds for the container and 11.9 pounds for the delivery rocket. Each balloon would be deployed separately so as to impact at an angle of 30° to the surface and at 6000 ft/second. The time of flight would be short, only 51 seconds. It would be possible to obtain only one high resolution picture of the impact area per photo pass. Pictures of the area before impact would need to be taken on an earlier pass.

Spherical Penetrometers

Two different spin stabilized delivery concepts were studied but rejected because variation of the impact velocity was too great. A third delivery system appeared to satisfy most of the impact requirements; it was the Space General "Lunar Surface Probe" concept developed under Contract NAS9-4897. A scaled down delivery system was defined which allowed a variable number of penetrometers to be carried as the payload. The objective was to deliver as many penetrometers to the surface as possible within the weight limitation of 1000 pounds for the total spacecraft.

Three different communication links were examined for their influence on maximizing the number of penetrometers.

The first link communicated the experiment data from the penetrometer to Lunar Orbiter to Earth. The key problem with this link was to keep the range within the capability of the 1/2 watt transmitter in the penetrometer. The controlled descent for the penetrometers took 110 seconds nominally. Time dispersions resulting from the controlled drop sequence were determined to be ± 13 seconds. Since the orbital velocity of the spacecraft was approximately 2 km per second, this dictated a range of at least ± 26 km. A 13 db gain link margin could be maintained over this range only by tracking the predicted impact point with a high gain directional antenna mounted on the spacecraft. This penalty of a tracking antenna was avoided in the two other links included in the study.

The second link transmitted data from penetrometer to delivery vehicle to Lunar Orbiter to Earth. This raised the communication range limit between delivery vehicle and Lunar Orbiter to 185 km which could be met readily. It required added communication equipment on the delivery vehicle and was included in the scaled weight model.

The third link eliminated the Lunar Orbiter and was from penetrometer to delivery vehicle to Earth.

Delivery orbits were studied in order to optimize orbital parameters on the basis of most penetrometers delivered. The best sequence was: 1) establish an initial orbit with perilune at 200 km and with apolune from 2000 to 3600 km, 2) establish a holding orbit with perilune at 5 to 20 km, and with apolune coincident with initial orbit, 3) stage the delivery vehicle from the spacecraft approximately at apolune of the holding orbit with a small $\triangle V$ increment sufficient to achieve a perilune of 4.6 km minimum altitude, and 4) thrust the Lunar Orbiter into a higher phasing orbit with perilune at 47 to 53 km which would allow the time lag of 110 seconds and put the Orbiter overhead at time of penetrometer impact.

This technique will permit delivery of three penetrometers using the Space General delivery vehicle and communication link number two. The spacecraft bus weight would be approximately 700 pounds, the delivery vehicle would weigh 300 pounds, giving a total launch weight of 1000 pounds.

STUDY GUIDELINES

- LUNAR ORBITER BASIC CONFIGURATION, NAS 1-3800
- DOSIMETER AND MICROMETEOROID DETECTORS MAY BE REMOVED
- SAME LAUNCH VEHICLE
- PRESENT SHROUD, IF POSSIBLE
- DSIF TRACKING ACCURACIES:

  R OF ± 5 M, $\dot{R}$ OF $\pm .001$ M/SEC., $\measuredangle$ OF $\pm 0.01°$

- CLARKE GRAVITATIONAL MODEL
- UNCERTAINTY BETWEEN ACIC REFERENCE DATUM AND TRAJECTORY MODELS WILL BE REDUCED TO $\pm$ 600 METERS.

SURFACE HARDNESS AND COMPOSITION

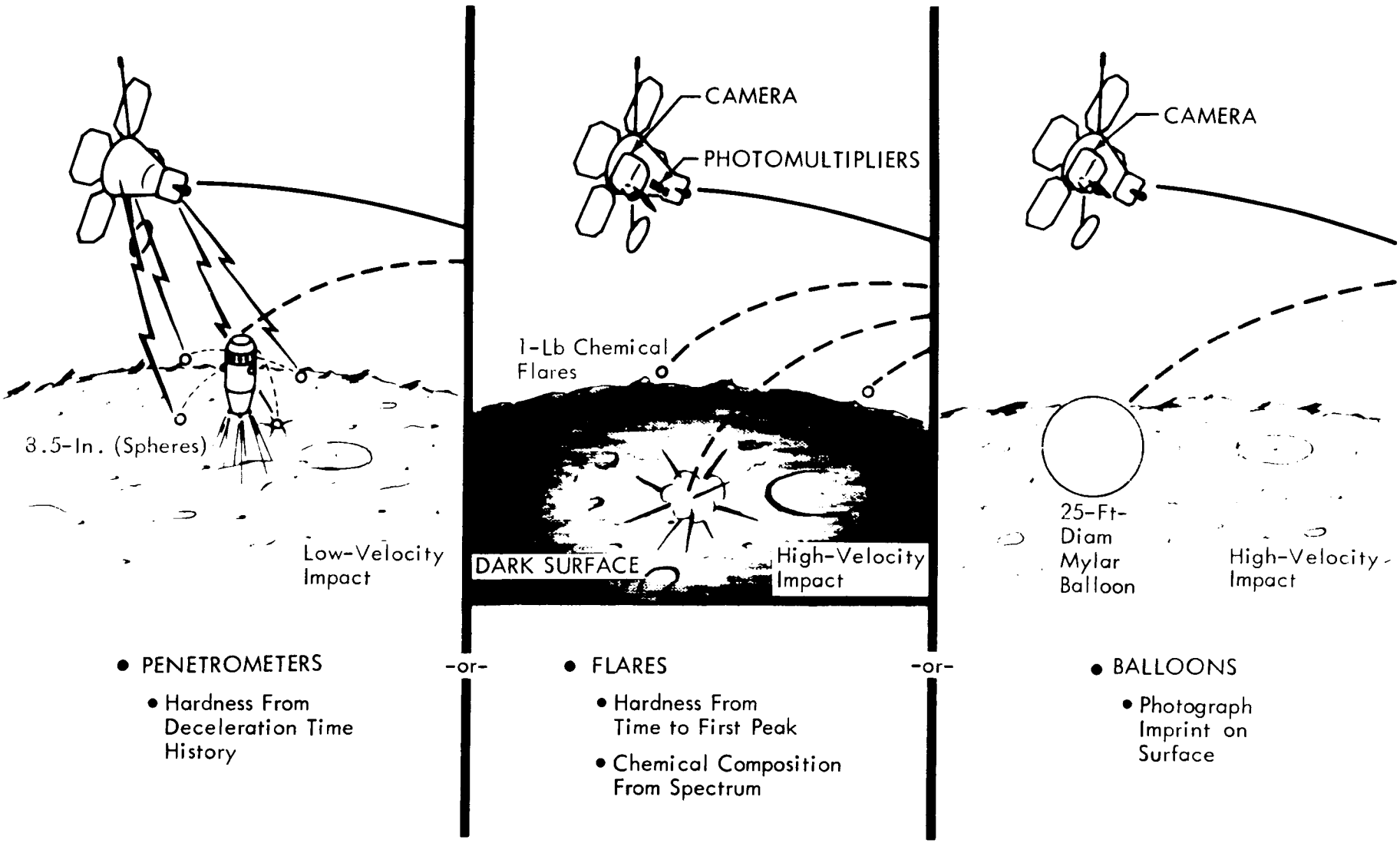
A RESEARCH STUDY OF THE LUNAR ORBITER SPACECRAFT REGARDING ITS ADAPTABILITY TO EXPERIMENTS FOR EVALUATING LUNAR-SURFACE CHARACTERISTICS WAS CONDUCTED BY THE BOEING COMPANY UNDER CONTRACT NAS1-6216. THE EXPERIMENTS WERE SELECTED BY THE NASA TO PROVIDE SOME MEASURE OF SURFACE BEARING STRENGTH. IN THIS APPLICATION, THE LUNAR ORBITER SERVES AS A BUS THAT CAN CARRY ADDITIONAL "PIGGY-BACK" EXPERIMENTS: IN THE CASE OF PENETROMETERS, THE NEW EXPERIMENT WOULD REPLACE THE PHOTOGRAPHIC SUBSYSTEM.

FLARE.- THE FLARE CONSISTS OF A MIXTURE OF CHLORINE TRIFLUORIDE AND FINE ALUMINUM WIRE. UPON IMPACT WITH THE SURFACE, A HIGH-SPEED CHEMICAL REACTION TAKES PLACE, RESULTING IN A "FLASH." STUDIES CONDUCTED BY DOUGLAS AIRCRAFT MSSD UNDER CONTRACT NAS7-327 INDICATED A CORRELATION BETWEEN RISE TIME AND TEMPERATURE OF FIRST PEAK WITH TARGET BEARING STRENGTH AND POROUS-LAYER DEPTH, RESPECTIVELY. SPECTRAL IRRADIANCE WITH RESPECT TO TIME IS MEASURED WITH A SERIES OF PHOTOTUBES AND NARROW BANDPASS FILTERS. RELATIVE CHEMICAL ABUNDANCES CAN BE DETERMINED BY RECORDING EMISSION LINE AND BAND STRUCTURE. THE SPECTRA CAN BE OBTAINED BY PLACING A DIFFRACTION GRATING IN FRONT OF THE CAMERA LENS.

BALLOON.- THE EXPERIMENT POSTULATES A SIGNATURE ON THE LUNAR SURFACE, EITHER BY AN IMPRESSION OR BY ALBEDO CHANGE CAUSED BY THE HIGH-VELOCITY IMPACT OF A 25-FOOT-DIAMETER BALLOON. THIS DEVICE MIGHT WEIGH 8 POUNDS AND WOULD USE TECHNOLOGY DEVELOPED FOR THE ECHO BALLOON. IT WOULD BE DEPLOYED FROM THE LUNAR ORBITER TO THE SURFACE BY A SOLID ROCKET MOTOR AND INFLATED ON THE WAY DOWN. PHOTOGRAPHS OF THE AREA BEFORE AND AFTER IMPACT PROVIDE A MEANS OF ASSESSING SURFACE CHARACTERISTICS.

PENETROMETERS.- THE DECELERATION TIME HISTORY OF A PENETROMETER IMPACTING ON THE LUNAR SURFACE OFFERS THE MOST READILY INTERPRETED DATA ON SURFACE HARDNESS. AERONUTRONIC DIVISION OF PHILCO HAS DEVELOPED A 5-POUND IMPACTOR UNDER CONTRACT TO NASA-LRC. THE PENETROMETERS ARE DELIVERED TO THE SURFACE BY AN INTERMEDIATE DELIVERY VEHICLE, WHICH ALLOWS THE PENETROMETERS TO FREE-FALL FROM A FIXED HEIGHT OVER THE LUNAR SURFACE. THE NOMINAL IMPACT VELOCITY OF THE PENETROMETER IS 150 FEET PER SECOND. THE DEVICE IS 8.5 INCHES IN DIAMETER AND HAS AN EXTERIOR BALSA COVER THAT ENCLOSES THE ACCELEROMETERS, TRANSMITTERS, AND BATTERY.

SURFACE HARDNESS AND COMPOSITION



- PENETROMETERS
  - Hardness From Deceleration Time History

-or-

- FLARES
  - Hardness From Time to First Peak
  - Chemical Composition From Spectrum

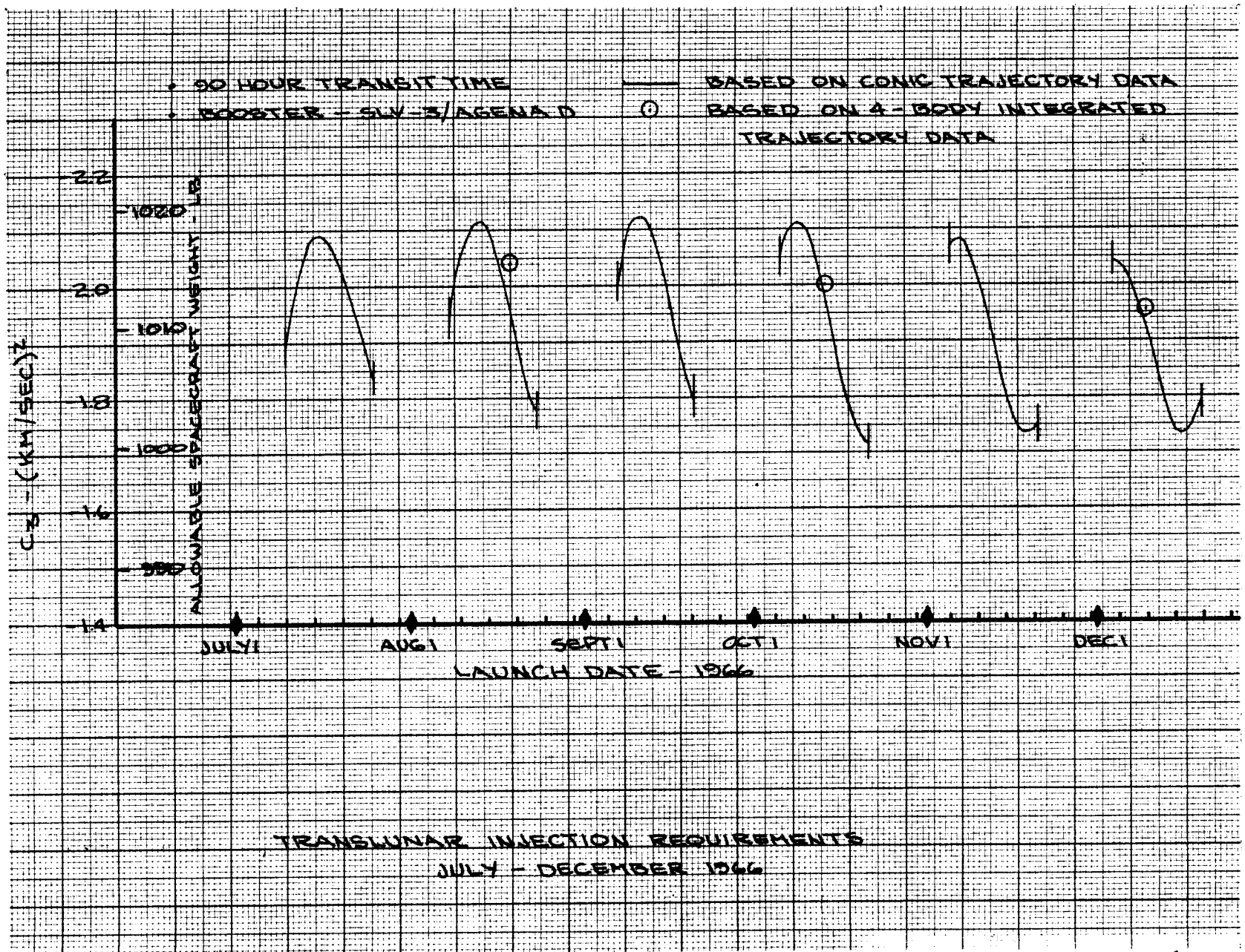
-or-

- BALLOONS
  - Photograph Imprint on Surface

TRANSLUNAR INJECTION REQUIREMENTS
JULY-DECEMBER 1966

TRANSLUNAR INJECTION REQUIREMENTS FOR LAUNCH IN THE SECOND HALF OF 1966 ARE PRESENTED ON THIS CHART. DATA ARE SHOWN ONLY FOR A 90-HOUR TRANSIT TIME. THESE CASES REPRESENT NEAR MINIMUM ENERGY TRANSITS. THE REQUIREMENTS ARE ILLUSTRATED WITH THE PARAMETER C_3 WHICH REPRESENTS TWICE THE TOTAL INJECTION ENERGY. THE LARGER NEGATIVE C_3 LEVELS CORRESPOND TO LOWER ENERGY LEVELS. THE ENERGY REQUIREMENTS WERE DETERMINED BY CONIC TRAJECTORY CALCULATIONS. SEVERAL SPECIFIC TRAJECTORIES, BASED ON 4-BODY INTEGRATED TRAJECTORY CALCULATIONS ILLUSTRATE THE VALIDITY OF THE CONIC DATA. THE C_3 VARIATION WITH LAUNCH DATE IS DUE TO THE MONTHLY VARIATION IN EARTH-MOON DISTANCE.

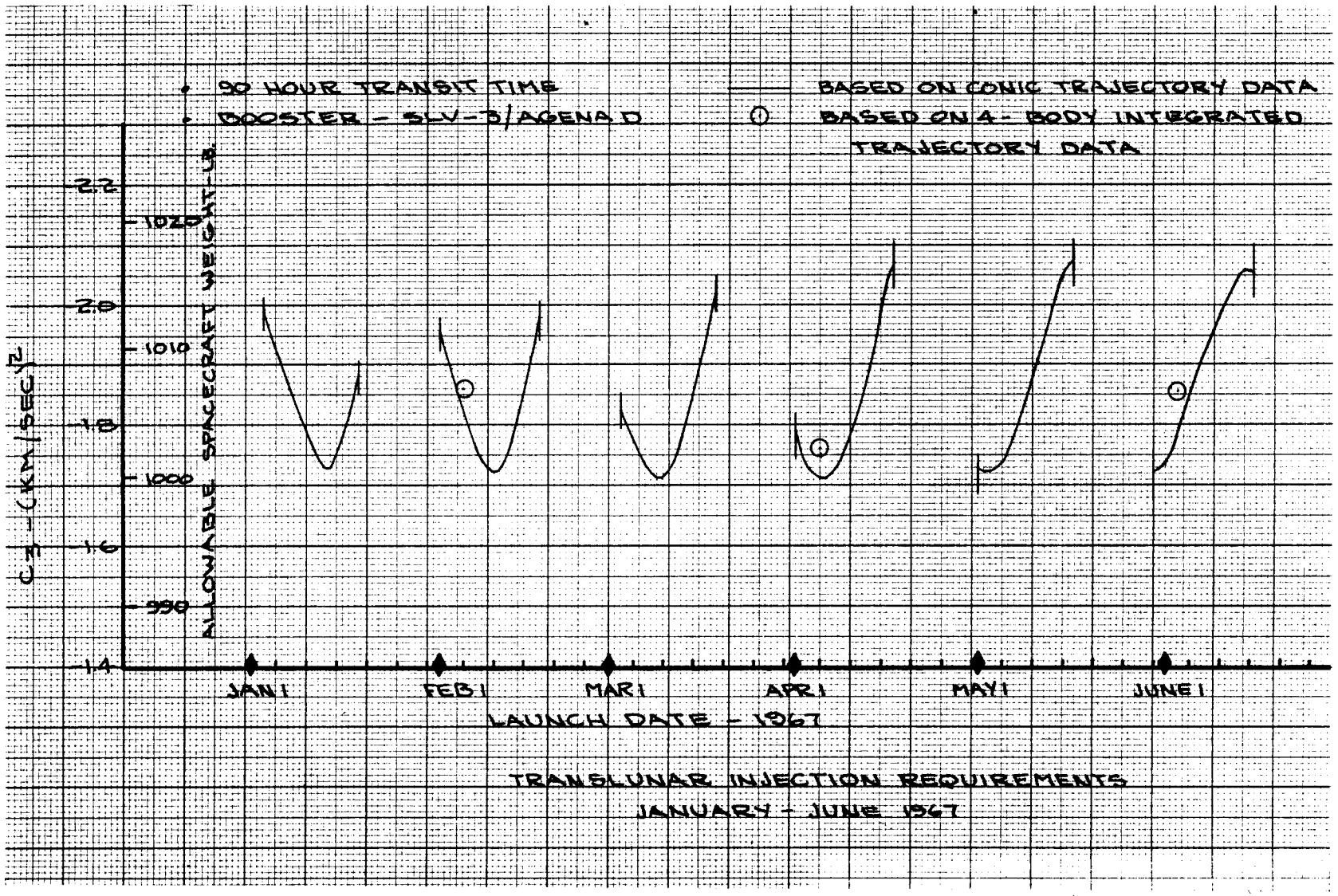
THE ALLOWABLE SPACECRAFT WEIGHTS ASSOCIATED WITH THE REQUIRED TRANSLUNAR ENERGY ARE INCLUDED ON THE C_3 AXIS. THESE WEIGHTS ARE BASED ON THE USE OF THE SLV-3 (ATLAS)/AGENA D BOOSTER. THIS IS THE BOOSTER PRESENTLY USED ON THE LUNAR ORBITER PROGRAM. NOTE THAT A SPACECRAFT WEIGHING 1000 POUNDS CAN BE PLACED ON THE DESIRED TRANSLUNAR TRAJECTORY ANY TIME IN THE SIX MONTH PERIOD SHOWN. HOWEVER, IF THE SPACECRAFT GROWS ONLY 20 POUNDS, NO ACCEPTABLE LAUNCH DATES ARE AVAILABLE.



TRANSLUNAR INJECTION REQUIREMENTS
JULY – DECEMBER 1966

TRANSLUNAR INJECTION REQUIREMENTS
JANUARY-JUNE 1967

THESE DATA EXTEND THE INFORMATION SHOWN ON THE PREVIOUS CHART TO JUNE 1967. THE MAXIMUM SPACECRAFT WEIGHT ALLOWING LAUNCH AT ANY TIME IS STILL 1000 POUNDS. EXTENSION OF THIS CHART TO LATER LAUNCH DATES WILL RESULT IN NO CHANGE IN THE MAXIMUM ALLOWABLE SPACECRAFT WEIGHT LEVELS.

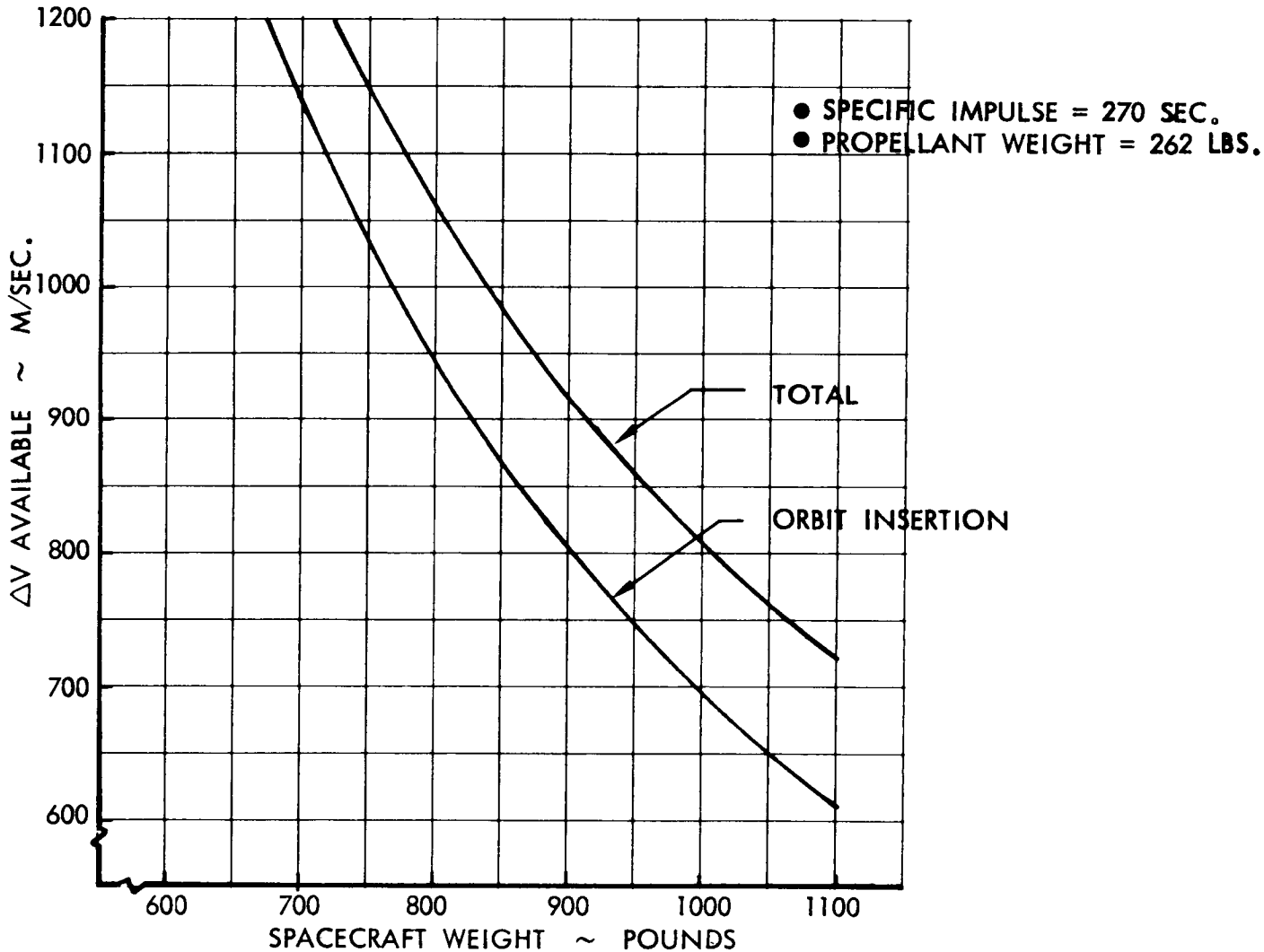


TRANSLUNAR INJECTION REQUIREMENTS
JANUARY – JUNE 1967

THREE-SIGMA MINIMUM LUNAR ORBITER VELOCITY CAPABILITY

THIS CHART ILLUSTRATES THE CAPABILITY OF THE LUNAR ORBITER VELOCITY CONTROL SYSTEM. THE DATA ARE BASED ON A SPECIFIC IMPULSE OF 270 SECONDS AND A USEABLE PROPELLANT WEIGHT OF 262 POUNDS. THESE VALUES REPRESENT THE THREE-SIGMA MINIMUM LEVEL FOR BOTH VARIABLES. CONSEQUENTLY, THE VELOCITY INCREMENT SHOWN IS CONSERVATIVE. THE DIFFERENCE BETWEEN THE TOTAL VELOCITY POTENTIAL AND THE AMOUNT AVAILABLE FOR ORBIT INSERTION IS 110 METERS/SECOND. THIS REPRESENTS THE ALLOWANCE REQUIRED FOR MIDCOURSE CORRECTIONS, ORBIT INJECTION ERRORS AND FINITE BURNING LOSSES.

3 σ MINIMUM LUNAR ORBITER VELOCITY CAPABILITY

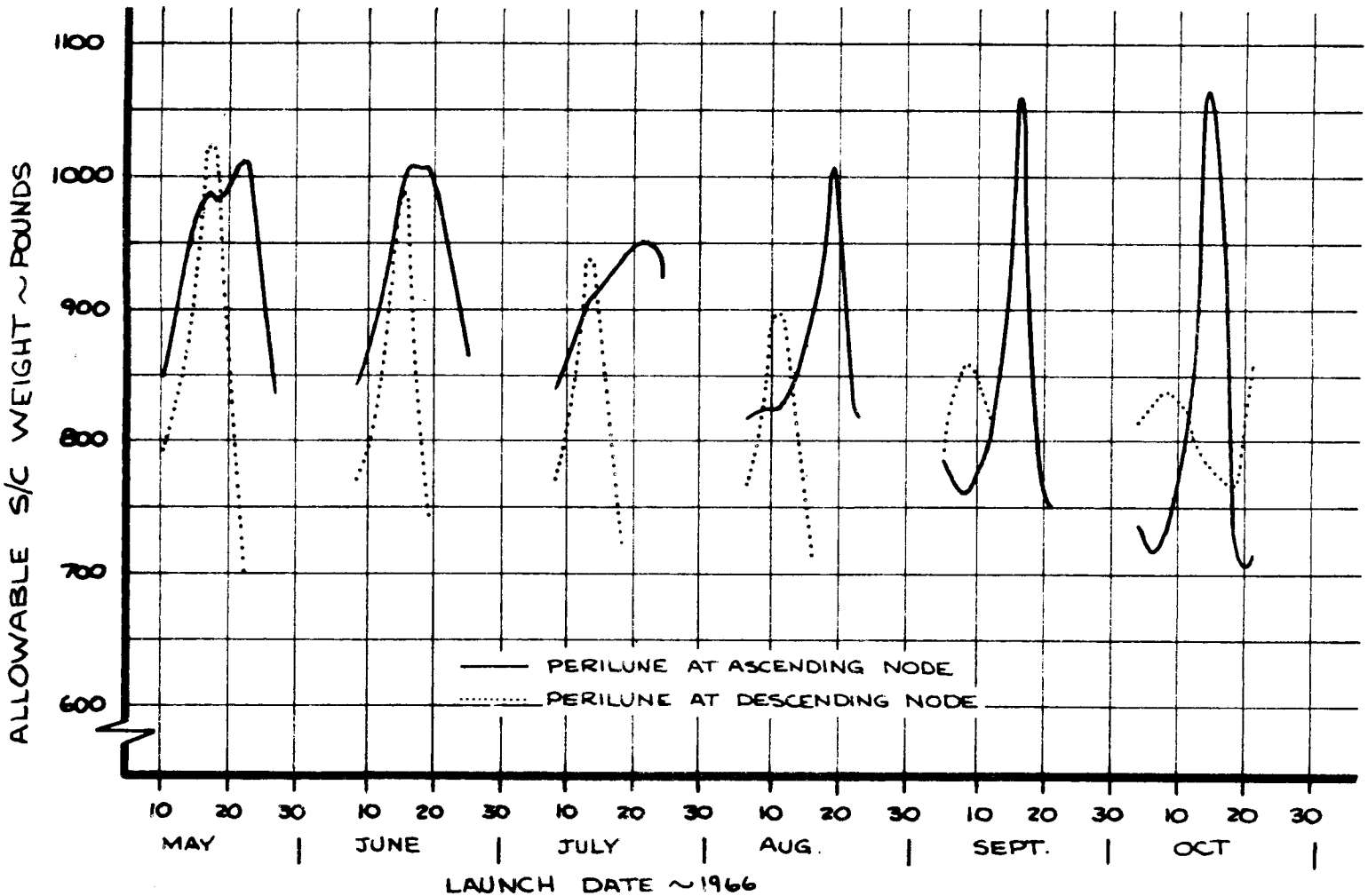


LAUNCH DATE EFFECTS ON SPACECRAFT WEIGHT, MAY THROUGH OCTOBER 1966

THESE DATA ARE BASED ON THE MAXIMUM INSERTION VELOCITIES REQUIRED FOR LUNAR ORBIT INJECTION AND TRANSFER ON THE SPECIFIED LAUNCH DAY. THE VELOCITY REQUIREMENTS WERE CONVERTED TO ALLOWABLE SPACECRAFT WEIGHT THROUGH THE ORBIT INSERTION VELOCITY AVAILABLE DATA SHOWN ON THE PREVIOUS CHART. THE SPACECRAFT WEIGHT DEFINED IS THE AMOUNT WHICH CAN BE INSERTED INTO THE NOMINAL LUNAR ORBITER PHOTOGRAPHIC ORBIT DEFINED ON THE CHART.

DATA ARE SHOWN FOR ESTABLISHMENT OF PERILUNE AT BOTH THE ASCENDING AND DESCENDING NODES. ONLY A FEW LAUNCH OPPORTUNITIES ARE AVAILABLE FOR SPACECRAFT WEIGHING 1000 POUNDS. HOWEVER, A MORE REALISTIC WEIGHT OF 950 POUNDS ALLOWS PHOTOGRAPHY AT EITHER THE ASCENDING OR DESCENDING NODE EVERY MONTH EXCEPT JULY AND NOVEMBER.

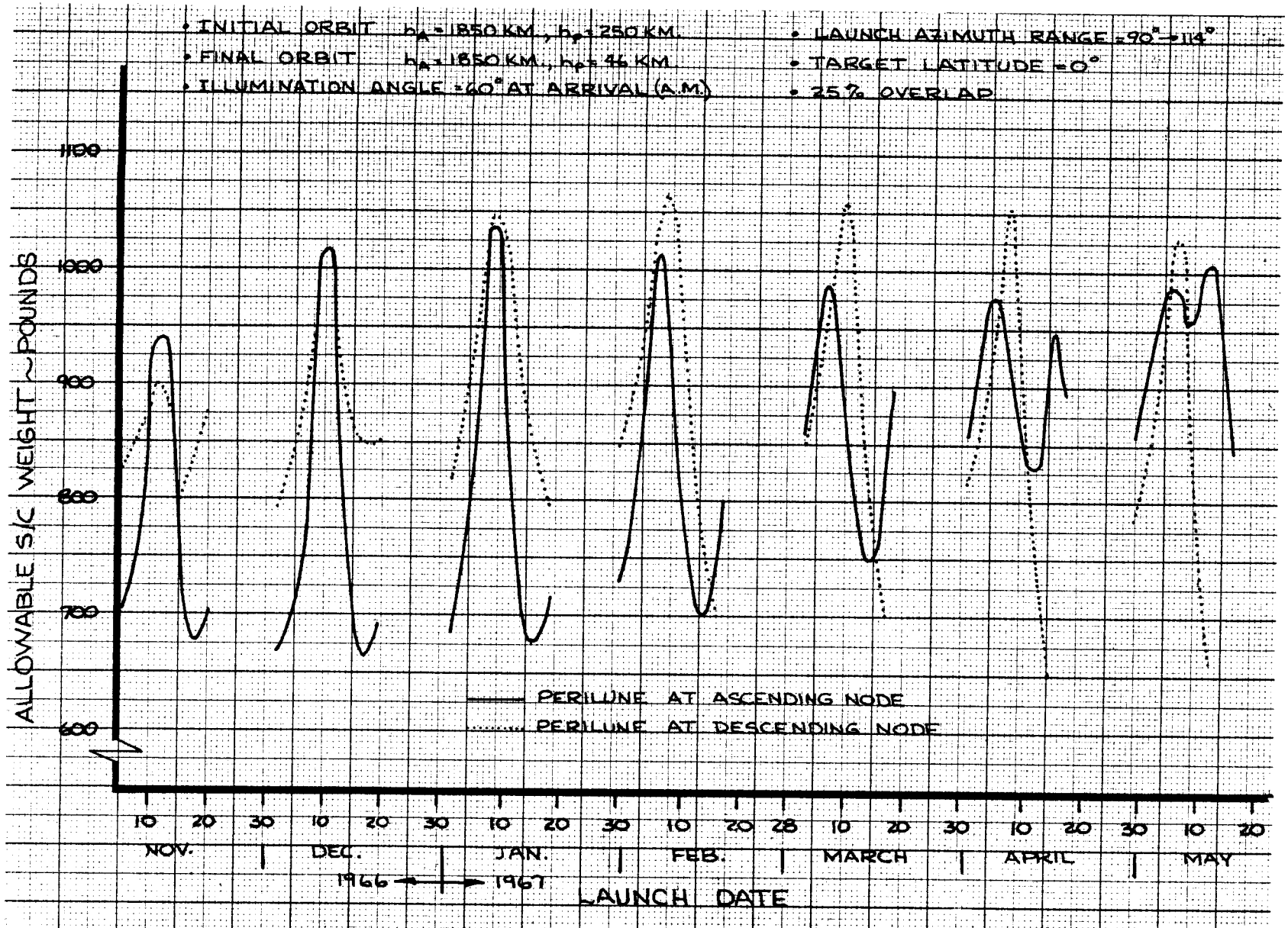
- INITIAL ORBIT $h_A = 1850$ KM, $h_p = 250$ KM
- FINAL ORBIT $h_A = 1850$ KM, $h_p = 46$ KM
- ILLUMINATION ANGLE = 60° AT ARRIVAL (AM.)
- LAUNCH AZIMUTH RANGE = 90° → 114°
- TARGET LATITUDE = 0°
- 25% OVERLAP



LAUNCH DATE EFFECTS ON SPACECRAFT WEIGHT ~ MAY THRU OCT, 1966

LAUNCH DATE EFFECTS ON SPACECRAFT WEIGHT,
NOVEMBER 1966 THROUGH MAY 1967

THIS CHART IS SIMPLY AN EXTENSION OF THE PREVIOUS CHART THROUGH MAY 1967.



LAUNCH DATE EFFECTS ON SPACECRAFT WEIGHT ~ NOV. 1966 THRU MAY 1967

LUNAR ORBITER WEIGHTS

THE SPACECRAFT AND SUBSYSTEM WEIGHTS SHOWN ARE BASED ON THE MAY 1966 WEIGHT STATEMENT. THIS LIST PROVIDES THE STARTING POINT FOR DETERMINING THE WEIGHT OF THE MODIFIED SPACECRAFT CARRYING THE EXPERIMENTS LISTED.

ALSO LISTED ON THE LOWER PORTION OF THE CHART IS A SHOPPING LIST OF ADDITIONAL POTENTIAL WEIGHT SAVING ITEMS.

LUNAR ORBITER WEIGHTS

CURRENT LUNAR ORBITER AT AGENA SEPARATION = 846 LBS.

		STRIPPED L.O. WEIGHT		
		FLARE	BALLOON	PENETROMETER
o AVAILABLE REDUCTIONS DEPENDING ON MISSION REQUIREMENTS :				
SCINTILLATION COUNTERS	- 1.32	X	X	X
MICROMETEROID DETECTORS	- 3.20	X	X	X
SCINTILLATION COUNTER SUPTS. & FASTENERS	- .57	X	X	X
ASSUME MICRO. DETEC. SUPTS. USED FOR THERMAL BARRIER SUPT.	--			
WIRING AND MISCELLANEOUS	- .91	X	X	X
PHOTOGRAPHIC SUBSYSTEM	-145.12			X
PHOTOGRAPHIC SUBSYSTEM COOLING FINS	- 1.71			X
PHOTOGRAPHIC SUBSYSTEM SUPTS.	- .79			X
CAMERA THERMAL DOOR	- 3.16			X
PHOTO SUBSYSTEM WIRING & CONN.	- 3.22			X
SUB-TOTAL OF WEIGHT REMOVED		6.0	6.0	160
TOTAL OF SPACECRAFT WEIGHT		840.0	840.0	686
o POSSIBLE ADDITIONAL CANDIDATE ITEMS :				
SOLAR PANELS INCL. SUPTS., ACT. & WIRING	- 17. LB.EA.			X
N_2 GAS FOR 11 MO. EXTENDED MISSION	- 4.29 LBS.			X
SHORT LIFE BATTERY OF REQ. VOLTAGE	- 10. LBS. (USE -4.71)			X
ELECTRICAL POWER CONTROLS = 10.17 LBS. (30%)	- 3. LBS.			
N_2 GAS FOR 30-DAY MISSION = 6.29 LBS. (30%)	- 2. LBS.			
DATA MEASUREMENT TRANSDUCERS	- 4. LBS.			
				660

EARLY LUNAR FLARE (ELF)

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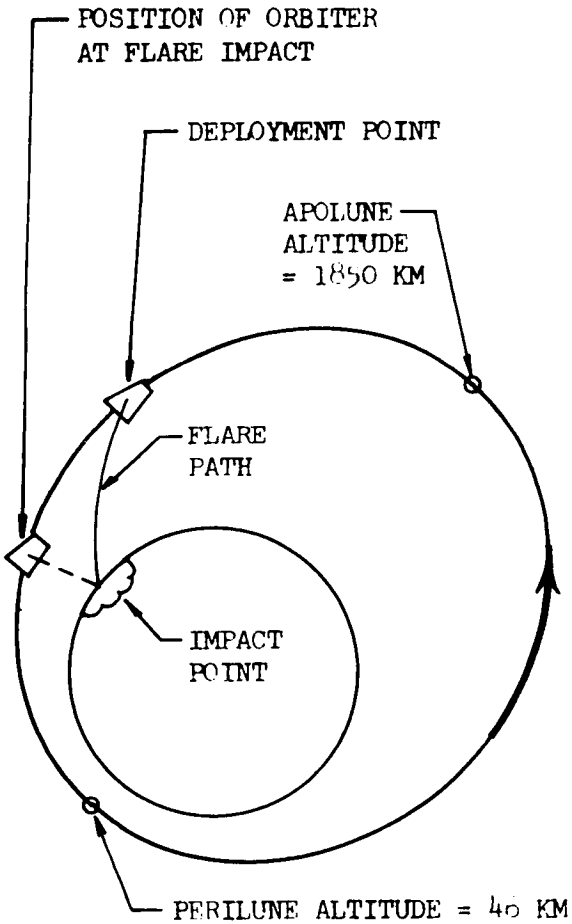
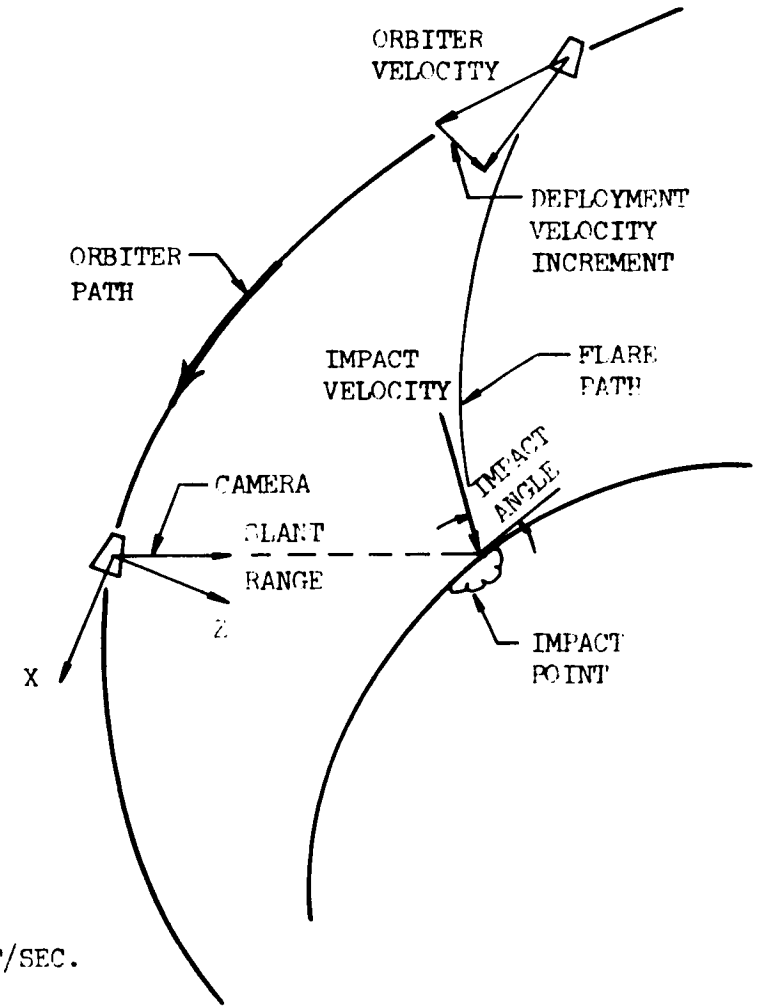
FLARE DELIVERY GEOMETRY

DELIVERY OF A FLARE TO THE SURFACE OF THE MOON INVOLVES DEPLOYMENT FROM SOME POINT ON THE NOMINAL LUNAR ORBITER ORBIT. THIS ORBIT HAS APOLUNE AND PERILUNE ALTITUDES OF 1850 KM AND 46 KM RESPECTIVELY. THE DEPLOYMENT DEVICE ASSUMED WAS A SMALL SPIN STABILIZED SOLID ROCKET MOTOR FIRED FROM A FIXED RECOILLESS TUBE MOUNTED ON THE ORBITER AND ORIENTED BY THE ORBITER ATTITUDE CONTROL SYSTEM. THE POSITION OF THE ORBITER AT DEPLOYMENT, THE DEPLOYMENT VELOCITY INCREMENT, AND THE REQUIRED FIRING ANGLE ARE FIXED BY THE SELECTION OF THE DESIRED IMPACT VELOCITY, ANGLE, AND THE SLANT RANGE BETWEEN THE ORBITER AND THE FLARE AT IMPACT.

THE DESIRED IMPACT VELOCITY FOR THE FLARE EXPERIMENTS IS BETWEEN 6000 FT/SECOND AND 6000 FT/SECOND WITH A CORRESPONDING IMPACT ANGLE $\geq 45^{\circ}$. IN ADDITION, THE ORBITER MUST BE ORIENTED TO PLACE THE CAMERA AXIS ALONG THE IMPACT LINE-OF-SIGHT WITH LOW VALUES OF SLANT RANGE DESIRED.

FLARE DELIVERY GEOMETRY



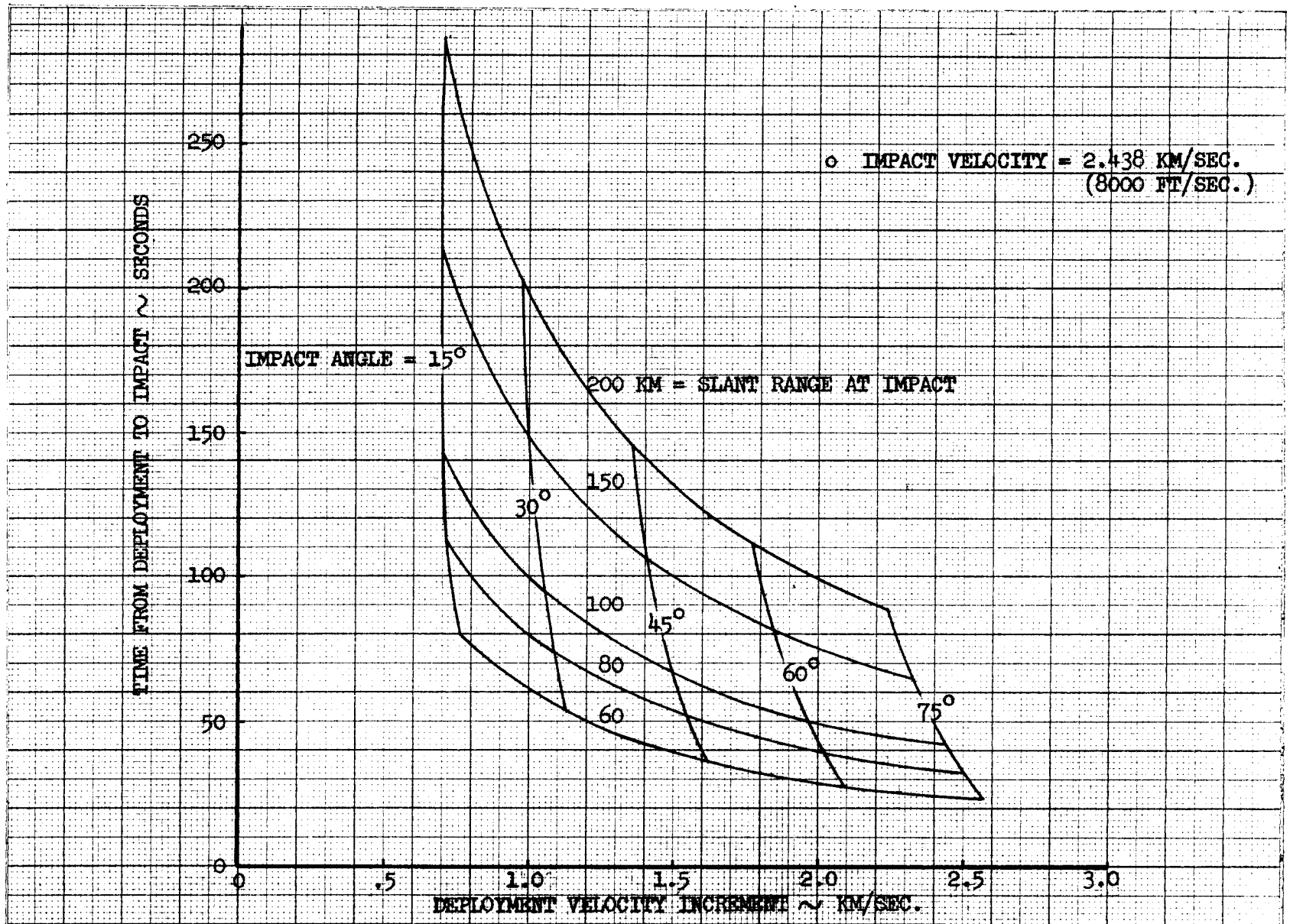


- REQUIRED IMPACT VELOCITY = 6000-8000 FT/SEC.
- IMPACT ANGLE $\geq 45^{\circ}$

FLARE TRAJECTORY DESIGN - 2.44 KM/SECOND

THIS CHART PROVIDES THE DATA REQUIRED TO DEFINE A FLARE DEPLOYMENT TRAJECTORY FOR AN IMPACT VELOCITY OF 2.44 KM/SECOND (8000 FT/SECOND). IMPACT ANGLES BETWEEN 15° AND 75° AND SLANT RANGES BETWEEN 60 KM AND 200 KM ARE DEPICTED. SELECTION OF THE DESIRED IMPACT CONDITIONS ALLOWS THE DETERMINATION OF THE REQUIRED DEPLOYMENT VELOCITY INCREMENT AND THE TIME FROM DEPLOYMENT TO IMPACT.

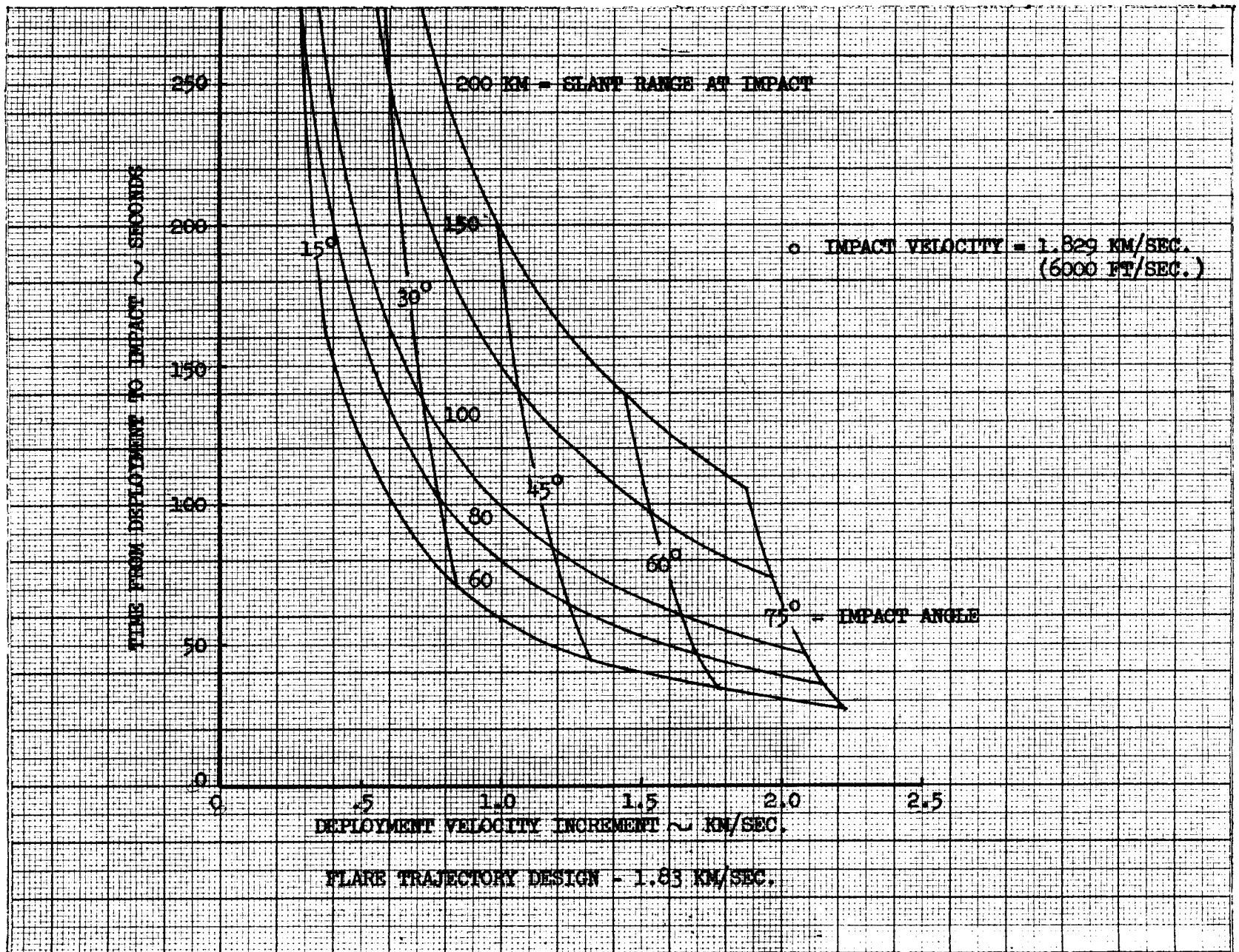
ALTHOUGH SHORTER SLANT RANGES, WITH A GIVEN IMPACT ANGLE, REQUIRE INCREASED DEPLOYMENT VELOCITY INCREMENTS, THE CHANGE IS VERY SMALL COMPARED TO THE EFFECT OF INCREASING THE DESIRED IMPACT ANGLE. NOTE THAT THE FLARE FLIGHT TIMES BECOME VERY SHORT AS THE SLANT RANGE REQUIREMENT REDUCES. FOR EXAMPLE, WITH A SLANT RANGE OF 60 KM AND AN IMPACT ANGLE OF 45°, THE FLARE FLIGHT TIME IS ONLY 36 SECONDS.



FLARE TRAJECTORY DESIGN - 2.44 KM/SEC.

FLARE TRAJECTORY DESIGN - 1.83 KM/SECOND

THIS CHART ILLUSTRATES THE SAME PARAMETERS AS THE PREVIOUS CHART FOR AN IMPACT VELOCITY OF 1.83 KM/SECOND (6000 FT/SECOND). IN GENERAL, THE EFFECT OF REDUCING THE DESIRED IMPACT VELOCITY IS TO REDUCE THE REQUIRED DEPLOYMENT VELOCITY AND INCREASE THE FLIGHT TIME. BASED ON A TYPICAL CASE WITH A 60 KM SLANT RANGE AND AN IMPACT ANGLE OF 45°, THE DEPLOYMENT VELOCITY INCREMENT IS 1.32 KM/SECOND AND THE FLIGHT TIME IS 45 SECONDS. WHEN COMPARED WITH THE CORRESPONDING INFORMATION FOR AN IMPACT VELOCITY OF 8000 FT/SECOND, THIS REPRESENTS A REDUCTION OF 0.29 KM/SECOND IN DEPLOYMENT VELOCITY AND AN INCREASE OF 9 SECONDS IN FLIGHT TIME.



FLARE TRAJECTORY DESIGN - 1.83 KM/SEC.

FLARE IMPACT ERRORS

THE ERRORS LISTED HERE REFER TO A TYPICAL FLARE DELIVERY TRAJECTORY FOR TWO DISTINCT IMPACT VELOCITIES. THE ERRORS WERE OBTAINED BY FIRST OBTAINING THE ERRORS IN STATE AT THE TIME OF THE INSERTION DUE TO ERRORS IN $\triangle V$ AND POINTING AND BY THEN AMPPING THESE DISPERSIONS TO THE GROUND. ERRORS IN KNOWLEDGE OF THE LUNAR SURFACE ALTITUDE WERE NOT INCORPORATED.

THE ERROR IN IMPACT TIME ASSOCIATED WITH THE ERRORS IN LOCAL COORDINATES REFLECTS THE PROBABILITY OF THE FLARE HAVING TO FLY A LONGER OR SHORTER TRAJECTORY THAN NOMINAL.

FLARE IMPACT ERRORS

- FIXED TRAJECTORY PARAMETERS

  SLANT RANGE = 80 KM

  IMPACT ANGLE = 45°

- ERROR SOURCES (3σ)

  POINTING ERROR = 0.88°

  ΔV SCALE FACTOR = 0.6%

- ERRORS AT IMPACT

	AT IMPACT VELOCITY = 2.44 KM/SEC. (8000 FT/SEC.)	AT IMPACT VELOCITY = 1.83 KM/SEC. (6000 FT/SEC.)
DOWNRANGE (KM) :	1.20	1.07
CROSS RANGE (KM) :	1.25	1.24
ALTITUDE RATE (M/SEC.) :	13.62	11.13
DOWNRANGE RATE (M/SEC.) :	12.28	17.52
CROSS RANGE RATE (M/SEC.) :	23.84	19.11
IMPACT TIME (SEC.) :	0.32	0.74

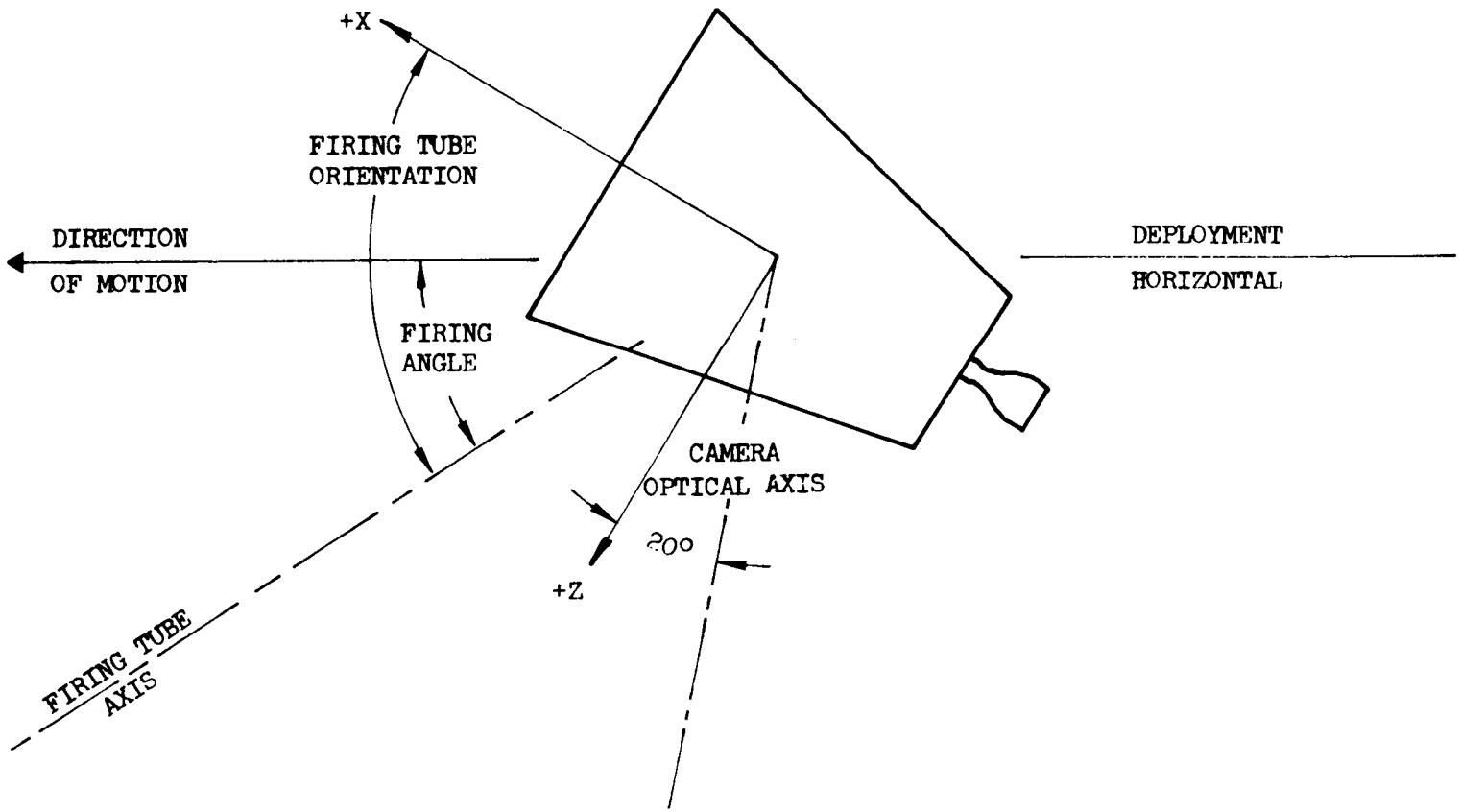
Revision 1, Dated 7-29-66

FIRING TUBE GEOMETRY

THE VERY SHORT FLIGHT TIMES ASSOCIATED WITH THE FLARE DEPLOYMENT TRAJECTORIES PRECLUDE ANY MANEUVERING OF THE SPACECRAFT BETWEEN DELIVERY VEHICLE DEPLOYMENT AND FLARE IMPACT. CONSEQUENTLY, IT WILL BE NECESSARY TO HAVE THE CAMERA PREALIGNED ALONG THE PREDICTED IMPACT LINE-OF-SIGHT AT THE TIME OF DEPLOYMENT. FOR A GIVEN SET OF DESIRED IMPACT CONDITIONS, THE RANGE ANGLE TRAVERSED BY THE SPACECRAFT DURING THE FLARE FLIGHT IS READILY DETERMINED. KNOWLEDGE OF THIS RANGE ANGLE AND THE ORIENTATION OF THE CAMERA SPECTROGRAPH AXIS RELATIVE TO THE SPACECRAFT IS SUFFICIENT TO DEFINE THE INERTIAL ORIENTATION OF THE SPACECRAFT AT DELIVERY VEHICLE DEPLOYMENT. THE DESIRED IMPACT CONDITION ALSO DEFINES THE REQUIRED DIRECTION OF THE DEPLOYMENT VELOCITY VECTOR. SINCE NO ROTATION OF THE ORBITER IS POSSIBLE AFTER DEPLOYMENT, THE ORIENTATION OF THE FIRING TUBE ON THE SPACECRAFT IS ALSO DEFINED. THE SKETCH SHOWN DEFINES BOTH THE FIRING ANGLE AND THE FIRING TUBE ORIENTATION RELATIVE TO THE DEPLOYMENT HORIZONTAL AND THE NOMINAL SPACECRAFT AXES.

THE OPTICAL AXIS OF THE CAMERA WITH THE DIFFRACTION GRATING INSTALLED MAY BE OFFSET IN THE ROLL DIRECTION BY SOME SMALL FIXED ANGLE AND DEPENDS ON THE PARTICULAR GRATING INSTALLATION SELECTED. THIS BIAS ANGLE HAS NOT BEEN ACCOUNTED FOR IN THE ILLUSTRATION.

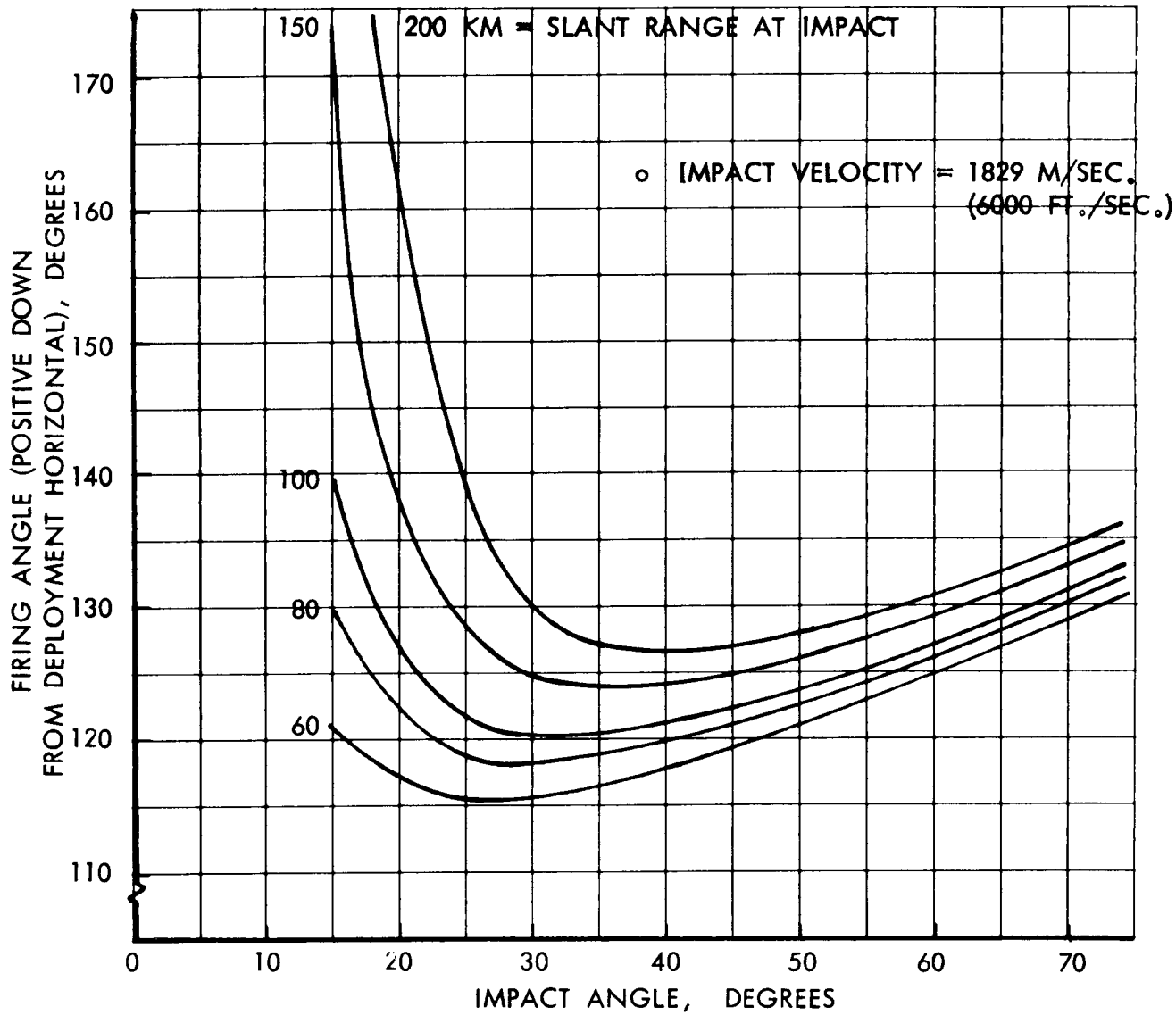
FIRING TUBE GEOMETRY



FLARE FIRING ANGLE REQUIREMENTS

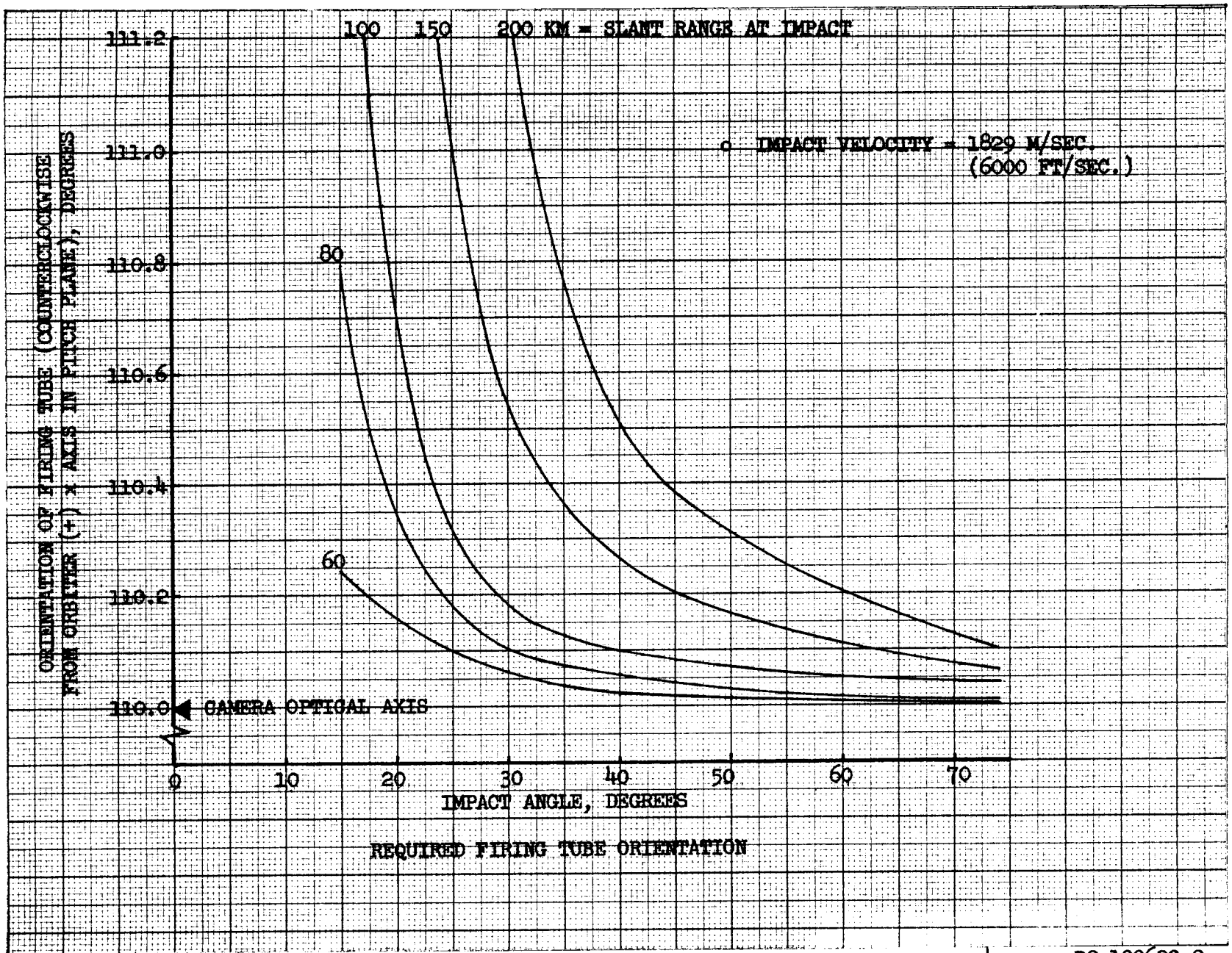
THE REQUIRED FIRING ANGLE DIRECTIONS FOR AN IMPACT VELOCITY OF 6000 FT/SECOND AND A RANGE OF IMPACT ANGLES BETWEEN 15° AND 75° ARE GIVEN. THE EFFECT OF IMPACT RANGE VARIATIONS FROM 60 TO 200 KM IS ALSO SHOWN. NOTE THAT THE FIRING DIRECTION IS ALWAYS DOWN AND BACK FROM THE VERTICAL FOR THIS RANGE OF IMPACT CONDITIONS.

FLARE FIRING ANGLE REQUIREMENTS



REQUIRED FIRING TUBE ORIENTATION

THE CAMERA SPECTROGRAPH AXIS MUST BE ALIGNED WITH THE SPACECRAFT/FLARE LINE-OF-SIGHT AT IMPACT. THE SHORT FLARE FLIGHT TIMES PRECLUDE ANY SPACECRAFT TURNING AFTER DEPLOYMENT. CONSEQUENTLY, THE FIRING TUBE MUST HAVE A SPECIFIC ORIENTATION RELATIVE TO THE SPACECRAFT AT DEPLOYMENT. TYPICAL VALUES OF REQUIRED ORIENTATION ARE ILLUSTRATED FOR VARIOUS VALUES OF IMPACT ANGLE AND SLANT RANGE. THE IMPACT VELOCITY IS A CONSTANT 6000 FT/SECOND. NOTE THAT THE ORIENTATION OF THE TUBE IS NEARLY CONSTANT AND IS ALIGNED VERY CLOSELY WITH THE CAMERA SPECTROGRAPH AXIS. THIS IS A DESIRABLE RESULT SINCE MANY IMPACT CONDITIONS ARE ACHIEVABLE WITH A FIXED TUBE ORIENTATION ON THE SPACECRAFT.



REQUIRED FIRING TUBE ORIENTATION

FLARE SENSOR EQUIREMENTS

THE SENSOR REQUIREMENTS FOR EACH KIND OF DATA TO BE OBTAINED FROM THE EXPERIMENT HAD TO BE ESTABLISHED.

FLARE SENSOR REQUIREMENTS

- TIME TO FIRST PEAK DETECTOR FOR HARDNESS
- SPECTRAL DATA FOR CHEMICAL COMPOSITION
- COLOR TEMPERATURE FOR DEPTH OF SURFACE LAYER

DATA RETRIEVAL

FAST RESPONSE PHOTO TUBE -- 222 KHz BANDWIDTH, REAL-TIME TRANSMISSION VIA. HIGH GAIN ANTENNA

DIFFRACTION GRATING IN PHOTOGRAPHIC SYSTEM AND MODIFIED CAMERA SHUTTER.

COLOR TEMPERATURE FROM FILTERED PHOTO TUBES, 6 CHANNELS ON LOOP TAPE RECORDER WITH PLAYBACK, 5 KHz BANDPASS.

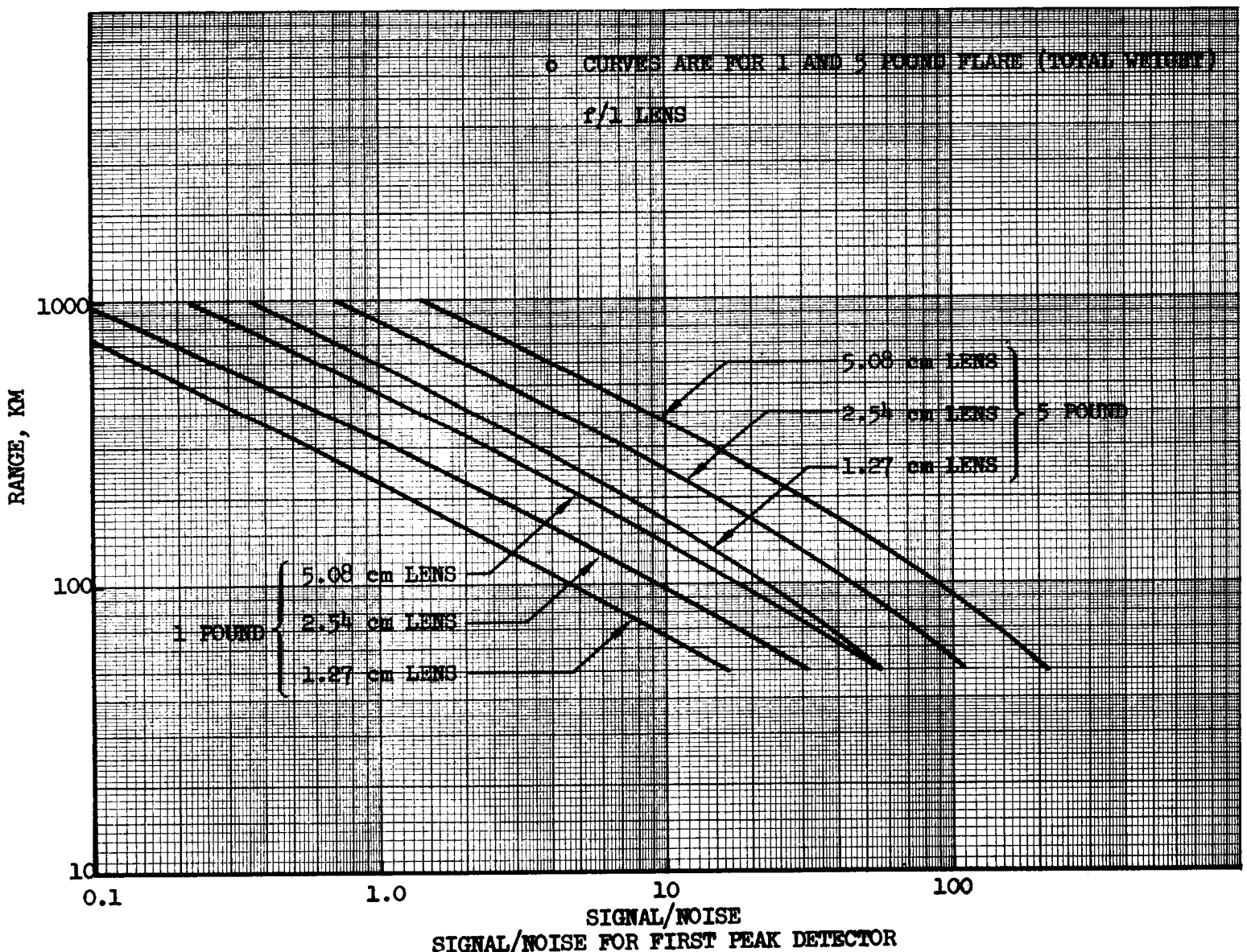
SIGNAL/NOISE FOR FIRST PEAK DETECTOR

THREE NOISE SOURCES ARE CONSIDERED IN THIS FIGURE. THEY ARE DETECTOR DARK NOISE, SCENE BACKGROUND NOISE, AND NOISE IN SIGNAL. THE DETECTOR DARK NOISE IS CONSTANT WHILE THE SCENE BACKGROUND NOISE INCREASES WITH RANGE AND THE NOISE IN SIGNAL INCREASES WITH DECREASING RANGE AND INCREASING FLARE SIZE.

TWO FLARE SIZES WERE CONSIDERED. THE ONE-POUND FLARE WAS CHOSEN AS BEING NEAR A PRACTICAL MINIMUM FROM THE STANDPOINT OF SIGNAL POWER WHILE THE FIVE-POUND FLARE APPROACHES THE MAXIMUM SIZE DUE TO DEPLOYMENT AND DELIVERY CONSTRAINTS.

THE THREE LENS SIZES WERE CHOSEN TO YIELD SIGNAL/NOISE FIGURES IN THE REGION OF 100 TO 1 FOR ORBITER TO FLARE IMPACT RANGES OF APPROXIMATELY 50 KM TO 1000 KM.

AS AN EXAMPLE, A ONE-POUND FLARE VIEWED FROM A RANGE OF 100 KM WITH A 2.54 CM F/1 LENS WILL PROVIDE A SIGNAL/NOISE OF APPROXIMATELY 10.


o CURVES ARE FOR 1 AND 5 POUND FLARE (TOTAL WEIGHT)
f/1 LENS
5.08 cm LENS
2.54 cm LENS
1.27 cm LENS
5 POUND
1 POUND
5.08 cm LENS
2.54 cm LENS
1.27 cm LENS
RANGE, KM
1000
100
10
0.1
1.0
10
100
SIGNAL/NOISE
SIGNAL/NOISE FOR FIRST PEAK DETECTOR

INSTRUMENTATION REQUIREMENTS
"FIRST PEAK" DETECTOR

THE CHARACTERISTICS OF THE INSTRUMENT TO BE INSTALLED ON THE SPACECRAFT TO RECORD "TIME TO FIRST PEAK" ARE SUMMARIZED ON THIS CHART.

INSTRUMENTATION REQUIREMENTS
"FIRST PEAK" DETECTOR

o	OPTICS	5.08 CM FOCAL LENGTH, f/1
o	DETECTOR	1 PHOTOMULTIPLIER TUBE (S-1 PHOTOCATHODE)
o	POWER	2 WATTS (AVERAGE)
o	WEIGHT	4 POUNDS
o	VOLUME	192 CU. IN. (12" X 4" X 4")
o	ENVIRONMENT	TEMPERATURE : 10° - 110°F
o	DATA RETRIEVAL	REAL-TIME -- HIGH GAIN ANTENNA READOUT

SPECTROGRAPH OPTICAL SCHEMATIC

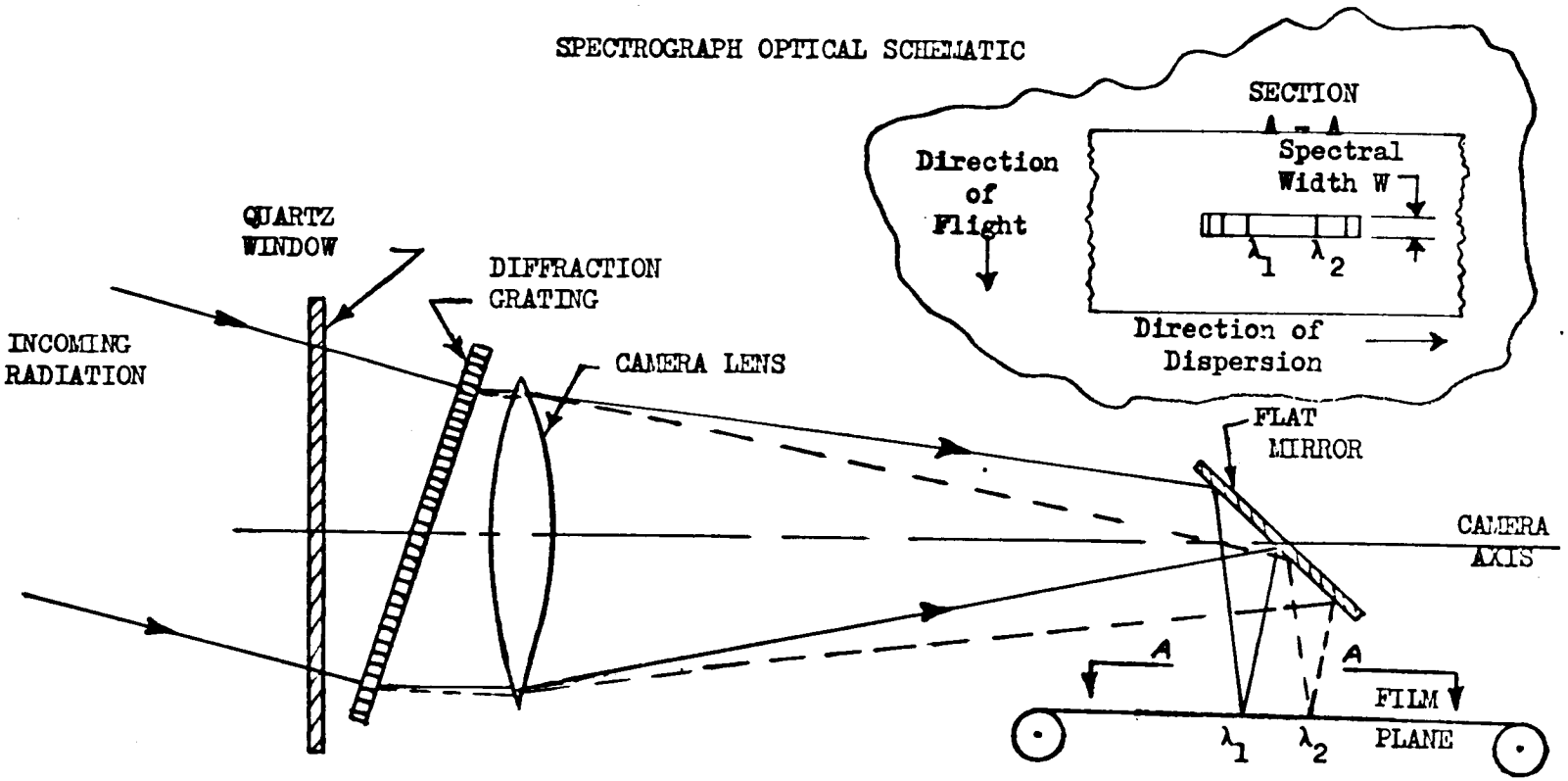
THIS FIGURE DEPICTS SCHEMATICALLY THE EXISTING LUNAR ORBITER CAMERA AS MODIFIED TO RECORD THE SPECTRAL CHARACTERISTICS OF THE LIGHT EMITTED DURING THE FLARE IMPACT.

THE DIFFRACTION GRATING SHOWN IS THE ONLY ADDITIONAL OPTICAL ELEMENT WHICH MUST BE INSTALLED IN THE EXISTING CAMERA.

AS IS SHOWN, THE CAMERA AXIS WILL BE SLIGHTLY OFFSET FROM THE LINE-OF-SIGHT TO THE FLARE IMPACT POINT TO PROVIDE THE PROPER ANGLE OF INCIDENCE TO THE GRATING.

THE DIRECTION OF DISPERSION IS ORIENTED NORMAL TO THE ORBITER LINE-OF-FLIGHT TO PREVENT LOSS OF SPECTRAL RESOLUTION DUE TO SMEAR. THE DISPERSION IS ORIENTED TO LIE ALONG THE LONG DIMENSION OF THE CAMERA HIGH-RESOLUTION FRAME TO PROVIDE A MAXIMUM DISPERSION RANGE WITHIN THE FILM FORMAT CONSTRAINTS.

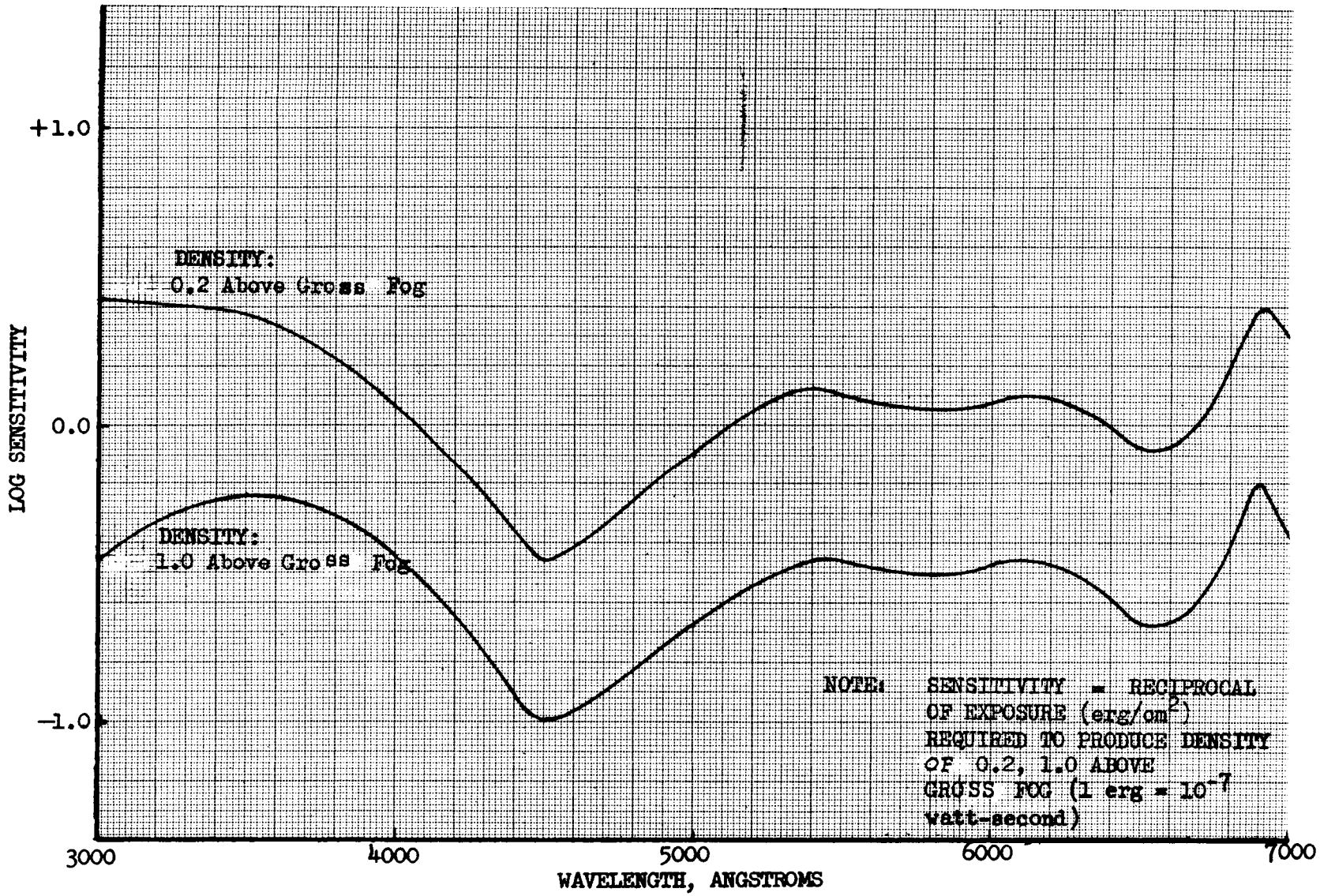
SPECTROGRAPH OPTICAL SCHEMATIC



NOTE: VEHICLE MOTION IS PERPENDICULAR TO DIRECTION OF DISPERSION

SPECTRAL SENSITIVITY SO-243 FILM

THIS FIGURE SHOWS THE SENSITIVITY OF THE LUNAR ORBITER CAMERA FILM AS A FUNCTION OF THE WAVELENGTH OF THE INCIDENT ILLUMINATION. THE CONSTANT DENSITY CURVES ARE USED TO ESTIMATE THE ENERGY DENSITY AT THE FILM PLANE REQUIRED TO PRODUCE A USEFUL SPECTROGRAPH.



SPECTRAL SENSITIVITY SO-243 FILM

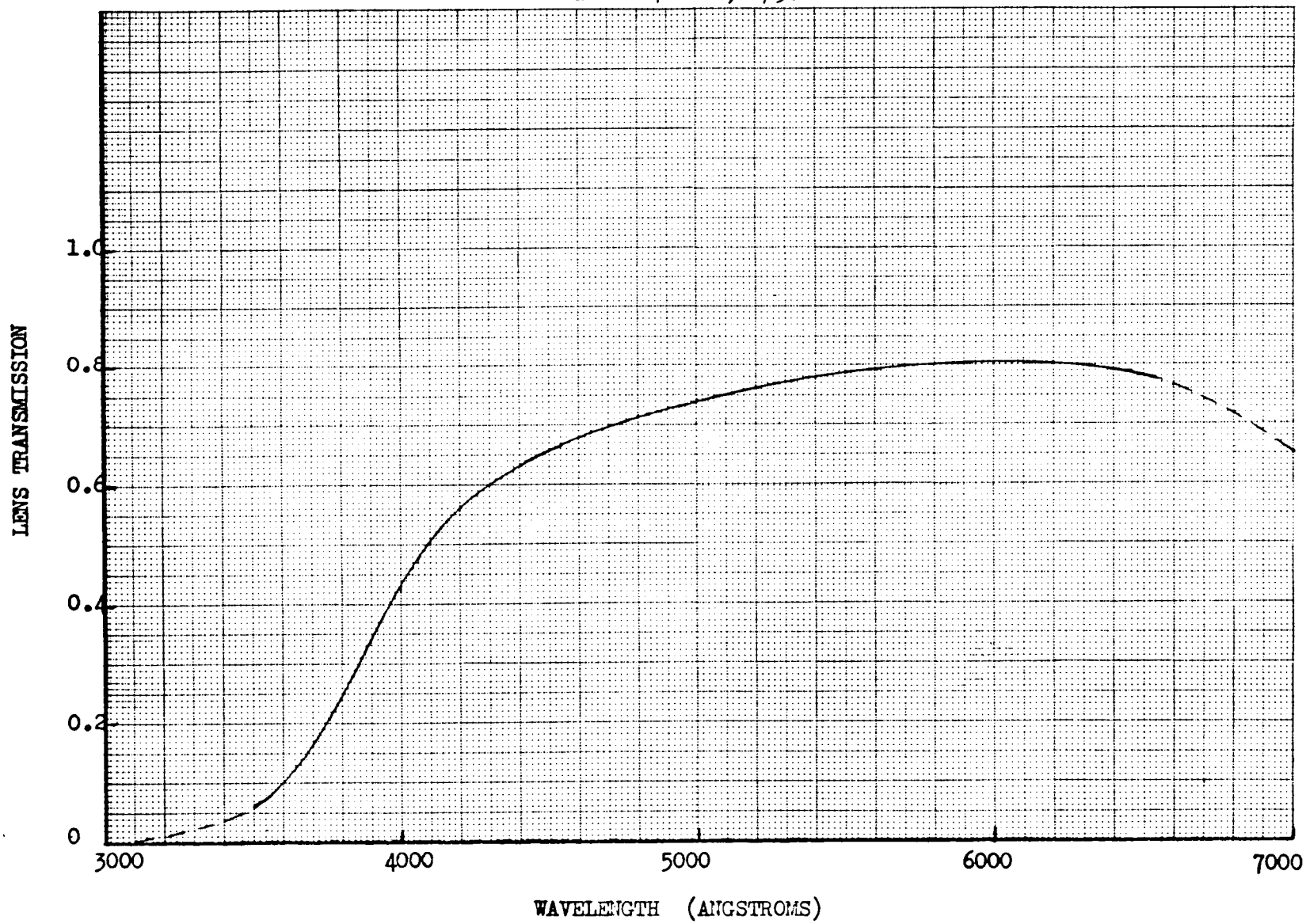
ON-AXIS TRANSMITTION - ORBITER 24-INCH F/5.6 LENS

THE LENS TRANSMISSION PLOTTED AS A FUNCTION OF WAVELENGTH IN THE FIGURE WAS OBTAINED BY MEASUREMENT AT EASTMAN KODAK. IT SHOULD BE NOTED THAT THE TRANSMISSION FALLS OFF RAPIDLY BELOW ABOUT 4200 ANGSTROMS AND THAT THE MEASURED DATA EXTENDS ONLY FROM 3500 ANGSTROMS TO 6500 ANGSTROMS.

MUCH OF THE SPECTRAL DATA OF INTEREST IN THE FLARE EXPERIMENT FALLS BELOW 4000 ANGSTROMS AND THE LOW LENS TRANSMISSION IN THIS REGION BECOMES A CRITICAL PARAMETER.

ON-AXIS TRANSMISSION

ORBITER 24-INCH, f/5.6 LENS

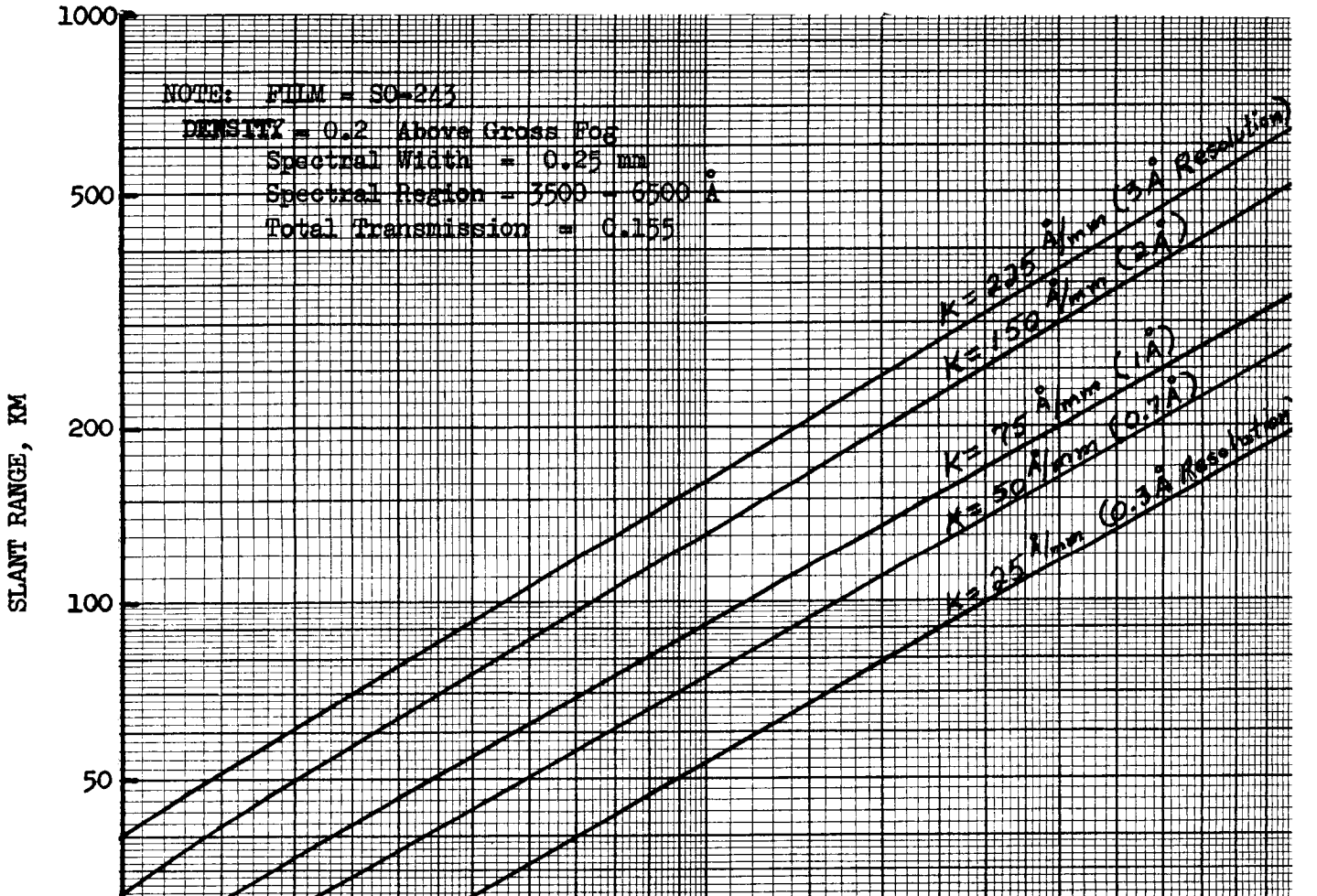


FLARE WEIGHT

BECAUSE THE OPTICAL PARAMETERS OF THE LUNAR ORBITER CAMERA ARE FIXED AS IS THE FILM TYPE, THE PRIMARY PARAMETER WHICH MAY BE VARIED ALONG WITH RANGE AND FLARE WEIGHT IS SPECTRAL DISPERSION.

FOR A FIVE-POUND FLARE AT AN OBSERVATION RANGE OF 50 KM, IT CAN BE SEEN THAT A GRATING YIELDING A DISPERSION IN THE FILM PLANE OF 50 ANGSTROMS/MM WILL PROVIDE A SPECTRAL RESOLUTION OF 0.7 ANGSTROMS.

THE DISPERSIONS CONSIDERED EXTEND FROM 225 ANGSTROMS/MM, WHICH WOULD PROVIDE SPECTROGRAMS OF MARGINAL USEFULNESS, TO 25 ANGSTROMS/MM WHICH WITH THE LUNAR ORBITER CAMERA IS STATE-OF-THE-ART.



FLARE WEIGHT

INSTRUMENTATION REQUIREMENTS
SPECTRAL ENERGY DETECTOR

IN ADDITION TO THE ITEMS LISTED ON THIS CHART, THE CAMERA WOULD NEED TO BE MODIFIED TO ALLOW USING AN "OPEN SHUTTER" TO RECORD THE FLARE EVENT.

INSTRUMENTATION REQUIREMENTS
SPECTRAL ENERGY DETECTOR

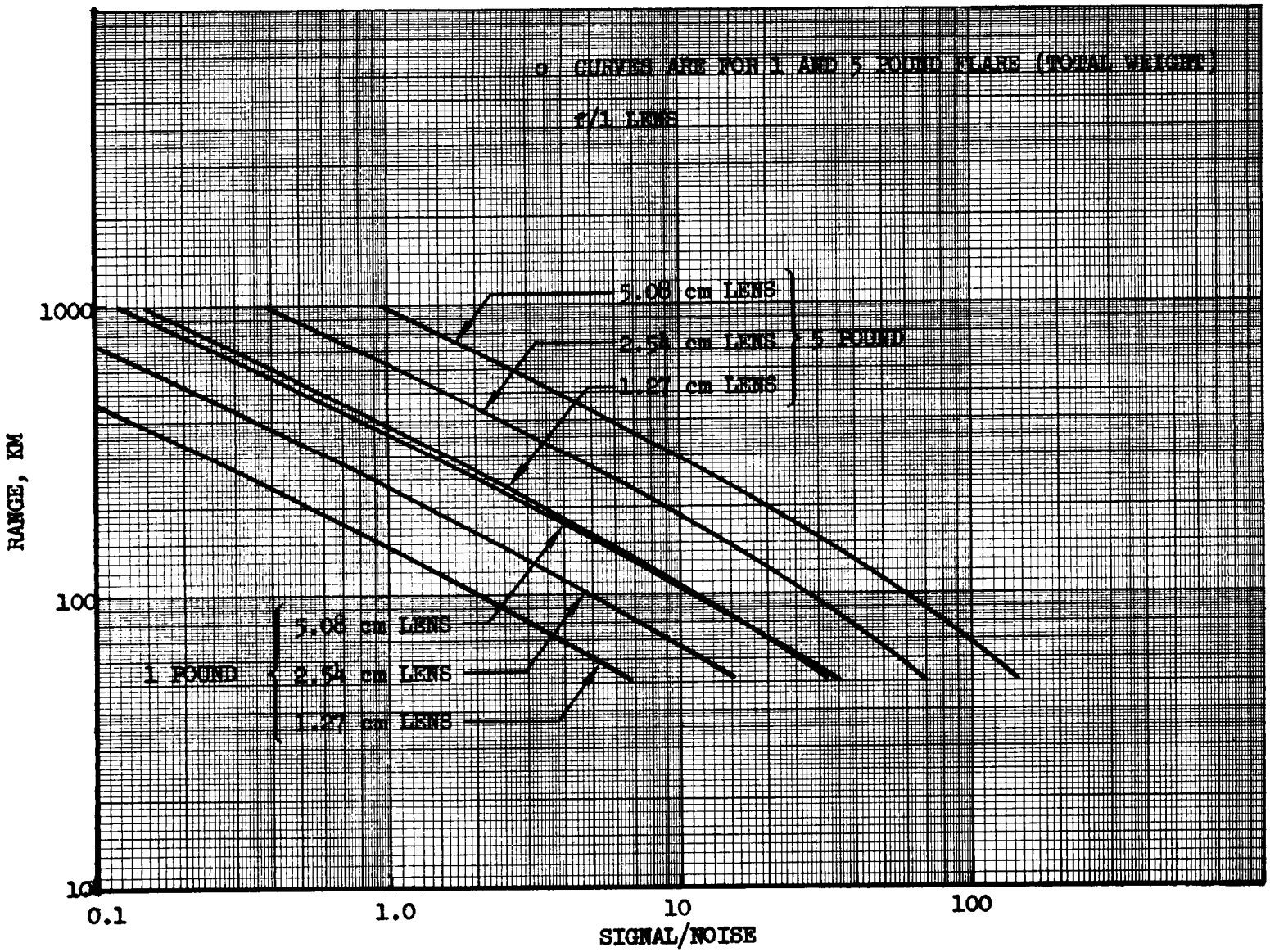
o	OPTICS	24-INCH, f/5.6 ORBITER LENS
o	DETECTOR	SO-243 FILM
o	POWER	MOMENTARY SURGE OF 100 - 200 WATTS
o	WEIGHT	6 POUNDS
o	VOLUME	72 CU. IN. (6" x 6" x 2")
o	ENVIRONMENT	TEMPERATURE : COMPATIBLE WITH EXISTING PHOTOGRAPHIC SUBSYSTEM
o	DATA RETRIEVAL	LINE SCAN READOUT - VIDEO LINK

SIGNAL/NOISE FOR TEMPERATURE DETECTOR

THIS FIGURE SHOWS THE SIGNAL/NOISE AS A FUNCTION OF RANGE, FLARE WEIGHT, AND LENS SIZE.

THIS FIGURE IS EQUIVALENT TO THAT FOR THE FIRST PEAK DETECTOR. THE NUMERICAL CHANGES ARE DUE TO REDUCED ENERGY BECAUSE OF SPECTRAL FILTERING, AND SMALLER REQUIRED BANDWIDTH BECAUSE OF REDUCED TIME RESOLUTION REQUIREMENTS OF THE COLOR TEMPERATURE MEASUREMENT.

FROM THIS FIGURE, A ONE-POUND FLARE VIEWED FROM A RANGE OF 100 KM WITH A 2.54 CM F/1 LENS WILL PROVIDE A SIGNAL/NOISE OF APPROXIMATELY 4.5.



SIGNAL/NOISE FOR TEMPERATURE DETECTOR

INSTRUMENTATION REQUIREMENTS
TEMPERATURE DETECTOR

TO OBTAIN THE TEMPERATURE DATA REQUIRES THE ADDITION TO THE SPACECRAFT OF THE INSTRUMENTS SUMMARIZED ON THIS CHART.

INSTRUMENTATION REQUIREMENTS
TEMPERATURE DETECTOR

o	OPTICS	5.08 CM FOCAL LENGTH, f/1
o	DETECTOR	6 PHOTOMULTIPLIER TUBES (S-1 PHOTOCATHODE)
o	POWER	10 WATTS (AVERAGE)
o	WEIGHT	15 POUNDS (TOTAL)
o	VOLUME	420 CU. IN. (5" x 7" x 12")
o	ENVIRONMENT	TEMPERATURE : 10 - 110°F
o	DATA RETRIEVAL	TAPE STORAGE - SEQUENTIAL READOUT

LUNAR ORBITER BUS WEIGHT
FLARE MISSION

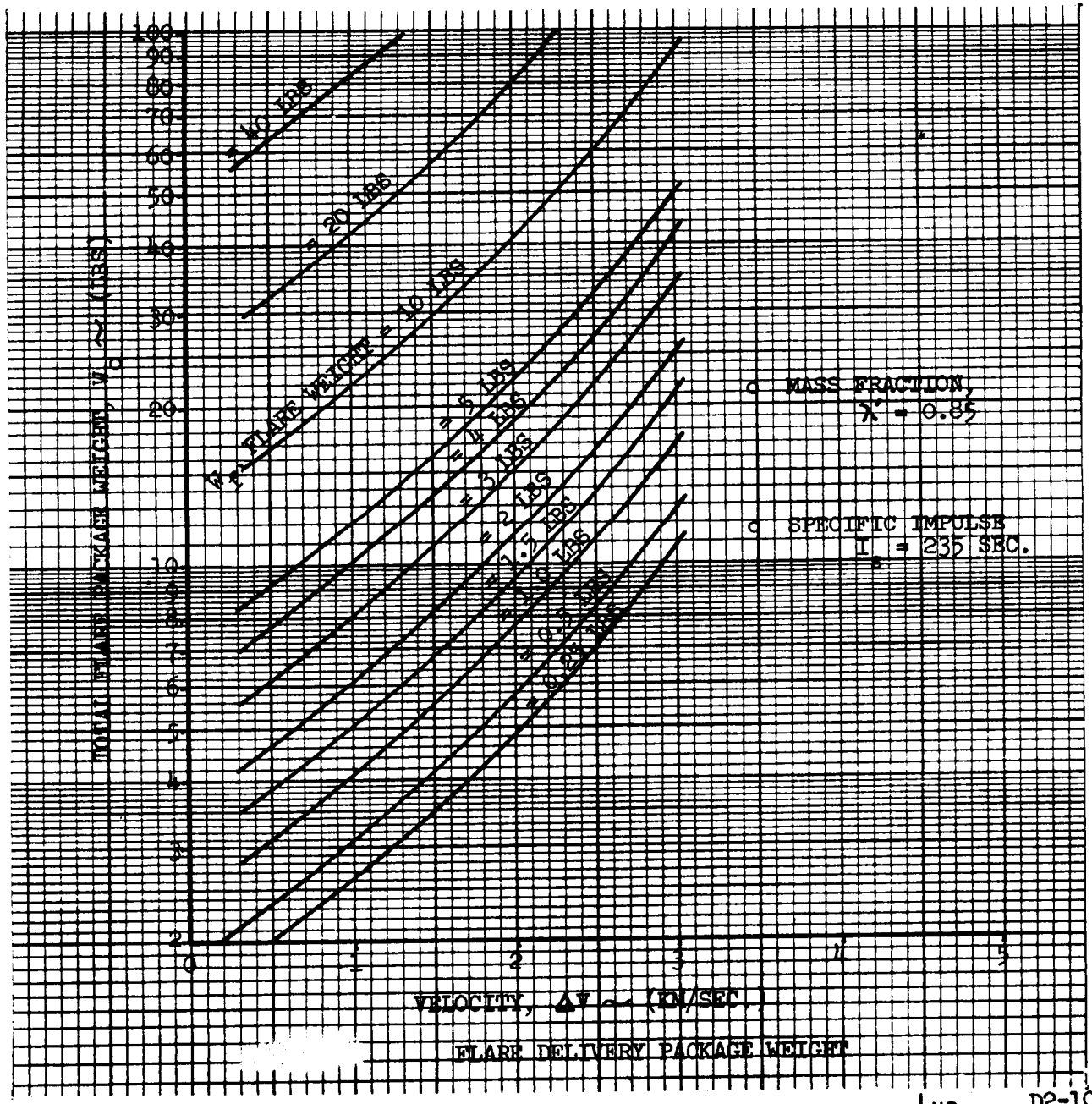
THE MAXIMUM ALLOWABLE WEIGHT ALLOTMENT FOR FLARE DELIVERY VEHICLES IS DEFINED HERE. THIS ALLOTMENT REPRESENTS THE ALLOWABLE WEIGHT REMAINING AFTER THE STRIPPED LUNAR ORBITER AND THE ADDITIONAL REQUIRED EQUIPMENT ARE REMOVED FROM THE MAXIMUM ALLOWABLE BUS WEIGHT.

LUNAR ORBITER BUS WEIGHT
FLARE MISSION

STRIPPED LUNAR ORBITER WEIGHT		840 POUNDS
ADDITIONAL EQUIPMENT FOR ELF MISSION :		
o DEPLOYMENT DEVICE AND MOUNTING STRUCTURE	20 POUNDS	
o DIFFRACTION GRATING AND ASSOCIATED HARDWARE	6	
o FIRST PEAK DETECTOR	4	
o TEMPERATURE DETECTOR	15	
o RECORDER AND RELATED INTERFACE EQUIPMENT	5	
TOTAL		50
LUNAR ORBITER BUS WEIGHT (FLARE)		890 POUNDS
MAXIMUM ALLOWABLE BUS WEIGHT		950
MAXIMUM ALLOTMENT FOR DELIVERY VEHICLES		60 POUNDS

FLARE DELIVERY PACKAGE WEIGHT

TWO BASIC PARAMETERS ARE PERTINENT TO THE SIZING OF THE FLARE DELIVERY ROCKET. THESE ARE THE WEIGHT OF THE FLARE AND THE REQUIRED DEPLOYMENT VELOCITY INCREMENT. THE TOTAL FLARE PACKAGE WEIGHT IS SHOWN HERE AS A FUNCTION OF FLARE WEIGHT AND DEPLOYMENT $\triangle V$. THE PACKAGE WEIGHT IS DEFINED AS THE WEIGHT OF THE FLARE PLUS THE DELIVERY ROCKET. THE DATA ARE BASED ON A PROPELLANT LOADING FRACTION OF 0.85 AND A SPECIFIC IMPULSE OF 235 SECONDS. THESE VALUES ARE REPRESENTATIVE OF THE PRESENT STATE-OF-THE-ART FOR SMALL ROCKET MOTORS.

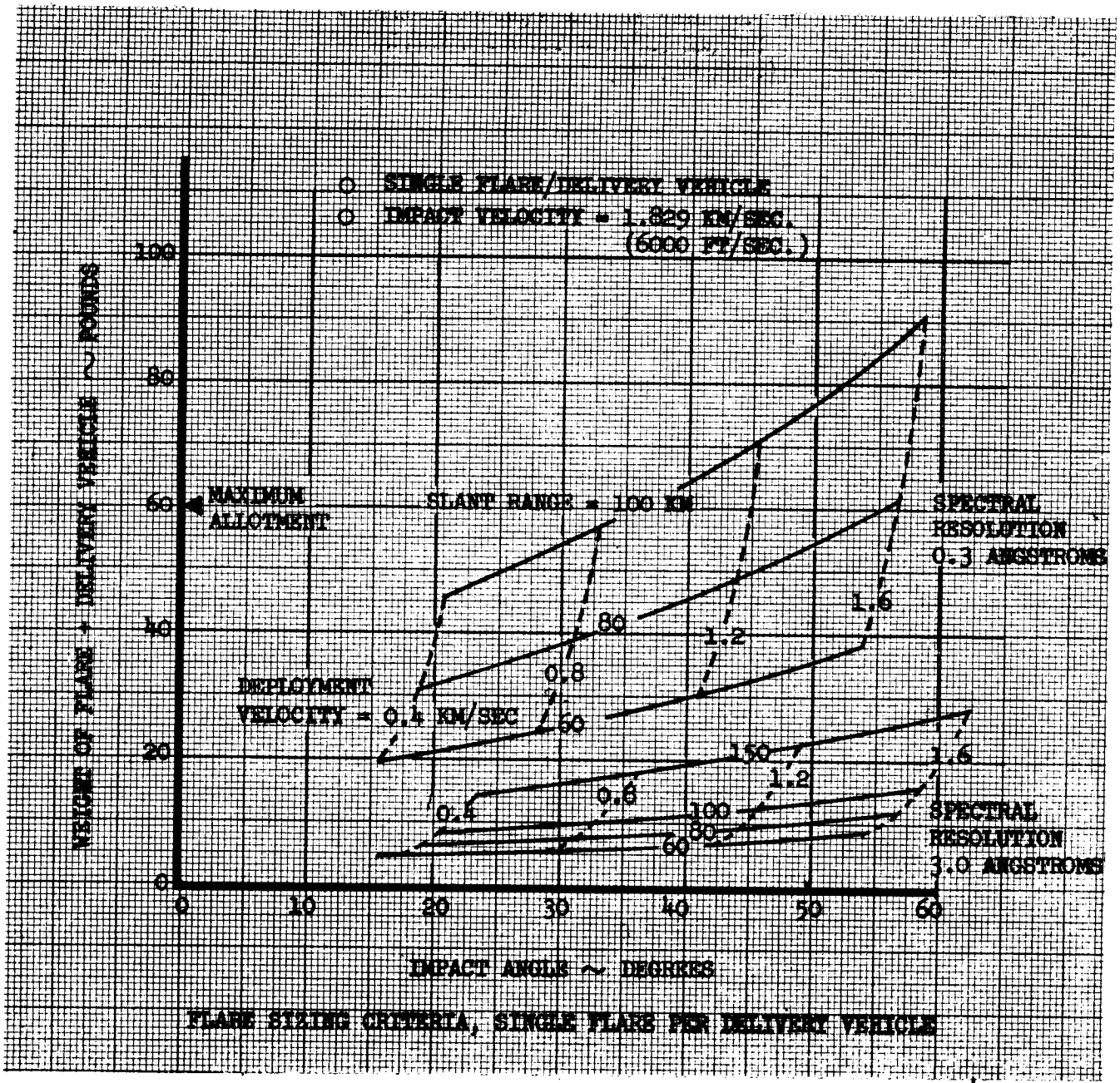


FLARE DELIVERY PACKAGE WEIGHT

FLARE SIZING CRITERIA, SINGLE FLARE PER DELIVERY VEHICLE

FLARE SYSTEM DEFINITION REQUIRES THAT SOME SIZING CRITERIA BE ESTABLISHED. THE DEPLOYMENT VELOCITY REQUIREMENT FOR GIVEN IMPACT CONDITIONS AND THE FLARE WEIGHT NECESSARY FOR GIVEN SLANT RANGE AND SPECTRAL RESOLUTION ALLOW SPECIFICATION OF THE DEPLOYMENT ROCKET SIZE. THESE DATA HAVE BEEN COMBINED TO ESTABLISH FLARE DELIVERY PACKAGE SIZES UNDER THE ASSUMPTION THAT EACH DELIVERY VEHICLE CARRIES ONLY ONE FLARE. THE DATA ARE BASED ON AN IMPACT VELOCITY OF 6000 FEET PER SECOND. TWO FAMILIES OF CURVES ARE SHOWN. ONE FAMILY IS BASED ON A RESOLUTION OF 0.3 ANGSTROMS WHICH IS ABOUT THE BEST POSSIBLE. THE OTHER FAMILY ASSUMES A RESOLUTION OF 3.0 ANGSTROMS, WHICH REPRESENTS THE WORST RESOLUTION WITH WHICH ACCEPTABLE DATA CAN BE OBTAINED. WITHIN EACH FAMILY OF CURVES, SLANT RANGE AND DEPLOYMENT VELOCITY VARIATIONS ARE SHOWN.

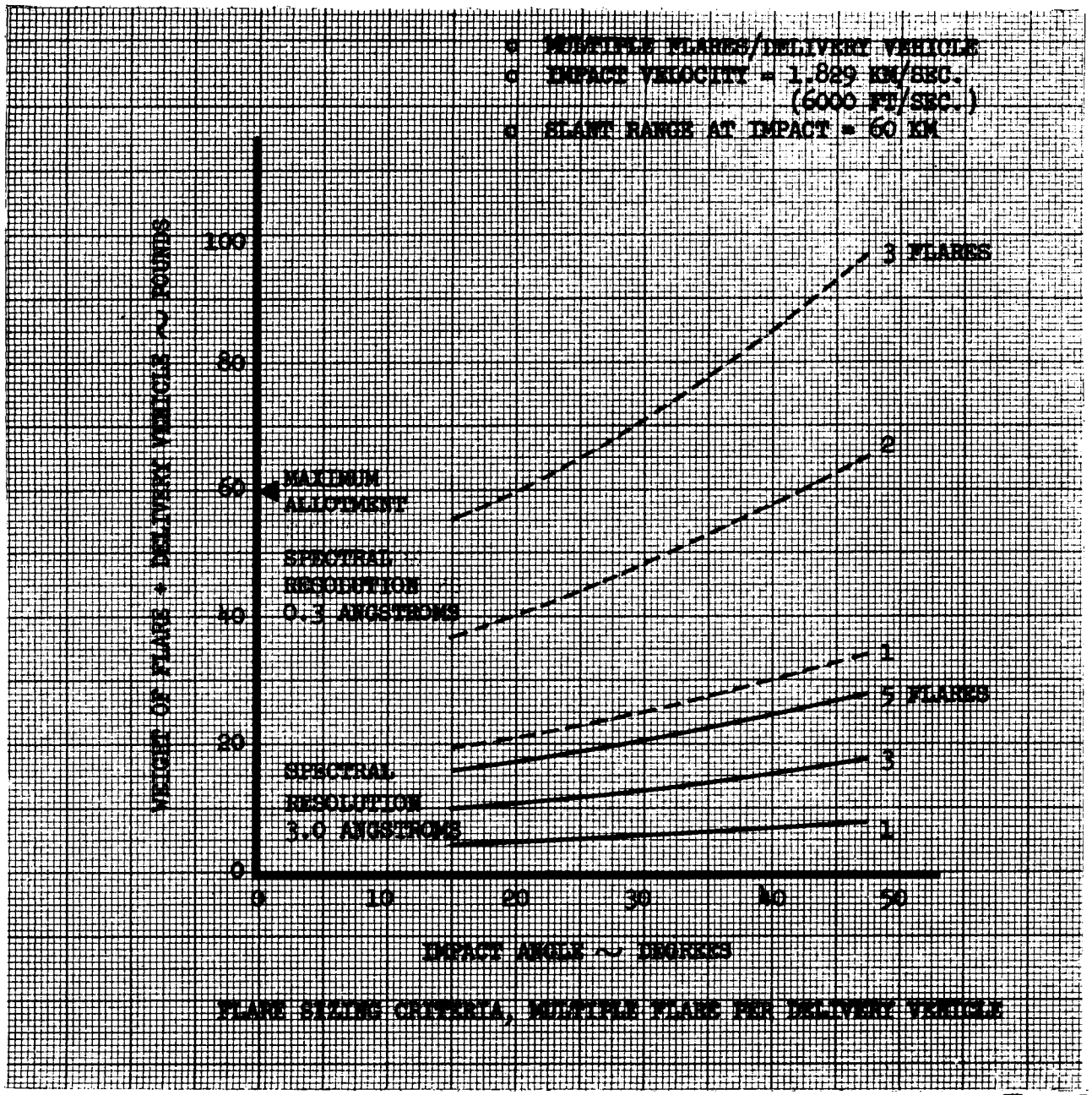
THE LARGE INCREASE IN DELIVERY VEHICLE SIZE FOR THE HIGHER RESOLUTION IS DUE TO THE MUCH LARGER FLARE WEIGHTS REQUIRED. NOTE THAT THE SHORTER IMPACT RANGES RESULT IN SMALLER DELIVERY VEHICLES. THIS IS AN APPARENT CONTRADICTION BECAUSE SHORTER RANGE REQUIRES HIGHER DEPLOYMENT VELOCITY INCREMENTS FOR GIVEN IMPACT ANGLE AND VELOCITY. HOWEVER, THE LARGE DECREASE IN FLARE SIZE ALLOWED AT SHORTER RANGES OVERRIDES THE VELOCITY INCREASE AND REDUCES THE DELIVERY ROCKET REQUIREMENTS.



FLARE SIZING CRITERIA, SINGLE FLARE PER DELIVERY VEHICLE

FLARE SIZING CRITERIA, MULTIPLE FLARE PER DELIVERY VEHICLE

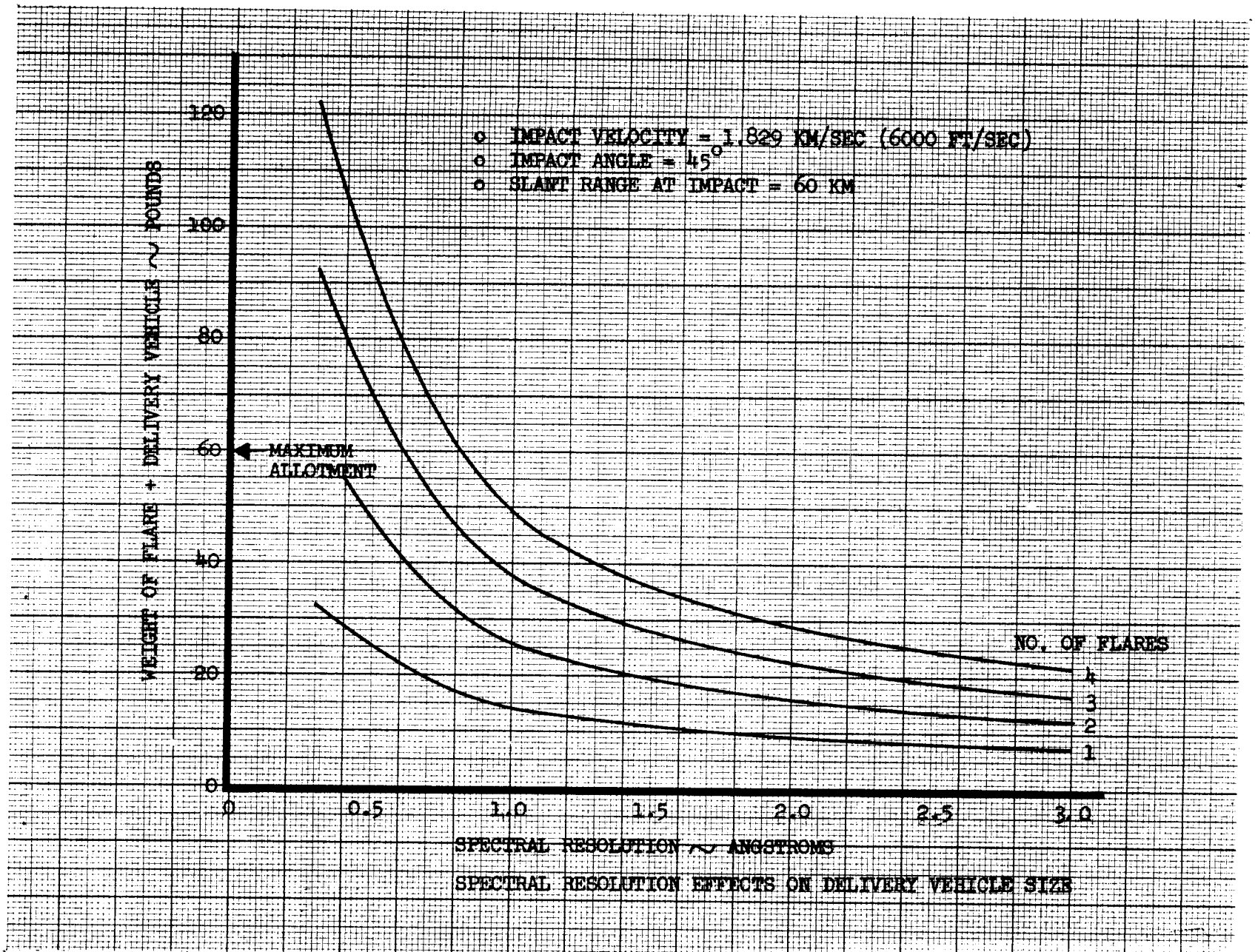
THIS CHART EXTENDS THE DATA SHOWN ON THE PREVIOUS CHART TO INCLUDE THE OPTION OF CARRYING MORE THAN ONE FLARE IN EACH DELIVERY VEHICLE. THE ENTIRE CHART IS BASED ON A CONSTANT SLANT RANGE AT IMPACT OF 60 KM.



FLARE SIZING CRITERIA, MULTIPLE FLARE PER DELIVERY VEHICLE

SPECTRAL RESOLUTION EFFECTS ON DELIVERY VEHICLE SIZE

IT IS SOMEWHAT UNLIKELY THAT EITHER EXTREME OF SPECTRAL RESOLUTION IS THE MOST DESIRABLE CHOICE FROM A SYSTEMS STANDPOINT. ALTHOUGH HIGH RESOLUTION IS DESIRABLE, VALUES AS LOW AS ONE ANGSTROM ARE MORE THAN ADEQUATE AND SIGNIFICANT REDUCTION IN DELIVERY VEHICLE SIZE IS POSSIBLE AT THESE RESOLUTIONS. TO ILLUSTRATE THE AVAILABLE GAINS IN DELIVERY VEHICLE SIZE, THE EFFECT OF SPECTRAL RESOLUTION VARIATION IS SHOWN HERE. DATA ARE BASED ON AN IMPACT ANGLE OF 45° AND AN IMPACT VELOCITY OF 6000 FEET PER SECOND. A SLANT RANGE OF 60 KM IS ASSUMED. NOTE THAT SIGNIFICANT REDUCTION IN DELIVERY VEHICLE SIZE IS ACHIEVED UNTIL RESOLUTIONS AS LOW AS ONE ANGSTROM ARE REACHED. MULTIPLE FLARES DELIVERED BY A SINGLE ROCKET AND BURSTING ALMOST SIMULTANEOUSLY WILL PRESENT DIFFICULTIES IN SENSING TIME TO FIRST PEAK. CONSEQUENTLY, NO ADDITIONAL CONSIDERATION OF THIS MODE IS MADE.

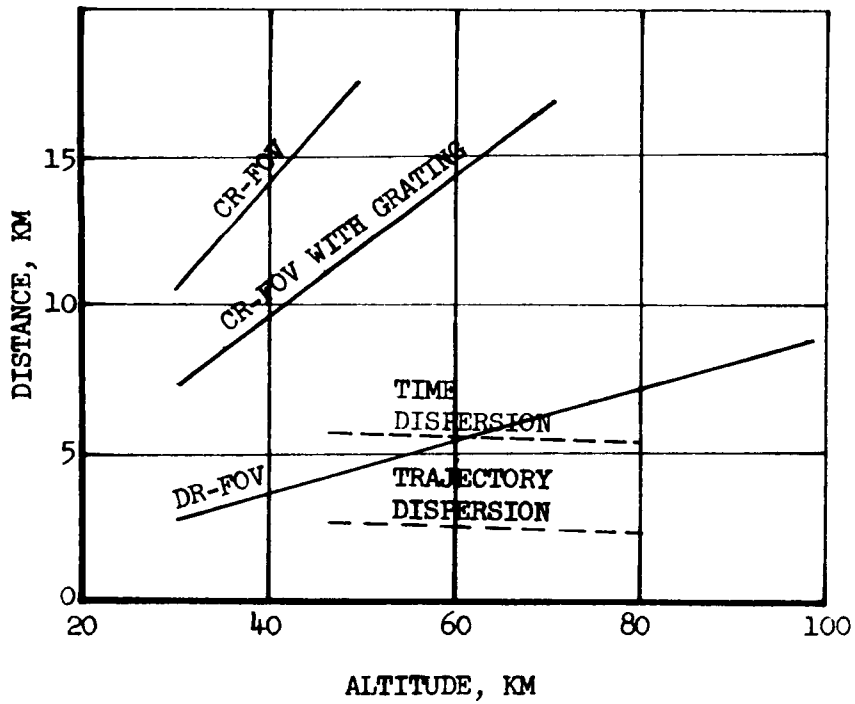


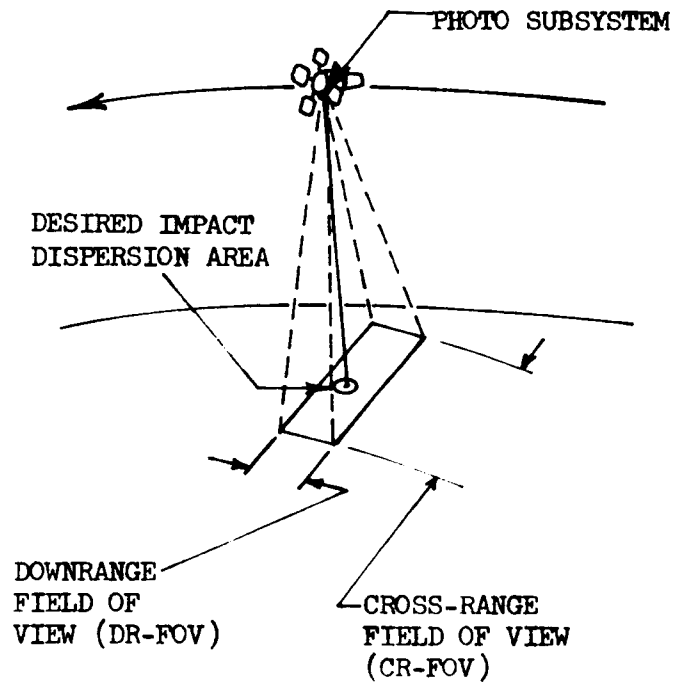
SPECTRAL RESOLUTION EFFECTS ON DELIVERY VEHICLE SIZE

FLARE DISPERSION REQUIREMENT FOR SPECTRAL DATA

ONE FINAL CRITERION IS CONSIDERED IN THE SELECTION OF A CANDIDATE FLARE DELIVERY SYSTEM; THAT OF TRAJECTORY DISPERSIONS DUE TO POINTING AND VELOCITY ERRORS IN THE DEPLOYMENT OF THE DELIVERY ROCKET. THIS CHART ILLUSTRATES THE NOMINAL DOWNRANGE AND CROSSRANGE FIELD OF VIEW AS A FUNCTION OF SPACECRAFT ALTITUDE. THIS FIELD OF VIEW MUST BE GREATER THAN THE TRAJECTORY DISPERSION TO INSURE THAT THE IMPACT IS SEEN. SINCE THE GRATING SPREADS THE SPECTRAL IMAGE ON A SIGNIFICANT PORTION OF THE FILM IN THE CROSSRANGE DIRECTION, THE AVAILABLE FIELD OF VIEW IS REDUCED. THE REDUCTION SHOWN IS BASED ON A RESOLUTION OF 0.7 ANGSTROMS.

THE 3-SIGMA TRAJECTORY DISPERSION SHOWN IS GOOD FOR BOTH THE DOWNRANGE AND CROSSRANGE DIRECTIONS. SINCE THIS IS THE DOMINANT CROSSRANGE ERROR, IT IS CLEAR THAT THE ALLOWABLE DISPERSION, REPRESENTED BY THE FIELD OF VIEW WITH GRATING, IS MORE THAN ADEQUATE. HOWEVER, AN ADDITIONAL DISPERSION EXISTS IN THE DOWNRANGE DIRECTION DUE TO THE UNCERTAINTY IN IMPACT TIME. THE ORBITER VELOCITY OF APPROXIMATELY 2 KM PER SECOND AT THE TIME OF IMPACT MAPS EACH SECOND OF IMPACT TIME UNCERTAINTY INTO AN ADDITIONAL 2 KM DOWNRANGE DISPERSION. THIS 3-SIGMA DISPERSION IS ALSO SHOWN ON THE CHART. THE ACTUAL TOTAL DISPERSION EXCEEDS ALLOWABLE FOR SPACECRAFT ALTITUDES LESS THAN 60 KM. ONE POSSIBLE MEANS OF ELIMINATING THIS CONSTRAINT WOULD BE TO TURN THE SPACECRAFT 90° TO THE LINE-OF-FLIGHT. THIS WOULD PLACE THE LARGE CROSSRANGE FIELD OF VIEW IN THE DIRECTION OF GREATEST DISPERSION AND RELIEVE THE DIFFICULTY.





FLARE DISPERSION REQUIREMENTS FOR SPECTRAL DATA

TYPICAL FLARE DELIVERY SYSTEM DEFINITION

BASED ON THE FLARE DELIVERY DATA PRESENTED, A TYPICAL FLARE DELIVERY SYSTEM HAS BEEN DEFINED. THE PARAMETERS WERE SELECTED TO RESULT IN THE GREATEST NUMBER OF FLARE DELIVERY VEHICLES POSSIBLE WHILE MAINTAINING A HIGH LEVEL OF SPECTRAL RESOLUTION.

TYPICAL FLARE DELIVERY SYSTEM DEFINITION

o IMPACT VELOCITY	- 1.829 KM/SEC. (6000 FT./SEC.)
o IMPACT ANGLE	- 45°
o SLANT RANGE AT IMPACT	- 60 KM
o SPECTRAL RESOLUTION	- 0.7 ANGSTROMS
o FLARE WEIGHT	- 6.8 POUNDS
o DELIVERY VEHICLE PLUS FLARE WEIGHT	- 19 POUNDS
o INITIAL LUNAR ORBITER WEIGHT (WITH 3 DELIVERY VEHICLES)	- 947 POUNDS
o ADDITIONAL PARAMETERS :	
- DEPLOYMENT VELOCITY	- 1.33 KM/SEC.
- FLIGHT TIME	- 44 SECONDS
- ORBITER POSITION AT DEPLOYMENT	- 12.7° FROM PERILUNE
- ORBITER POSITION AT FLARE IMPACT	- 10° FROM PERILUNE AND .88° AHEAD OF IMPACT POINT
- VIEW ANGLE	- -26.6° FROM VERTICAL
- FIRING ANGLE	- 119.5°
- FIRING TUBE ORIENTATION	- ALONG CAMERA SPECTROGRAPH AXIS

BALLOON

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BALLOON REQUIREMENTS

THE REQUIREMENTS DEFINED FOR THE BALLOON MISSION ARE LISTED. THESE REQUIREMENTS WERE ESTABLISHED BY NASA LRC AT THE INITIATION OF THE STUDY.

BALLOON REQUIREMENTS

- IMPACT VELOCITY, 6000 TO 8000 FT./SECOND.
- IMPACT ANGLE > 30° TO SURFACE HORIZONTAL.
- DATA RETRIEVAL BY HIGH RESOLUTION PHOTOS BEFORE, DURING AND AFTER IMPACT.
- DEVICE IS A 25 FOOT DIAMETER MYLAR BALLOON, .5 MIL THICK, WEIGHT IS 8 POUNDS.
- INFLATE IN SUN AND $T_{INFLATE}$ > 3 SECONDS.
- LAUNCHED FROM TUBE ON SPACECRAFT.

LUNAR ORBITER BUS WEIGHT
BALLOON MISSION

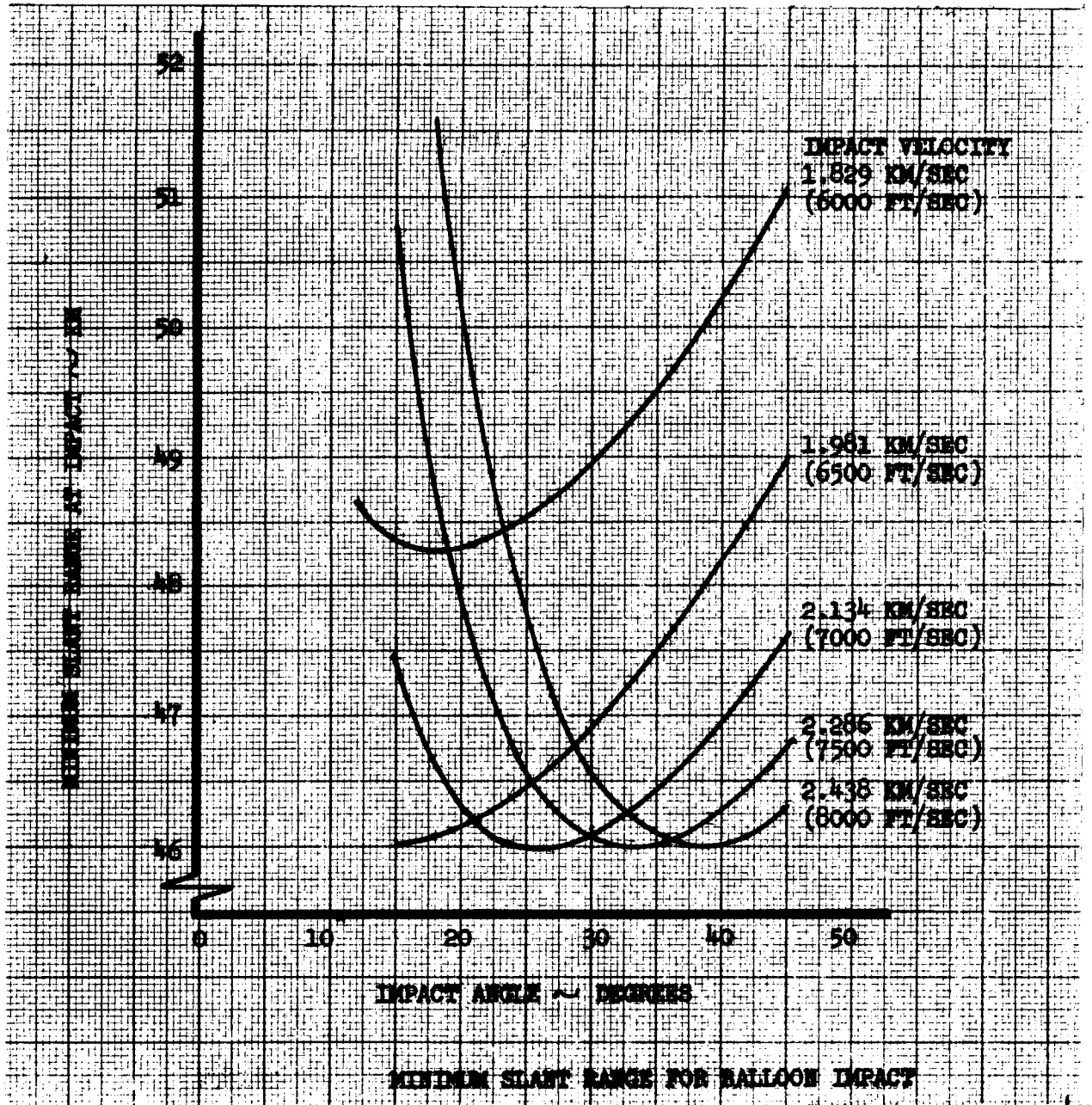
THE MAXIMUM ALLOWABLE WEIGHT ALLOTMENT FOR BALLOON DELIVERY VEHICLES IS DEFINED HERE. THIS ALLOTMENT REPRESENTS THE ALLOWABLE WEIGHT REMAINING AFTER THE STRIPPED LUNAR ORBITER AND THE ADDITIONAL REQUIRED EQUIPMENT ARE REMOVED FROM THE MAXIMUM ALLOWABLE BUS WEIGHT.

LUNAR ORBITER BUS WEIGHT
BALLOON MISSION

STRIPPED LUNAR ORBITER WEIGHT	840 POUNDS
ADDITIONAL EQUIPMENT FOR BALLOON MISSION DEPLOYMENT DEVICE AND MOUNTING STRUCTURE	20
LUNAR ORBITER BUS WEIGHT (BALLOON)	860 POUNDS
MAXIMUM ALLOWABLE BUS WEIGHT	950
MAXIMUM ALLOTMENT FOR DELIVERY VEHICLES	90 POUNDS

MINIMUM SLANT RANGE FOR BALLOON IMPACT

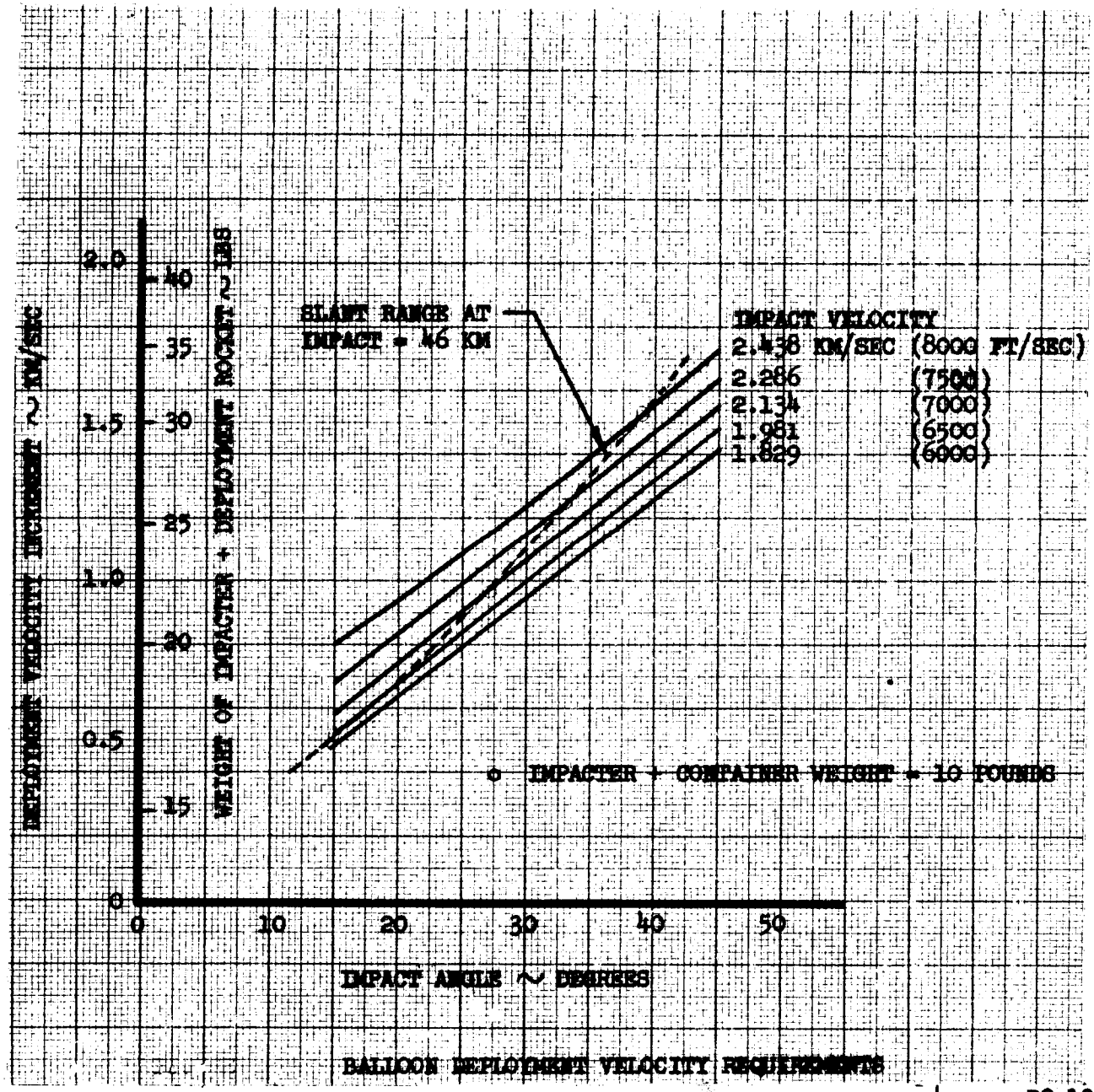
THE DELIVERY MECHANICS REQUIRED FOR THE BALLOON MISSION ARE ASSUMED IDENTICAL TO THE FLARE. HOWEVER, THE HIGH RESOLUTION PHOTOGRAPHIC REQUIREMENT DICTATES A NEED FOR VERY SHORT SLANT RANGE BETWEEN THE SPACECRAFT AND THE BALLOON AT IMPACT. THIS FACTOR PROMPTED DETERMINATION OF THE MINIMUM POSSIBLE SLANT RANGE OBTAINABLE. OBVIOUSLY, THIS WILL BE THE 46 KM PERILUNE ALTITUDE OF THE LUNAR ORBITER. HOWEVER, THIS REPRESENTS A UNIQUE SOLUTION REQUIRING THE SPACECRAFT TO BE AT PERILUNE AND DIRECTLY OVERHEAD WHEN THE IMPACT OCCURS. THIS CHART ILLUSTRATES THE MINIMUM SLANT RANGE ASSOCIATED WITH THE DESIRED IMPACT ANGLES AND VELOCITIES. NOTE THAT ONLY PARTICULAR COMBINATIONS OF IMPACT VELOCITY AND ANGLE RESULT IN 46 KM SLANT RANGE. IN FACT, NO SOLUTION IS POSSIBLE FOR AN IMPACT VELOCITY OF 6000 FEET PER SECOND. HOWEVER, LITTLE DEGRADATION IN RESOLUTION WOULD RESULT FROM ANY OF THE SLANT RANGES SHOWN.



MINIMUM SLANT RANGE FOR BALLOON IMPACT

BALLOON DEPLOYMENT VELOCITY REQUIREMENTS

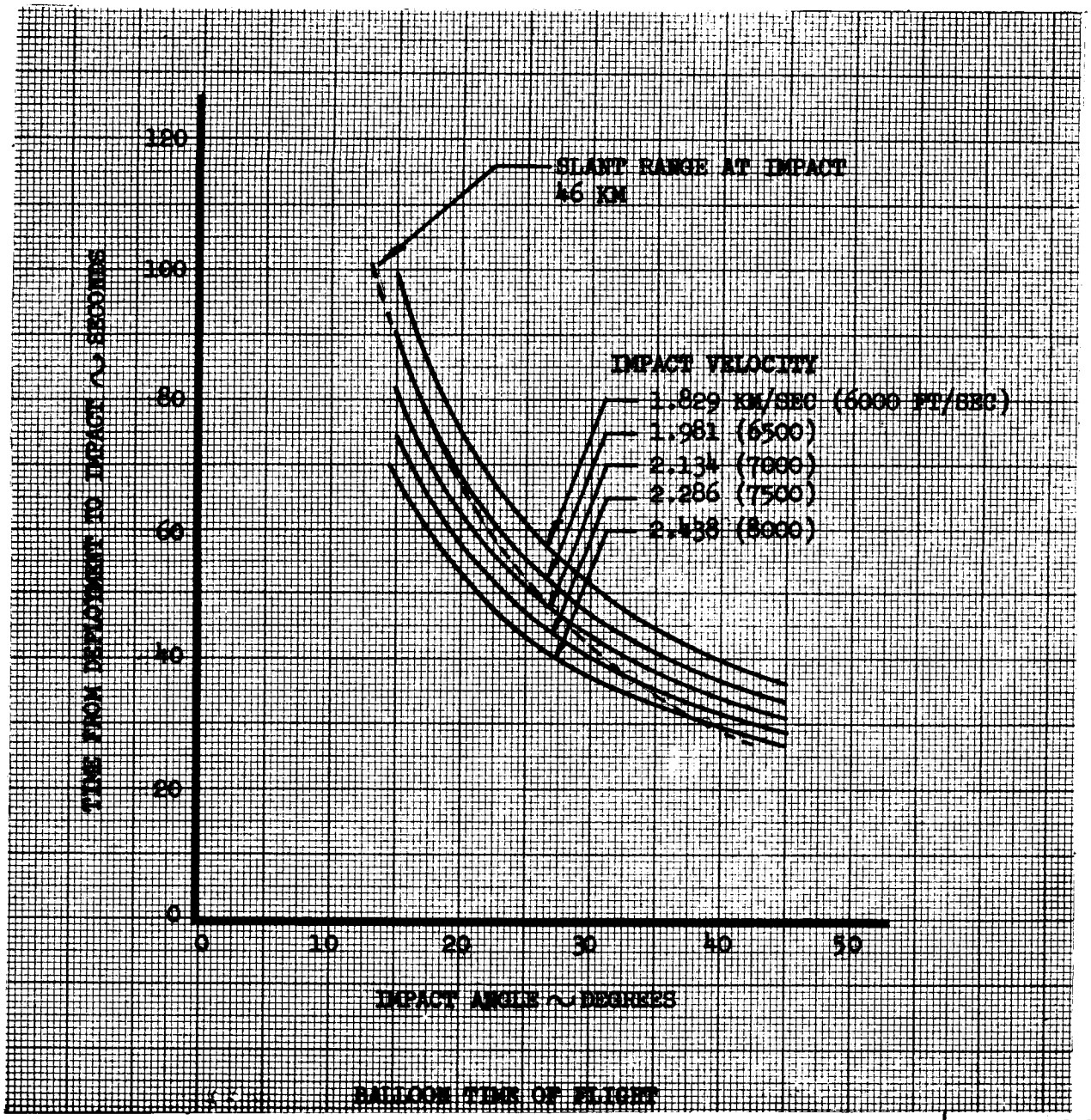
THE REQUIRED DEPLOYMENT VELOCITIES CORRESPONDING TO THE MINIMUM SLANT RANGES SHOWN ON THE PREVIOUS CHART ARE GIVEN HERE. THE LOCUS OF IMPACT CONDITIONS FOR THE 46 KM SLANT RANGE IS SUPERIMPOSED. SINCE ONLY ONE BALLOON WEIGHT IS CONSIDERED, THE DEPLOYMENT VELOCITIES REPRESENT SPECIFIC DEPLOYMENT ROCKET SIZES. THE ROCKETS WERE SIZED UNDER THE SAME ASSUMPTIONS AS THE FLARE DELIVERY ROCKETS. THE WEIGHT OF THE BALLOON PLUS DELIVERY VEHICLE REQUIRED FOR ANY DEPLOYMENT VELOCITY INCREMENT IS INCLUDED ON THE ORDINATE.



BALLOON DEPLOYMENT VELOCITY REQUIREMENTS

BALLOON TIME OF FLIGHT

THE TIME FROM BALLOON DEPLOYMENT TO IMPACT FOR THE COMBINATIONS OF IMPACT CONDITIONS STUDIED IS GIVEN HERE. THE TIMES ARE QUITE SHORT, RANGING BELOW 60 SECONDS IN MOST CASES. CONSEQUENTLY, AS IN THE CASE OF THE FLARE DELIVERIES, THE CAMERA LENS MUST BE ORIENTED ALONG THE IMPACT LINE-OF-SIGHT AT THE TIME OF DEPLOYMENT.



BALLOON TIME OF FLIGHT

BALLOON IMPACT ERRORS

THE ERRORS PRESENTED REFER TO TWO TYPICAL BALLOON TRAJECTORIES WITH A SLANT RANGE BETWEEN THE SPACECRAFT AND THE IMPACT POINT OF 46 KM. THE PROCEDURE UTILIZED TO OBTAIN THESE IS THE SAME AS THAT DESCRIBED EARLIER FOR THE FLARE. IN GENERAL, THE MAGNITUDE OF THE ERRORS IS SIMILAR TO THE FLARE DATA.

BALLOON IMPACT ERRORS

- FIXED TRAJECTORY PARAMETER

  SLANT RANGE = 46 KM

- ERROR SOURCES (3 σ)

  POINTING ERROR = 0.88°

  ΔV SCALE FACTOR = 0.6%

- ERRORS AT IMPACT

	AT IMPACT VELOCITY = 2.44 KM/SEC. (8000 FT/SEC) AND IMPACT ANGLE = 39°	AT IMPACT VELOCITY = 2.13 KM/SEC. (7000 FT/SEC.) AND IMPACT ANGLE = 26°
DOWNRANGE (KM) :	1.08	1.25
CROSS RANGE (KM) :	0.71	0.71
ALTITUDE RATE (M/SEC.) :	9.24	5.54
DOWNRANGE RATE (M/SEC.) :	13.33	13.75
CROSS RANGE RATE (M/SEC.) :	23.53	14.32
IMPACT TIME (SEC.) :	0.2	0.3

Revision 1, Dated 7-29-66

TYPICAL BALLOON DELIVERY SYSTEM DEFINITION

BASED ON THE BALLOON DELIVERY DATA PRESENTED, A TYPICAL BALLOON DELIVERY SYSTEM HAS BEEN DEFINED. THE PARAMETERS WERE SELECTED TO RESULT IN THE GREATEST NUMBER OF BALLOON DELIVERY VEHICLES POSSIBLE WHILE MAINTAINING A MINIMUM SLANT RANGE BETWEEN THE SPACECRAFT AND THE IMPACT POINT.

TYPICAL BALLOON DELIVERY SYSTEM DEFINITION

o IMPACT VELOCITY	-	1.829 KM/SEC. (6000 FT./SEC.)
o IMPACT ANGLE	-	30°
o SLANT RANGE AT IMPACT	-	48.95 KM
o BALLOON WEIGHT	-	8 LBS. PLUS 2 POUND CONTAINER
o DELIVERY VEHICLE PLUS BALLOON WEIGHT	-	21.9 POUNDS
o INITIAL LUNAR ORBITER WEIGHT (WITH 4 DELIVERY VEHICLES)	-	947.6 POUNDS
o ADDITIONAL PARAMETERS :		
- DEPLOYMENT VELOCITY	-	0.95 KM/SEC.
- FLIGHT TIME	-	51 SECONDS
- ORBITER POSITION AT DEPLOYMENT	-	2.75° FROM PERILUNE
- ORBITER POSITION AT BALLOON IMPACT	-	0.4° BEYOND PERILUNE AND 0.55° AHEAD OF IMPACT POINT
- VIEW ANGLE	-	-19.8° FROM VERTICAL
- FIRING ANGLE	-	113°
- FIRING TUBE ORIENTATION	-	ALONG CAMERA AXIS

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SPHERICAL PENETROMETERS

THE PRINCIPAL FEATURES OF THE PENETROMETER STUDY ARE LISTED.

SPHERICAL PENETROMETERS

- DEVELOPED BY AERONUTRONIC UNDER CONTRACT NAS 1-4923.
- ACCELEROMETER, BATTERY, AND TRANSMITTER ENCLOSED BY BALSA COVER.
- WEIGHT 5 POUNDS, 8-1/2 INCH DIAMETER.
- IMPACT VELOCITY DESIRED -- 150' / SECOND.
- KNOWLEDGE OF VELOCITY AND ANGLE 10% OR BETTER DESIRED.
- 1/2 WATT TRANSMITTER

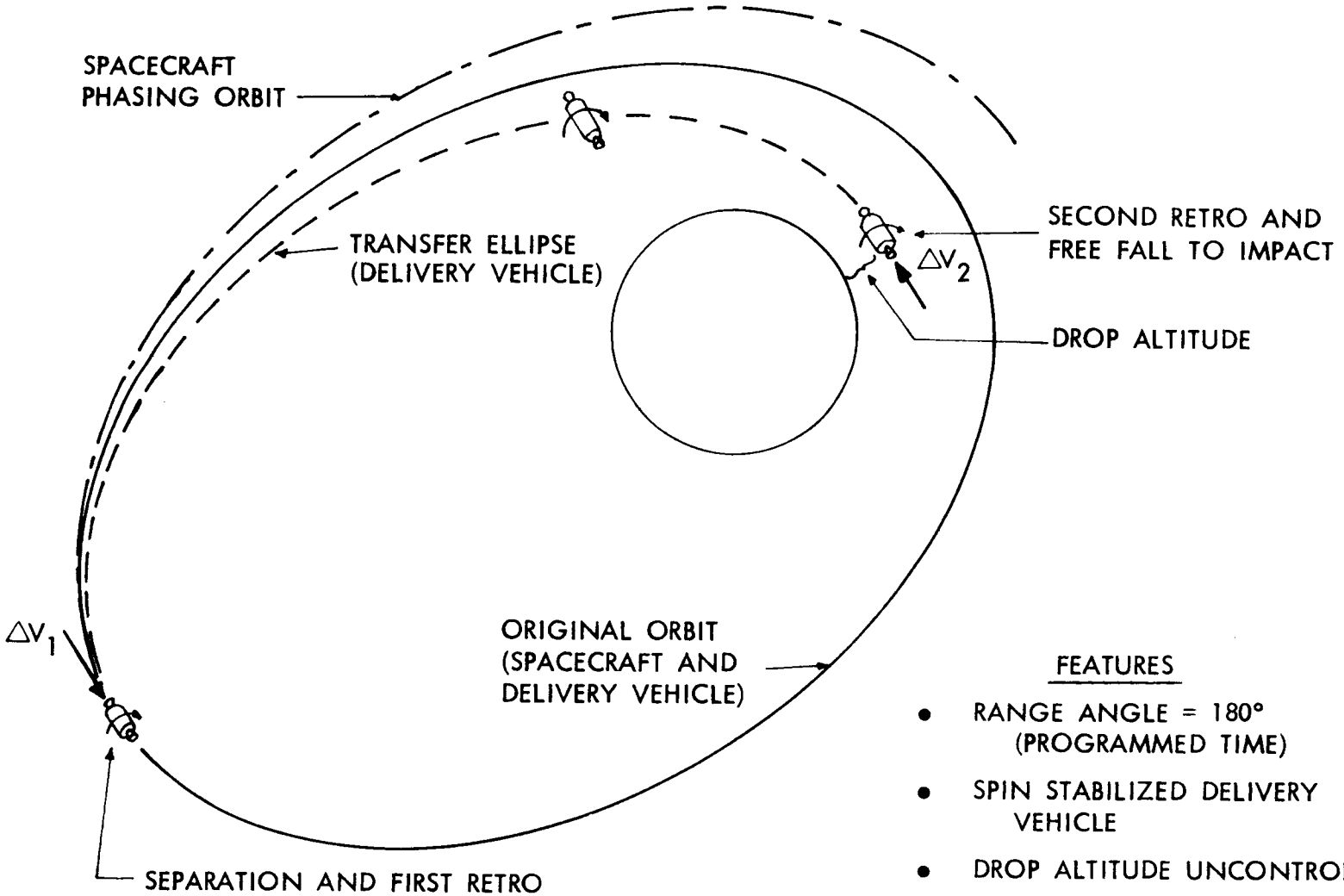
DELIVERY CONCEPTS STUDIED

- "SIMPLE" SPIN STABILIZED SYSTEM
- SPIN STABILIZED "PROXIMITY FUZE" SYSTEM WITH VELOCITY INCREMENT CONTROL.
- CONTROLLED DESCENT USING ALTITUDE MARKS -- "SPACE GENERAL" CONCEPT UNDER CONTRACT NAS 9-4897.

SPIN STABILIZED PROGRAMMED DELIVERY

THE SIMPLEST SCHEME FOR SEMISOFT LANDING OF PAYLOADS FROM ORBIT IS THE SPIN STABILIZED, PROGRAMMED DELIVERY CONCEPT. THE DELIVERY VEHICLE IS SPUN UP AND SEPARATED AT APOLUNE. AT THE SAME TIME OR IMMEDIATELY AFTER, DEPENDING ON THE SEPARATION TECHNIQUE, THE FIRST RETRO IS FIRED WHICH PUTS THE DELIVERY VEHICLE INTO A TRANSFER TRAJECTORY HAVING A PERILUNE ALTITUDE EQUAL TO THE DESIRED PENETROMETER DROP ALTITUDE. THE ORBITING SPACECRAFT MAY REQUIRE A POSIGRADE VELOCITY CHANGE TO BOOST IT INTO A PHASING ORBIT SO THAT IT WILL BE IN A POSITION TO RECEIVE THE ACCELEROMETER SIGNALS AT IMPACT. AT DELIVERY VEHICLE PERILUNE THE THRUST AXIS IS ONCE AGAIN ALIGNED WITH THE VELOCITY VECTOR AND NORMAL TO THE LOCAL VERTICAL. IN RESPONSE TO A PREPROGRAMMED TIMING SIGNAL, THE SECOND RETRO (LOCATED AT THE OPPOSITE END OF THE VEHICLE FROM THE FIRST) IS FIRED TO REMOVE THE ORBITAL VELOCITY AND ALLOW THE VEHICLE TO FREE-FALL TO THE SURFACE.

SPIN STABILIZED PROGRAMMED DELIVERY



FEATURES

- RANGE ANGLE = 180° (PROGRAMMED TIME)
- SPIN STABILIZED DELIVERY VEHICLE
- DROP ALTITUDE UNCONTROLLED

ERRORS FOR SPIN STABILIZED PROGRAMMED DELIVERY

VARIATIONS IN PENETROMETER IMPACT VELOCITY ARE DUE TO DISPERSIONS IN DROP ALTITUDE AND TO RESIDUAL VELOCITY AT THE START OF FREE-FALL. SINCE THE DROP ALTITUDE IS THE DIFFERENCE BETWEEN THE ORBITAL RADIUS AND THE "ALTITUDE" OF THE LUNAR SURFACE, ERRORS IN EITHER WILL CONTRIBUTE TO IMPACT VELOCITY DISPERSIONS. THE MAGNITUDE OF THE FIRST RETRO-VELOCITY CHANGE CAN BE CONTROLLED TO $\pm 1\%$. HOWEVER, DUE TO THE HIGH SENSITIVITY OF DESCENT TRAJECTORY PERILUNE (5500 METERS PER M/SECOND OF VELOCITY ERROR), THIS RESULTS IN AN APPRECIABLE ERROR IN DROP ALTITUDE. KNOWLEDGE OF THE ALTITUDE OF THE LUNAR SURFACE VARIES WITH LOCATION. IT IS ASSUMED THAT LOCAL IRREGULARITIES WILL BE UNPREDICTABLE AND THAT THE MEAN ALTITUDE FOR A GIVEN TARGET AREA WILL BE KNOWN TO A 3σ UNCERTAINTY OF ± 600 METERS BY THE TIME THE PENETROMETER MISSION TAKES PLACE. ORBITAL TRACKING AND INJECTION ACCURACIES ARE CURRENT LUNAR ORBITER ESTIMATES. CONTRIBUTING DIRECTLY TO THE IMPACT VELOCITY ERROR IS THE VERTICAL COMPONENT OF THE SECOND RETRO-VELOCITY CHANGE WHICH IS DUE TO THE THRUST POINTING ERRORS ASSOCIATED WITH VEHICLE ATTITUDE ORIENTATION AND THRUST MISALIGNMENTS.

IT WAS CONCLUDED THAT THIS DELIVERY TECHNIQUE LACKED THE DESIRED PRECISION IN IMPACT VELOCITY.

ERRORS FOR SPIN STABILIZED PROGRAMMED DELIVERY

NOMINAL IMPACT VELOCITY

150 ± 15 FT./SEC. (± 10%) = 45.7 ± 4.57 M/SEC.

NOMINAL DROP ALTITUDE

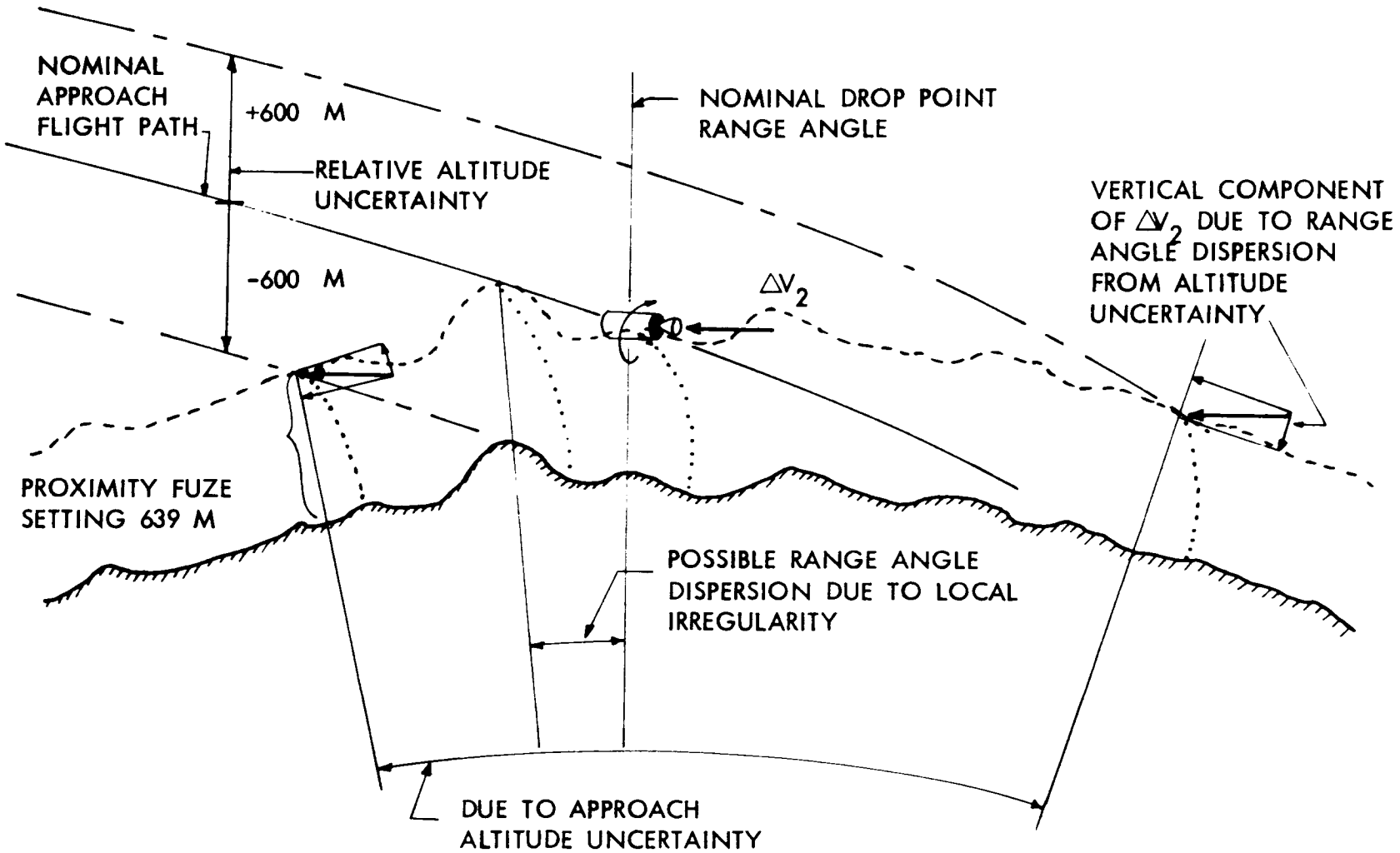
$2100 \begin{smallmatrix} +430 \\ -400 \end{smallmatrix}$ FT. = $639 \begin{smallmatrix} +132 \\ -121 \end{smallmatrix}$ METERS

ERROR SOURCE	CURRENTLY ACHIEVABLE ACCURACIES 3σ	
	DROP ALTITUDE	IMPACT VELOCITY
ΔV_1 MAGNITUDE (± 1% OF 8 M/SEC.)	± 440 METERS	
ΔV_1 DIRECTION	0	
LOCAL TOPOGRAPHICAL IRREGULARITIES	± 100	
"ALTITUDE" OF LUNAR SURFACE	± 600	
ORIGINAL ORBIT TRACKING ACCURACY	± 300	
ORIGINAL ORBIT TOLERANCE AFTER ADJUSTMENT (± 5 KM WITHOUT ADJUSTMENT)	± 150	
ROOT - SUM - SQUARE	± 822 METERS	+ 23.1 (-56) M/SEC.
ΔV_2 DIRECTION (1957 M/SEC. SIN .88°)		± 30
RSS OF IMPACT VELOCITY ERRORS		± 37.9 M/SEC.

SPIN STABILIZED, PROXIMITY FUZE CONCEPT

THIS CONCEPT IS SIMILAR TO THE PROGRAMMED DELIVERY EXCEPT THAT TRUE ALTITUDE SENSING IS USED TO CONTROL THE SECOND RETRO-FIRING. IN THIS WAY THE MAJOR SOURCES OF DROP ALTITUDE UNCERTAINTY ARE ELIMINATED BUT THE IMPACT POINT IS LESS PREDICTABLE. THE VEHICLE IS SPIN STABILIZED AND TRUE ALTITUDE SENSING IS BY MEANS OF A SIMPLIFIED RADAR ALTIMETER OR PROXIMITY FUZE TO INDICATE WHEN THE DESIRED DROP ALTITUDE IS REACHED. THE EXAGGERATED FEATURES OF THE DIAGRAM ILLUSTRATE THE SOURCES OF ERROR IN THIS DELIVERY CONCEPT. IT CAN BE SEEN THAT LOCAL SURFACE IRREGULARITIES OR ORBITAL ALTITUDE UNCERTAINTIES WHEN COMBINED WITH A LOW APPROACH ANGLE CAN CAUSE LARGE DISPERSIONS IN RANGE ANGLE. BECAUSE THE VEHICLE IS INERTIALLY ORIENTED, THESE RANGE ANGLE DISPERSIONS BECOME THRUST POINTING ERRORS WITH RESPECT TO THE LOCAL LUNAR HORIZONTAL. THE RESULTING VERTICAL COMPONENTS OF THE SECOND RETRO-VELOCITY CONTRIBUTE DIRECTLY TO IMPACT VELOCITY ERRORS. IN ADDITION, THERE IS A DROP ALTITUDE UNCERTAINTY BECAUSE THE PENETROMETERS IMPACT AT A POINT OTHER THAN THAT WHICH GIVES THE ALTITUDE INDICATION.

SPIN STABILIZED, PROXIMITY FUZE CONCEPT



MAJOR ERRORS FOR SPIN STABILIZED PROXIMITY FUZE CONCEPT

THIS CHART PRESENTS A TABULATION OF THE ERRORS ILLUSTRATED IN THE PREVIOUS CHART. IT IS APPARENT THAT THIS DELIVERY CONCEPT DOES NOT MEET THE PENETROMETER IMPACT VELOCITY ACCURACY REQUIREMENT.

IN ADDITION, IT APPEARS THAT IMPACT POINT UNCERTAINTIES WOULD REQUIRE COMMUNICATION CAPABILITY OVER A GREATER RANGE THAN AVAILABLE FROM THE 1/2 WATT TRANSMITTER ON BOARD THE PENETROMETER.

MAJOR ERRORS FOR SPIN STABILIZED PROXIMITY FUZE CONCEPT

	ALTITUDE	VERTICAL VELOCITY
DROP ALTITUDE DISPERSIONS :		
o PROXIMITY FUZE ACCURACY (5%)	32.0 M	10.4 M/SEC.
o LOCAL SURFACE IRREGULARITIES (PENETROMETERS IMPACT BEYOND ALTITUDE SENSING POINT)	50.0	12.7
THRUST POINTING DISPERSIONS :		
o RANGE ANGLE UNCERTAINTY DUE TO APPROACH ALTITUDE DISPERSION (± 600 M)		3.0 *
o ATTITUDE CONTROL ERRORS (.88°)		30.7 *
RSS		34.0 M/SEC.

REQUIREMENT : $V_z = 45.7 \pm 4.6$ M/SEC.

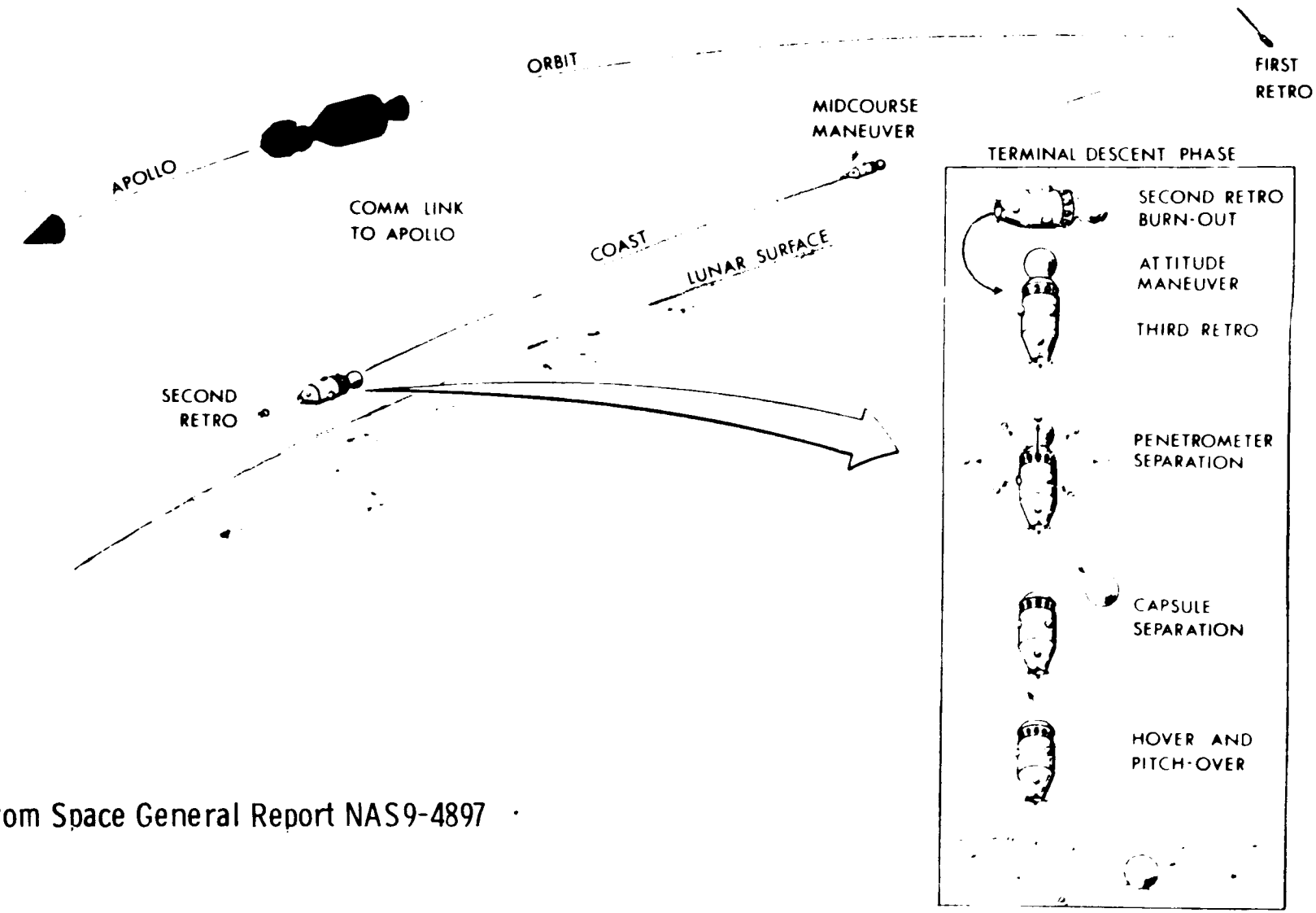
ERRORS NOT ASSESSED ARE : HORIZONTAL VELOCITY ERRORS AND THE EFFECT ON RANGE ANGLE OF LOCAL SURFACE IRREGULARITIES.

* ASSUMES : $\Delta V_2 = 2000$ M/SEC. AND FLIGHT PATH ANGLE = 13°

LSP (SPACE GENERAL) DELIVERY SEQUENCE

ONE CLASS OF PENETROMETER DELIVERY METHODS IS TYPIFIED IN THE SPACE GENERAL CORPORATION SCHEME FOR THE SEMISOFT LANDING OF LUNAR PAYLOADS FROM THE APOLLO CSM. THE DELIVERY VEHICLE IS RATHER SOPHISTICATED, HAVING SUCH FEATURES AS: ACTIVE 3-AXIS ATTITUDE CONTROL, RADAR FOR TRUE ALTITUDE SENSING, AND CLOSED-LOOP VELOCITY CONTROL. THE FIRING OF THE FIRST RETRO-MOTOR TAKES PLACE SHORTLY AFTER SEPARATION FROM THE PARENT VEHICLE. A LINEAR ACCELEROMETER MONITORS THE VELOCITY INCREMENT AND PROVIDES A CUTOFF SIGNAL FOR THE LIQUID MONOPROPELLANT VERNIER ENGINES. WHILE LOSING ALTITUDE IN A TRANSFER TRAJECTORY, A 180° ATTITUDE MANEUVER IS PERFORMED TO ORIENT THE SECOND SOLID RETRO-MOTOR. AT THE PROGRAMMED TIME, SECOND RETRO-FIRING IS INITIATED. THE SOLID PROPELLANT MOTOR PROVIDES MOST OF THE $\triangle V$ AND THE LIQUID VERNIER ENGINES PROVIDE THE PRECISE CUTOFF CONTROL. ATTITUDE IS CONTROLLED BY A LOW LEVEL COLD GAS N_2 SYSTEM AND A HIGH LEVEL MONOPROPELLANT SYSTEM IN PITCH AND YAW. AT BURNOUT, THE SECOND SOLID RETRO-CASE IS JETTISONED, THE VEHICLE UNDERGOES A 90° ATTITUDE MANEUVER AND THE ALTITUDE MARKING UNIT IS ACTIVATED FOR THE VERNIER DESCENT PHASE OF THE MISSION.

LSP DELIVERY SEQUENCE



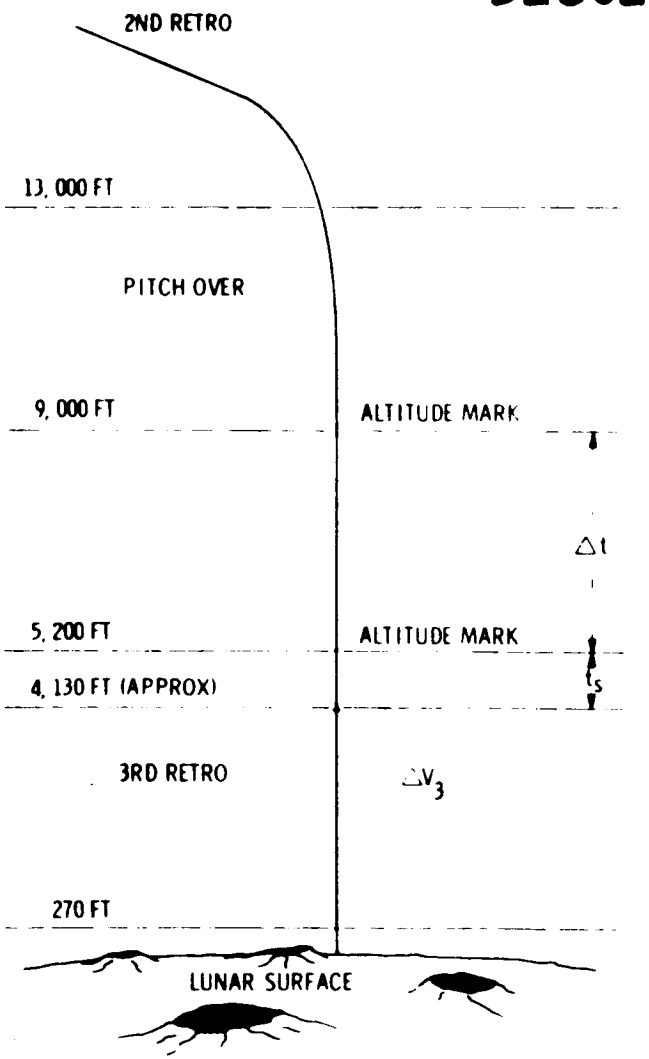
From Space General Report NAS9-4897

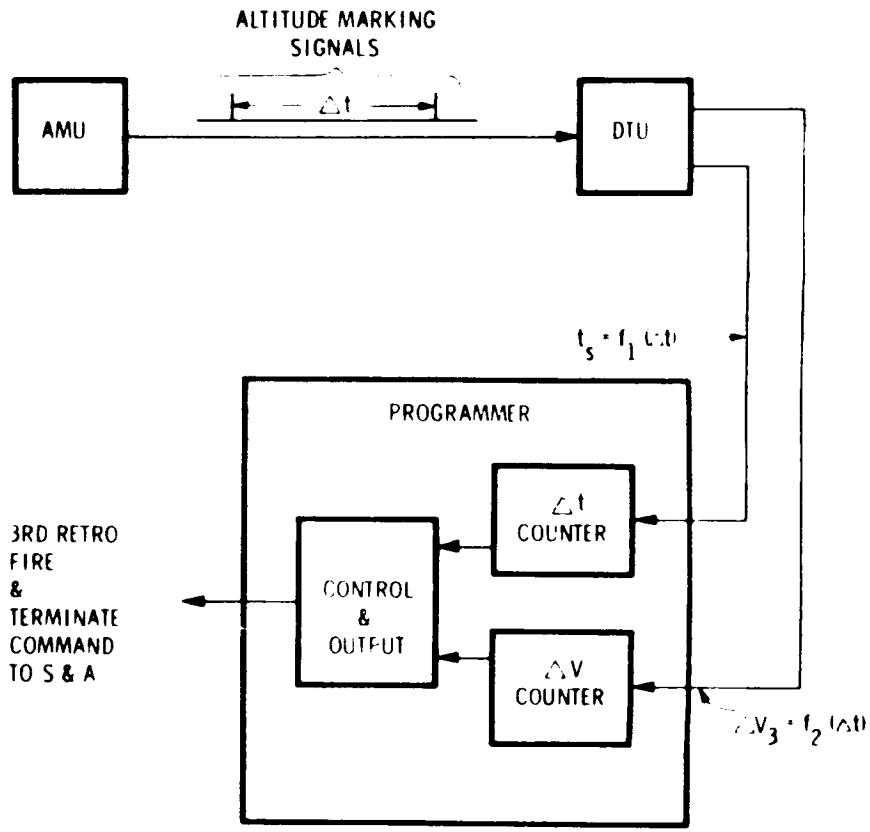
LSP (SPACE GENERAL) GUIDANCE & CONTROL
DESCENT CONTROL

WITH THE COLD GAS REACTION CONTROL SYSTEM PROVIDING ATTITUDE CONTROL, THE DELIVERY VEHICLE FALLS TOWARD THE LUNAR SURFACE. THE ALTITUDE MARKING UNIT GIVES A SIGNAL AS TWO PRESET ALTITUDE LEVELS ARE REACHED. USING THIS INFORMATION, THE REQUIRED VALUE OF $\triangle V_3$ IS COMPUTED AS WELL AS THE DELAY TIME AFTER THE SECOND ALTITUDE MARK FOR PROPER FIRING OF THE THIRD RETRO-ENGINE. THESE ARE THE SAME ENGINES USED FOR VELOCITY TRIM IN THE FIRST TWO RETRO-FIRINGS. AS BEFORE, THE VELOCITY INCREMENT IS MONITORED BY THE LINEAR ACCELEROMETER AND PENETROMETER DEPLOYMENT IS SIGNALED WHEN THE COMPUTED VALUE OF $\triangle V_3$ IS ACHIEVED. WITH THIS TECHNIQUE, THE EFFECTS OF THE MAJOR UNCERTAINTIES IN VEHICLE AND TRAJECTORY PARAMETERS ON IMPACT CONDITIONS ARE ELIMINATED.

LSP GUIDANCE & CONTROL

DESCENT CONTROL





From Space General Report NAS9-4897

ERRORS FOR SPACE GENERAL CONCEPT

AMONG THOSE ERRORS WHICH ARE NOT ELIMINATED IN THE SPACE GENERAL DESCENT CONTROL TECHNIQUE ARE VARIATIONS IN THIRD STAGE IGNITION WEIGHT, THIRD STAGE THRUST, AND ALTITUDE MARK SETTINGS. DEVIATIONS FROM THE NOMINAL VALUES USED IN THE COMPUTATIONS OF ΔV_3 AND IGNITION TIME WILL RESULT IN DEPARTURES FROM THE NOMINAL DESCENT PROFILE. THE ACHIEVEMENT OF ZERO VERTICAL VELOCITY AND PENETROMETER RELEASE AT PRECISELY THE DESIRED DROP ALTITUDE IS NOT ASSURED. THE VERNIER DESCENT CONTROL SCHEME DOES AFFECT HORIZONTAL VELOCITY COMPONENTS AND THESE ARE CARRIED THROUGH TO IMPACT. THE MAJOR SOURCES OF HORIZONTAL VELOCITY ERRORS ARE THE DIFFERENCES BETWEEN THE REQUIRED AND DELIVERED VALUE OF ΔV_2, AND THE ANGULAR MISALIGNMENT OF THE THRUST VECTOR DURING ENGINE FIRING.

THE DISPERSION IN ELAPSED TIME FROM SEPARATION OF DELIVERY VEHICLE TO PENETROMETER IMPACT IS OF IMPORTANCE BECAUSE OF THE EFFECT ON COMMUNICATION RANGE. THE DISPERSIONS IN TIME PRIOR TO ΔV_2 BURNOUT ARE COVERED IN ANOTHER CHART. THE INCREASED DESCENT TIME OVER NOMINAL ASSUMES A WORST CASE HIGH VALUE FOR ΔV_2 BURNOUT ALTITUDE (NOMINAL <u>PLUS</u> THE ROOT-SUM-SQUARE OF THE TOLERANCES) AND ALSO ASSUMES THE 3σ VERTICAL VELOCITY DISPERSION IS UPWARD. THE NOMINAL DESCENT TIME IS BASED ON A NOMINAL ΔV_2 BURNOUT ALTITUDE OF 4.6 KM AND A DROP ALTITUDE OF 639 METERS. THE VALUE OF 107.5 SECONDS, USED IN TRAJECTORY COMPUTATIONS, IS BASED ON THE INITIAL ALTITUDE OF 3.96 KM (13,000 FEET) USED IN THE SPACE GENERAL REPORT AND THE 639 METER DROP ALTITUDE OF THE CURRENT STUDY. IT WAS CONCLUDED THAT THIS DELIVERY TECHNIQUE SATISFIES THE IMPACT REQUIREMENTS FOR THE PENETROMETERS.

ERRORS FOR SPACE GENERAL CONCEPT

ERROR SOURCE	IMPACT VELOCITY DISPERSIONS $\Delta\dot{X}$	$\Delta\dot{Y}$	$\Delta\dot{Z}$
INITIAL SPACECRAFT ALIGNMENT	.37	2.88	--
ΔV_2 TOTAL IMPULSE	1.65		
THIRD STAGE IGNITION WEIGHT UNCERTAINTY	--	--	3.26
THIRD STAGE THRUST LEVEL	--	--	3.72
ALTITUDE MARK SETTING	—	—	2.40
RSS	1.69	2.88	5.18 M/SEC.
RSS TOTAL		6.16 M/SEC.	

	ELAPSED TIME DISPERSIONS
o 3 σ TIME DISPERSION THROUGH ΔV_2 BURNOUT	.7 SECONDS
o ADDITIONAL DESCENT TIME* DUE TO h AND $\dot{h}$ DISPERSIONS AT ΔV_2 BURNOUT.	13.3 SECONDS
RSS TOTAL	13.3 SECONDS

* (NOMINAL DESCENT TIME = 107.5 SECOND)

ERRORS OF DELIVERY VEHICLE AFTER $\triangle V_2$ BURN

THE FOLLOWING CHART REPRESENTS THE ERRORS IN POSITION AND VELOCITY CAUSED BY $\triangle V_2$ AND ITS ASSOCIATED ERROR SOURCES. A SEARCHING PROCEDURE WAS UTILIZED TO OBTAIN THE CORRECT NOMINAL PARAMETERS AFTER BURN ALLOWING FOR A FINITE THRUST TIME. THE RESULTS REPRESENT A SINGLE ENGINE FURNISHING THE DESIRED VELOCITY CHANGE TO AN ACCURACY CLOSE TO THAT OF THE DOUBLE THRUST LEVEL INHERENT IN THE BASIC PENETROMETER CONCEPT. THE ERRORS WERE THEN TRANSFORMED INTO THE DESIRED COORDINATES BY THE METHOD PREVIOUSLY DESCRIBED. THE ERROR IN BURN TIME ACCRUES SINCE THE ENGINE IS ASSUMED TO PROVIDE THE CORRECT $\triangle V$ RATHER THAN A FIXED BURN TIME.

ERRORS OF DELIVERY VEHICLE AFTER ΔV_2 BURN

- ERROR SOURCES (3σ):

PITCH AND YAW GYRO MISALIGNMENT	0.4°
PITCH AND YAW GYRO DRIFT	0.5° / HR.
INITIAL MASS UNCERTAINTY	6.18 LBS.
I_{SP} UNCERTAINTY	0.5 %
MASS RATE UNCERTAINTY	0.5 %
ACCELEROMETER NULL BIAS	1.4×10^{-4} g
ACCELEROMETER SCALE FACTOR	0.2 %

- POSITION AND VELOCITY ERRORS:

ALTITUDE	DOWNRANGE	CROSSRANGE	ALTITUDE RATE	DOWNRANGE RATE	CROSSRANGE RATE
97.1 M	443.3 M	97.1 M	11.96 M/SEC.	2.69 M/SEC.	11.94 M/SEC

- ERROR IN BURN TIME .7 SECONDS

SPACE GENERAL CONCEPT: EVENT SEQUENCE

THE TIMING OF THE DESCENT SEQUENCE IN THE SPACE GENERAL CONTROL CONCEPT DEPENDS ON THE INITIAL CONDITIONS. SHOWN ARE THE DESCENT PARAMETERS - ALTITUDE, TIME, AND VERTICAL VELOCITY FOR DESCENT SEQUENCES BASED ON THE NOMINAL, 3σ HIGH, AND 3σ LOW VALUES OF INITIAL CONDITIONS. VARIATIONS SHOWN ARE DUE TO INITIAL CONDITIONS ONLY. THE CONTROL SYSTEM IS ASSUMED TO OPERATE WITHOUT ERROR. THE UPPER ALTITUDE MARKER MUST BE SET FAR ENOUGH BELOW THE 3σ LOW INITIAL ALTITUDE TO ALLOW TIME FOR THE 90 DEGREE ATTITUDE MANEUVER. THE LOWER ALTITUDE MARKER MUST BE ABOVE THE REQUIRED IGNITION ALTITUDE FOR THE 3σ HIGH CASE BY A SUFFICIENT AMOUNT TO ALLOW TIME FOR COMPUTATION OF ΔV_3 MAGNITUDE AND THIRD RETRO-IGNITION DELAY. AT THE SAME TIME, IT IS DESIRED TO KEEP THE INTERVAL BETWEEN MARKERS AS GREAT AS POSSIBLE TO KEEP ABSOLUTE ERRORS IN ALTITUDE MARK SETTING AND TIME INTERVAL RESOLUTION FROM CAUSING LARGE PERCENTAGE ERRORS IN THE CONTROL COMPUTATIONS. THE ALTITUDE OF THE LOWER MARK CAN BE REDUCED IF THE THIRD RETRO-THRUST TO LUNAR WEIGHT RATIO IS INCREASED BEYOND THE ASSUMED VALUE OF 3.5.

IT WAS CONCLUDED THAT THE TIME DISPERSIONS ON PENETROMETER IMPACT WERE ABOUT ± 13 SECONDS. THIS IMPOSES A SEVERE COMMUNICATION RANGE REQUIREMENT ON THE SYSTEM SINCE THE LUNAR ORBITER PASSES OVERHEAD AT ORBITAL VELOCITY, APPROXIMATELY 2 KM PER SECOND.

SPACE GENERAL CONCEPT : EVENT SEQUENCE

3σ HIGH			EVENT	NOMINAL			3σ LOW		
Vz M/SEC.	t (SEC.)	h (M)		h	t	Vz	h	t	Vz
11.9 UPWARD	0	5200	ΔV_2 BURNOUT	4600	0	0	4000	0	11.9 DOWN
99.6	68.4	2219	ΔV_3 IGNITION	1800	58.9	95.3	1540	48.5	91
0	95.8	639	PENETROMETER RELEASE	639	82.5	0	639	70.8	0
45.7	123.8	0	IMPACT	0	110.5	45.7	0	98.8	45.7

COMMUNICATION LINKS

AN ORDER OF PRIORITY FOR CONSIDERATION OF THE COMMUNICATION LINKS FOR DATA RETRIEVAL WAS ESTABLISHED BY THE WORK STATEMENT AS INDICATED ON THIS CHART.

COMMUNICATION LINKS

PRIORITY 1:

PENETROMETERS TO LUNAR ORBITER TO EARTH

PRIORITY 2:

PENETROMETERS TO DELIVERY RELAY TO LUNAR ORBITER TO EARTH.

PRIORITY 3:

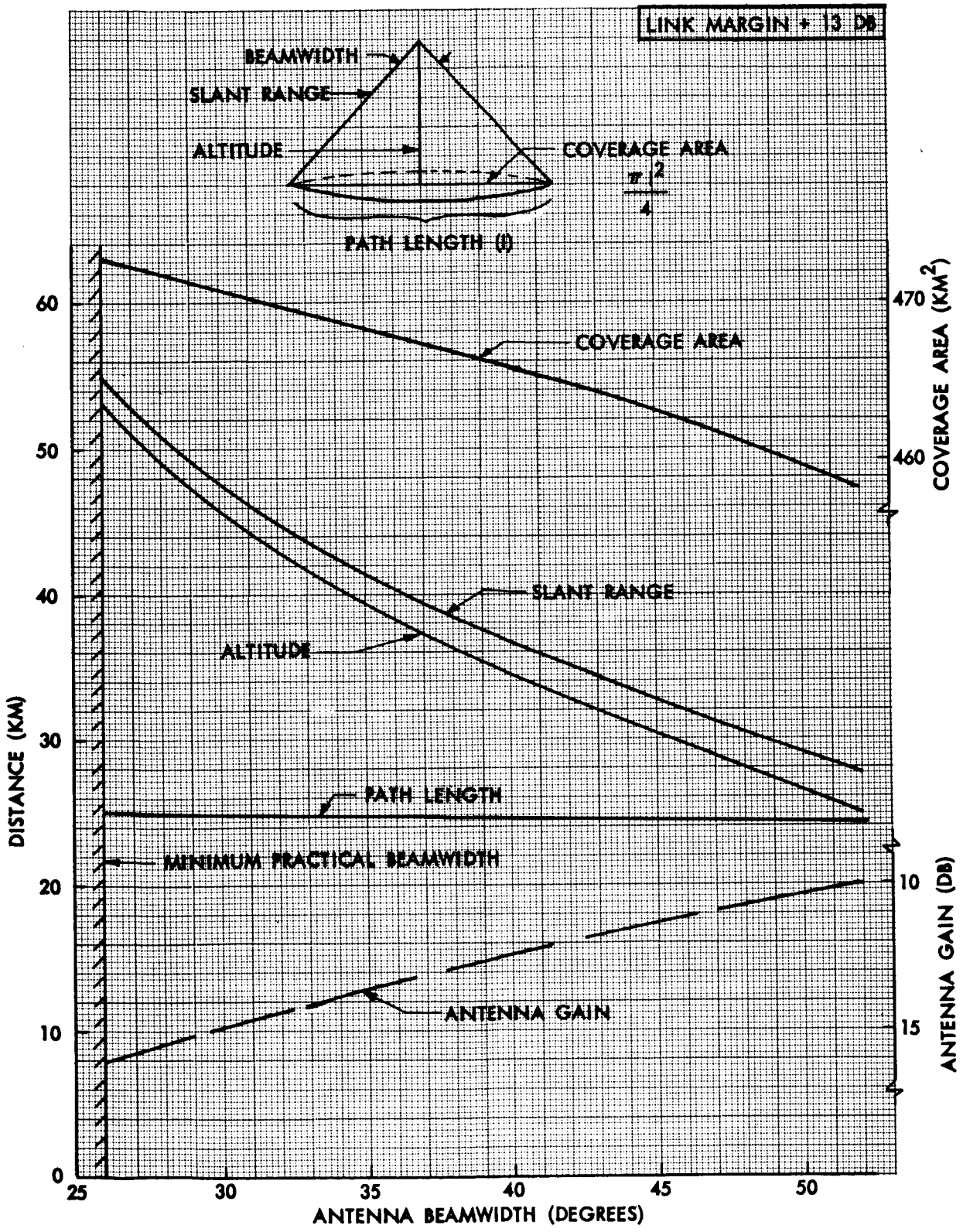
PENETROMETERS TO DELIVERY RELAY TO EARTH.

PENETROMETER - LUNAR ORBITER UHF ANTENNA CONSTRAINTS
ANTENNA NORMAL TO SURFACE

THIS FIGURE ILLUSTRATES THE EFFECT OF ANTENNA BEAMWIDTH ON THE MAXIMUM DISTANCE ALLOWED WHILE HOLDING A CONSTANT LINK GAIN MARGIN OF +13 DB.

THE CASE OF THE LUNAR ORBITER UHF RECEIVING ANTENNA POINTING NORMAL TO THE LUNAR SURFACE AT THE TIME OF PENETROMETER IMPACT WAS SELECTED BECAUSE IT ALLOWS GREATEST VISIBILITY OF THE OVERALL COMMUNICATIONS PROBLEM. THE ANTENNA IS CONICAL WITH A CIRCULAR CROSS-SECTION. THE MAXIMUM UHF ANTENNA GAIN IS LIMITED TO +16 DB (26 DEGREES BEAMWIDTH) TO PROVIDE REALISTIC LIMITATIONS OF SIZE AND WEIGHT TO BE ADDED TO THE LUNAR ORBITER SPACECRAFT.

THE CONCLUSION WAS THAT GROUND COVERAGE WAS APPROXIMATELY CONSTANT AND COULD NOT BE IMPROVED SIGNIFICANTLY BY CHANGING ANTENNA BEAMWIDTH.

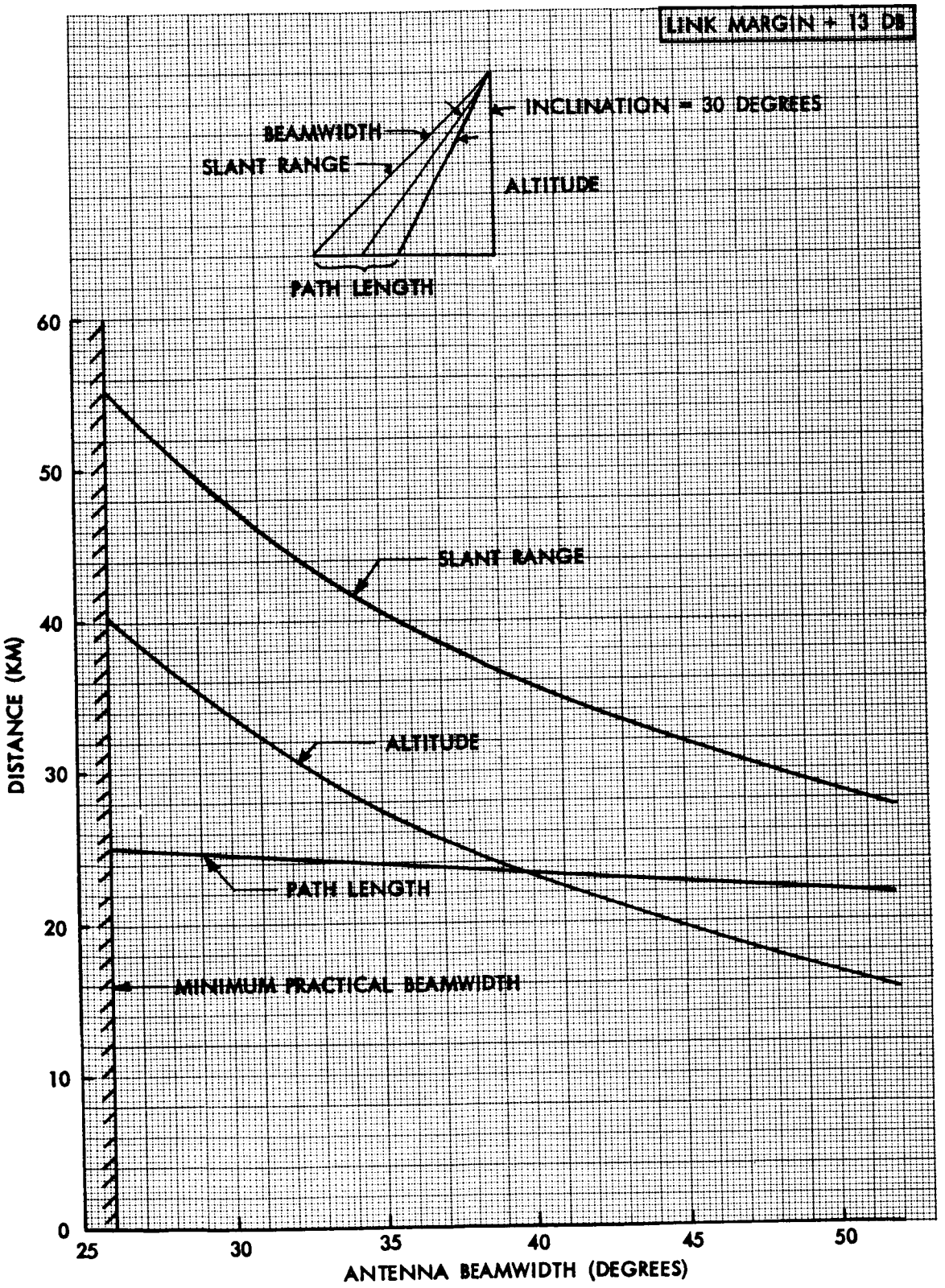


PENETROMETER - LUNAR ORBITER
UHF ANTENNA CONSTRAINTS

PENETROMETER - LUNAR ORBITER UHF ANTENNA CONSTRAINTS
INCLINED ANTENNA

THIS FIGURE INDICATES THE EFFECT OF ANTENNA GAIN AND BEAMWIDTH ON RANGE AND PATH LENGTH AT VARIOUS SPACECRAFT ALTITUDES NECESSARY TO HOLD A CONSTANT LINK GAIN MARGIN OF +13 DB FOR THE LUNAR ORBITER UHF ANTENNA POINTING 30 DEGREES FROM THE NORMAL TO THE LUNAR SURFACE AT THE TIME OF PENETROMETER IMPACT. THIS POINTING ANGLE ALLOWS THE PENETROMETER IMPACT AREA TO LIE AHEAD OR BEHIND THE SPACECRAFT POSITION AT THE TIME OF PENETROMETER IMPACT.

THE INCLINED POINTING ANTENNA DOES NOT ALLOW A GREATER GROUND PATH COVERAGE THAN THE FIXED NORMAL POINTING ANTENNA DUE TO THE REQUIREMENT OF A LOWER ALTITUDE FOR THE SAME BEAMWIDTH.



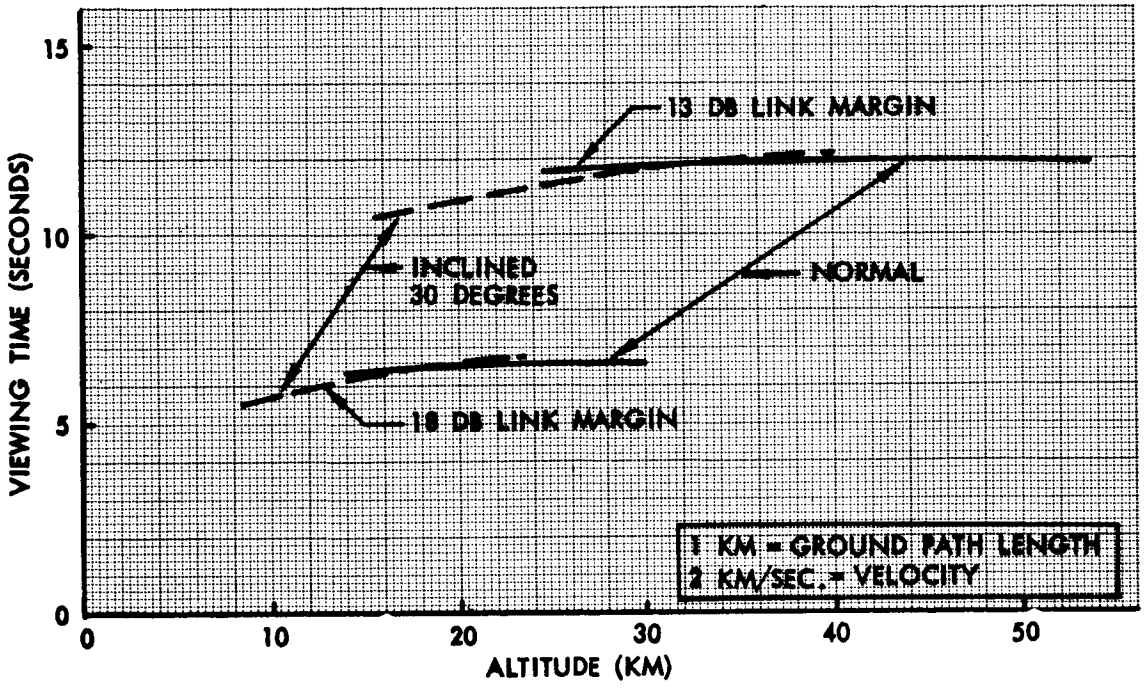
PENETROMETER - LUNAR ORBITER
UHF ANTENNA CONSTRAINTS

VIEWING TIME CAPABILITY - REALIZABLE, FIXED ANTENNAS

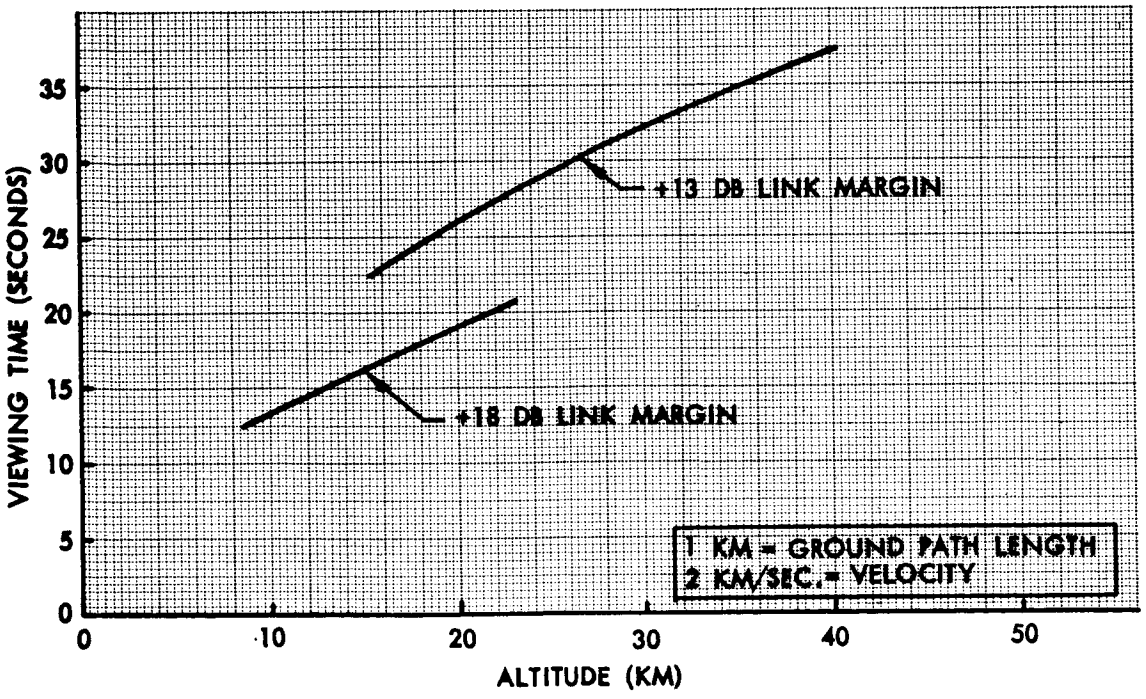
THE VIEWING TIME IS THE TIME PERIOD WHEN THE PROJECTED ANTENNA PATTERN COMPLETELY COVERS A ONE KILOMETER PATH LENGTH ON THE LUNAR SURFACE. THE MAXIMUM VIEWING TIME FOR EITHER THE 30 DEGREES INCLINED OR THE NORMAL POINTING FIXED ANTENNAS IS APPROXIMATELY 12 SECONDS. FOR THE SAME ALTITUDES, THERE IS NO SIGNIFICANT ADVANTAGE OF THE INCLINED OR NORMAL POINTING ANTENNA. THE REASON IS THAT, FOR THE SAME ALTITUDE, A NARROWER BEAM ANTENNA IS REQUIRED FOR THE INCLINED ANTENNA IN ORDER TO MEET THE GREATER RANGE REQUIREMENTS.

VIEWING TIME CAPABILITY - REALIZABLE, SWEEP ANTENNAS

THIS FIGURE INDICATES THE VIEWING TIME CAPABILITY FOR DIFFERENT BEAMWIDTH ANTENNAS, WHICH COMPLETELY COVER A ONE KILOMETER PATH LENGTH WITH A SWEEP FROM -30 DEGREES FROM THE NORMAL TO +30 DEGREES FROM THE NORMAL TO THE LUNAR SURFACE. THE SWEEPING ANTENNA ALLOWS SIGNIFICANT VIEWING TIME INCREASE OVER THE FIXED ANTENNA CASE (35 SECONDS VERSUS 12 SECONDS) BUT REQUIRES ADDED SPACECRAFT COMPLEXITY DUE TO THE SWEEP MECHANISM AND PROGRAMMING.



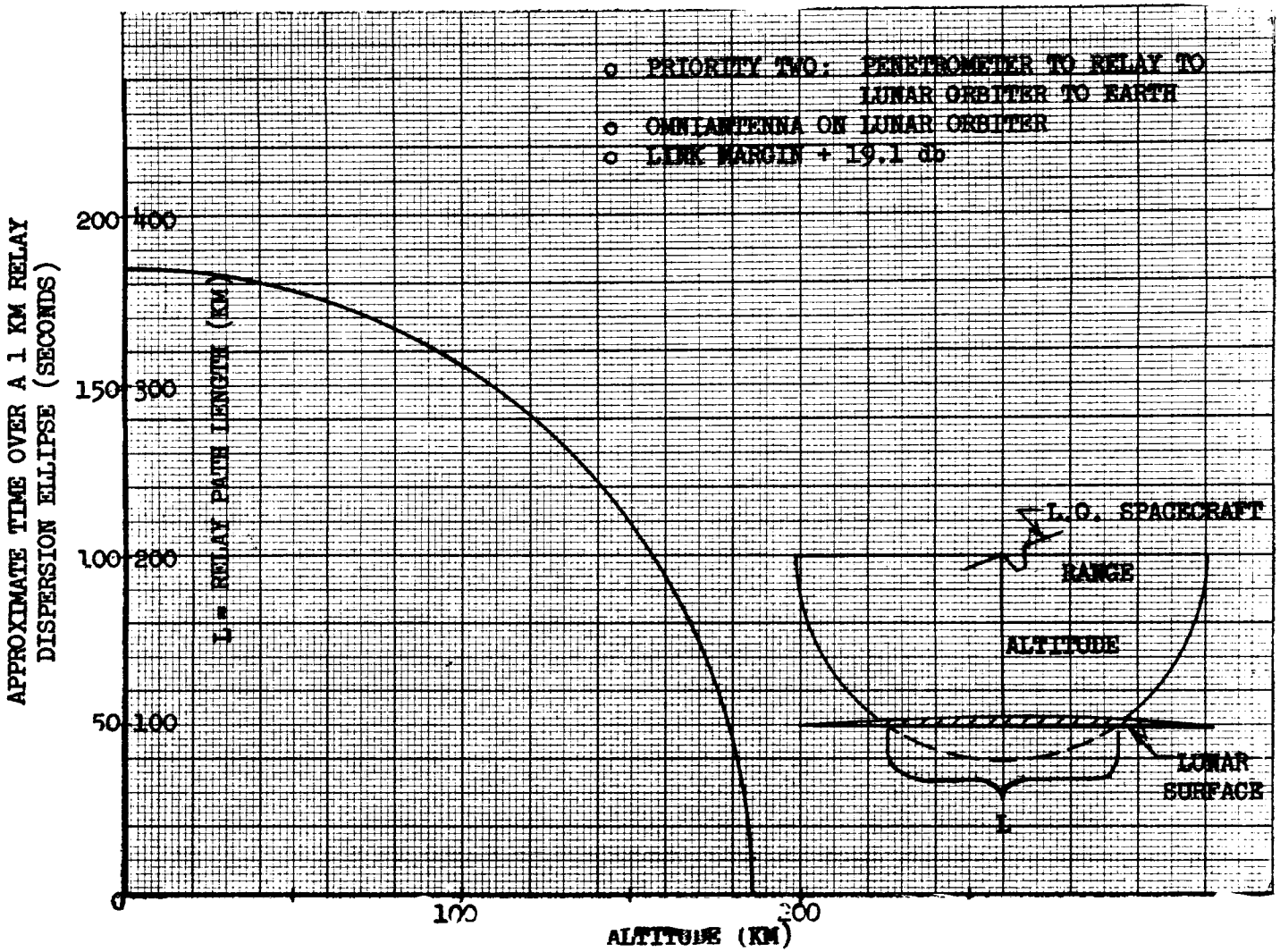
VIEWING TIME CAPABILITY - REALIZABLE, FIXED ANTENNAS



VIEWING TIME CAPABILITY - REALIZABLE, SWEEP ANTENNAS

VHF LUNAR ORBITER RECEIVING ANTENNA COVERAGE

PRIORITY TWO REQUIRES A VHF RECEIVING ANTENNA ON THE LUNAR ORBITER. INCREASED RANGE BETWEEN THE ORBITER AND RELAY VEHICLE RESULTS FROM THE HIGHER POWER TRANSMITTER ON THE RELAY. AN OMNIDIRECTIONAL ANTENNA CAN BE USED ON THE ORBITER.



VHF LUNAR ORBITER RECEIVING ANTENNA COVERAGE

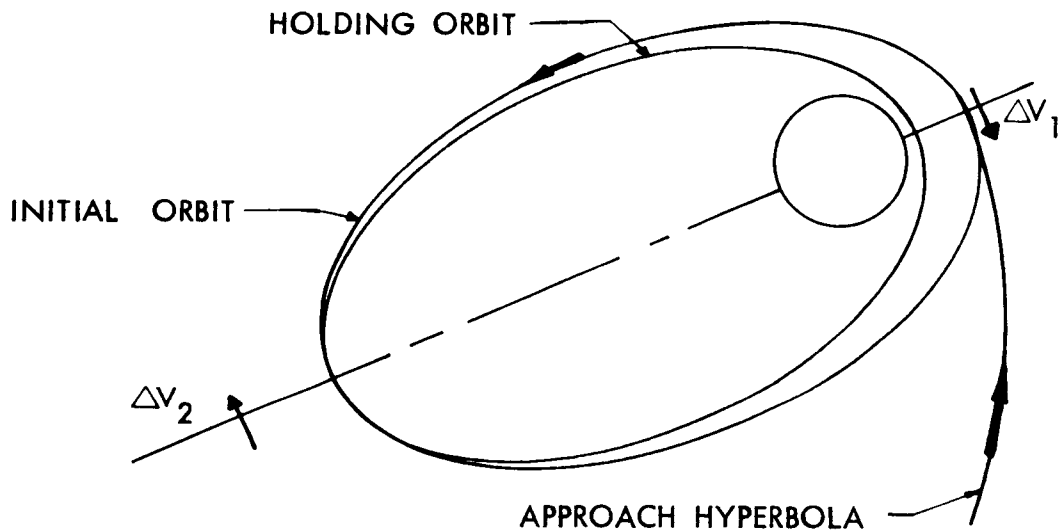
ORBIT DEFINITION

THE BULK OF THE PENETROMETER STUDY IS BASED ON A NOMINAL SET OF ORBITS AND TRANSFER MANEUVERS AS ILLUSTRATED IN THIS CHART. WHEN PERILUNE OF THE APPROACH HYPERBOLA IS REACHED, THE SPACECRAFT IS INSERTED INTO AN INITIAL ORBIT. PERILUNE OF THE APPROACH HYPERBOLA AND THE INITIAL ORBIT ARE ASSUMED TO BE COINCIDENT. PERILUNE ALTITUDE IS FIXED AT 200 KM, WHICH IS CONSISTENT WITH THE CURRENT NOMINAL LUNAR ORBITER OPERATION.

A C_3 OF 1.0 $KM^2/SECOND^2$ WAS CHOSEN FOR THE APPROACH HYPERBOLA. THE ORBITAL ENERGY ASSOCIATED WITH THIS C_3 IS NEAR THE MAXIMUM PREDICTED FOR A 90 HOUR TRANSIT TIME. INSERTION VELOCITY REQUIREMENTS BASED ON THIS ENERGY ARE ALSO MAXIMUM. THE VELOCITY BUDGETED FOR THESE MAXIMUM REQUIREMENTS WILL BE SUFFICIENT FOR ALL ARRIVAL TIMES.

AFTER AN APPROPRIATE PERIOD OF TIME IN THE INITIAL ORBIT, A RETRO-MANEUVER IS MADE AT APOLUNE OF THE INITIAL ORBIT, TRANSFERRING THE SPACECRAFT FROM APOLUNE OF THE INITIAL ORBIT TO APOLUNE OF THE HOLDING ORBIT. PERILUNE ALTITUDE OF THE HOLDING ORBIT IS BETWEEN 5 AND 20 KM.

ORBIT DEFINITION



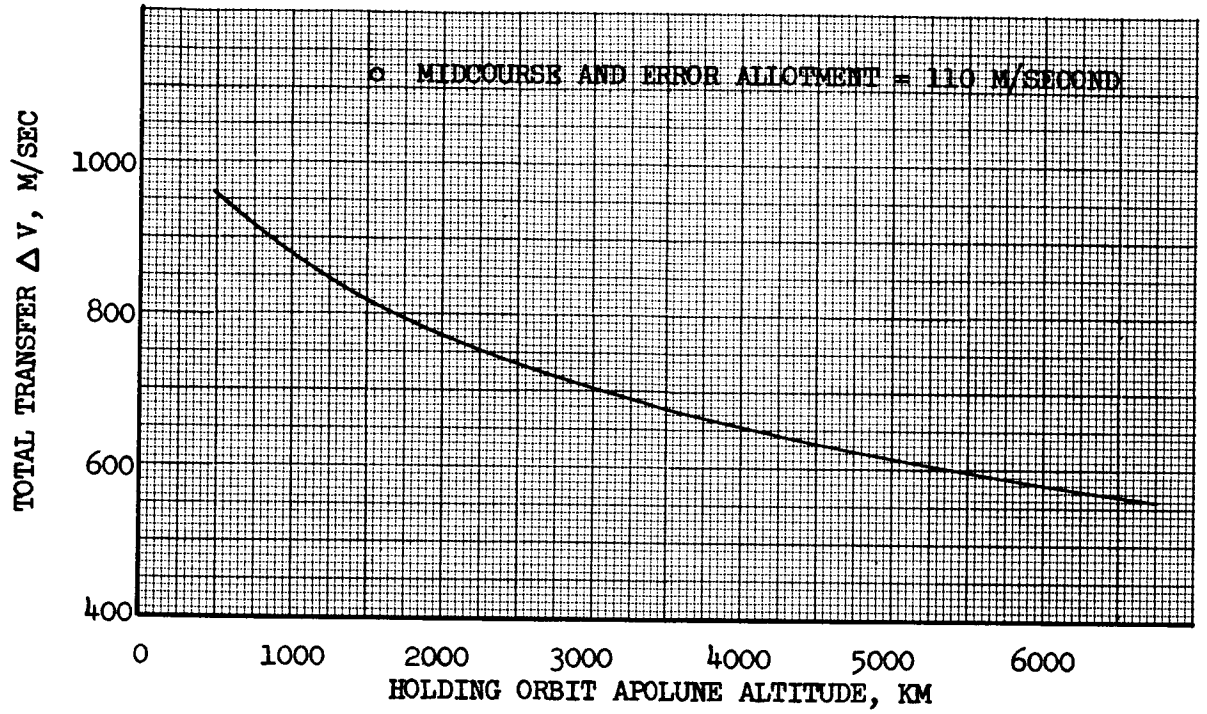
- APPROACH HYPERBOLA - C_3 = 1.0 $KM^2/SEC.^2$ (APPROXIMATE MAXIMUM)
  PERILUNE ALTITUDE = 200 KM
- INITIAL ORBIT - PERILUNE ALTITUDE = 200 KM
  INSERTION MANEUVER AT PERILUNE
- HOLDING ORBIT - PERILUNE ALTITUDE = 5 TO 20 KM
  APOLUNE COINCIDENT WITH APOLUNE OF INITIAL ORBIT
  TRANSFER MANEUVER AT APOLUNE

ΔV REQUIRED FOR TRANSFER TO HOLDING ORBIT

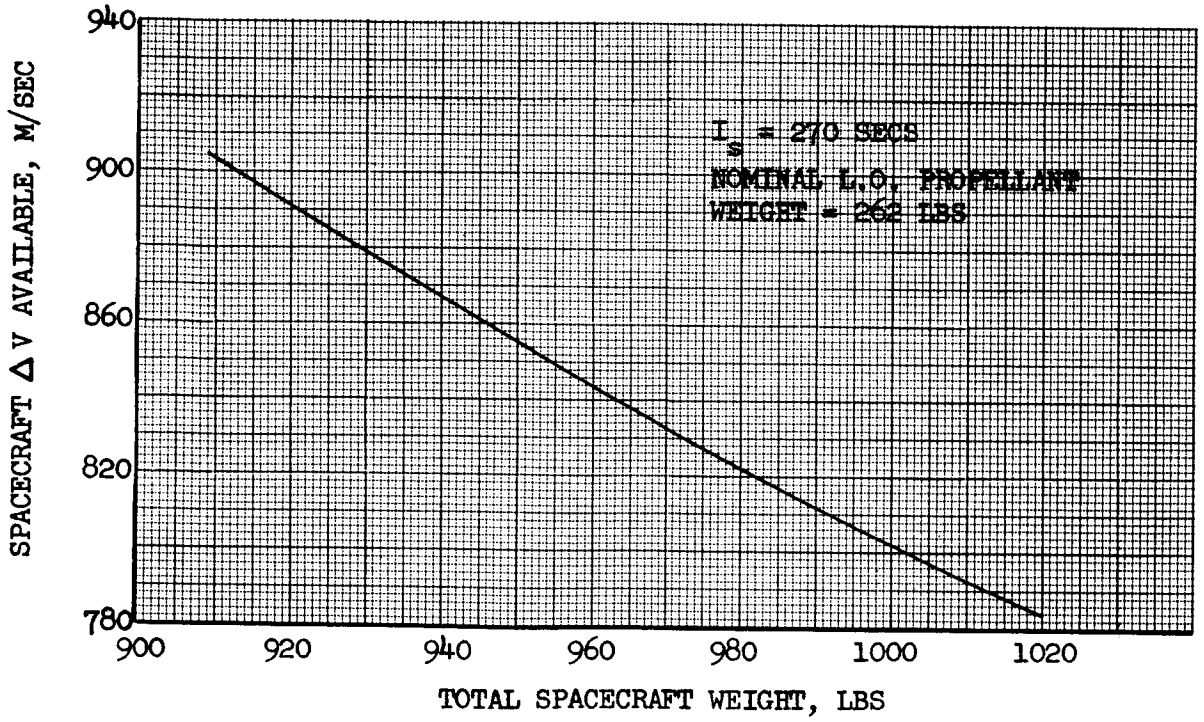
THE VELOCITY REQUIRED TO TRANSFER FROM THE APPROACH HYPERBOLA TO THE HOLDING ORBIT (THE TOTAL TRANSFER ΔV) IS SHOWN. THIS REQUIREMENT IS THE SUM OF THE REQUIREMENTS FOR INSERTION (ΔV_1), TRANSFER TO THE HOLDING ORBIT (ΔV_2), AND 110 METERS PER SECOND FOR MIDCOURSE AND ORBITAL CORRECTIONS.

SPACECRAFT ΔV AVAILABLE

THIS FIGURE GIVES THE SPACECRAFT ΔV CAPABILITY FOR SPACECRAFT WEIGHT RANGING FROM 900 TO 1020 POUNDS. THE TERM SPACECRAFT WEIGHT, AS USED HERE, INCLUDES THE DELIVERY VEHICLE WEIGHT. THE VELOCITY CAPABILITY IS BASED ON THE NOMINAL LUNAR ORBITER PROPELLANT WEIGHT OF 262 POUNDS, AND THE GUARANTEED MINIMUM I_s OF 270 SECONDS.



Δ V REQUIRED FOR TRANSFER TO HOLDING ORBIT



SPACECRAFT Δ V AVAILABLE

Revision 1, Dated 7-29-66

PENETROMETER DELIVERY

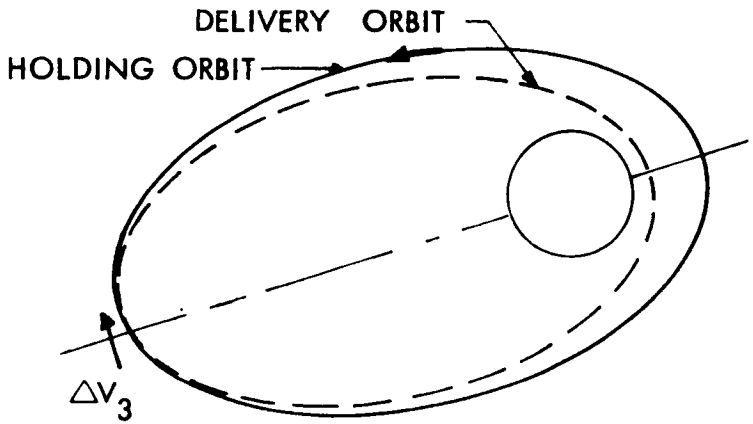
AFTER THE SPACECRAFT HAS BEEN IN THE HOLDING ORBIT LONG ENOUGH TO PHASE WITH A TARGET ON THE SURFACE, THE DELIVERY VEHICLE IS TRANSFERRED FROM THE HOLDING TO THE DELIVERY ORBIT. THIS TRANSFER MANEUVER OCCURS AT APOLUNE. APOLUNE OF THE DELIVERY ORBIT WAS ASSUMED COINCIDENT WITH THE APOLUNE OF BOTH THE INITIAL AND THE HOLDING ORBITS. IT WAS ALSO ASSUMED THAT THE DELIVERY VEHICLE'S FINAL DESCENT IS INITIATED AT PERILUNE. THIS ASSUMPTION THEN REQUIRES THAT THE PERILUNE ALTITUDE OF THE DELIVERY ORBIT MUST BE 4.6 KM.

ORBITER BUS PHASING

THE $\triangle V$ REQUIRED TO SEPARATE THE DELIVERY VEHICLE FROM THE ORBITER BUS ($\triangle V_3$) SIZES THE SEPARATION MECHANISM. A LIGHTWEIGHT MECHANISM REQUIRES THAT THE SEPARATION $\triangle V$ BE AS LOW AS PRACTICAL. THE HOLDING AND DELIVERY ORBITS WILL THEN BE QUITE CLOSE IN BOTH SIZE AND ORBITAL PERIOD. IF THE BUS WERE TO REMAIN IN THE HOLDING ORBIT IN PRIORITIES ONE AND TWO, IT WOULD BE OUTSIDE THE COMMUNICATION RANGE BY THE TIME THE PENETROMETERS IMPACT. IT IS THEN NECESSARY TO PLACE THE BUS IN A SLOWER ORBIT SO THAT IT LAGS BEHIND THE DELIVERY VEHICLE. IF THE PROPER PHASING ORBIT IS SELECTED, THE BUS WILL BE DIRECTLY OVER THE PENETROMETERS AS THEY IMPACT. THE PHASING $\triangle V$ REQUIRED ($\triangle V_4$) DEPENDS ON THE LENGTH OF TIME THE DELIVERY VEHICLE CAN BE SEPARATED FROM THE BUS.

ORBIT DEFINITION

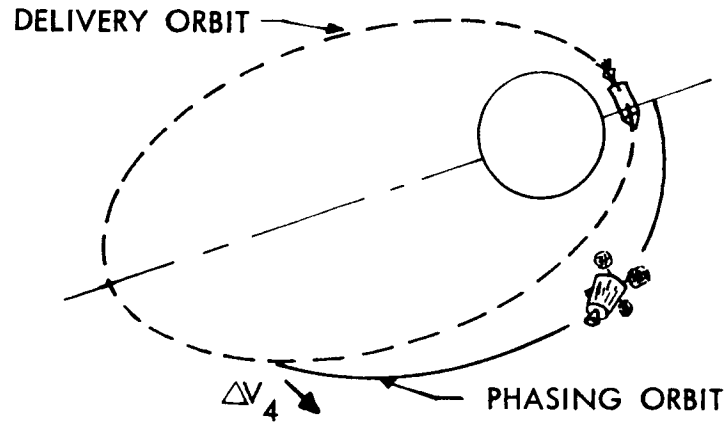
PENETROMETER DELIVERY
(SPACE GENERAL CONCEPT)



TRANSFER FROM HOLDING ORBIT TO DELIVERY ORBIT :

- DELIVERY ORBIT APOLUNE COINCIDENT WITH HOLDING ORBIT APOLUNE
- PERILUNE ALTITUDE = 4.6 KM

ORBITER BUS PHASING



BUS LAGS DELIVERY VEHICLE TO ALLOW FOR DESCENT TIME.

DESCENT TIMES :

- PRIORITY ONE : 107.5 SECONDS
- PRIORITY TWO : 114.5 SECONDS

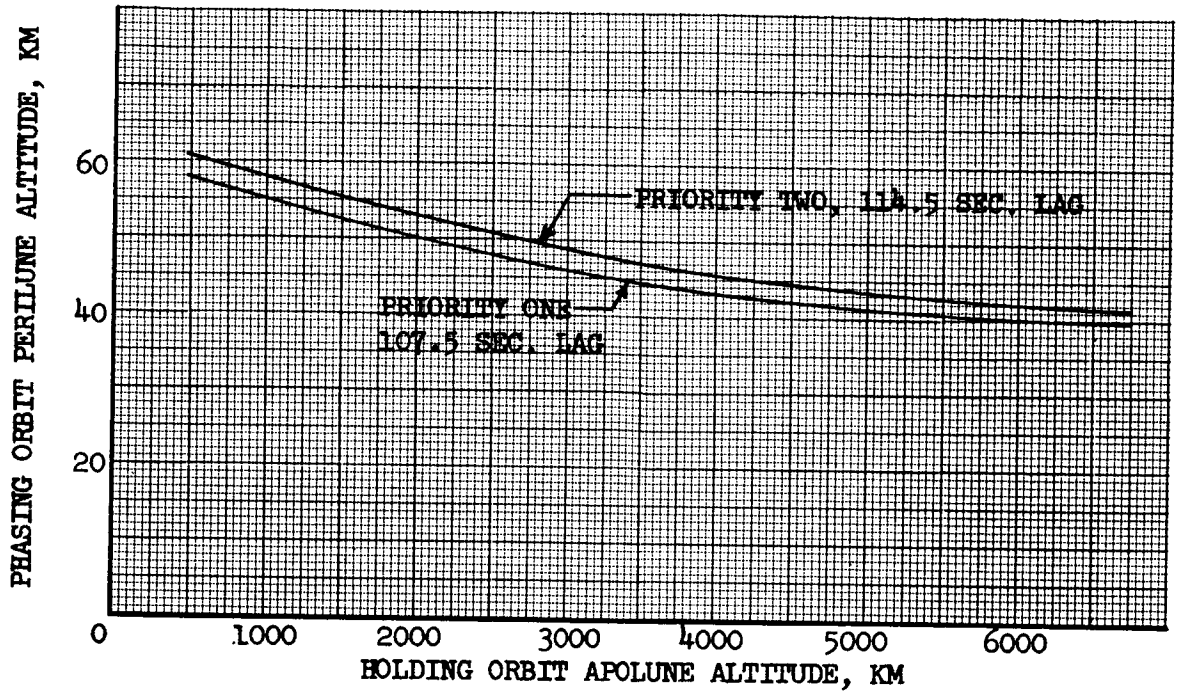
HALF ORBIT PHASING
PHASING ORBIT PERILUNE ALTITUDE

THIS CHART GIVES THE PHASING ORBIT PERILUNE ALTITUDES FOR PRIORITIES ONE AND TWO. THERE IS NO PHASING REQUIREMENT FOR PRIORITY THREE. THE SEVEN-SECOND DIFFERENCE IN DESCENT TIMES REQUIRES THE PRIORITY TWO PHASING ORBIT PERILUNE ALTITUDE TO BE ABOUT 2 KILOMETERS HIGHER THAN THE PRIORITY ONE ALTITUDES. DEPENDING ON THE APOLUNE ALTITUDE, PHASING ORBIT PERILUNE ALTITUDES RANGE FROM ABOUT 40 TO 60 KILOMETERS. USING THIS PHASING SCHEME, THE BUS IS BARELY WITHIN COMMUNICATION RANGE FOR PRIORITY ONE, BUT WELL WITHIN RANGE FOR PRIORITY TWO.

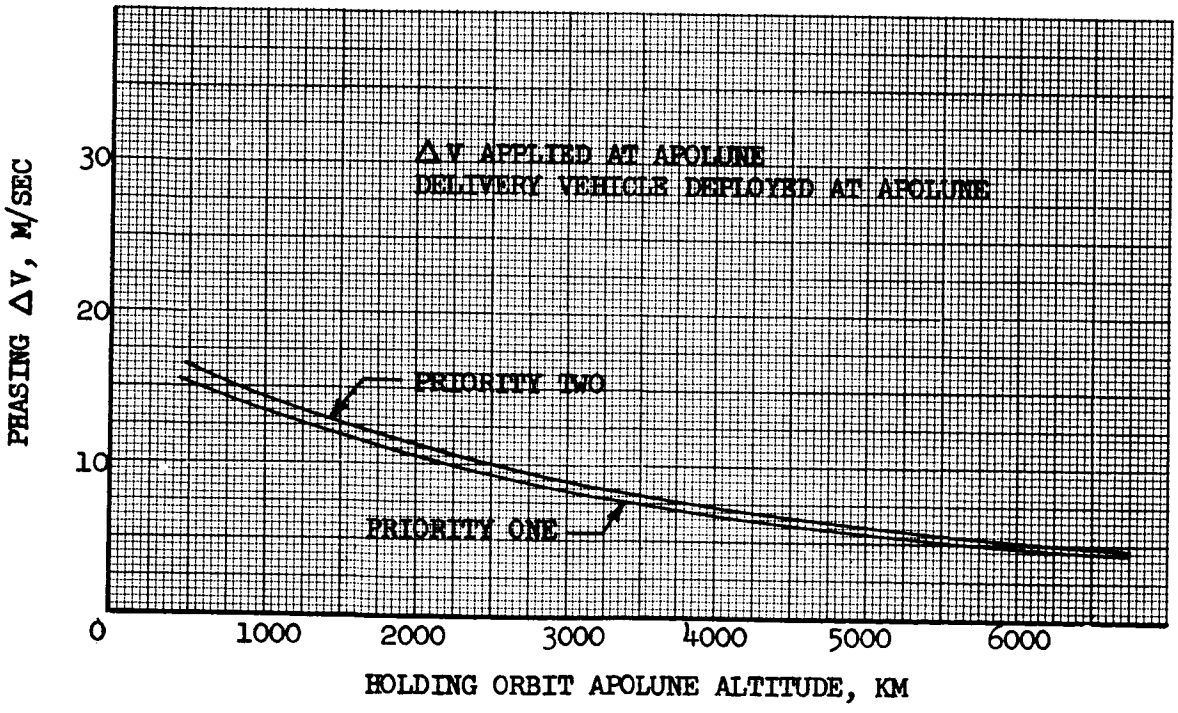
HALF ORBIT PHASING
PHASING ΔV REQUIRED

THE HALF ORBIT PHASING IS REPRESENTATIVE OF SCHEMES WITH LOW VELOCITY REQUIREMENTS. THE DELIVERY VEHICLE IS SEPARATED FROM THE BUS NEAR APOLUNE OF THE HOLDING ORBIT. WITH THE SAME IMPULSE, THE DELIVERY VEHICLE IS PLACED IN THE DELIVERY ORBIT. IMMEDIATELY AFTER THIS, THE BUS TRANSFERS TO THE PHASING ORBIT. HALF AN ORBIT LATER, AT PERILUNE, THE BUS IS OVERHEAD AS THE PENETROMETERS STRIKE THE SURFACE.

THIS CHART GIVES THE VELOCITY REQUIRED TO TRANSFER FROM THE HOLDING ORBIT TO THE PHASING ORBIT. SINCE THE PRIORITY TWO ORBITS ARE LARGER THAN THE PRIORITY ONE ORBITS, THE VELOCITY REQUIREMENTS ARE FROM 0.5 TO 1.0 METER PER SECOND HIGHER.



HALF ORBIT PHASING
PHASING ORBIT PERILUNE ALTITUDE



HALF ORBIT PHASING
PHASING ΔV REQUIRED

POSITION AND VELOCITY ERRORS OF PHASING ORBIT AT PERILUNE

THE ERRORS SHOWN RESULT FROM AN IMPULSE APPLIED TO THE ORBITER AT APOLUNE ON THE HOLDING ORBIT. THESE ERRORS WERE PROPAGATED ALONG THE RESULTING PHASING ORBIT TO PERILUNE.

THE SIMILAR ERRORS IN POSITION AND VELOCITY OF THE DELIVERY VEHICLE WERE SHOWN TO BE OF THE ORDER OF METERS AND MILLIMETERS/SECOND AND THEREFORE CONSIDERED TO BE NEGLIGIBLE.

POSITION AND VELOCITY ERRORS OF PHASING ORBIT AT PERILUNE

- ERROR SOURCES (3σ) : POINTING = 0.88° ΔV SCALE FACTOR = 0.6%

- ERRORS AT PERILUNE PROPOGATED FROM APOLUNE :

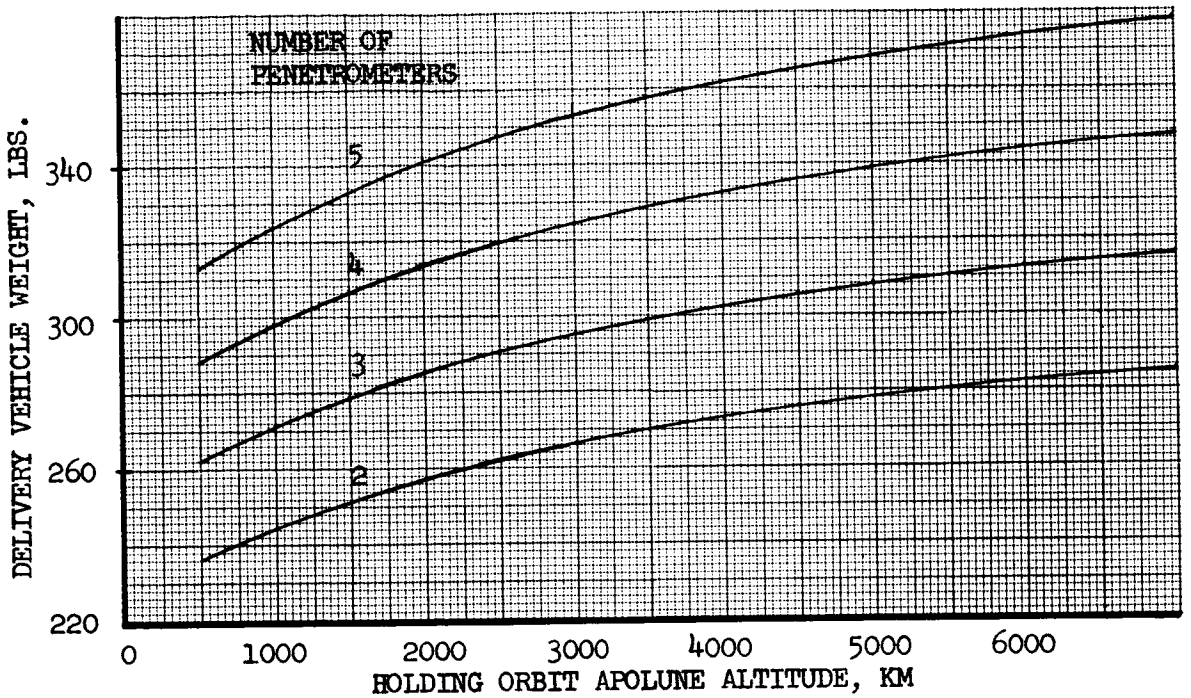
APOLUNE ALTITUDE (KM) :	2300	4000	6000
PERILUNE ALTITUDE (KM) :	49.6	44.2	40.76
ALTITUDE (KM) :	0.57	0.27	0.26
DOWNRANGE (KM) :	3.62	2.21	2.87
CROSS RANGE (KM) :	0.018	0.011	0.005
ALTITUDE RATE (M/SEC.) :	2.65	1.59	2.04
DOWNRANGE RATE (M/SEC.) :	0.37	0.16	0.16
CROSS RANGE RATE (M/SEC.) :	0.52	0.27	0.29
ERROR IN TIME FROM PERILUNE (SEC.) :	2.2	1.2	1.4

PRIORITY TWO: DELIVERY VEHICLE WEIGHT

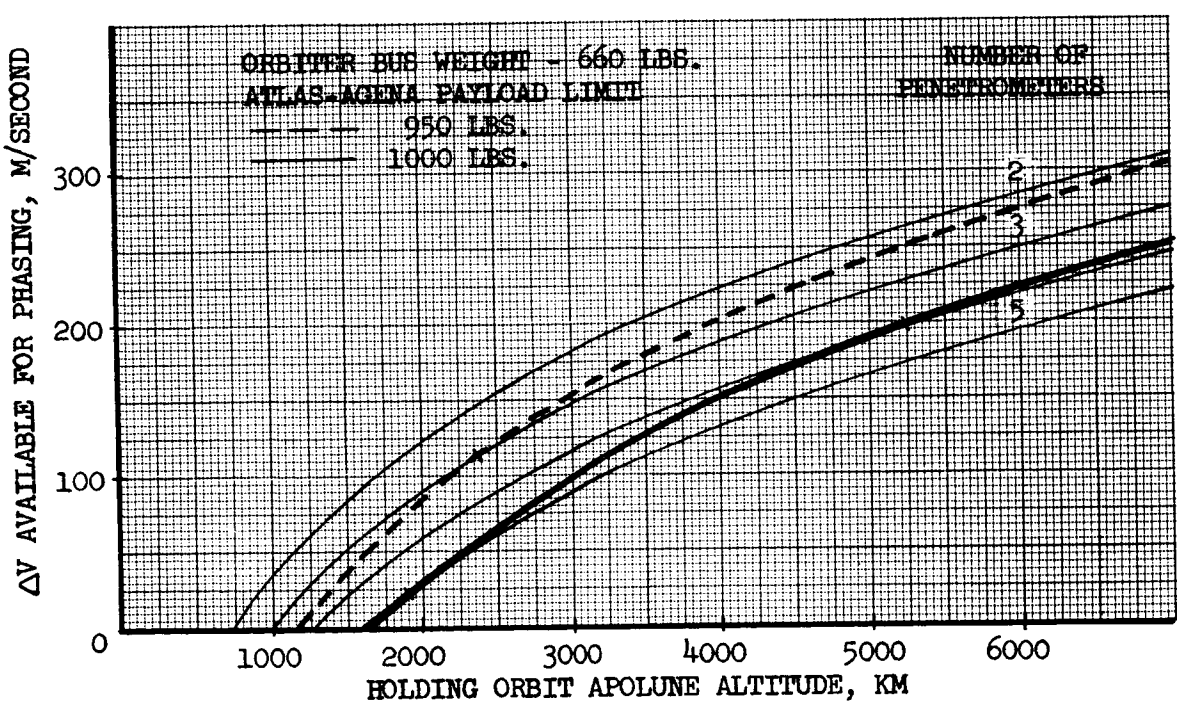
THE DELIVERY VEHICLE WEIGHTS USED IN THIS STUDY WERE GENERATED FROM AN ANALYTICAL MODEL. USING THIS MODEL, THE WEIGHT IS DETERMINED BY THE NUMBER OF PENETROMETERS CARRIED, THE COMMUNICATION WEIGHT REQUIRED, AND THE REQUIRED VELOCITY CAPABILITY. BECAUSE THE DELIVERY VEHICLE DESCENT BEGINS AT PERILUNE, THE VELOCITY REQUIRED TO INITIATE THE FREE-FALL IS EQUAL TO THE PERILUNE VELOCITY OF THE DELIVERY ORBIT. FOR A FIXED PERILUNE ALTITUDE, PERILUNE VELOCITY AND CONSEQUENTLY DELIVERY VEHICLE WEIGHT INCREASE WITH APOLUNE ALTITUDE. THIS FIGURE GIVES THE DELIVERY VEHICLE WEIGHT AS A FUNCTION OF APOLUNE ALTITUDE FOR THE PRIORITY TWO MODE.

PRIORITY TWO: ΔV AVAILABLE FOR PHASING

IN ADDITION TO THE TOTAL TRANSFER ΔV REQUIREMENT, VELOCITY MUST BE BUDGETED FOR THE PHASING MANEUVERS NECESSARY IN PRIORITIES ONE AND TWO. ANY DIFFERENCE BETWEEN THE REQUIRED AND AVAILABLE ΔV THEN REPRESENTS THE ΔV AVAILABLE FOR PHASING. FOR PRIORITY THREE MISSIONS, THIS ΔV DIFFERENCE IS EXCESS CAPABILITY. THE VARIATION OF THIS AVAILABLE PHASING ΔV WITH APOLUNE ALTITUDE IS SHOWN FOR A PRIORITY TWO CASE. VARIATIONS FOR TWO ATLAS/AGENA PAYLOAD LIMITS, 950 AND 1000 POUNDS, ARE ALSO SHOWN. IT CAN BE SEEN THAT THE PAYLOAD LIMIT IS THE "OPTIMUM" SPACECRAFT WEIGHT. FOR A SPECIFIED PHASING ΔV REQUIREMENT, THE MAXIMUM NUMBER OF PENETROMETERS CAN BE DELIVERED IF THE SPACECRAFT IS AS HEAVY AS POSSIBLE. IN THE REMAINDER OF THE STUDY, THE 1000 POUND ATLAS/AGENA PAYLOAD LIMIT IS USED.



PRIORITY TWO: PENETROMETER-RELAY-BUS-EARTH DELIVERY VEHICLE WEIGHT



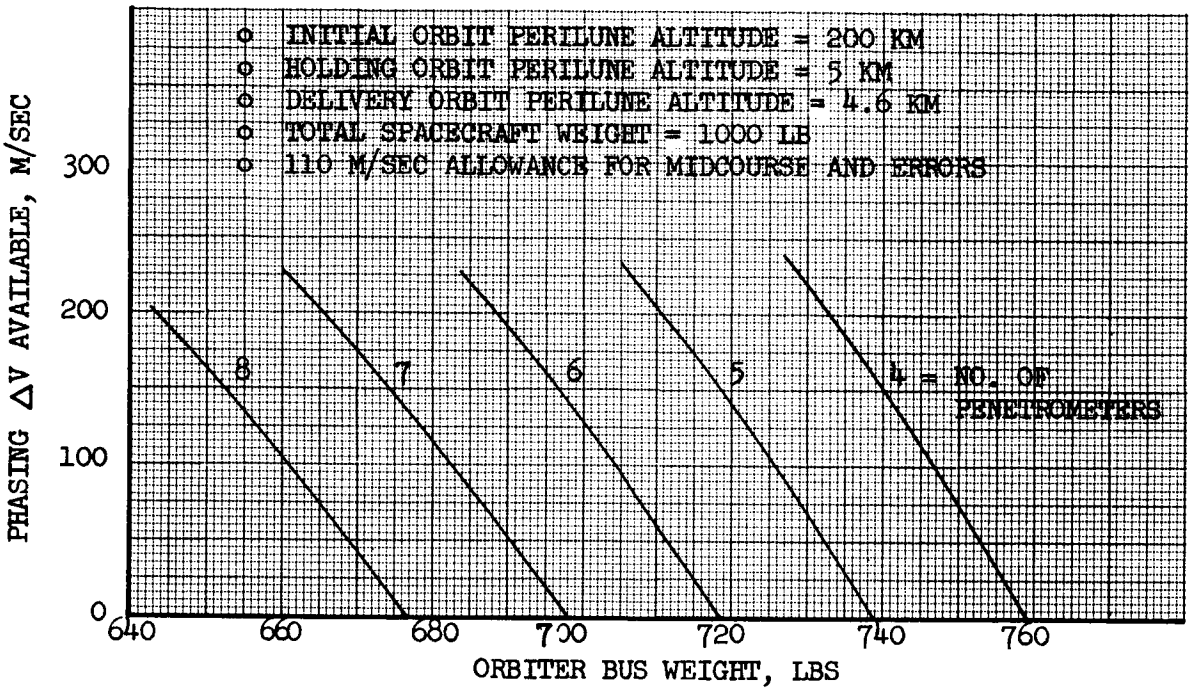
PRIORITY TWO: PENETROMETER-RELAY-ORBITER-EARTH
ΔV AVAILABLE FOR PHASING

PRIORITY ONE: MAXIMUM ΔV AVAILABLE FOR PHASING

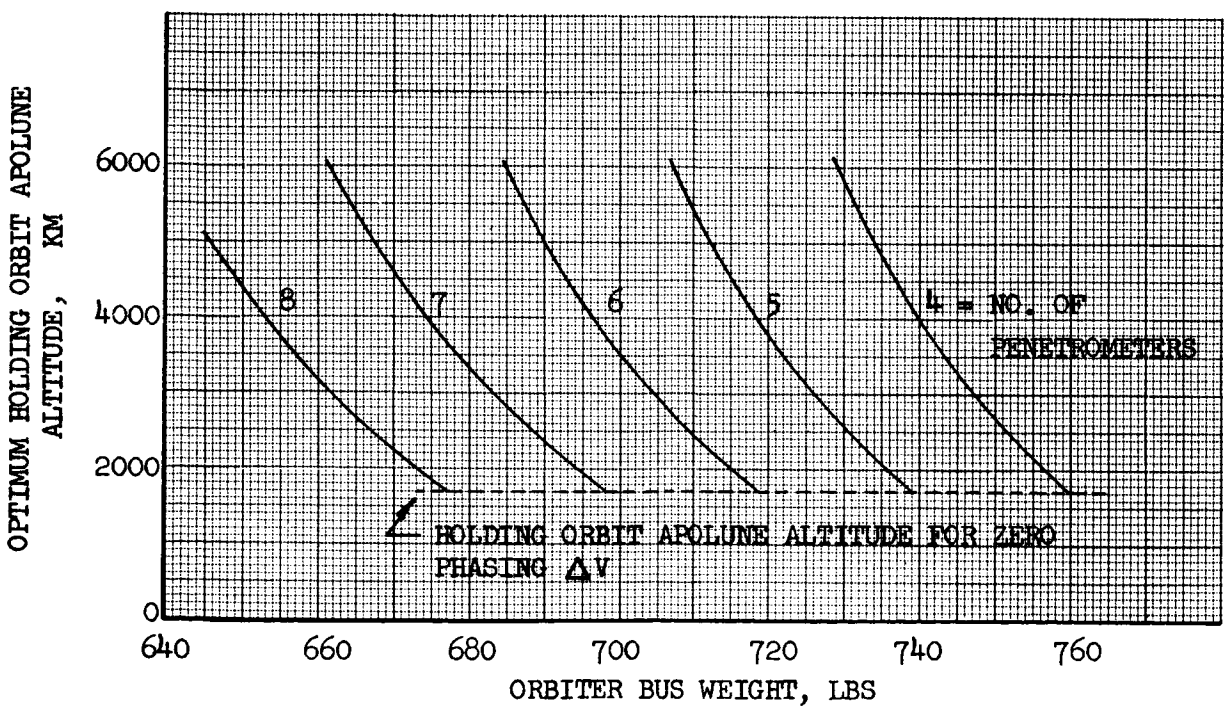
THIS CHART PRESENTS THE MAXIMUM PHASING ΔV AVAILABLE FOR A PRIORITY ONE MISSION. SINCE THE ORBITER BUS WEIGHT HAS NOT BEEN COMPLETELY DETERMINED, IT MUST BE TREATED AS AN INDEPENDENT VARIABLE. ONCE THE BUS WEIGHT IS DETERMINED, THE NUMBER OF PENETROMETERS THAT CAN BE DELIVERED AND THE PHASING ΔV ARE DEFINED.

PRIORITY ONE: OPTIMUM HOLDING ORBIT APOLUNE ALTITUDE

BECAUSE THE SPACECRAFT WEIGHT IS FIXED AT 1000 POUNDS, EACH APOLUNE SHOWN IS UNIQUELY RELATED TO A BUS WEIGHT AND ΔV COMBINATION. INCREASING THE APOLUNE ALTITUDE REQUIRES AN INCREASE IN THE DELIVERY VEHICLE WEIGHT. FOR A SPECIFIC BUS WEIGHT, THE NUMBER OF PENETROMETERS MUST BE DECREASED TO MAINTAIN THE 1000 POUND SPACECRAFT WEIGHT LIMIT. IF APOLUNE ALTITUDE IS DECREASED, THE DELIVERY VEHICLE VELOCITY CAPABILITY EXCEEDS THE REQUIREMENTS.



PRIORITY ONE: PENETROMETER-BUS-EARTH
MAXIMUM ΔV AVAILABLE FOR PHASING



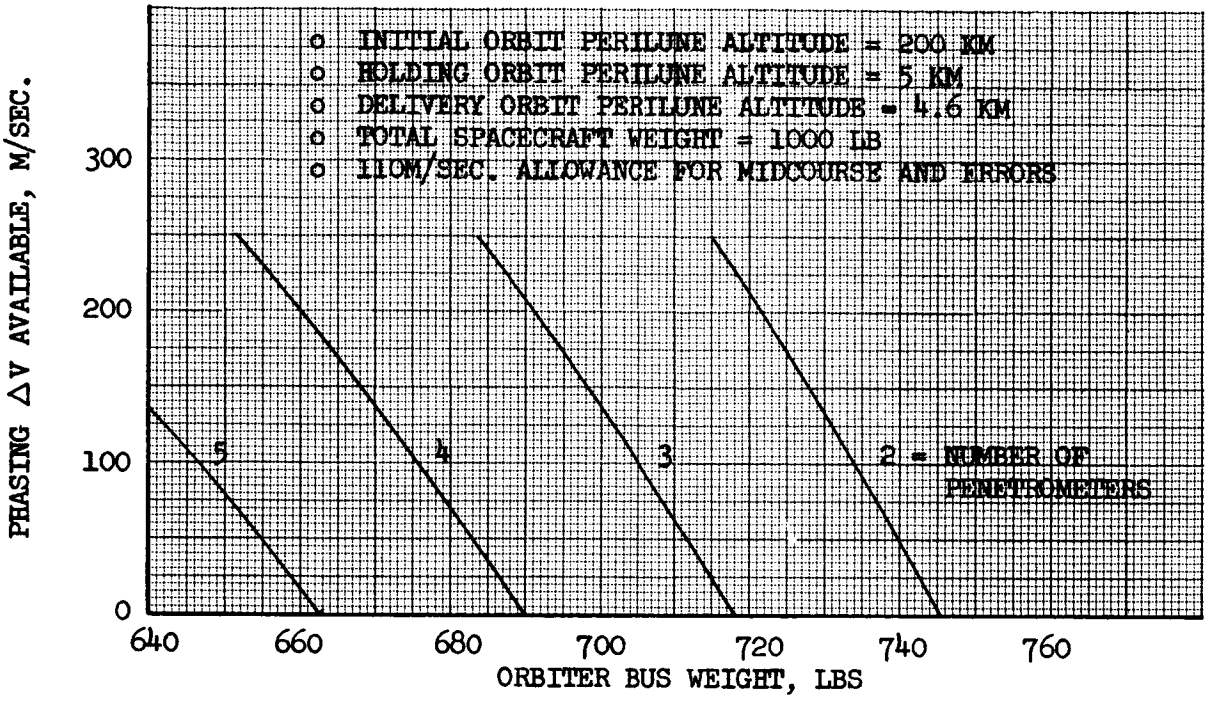
PRIORITY ONE: PENETROMETER-BUS-EARTH
OPTIMUM HOLDING ORBIT APOLUNE ALTITUDE

PRIORITY TWO: MAXIMUM ΔV AVAILABLE FOR PHASING

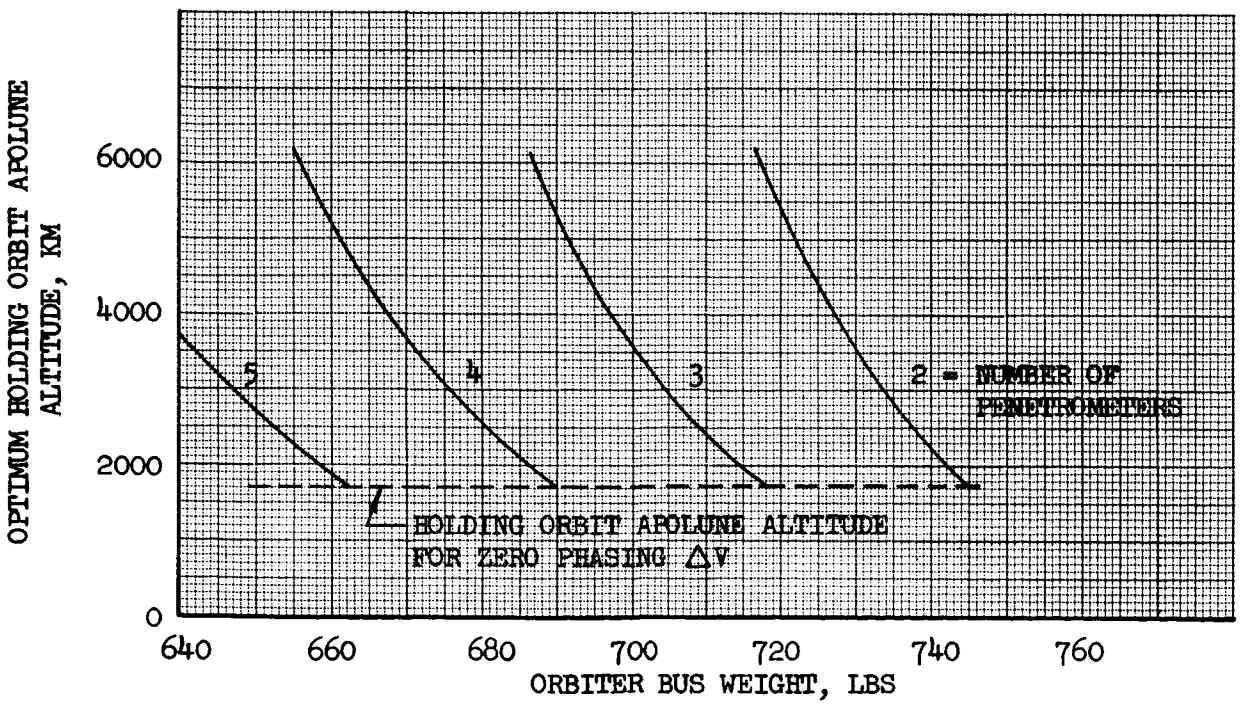
THIS CHART GIVES THE SAME INFORMATION FOR THE PRIORITY TWO MISSION AS SHOWN PREVIOUSLY FOR PRIORITY ONE. SINCE THE REQUIRED COMMUNICATION EQUIPMENT IS HEAVIER, FEWER PENETROMETERS CAN BE DELIVERED FOR A SPECIFIED BUS WEIGHT.

PRIORITY TWO: OPTIMUM HOLDING ORBIT APOLUNE ALTITUDE

APOLUNE ALTITUDES CORRESPONDING TO THE PHASING ΔV ARE SHOWN.



PRIORITY TWO: PENETROMETER-RELAY-BUS-EARTH
MAXIMUM ΔV AVAILABLE FOR PHASING



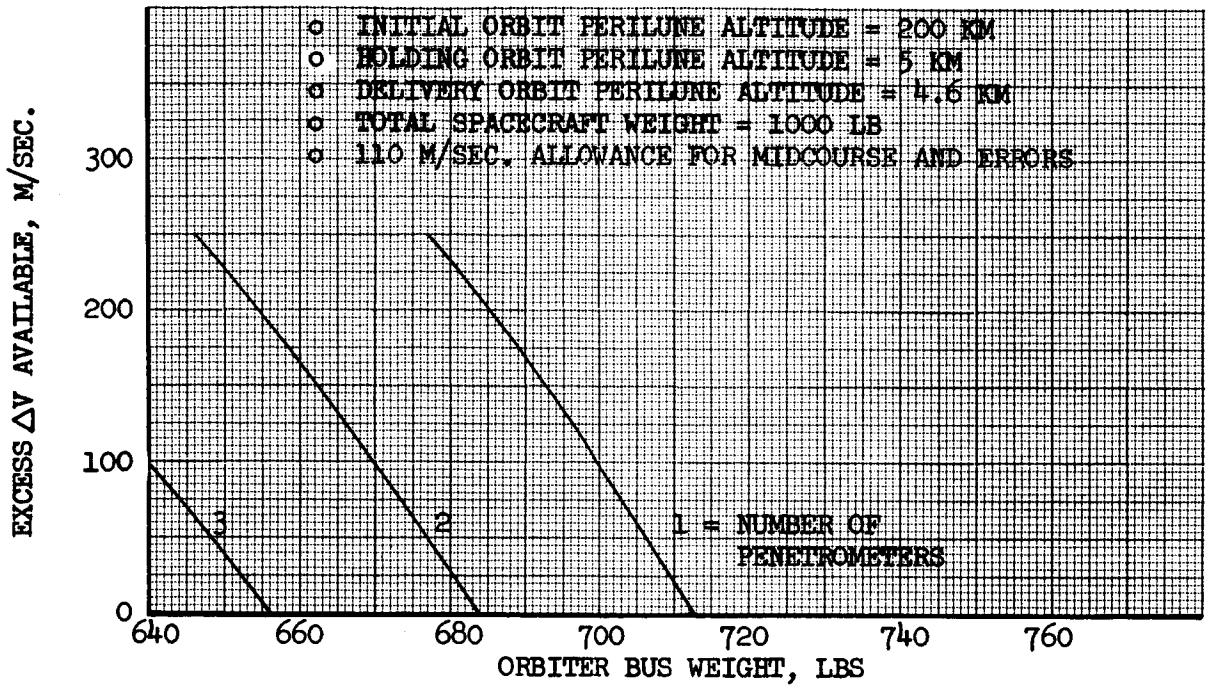
PRIORITY TWO: PENETROMETER-RELAY-BUS-EARTH
OPTIMUM HOLDING ORBIT APOLUNE ALTITUDE

PRIORITY THREE: MAXIMUM ΔV AVAILABLE

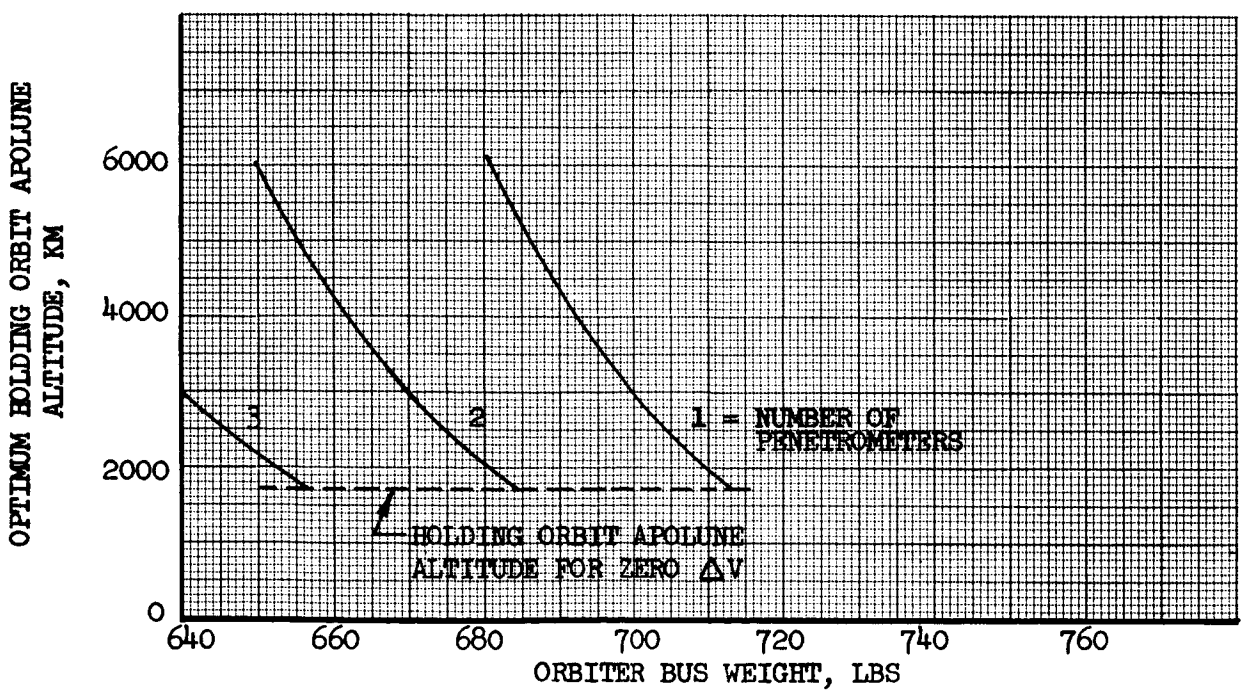
THE DATA SHOWN HERE ARE SIMILAR TO THE PHASING ΔV DATA GIVEN FOR PRIORITIES ONE AND TWO. SINCE THERE IS NO PHASING REQUIREMENT, THE DIFFERENCE BETWEEN THE AVAILABLE AND REQUIRED ΔV IS EXCESS. THE HEAVIER COMMUNICATION EQUIPMENT REQUIRED FURTHER REDUCES THE NUMBER OF PENETROMETERS THAT CAN BE DELIVERED.

PRIORITY THREE: OPTIMUM HOLDING ORBIT APOLUNE ALTITUDE

THE APOLUNE ALTITUDES SHOWN HERE FOR THE GIVEN BUS WEIGHTS CORRESPOND TO THE EXCESS ΔV DATA DESCRIBED ABOVE.



PRIORITY THREE: PENETROMETER-RELAY-EARTH
MAXIMUM ΔV AVAILABLE

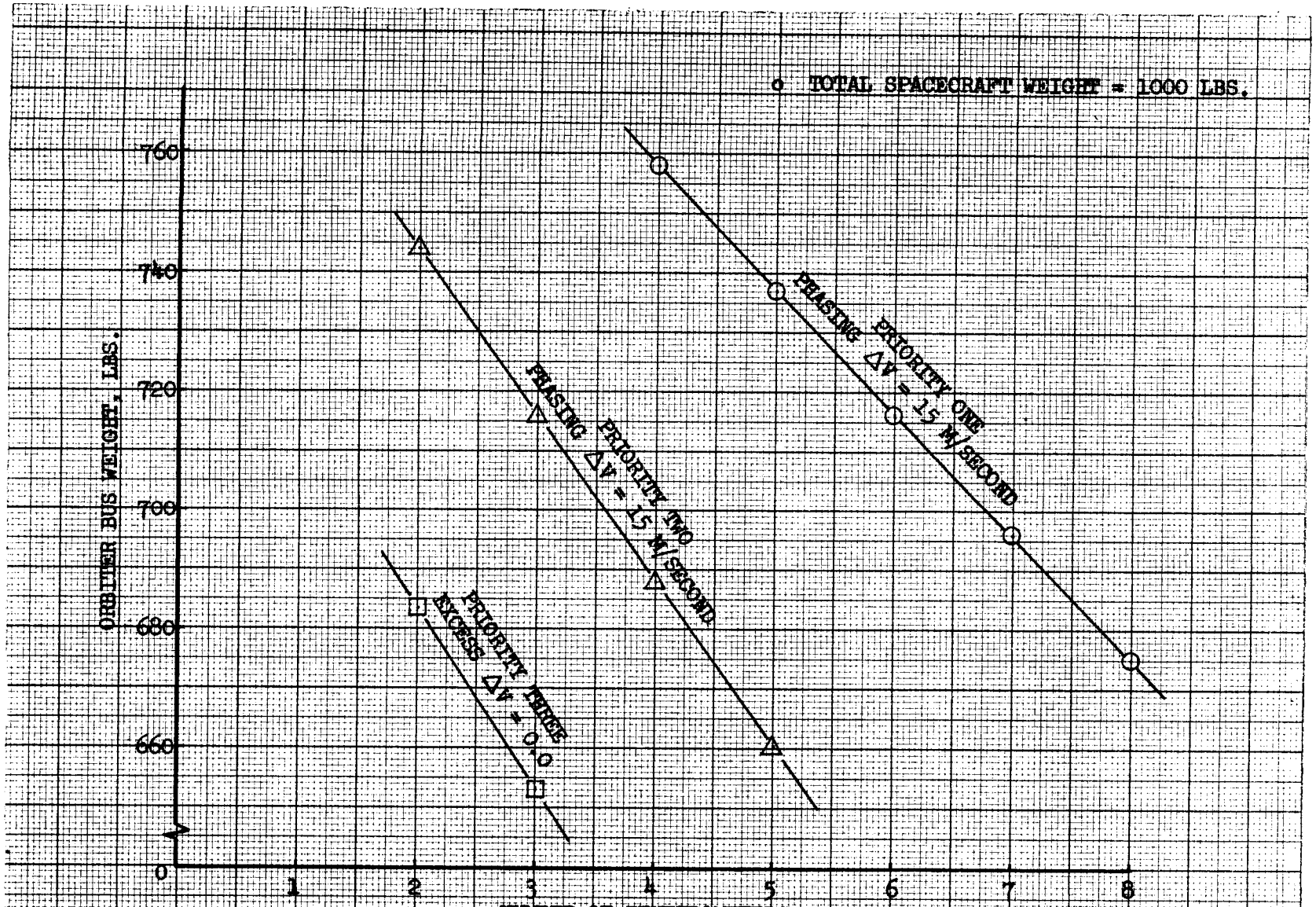


PRIORITY THREE: PENETROMETER-RELAY-EARTH
OPTIMUM HOLDING ORBIT APOLUNE ALTITUDE

Revision 1, Dated 7-29-66

MAXIMUM ALLOWABLE ORBITER BUS WEIGHT

THIS CHART IS A SUMMARY OF THE PARAMETRIC DATA. FOR EACH COMMUNICATION MODE, THE MAXIMUM BUS WEIGHT ALLOWABLE FOR A GIVEN NUMBER OF PENETROMETERS IS SHOWN. IN THIS DATA, 15 M/S ARE ALLOWED FOR PHASING IN THE PRIORITY ONE AND TWO MODES, AND ZERO EXCESS ΔV FOR PRIORITY THREE.



MAXIMUM ALLOWABLE ORBITER BUS WEIGHT

Revision 1, Dated 7-29-66

CANDIDATE PENETROMETER SYSTEM

IF THE 700 TO 715 POUND BUS WEIGHT IS REALISTIC, 6 PENETROMETERS CAN BE DELIVERED USING THE PRIORITY ONE COMMUNICATION MODE, 3 PENETROMETERS USING PRIORITY TWO, AND ONLY 1 PENETROMETERS WITH PRIORITY THREE. IF THE HALF ORBIT PHASING SCHEME IS EMPLOYED, THE BUS IS BARELY WITHIN COMMUNICATION IN THE PRIORITY ONE MODE, BUT WELL WITHIN RANGE FOR THE PRIORITY TWO MISSIONS. THE PRIORITY TWO MISSION THEN DELIVERS THE MOST PENETROMETERS WITH A RELIABLE COMMUNICATION LINK.

CANDIDATE PENETROMETER SYSTEM

- PRIORITY TWO : PENETROMETER TO RELAY TO ORBITER TO EARTH
- THREE PENETROMETERS
- BUS WEIGHT : 700 TO 715 LBS.
- HOLDING ORBIT : PERILUNE ALTITUDE - 5 TO 20 KM
  APOLUNE ALTITUDE - 2000 TO 3600 KM
- HALF ORBIT PHASING MANEUVER
  - ORBITER DIRECTLY OVERHEAD AT PENETROMETER IMPACT
  - ORBITER ALTITUDE AT PENETROMETER IMPACT : 47 TO 53 KM
  - DELIVERY VEHICLE AND ORBITER SEPARATED 1.8 TO 2.6 HRS.
  - ΔV REQUIRED FOR PHASING : 8 TO 11 METERS PER SECOND
  - STAGING ΔV REQUIRED FOR DELIVERY RELAY VEHICLE :
    0.5 TO 2.5 METERS PER SECOND

Revision 1, Dated 7-29-66

APPENDIX A

EXPERIMENT DESCRIPTIONS

THE BOEING COMPANY AEROSPACE GROUP

SPACE DIVISION · P. O. BOX 3995 · SEATTLE WASHINGTON 98124

LUNAR ORBITER PROGRAM

MAY 19 1966

IN REPLY REFER TO
2-1590-00-049

To: National Aeronautics and Space Administration
Langley Research Center
Lunar Orbiter Program
ATTN: L. W. Fitchett, Contracting Officer
Building 1221
I. O. MacConochie, Technical Manager
Building 1232
Langley Station
Hampton, Virginia 23365

Subject: Contract NAS1-6216, Report of Meetings between NASA and Contractor personnel held 26-27 April 1966 concerning Study Contract of Lunar Orbiter Adaptability to Experiments to Evaluate Lunar Surface Characteristics

1. The purpose of this letter is to submit, for NASA's concurrence, subject meeting report which describes the experiments to be used by the Contractor in the study of the Lunar Orbiter's adaptability to perform experiments to evaluate Lunar surface characteristics.

2. The attachment to this letter is the report of the subject meetings and the Contractor considers the report to be a complete and accurate description of those experiments.

3. The Contractor requests formal NASA concurrence that the experiments described in the attachment are those to be studied by the Contractor under Contract NAS1-6216.

THE BOEING COMPANY
Space Division

E. J. Nicholson
Contracts Manager
Spacecraft Systems

Attachment

cc: K. L. Wadlin
R. D. Eisenhart
AFPR

REPORT OF MEETINGS BETWEEN NASA, LRC AND BOEING PERSONNEL

APRIL 26 and 27, 1966

concerning study contract of Lunar Orbiter Adaptability to experiments to evaluate lunar surface characteristics

1. NASA representatives reviewed the current status of each experiment. Boeing representatives reviewed the experiment data to be obtained and discussed the manner in which the planned study would be conducted. This was followed by a question and answer session and finally a detailed listing of the characteristics of each experiment which was to be used in the study. The above meetings covered two days. The following paragraphs briefly describe the discussions.

2. Ground Rules

The following general ground rules were discussed:

a) This study is of the Lunar Orbiter adaptability to the experiments.

b) Each of the experiments should be treated impartially as though the data obtained were of equal value.

c) The funding level of the study does not permit including work on experiment design except in a parametric manner, e.g., to size the required rocket motors. Evaluation of the scientific aspects of the experiments is the responsibility of the NASA-LRC.

3. Penetrometers

Aeroneutronics Division of Philco is presently under contract and will deliver a final report soon. The LEM application for the penetrometer is being funded on a continuing basis. The Space General work on a delivery system from a CSM is being dropped.

In the LEM application, 8 of the devices are kicked out 1500 to 2000 ft. over the lunar surface and are intended to impact in the landing area (although the trajectories are a current problem area) and telemeter data back to LEM to be evaluated prior to the LEM "decision to abort" altitude is reached.

There is some work going on at Langley on a lighter weight device, but it is not available for this study.

Boeing personnel discussed the flight mechanics of delivering the penetrometers to the lunar surface and the errors involved in an open loop spin stabilized concept. The conclusion was reached that errors in knowledge of the lunar reference datum precluded this approach from giving acceptable impact velocity dispersion. Also that semi-soft landing techniques sensing altitude were required for the delivery vehicle.

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3. Penetrometers (Continued)

Appendix A is a detailed description of the penetrometer experiment and is to be the basis of the Boeing study. Concurrence by the NASA is requested. Appendix D explains the weight calculations.

4. Flare

Douglas has completed two contracts on the "Early Lunar Flare;" the latter was to provide calibration data. The primary objective is to provide surface bearing strength. A secondary objective is to provide chemical composition. It is to be considered as an added payload to the present L/O mission to obtain high resolution photos. In concept the flare could be implemented by an arrangement similar to Figures 1 and 2. The principal problems to be studied by The Boeing Company are to determine the ΔV requirements such that the flare can be seen from the orbiter and the data can be retrieved.

Appendix B is a detailed description of the Flare experiment and is to be the basis of the Boeing study. Concurrence by the NASA is requested.

5. Balloon

The balloon (low density impacter) was conceived at Langley and postulates a signature on the lunar surface - either by an impression or by albedo change - caused by the impact. High resolution photography of the impact area before and after could provide information about the surface. No calibration test work has been done. No plan exists at this time to develop the concept further.

Boeing presented an analysis of cratering depth showing a range from .0006 cm for a granite surface to 30 cm for a polyurthane foam material. The paper was taken back to Langley for the information of Dr. Brooks and Dr. Katzoff.

Appendix C contains a detailed description of the Balloon experiment and is to be the basis of the Boeing study. Concurrence by the NASA is requested.

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APPENDIX A

A. PENETROMETER EXPERIMENT DESCRIPTION

1. Objective

The objective is to obtain time history of deceleration of impacting accelerometer to obtain bearing strength. Priority of studies with regard to communications link is: (1) Penetrometer to Lunar Orbiter to Earth, (2) Penetrometer to delivery vehicle relay to Lunar Orbiter to Earth, and (3) Penetrometer to relay to Earth.

2. Mission

The principle mission is to obtain penetrometer data. No photographic capability is required.

3. Experiment Constraints

a. Nominal impact velocity of 150 ft/sec $\pm$ 5%, knowledge of velocity.

b. Impact angle $\leq$ 30 degrees to vertical, knowledge of angle $\pm$ 10%.

c. Impact within 10 KM area limit of site.

d. Spin rate up to 50 rpm permitted.

4. Government Furnished Information and Equipment

a. Results of Philco study and copy of final report. Use Philco penetrometers for study.

b. Results of Space General study and one copy of oral presentation booklet. Scale down payload to penetrometers only for this study.

c. Langley memorandum of January 6, 1966 and other notes and drawings.

5. Additional Equipment

a. Antennas for first and second priority objectives are to be sized by Boeing.

b. Requirements for a recorder are to be determined.

6. Detailed Characteristics of the Penetrometer Pertinent to the Study

a. Number of Penetrometers	8	Philco Study
b. Nominal Diameter (each)	8.5 Inches	

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A. 6. (Continued)

c.	Sensing and Transmitting Error	± 7.4%
c.	Transmitting Frequency	435 MH_Z - 456 MH_Z in 3 MH_Z increments
d.	Transmitter Power	0.5 Watts
f.	Weight	2.3 Kilograms/Penetrometer
g.	Modulation	AM/FM
h.	Transmitter Peak Deviation	± 160 KHZ
i.	Transmitting Antenna Gain	- 3.0 db, ± 1.5 db
j.	Transmission Circuit Loss	- 1.0 db ± 0.5 db
k.	Polarization Loss	- 3.0 db, + 3 db - 0 db

7. Penetrometer Communication Requirements for Priority (1) - Penetrometer - Lunar Orbiter - Earth Option

a. The communication link characteristics are based on the Philco study:

Receiver Threshold SNR_{IF}	12 db
System Noise Figure	6 db
Desired Margin (IF)	13 db minimum
Total Error - RMS	± 10%

b. The basis for the Boeing study is the following:

1. Determine suitable VHF antennas for the Lunar Orbiter with the following assumptions.

Frequency Range	430 MH_Z to 460 MH_Z
Gain	Maximum, consistent with coverage requirements
Polarization	V - Linear
	H - Linear
Load Impedance	50 ohms
Allowable VSWR Receiver	2:1

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A. 7. b. (Continued)

2. A baseband of eight subcarriers between 12.8 KHZ and 70.8 KHZ each FM with deviation ± 2 KHZ.

3. Determine need and characteristics of interface equipment.

4. Determine need and characteristics of recorder.

USE FOR TYPEWRITTEN MATERIAL ONLY

A. 7. (Continued)

c. Additional equipment weight to be installed on Lunar Orbiter:

Weight Summary

	No. of Penetrometers			
	2	4	6	8
GFE (Philco)	8.2	14.4	20.6	27.0
Additional Equipment	6.2	7.2	8.1	9.0
Total, Lbs.	14.4	21.6	28.7	36.0

d. Detailed breakdown:

	2	4	6	8
GFE Philco				
Preamplifier	0.4	0.4	0.4	0.4
Power Divider	0.4	0.4	0.4	0.4
RF Switches	0.4	0.4	0.4	0.4
Receivers	2.0	4.0	6.0	8.0
Diversity Sel.	0.5	1.0	1.5	2.0
Subcarrier Demod.	0.4	0.8	1.2	1.6
Subcarrier Osc.	0.5	1.0	1.5	2.0
Power Reg.	0.3	0.3	0.3	0.3
Battery, 28 V, 67 W - Hrs	0.9	1.75	2.6	3.5
Control and J Box	0.3	0.3	0.3	0.3
Connectors and Wiring	1.4	2.7	4.0	5.4
Miscellaneous	0.7	1.3	2.0	2.7
Subtotal, Lbs.	8.2	14.35	20.6	27.0
Additional Equipment				
(2) Rec. Antennas	2.0	2.0	2.0	2.0
Interface Equipment	0.7	1.3	1.9	2.5
Recorder	3.0	3.0	3.0	3.0
Miscellaneous	0.5	0.9	1.2	1.5
Subtotal, Lbs.	6.2	7.2	8.1	9.0

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A. 7. (Continued)

e. Langley Research Center will provide output voltage level range requirements and the Philco Final Report on Penetrometer Study.

8. Delivery Vehicle Weight for Priority (1) - Penetrometer to Lunar Orbiter to Earth

Weight Summary

	No. of Penetrometers			
	2	4	6	8
Payload	(15)	(30)	(45)	(60)
Penetrometers	10	20	30	40
Structures, Mechanisms	5	10	15	20
Guidance and Control	(40)	(40)	(40)	(40)
Communications	(0)	(0)	(0)	(0)
Propulsion Systems (See Appendix D)	(154.3)	(181.4)	(208.5)	(235.6)
Solid Propellant*	111.5	134.0	156.4	178.9
Solid Motor Inerts	11.2	13.4	15.7	17.9
Hydrazine Propellants	7.9	9.5	11.1	12.7
Hydrazine System Inerts	18.4	18.9	19.3	19.8
N_2 Reaction Control System	5.3	5.6	6.0	6.3
Total Delivery Vehicle Weight, lbs	(209.3)	(251.4)	(293.5)	(335.6)

* Retropropulsion allocations are based on an ideal velocity capability of 6600 fps (2.01 Km/Sec) and a solid propellant specific impulse of 270 lb_f-sec/lb_m which is consistent with existing technology. Parametric studies will be conducted to evaluate retropropulsion requirements over the velocity range considered for this mission.

9. Communication Requirements for Priority (2) - Penetrometer-Relay-Lunar Orbiter-Earth

a. Relay characteristics are:

Philco Study:

Transmitter Frequency	260 MH_Z
Modulation	FM/PM
Transmitter Power Output	40 Watts $\pm$ 1 db

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A. 9. a. (Continued)

Receiving Antenna Gain	4.5 db ± 1.5 db
Receiving Circuit Loss	- 1.0 db ± 0.5 db
Transmitting Antenna Gain	0.0 db ± 3.0 db
Transmitting Circuit Loss	- 1.0 db ± 0.5 db
Polarization Loss	- 3.0 db, + 3 db - 1.0 db
Threshold SNR_{IF}	+ 12 db
Subsystem Noise Figure	+ 6 db
Desired Margin	+ 13 db minimum
DC Power Provided	67 Watt - Hours

b. The basis for the Boeing study includes the following:

1. Determine a suitable antenna for the Lunar Orbiter with the following characteristics:

Frequency	260 MHZ ± 500 KHZ
Gain	Maximum, consistent with coverage requirements.
Polarization	RH Circular
Load Impedance	50 ohms
Allowable VSWR at Receiver	2:1

2. A baseband of eight subcarriers between 12.8 KHZ and 70.8 KHZ, each FM with deviation ± 2 KHZ.

3. Determine need and characteristics of interface equipment.

4. Determine need and characteristics of [illegible] recorder.

USE FOR TYPEWRITTEN MATERIAL ONLY

A. 9. (Continued)

c. Equipment weight:

Weight Summary

	No. of Penetrometers			
	2	4	6	8
On Relay, GFE (Philco)	18.7	24	29.4	34.9
On Lunar Orbiter (Additional)	8.9	12.8	16.6	20.4
Total, Lbs.	27.6	36.8	46.0	55.3

USE FOR TYPEWRITTEN MATERIAL ONLY

A. 9. (Continued)

d. Detailed breakdown:

	2	4	6	8
On Relay, GFE (Philco)				
Rec. Antenna	1.2	1.2	1.2	1.2
Preamplifier	0.4	0.4	0.4	0.4
Power Divider	0.4	0.4	0.4	0.4
RF Switches	0.4	0.4	0.4	0.4
Receivers	2.0	4.0	6.0	8.0
Diversity Sel.	0.5	1.0	1.5	2.0
Subcarrier Demod.	0.4	0.8	1.2	1.6
Subcarrier Osc.	0.5	1.0	1.5	2.0
40 Watt Trans.	6.0	6.0	6.0	6.0
Power Reg.	0.3	0.3	0.3	0.3
Battery, 28 V, 67 W - Hrs	3.5	3.5	3.5	3.5
Trans. Antenna	0.7	0.7	0.7	0.7
Control and J Box	0.3	0.3	0.3	0.3
Connectors and Wiring	1.4	2.7	4.0	5.4
Miscellaneous	0.7	1.3	2.0	2.7
Subtotal, Lbs.	18.7	24.0	29.4	34.9
On L/O Additional Equip.				
Rec. Antenna	1.0	1.0	1.0	1.0
Preamplifier	0.4	0.4	0.4	0.4
Power Divider	0.4	0.4	0.4	0.4
Receivers	2.0	4.0	6.0	8.0
Subcarrier Demod.	0.4	0.8	1.2	1.6
Subcarrier Osc.	0.5	1.0	1.5	2.0
Interface Equipment	0.7	1.3	1.9	2.5
Recorder	3.0	3.0	3.0	3.0
Miscellaneous	0.5	0.9	1.2	1.5
Subtotal, Lbs.	8.9	12.8	16.6	20.4

USE FOR TYPEWRITTEN MATERIAL ONLY

A. 9. (Continued)

e. Langley Research Center will provide output level range requirements and Philco Final Report on penetrometer study.

10. Delivery Vehicle Weight for Priority #2 - Penetrometer to Relay to Lunar Orbiter to Earth

Weight Summary

	No. of Penetrometers			
	2	4	6	8
Payload	(15)	(30)	(45)	(60)
Penetrometers	10	20	30	40
Structures, Mechanisms	5	10	15	20
Guidance and Control	(40)	(40)	(40)	(40)
Communications	(18.7)	(24)	(29.4)	(34.9)
Propulsion Systems (See Appendix D)	(191.5)	(229.1)	(266.7)	(304.6)
Solid Propellant*	141.4	172.2	203.2	234.3
Solid Motor Inerts	14.8	17.2	20.3	23.5
Hydrazine Propellant	10.8	13.2	15.5	17.9
Hydrazine System Inerts	19.2	20.0	20.7	21.4
N_2 Reaction Control System	5.9	6.5	7.0	7.5
Total Delivery Vehicle Weight, Lbs	(265.2)	(323.1)	(381.1)	(439.5)

* Retropropellant allocations are based on an ideal velocity capability of 6600 fps (2.01 Km/Sec) and a solid propellant specific impulse of 270 lb_f-sec/lb_m which is consistent with existing technology. Parametric studies will be conducted to evaluate retropropulsion requirements over the velocity range considered for this mission.

11. Communication Requirements for Priority (3) - Penetrometer-Relay-Earth

a. Relay characteristics are:

Philco Study:

Receiving Antenna Gain	4.5 db ± 1.5 db
Receiving Circuit Loss	- 1.0 db ± 0.5 db

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A. 11. a. (Continued)

Threshold SNR_{IF}	+ 12 db
System Noise Figure	+ 6 db
Desired Margin	+ 13 db minimum

b. Boeing study: No communications required.

c. Equipment weight

Weight Summary

	No. of Penetrometers			
	2	4	6	8
GFE (Philco)	7.3	12.2	17.1	22.0
GFE (Additional)	33.7	34.3	34.9	35.5
Total, Lbs.	41.0	46.5	52.0	57.5

USE FOR TYPEWRITTEN MATERIAL ONLY

A. 11. (Continued)

d. Detailed breakdown:

	No. of Penetrometers			
	2	4	6	8
Relay - Philco				
Rec. Antenna	1.2	1.2	1.2	1.2
Preamplifier	0.4	0.4	0.4	0.4
Power Divider	0.4	0.4	0.4	0.4
RF Switches	0.4	0.4	0.4	0.4
Receivers	2.0	4.0	6.0	8.0
Diversity Sel.	0.5	1.0	1.5	2.0
Subcarrier Demod.	0.4	0.8	1.2	1.6
Subcarrier Osc.	0.5	1.0	1.5	2.0
Connectors and Wiring	1.0	2.0	3.0	4.0
Miscellaneous	0.5	1.0	1.5	2.0
Subtotal, Lbs.	7.3	12.2	17.1	22.0
Additional GFE				
Interface Equipment	0.7	1.3	1.9	2.5
Transponder	12.0	12.0	12.0	12.0
High Gain S Band Antenna	7.0	7.0	7.0	7.0
Miscellaneous	5.5	5.5	5.5	5.5
Traveling Wave Tube (10 watt)	4.5	4.5	4.5	4.5
Battery	4.0	4.0	4.0	4.0
Subtotal, Lbs.	33.7	34.3	34.9	35.5

USE FOR TYPEWRITTEN MATERIAL ONLY

A. 12. Delivery Vehicle Weight for Priority #3 - Penetrometer to Relay to Earth

Weight Summary

	No. of Penetrometers			
	2	4	6	8
Payload	(15)	(30)	(45)	(60)
Penetrometer	10	20	30	40
Structures, Mechanisms	5	10	15	20
Guidance and Control	(40)	(40)	(40)	(40)
Communications	(41.0)	(46.5)	(52.0)	(57.5)
Propulsion Systems (See Appendix D)	(232.7)	(270.6)	(308.1)	(346.2)
Solid Propellant*	175.2	206.3	237.2	268.5
Solid Motor Inerts	17.6	20.7	23.7	26.9
Hydrazine Propellant	13.4	15.8	18.1	20.5
Hydrazine System Inerts	20.0	20.7	21.5	22.2
N_2 Reaction Control System	6.5	7.1	7.6	8.1
Total Delivery Vehicle Weight, Lbs.	(328.7)	(387.1)	(445.1)	(503.7)

* Retropropellant allocations are based on an ideal velocity capability of 6600 fps (2.01 Km/Sec) and a solid propellant specific impulse of 270 lb_f-sec/lb_m which is consistent with existing technology. Parametric studies will be conducted to evaluate retropropulsion requirements over the velocity range considered for this mission.

USE FOR TYPEWRITTEN MATERIAL ONLY

APPENDIX B

B. FLARE EXPERIMENT DESCRIPTION

1. Objective

The first objective of the flare experiment is to obtain bearing strength data. Second is to obtain chemical composition. The goal for this study is to identify the incremental weight per flare which could be carried by the Lunar Orbiter. Up to four flares are of interest. Interpretation of the surface layer thickness is desired. Required sensors to obtain the above data are: (1) Record time to first peak, (2) Record temperature of first peak - three photo multipliers are required, and (3) A diffraction grating could be used to give chemical composition.

2. Mission Considerations

The flare experiment should be considered for application later in time than the Block I spacecraft photo objectives. The mission should select experiment conditions which satisfy the flare first. The area of interest is the Apollo band, ± 45 degrees longitude and ± 5 degrees latitude.

The principle constraints are that the flare should impact in the dark surface at an impact angle greater than 45 degrees to the horizontal. The slant range is limited to 80 KM to obtain chemical composition by the diffraction grating.

3. Government Furnished Equipment or Information

The experiment should be considered as government furnished and the Douglas report was supplied. Assume the flash intensity scales with weight directly.

4. Additional Equipment

The additional equipment requirements which are to be defined by the study include, (1) Diffraction grating, (2) Photo multipliers, (3) Recorder, and (4) Delivery system.

5. Flare Description

a.	Weight	3/8 lb per flare (minimum)
b.	Shape	Spherical size and mass
c.	Temperature	0 degrees to 36 degrees C prelaunch and transit - 76 degrees C to + 11.8 degrees C must be in liquid phase for deployment and impact.

USE FOR TYPEWRITTEN MATERIAL ONLY

B. 5. (Continued)

d.	Impact Velocity	From 6,000 to 8,000 ft/sec is acceptable, but testing to date has been near 8,000 ft/sec
e.	Vibration	Assume compatible to Lunar Orbiter
f.	Number	Up to four flares are of interest

6. Delivery System Description

The Boeing Company will identify the delivery system requirements based on using idealized velocity. Assume an I_{sp} of 235 and propellant mass fraction of .85. Refer to the Chemical Propulsion Information Agency Handbook, Vol. II.

USE FOR TYPEWRITTEN MATERIAL ONLY

APPENDIX C

C. BALLOON EXPERIMENT DESCRIPTION

1. Objective

The primary objective of the experiment is to obtain high resolution photographs of the impact area before and after the balloon impacts. It is desired to obtain a photographic record of balloon inflation if possible.

2. Mission Considerations

The balloon experiment is considered applicable to missions after Block I photo missions have been satisfied. The balloon must be deployed and inflated in the sun. Inflation time can be anything greater than three seconds. The lighting angles at impact should be the same as for high resolution photos, i.e. 55 degrees - 70 degrees. The balloon should impact at an angle greater than 30 degrees to the horizontal. The impact velocity should be 6000 to 8000 ft/sec.

3. Government Furnished Equipment or Information

The experiment is considered to be government furnished. The reference material supplied is the Humble memo. Assume a tube launcher for the delivery mechanism. Mechanisms and experiments for the delivery system will be supplied by NASA-LRC.

4. Additional Equipment

As a result of the study of delivery flight mechanics Boeing will define ΔV requirements and size the rocket motor based on an I_{sp} of 235 and λ' of .85.

5. Balloon

The balloon is an inflatable aluminized mylar, 25 ft diameter, .5 mil thick mylar which weighs 8 lbs. It can be packaged in either a spherical or cylindrical can of 470 cubic inches, the total weighing 10 lbs. This includes the gas inflation system for the balloon. It must be inflated in the sun. The thermal requirements are to maintain temperature between - 60 degrees C to + 150 degrees C prior to deployment.

The electrical interface requires two squibs (one is redundant), which require 5 volts 10 amps to fire. A typical rocket motor for this application is the Atlantic Research MK III five inch spherical rocket motor.

A suitable spin up motor power requirement is 0.56 volts .8 amps. If it is spun, it requires arming.

USE FOR TYPEWRITTEN MATERIAL ONLY

APPENDIX D

D. DELIVERY VEHICLE MODEL

The penetrometer delivery vehicle is conceptually a scaled version of the Space General delivery vehicle described in Space General Report SGC 902R-6, "Lunar Survey Probe Delivery Vehicle Study, Final Report, Oral Briefing," Contract NAS9-4897, dated November 29, 1965 and Langley Research Center memorandum, "Possible Penetrometer-Type Experiments for Follow-On or Upgrated Lunar Orbiter," dated January 6, 1966. The delivery vehicle is herein defined by the following analytical model:

$$W_{DV} = \frac{1240\beta_R \left[59.43 + W_C + 7.5\,(P)\right]}{\left[1261 - 147\beta_R - 1.683\,t_v\right]} \qquad (1)$$

Where: $$R = e^{\left[\Delta V_R / g_e\, I_{S_{SP}}\right]} \qquad (2)$$

W_{DV} = Delivery Vehicle Weight (lbs.)

W_C = Delivery Vehicle Communications Weight (lbs.)

t_v = Hover time (seconds). In the example, one second was used for priority 1 missions where the communication relay is in the Lunar Orbiter, and 8 seconds where it is in the delivery vehicle.

P = Number of penetrometers.

ΔV_R = Delivery vehicle retro velocity measurement (fps). The example was based in a value of 6600 fps.

g_e = Gravity Constant, Earth (ft/sec^2)

$I_{S_{SP}}$ = Solid motor retropropellant specific impulse (lb_f-sec/1 b_m). A value of 270 lb_f-sec/1 b_m was used in the example.

Systems weight breakdown of the delivery vehicle is defined by the following analytical models used to derive expression (1).

$$W_{DV} = W_Y + W_C + W_{GC} + W_{PS_T} \qquad (3)$$

Where: W_Y = Delivery Vehicle Payload Weight (lbs.)

W_C = Delivery Vehicle Communications Weight (lbs.)

W_{GC} = Delivery Vehicle Guidance and Control Weight (lbs.)

W_{PS_T} = Delivery Vehicle Total Propulsion System Weight (lbs.)

USE FOR TYPEWRITTEN MATERIAL ONLY

D. (Continued)

$$W_Y = 7.5(P) = W_{YS} + W_{YP} \quad (4)$$

Where: P = Number of Penetrometers

W_{YS} = Payload structure and mechanisms weight (lbs.)

W_{YP} = Penetrometer weight (lbs.)

$$W_{YS} = 2.5\ (P) \quad (5)$$

$$W_{YP} = 5\ (P) \quad (6)$$

W_C = Communications weight described in Appendices A, B, and C (7)

$$W_{GC} = 40 \quad (8)$$

$$W_{PS_T} = 19.43 + \frac{W_{DV}}{1240\beta_R}\left[1387\beta_R - 1261 + 1.683\ t_v\right] \quad (9)$$

The total propulsion system weight breakdown is defined by the following analytical models used to derive expression (9).

$$W_{PS_T} = W_M + W_H + W_{RCS} \quad (10)$$

Where: W_M = Solid propellant retromotor weight (lbs.)

W_H = Hydrazine system weight (lbs.)

W_{RCS} = Nitrogen reaction control system weight (lbs.)

$$W_M = 1.1\ W_{SP} \quad (11)$$

Where: W_{SP} = Propellant weight, Solid propellant retromotor

$$W_{SP} = W_{DV}\ \frac{\left[\beta_R - 1\right]}{\beta_R} \quad (12)$$

$$W_H = 16 + 1.3\ W_{HP} \quad (13)$$

Where: W_{HP} = Hydrazine propellant weight (lbs.)

$$W_{HP} = \frac{W_{DV}}{\beta_R}\left[\frac{13.727\beta_R - 61.045 + t_v}{1127.3}\right] \quad (14)$$

USE FOR TYPEWRITTEN MATERIAL ONLY

D. (Continued)

The hydrazine propellant weight breakdown is defined by the following analytical models used to derive expression (14):

$$W_{HP} = 1.1 \left[W_{P_\Delta} + W_{P_A} + W_{P_R} + W_{P_H} \right] \tag{15}$$

Where: W_{P_Δ} = Hydrazine propellant weight (lbs.) required for vernier correction to account for 3-sigma total impulse tolerances in the solid propellant retrometer.

W_{P_A} = Hydrazine propellant weight (lbs.) required for thrust vector control due to thrust vector misalignment.

W_{P_R} = Hydrazine propellant weight (lbs.) required to nullify velocity accumulated in the free fall from 13000 feet to 270 feet.

W_{P_H} = Hydrazine propellant weight (lbs.) required for hovering at 270 feet during penetrometer ejection.

$$W_P = 0.006\left(I_{S_{SP}} W_{SP}\right) \tag{16}$$

$$W_{P_A} = 0.0022\left(I_{S_{SP}} W_{SP}\right) \tag{17}$$

$$W_{P_R} = .0603 \left(W_{D_V} - W_{SP}\right) \tag{18}$$

$$W_{P_H} = \frac{\left(W_{DV} - W_{SP}\right)}{1240} t_v \tag{19}$$

The nitrogen reaction control system weight and weight breakdown are defined by the following analytical models used to derive expression (20).

$$W_{RCS} = 3.43 + 2.5 W_{N_2} \tag{20}$$

Where: W_{N_2} = Reaction control system nitrogen gas weight (lbs.)

$$W_{RCS} = W_F + W_B + W_{N_2} \tag{21}$$

Where: W_F = Reaction control system fixed weights (lbs.) for valves, plumbing and thrusters.

W_B = Reaction control system nitrogen gas bottle weight (lbs.)

USE FOR TYPEWRITTEN MATERIAL ONLY

D. (Continued)

$$W_F = 3.43 \tag{22}$$

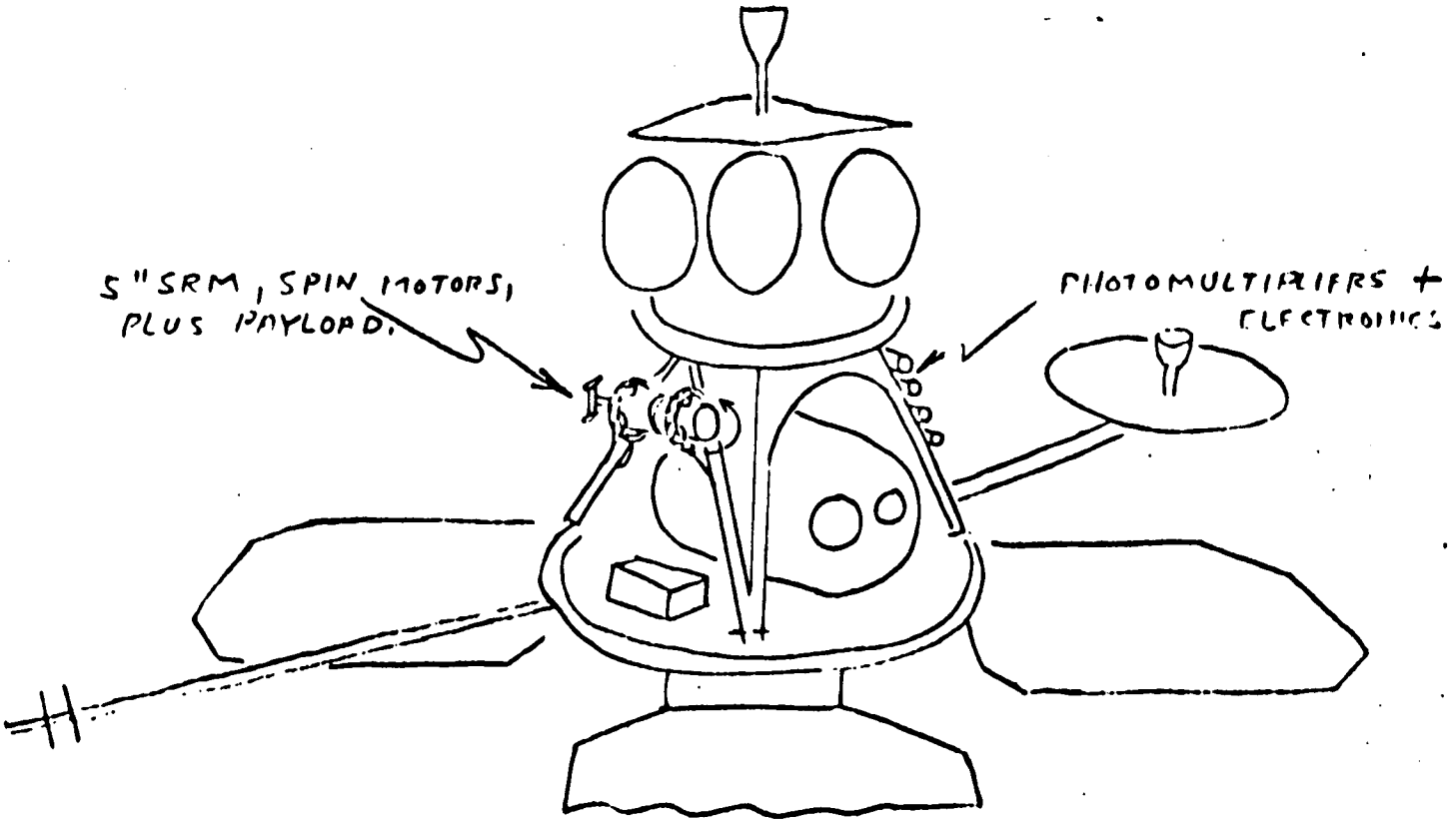
$$W_B = 1.5\ W_{N_2} \tag{23}$$

$$W_{N_2} = 0.0914\ W_{HP} \tag{24}$$

USE FOR TYPEWRITTEN MATERIAL ONLY

BY ..... DATE 4/21/66 SUBJECT ..... SHEET NO. ..... OF .....
CHKD BY ..... DATE ..... JOB NO. .....

L/O WITH ELF OR BALLOON DEPLOYMENT MECHANISM (OR ELF)

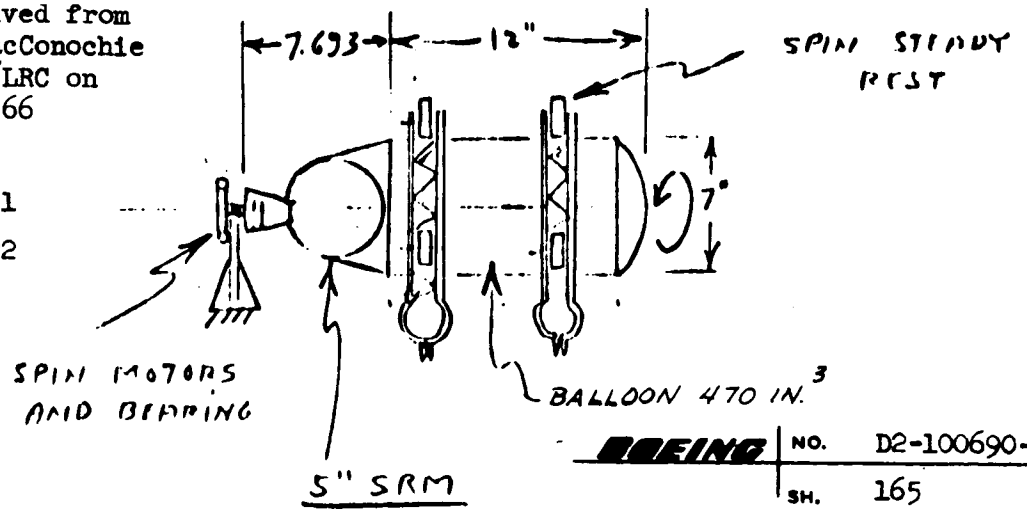


LOW DENSITY IMPACTER

NOTE: Sketches received from I. MacConochie NASA/LRC on 4-27-66

FIGURE 1

Page 22



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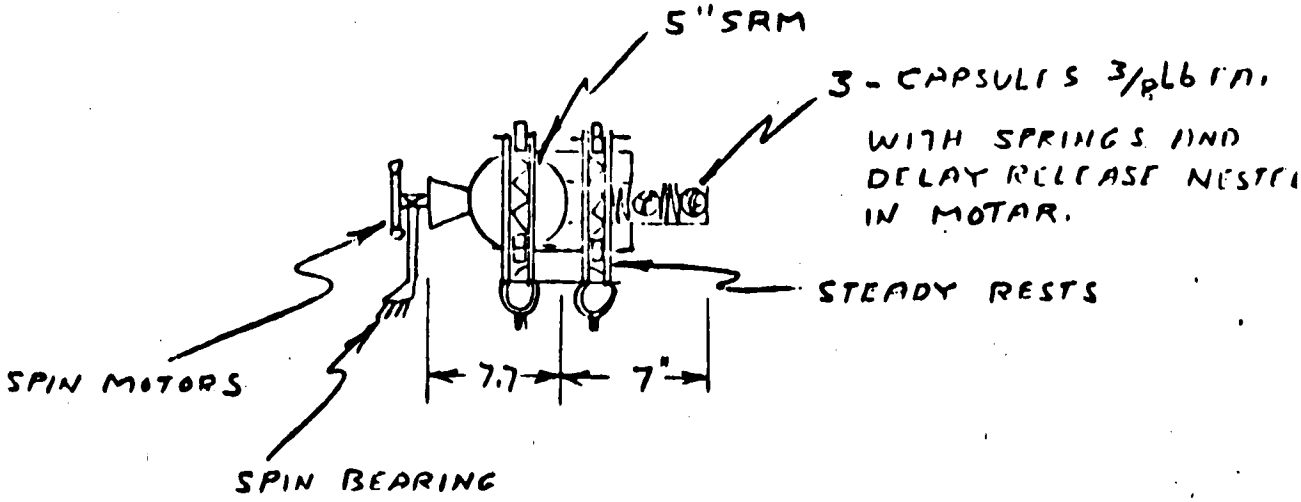
ELF DEPLOYMENT MECHANISM



DELIVERY SYSTEM - SPHERICAL PENETROMETERS

NOTE: Sketches received from I. MacConochie NASA/LRC on 4-27-66

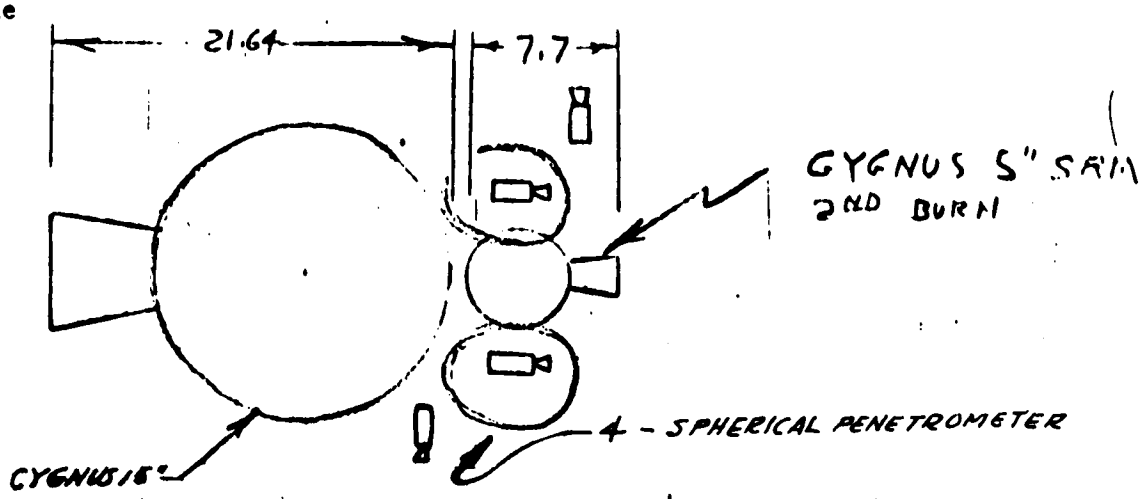


FIGURE 2

Rec'd 6/20/66

LUNAR ORBITER PMO
CORRESPONDENCE
ORIGINAL ID: ORS
COPIES ID:

UNITED STATES GOVERNMENT

Memorandum

Langley Research Center

Helberg

TO : Boeing Company, Aerospace Division
Attention: Mr. E. J. Nicholson
P. O. Box 3995, Seattle, Washington 98124

DATE: June 13, 1966

FROM : Langley Research Center, Flight Vehicles and Systems Division
Spacecraft Structures Section - I. O. MacConochie - TRCO

Karpt
Rawls
Smith, W
Nicholson
Nicholas
Aschke

SUBJECT: Response to "Report of Meeting Between NASA and Contractor Personnel Held 26-27 of April Concerning Study Contract of Lunar Orbiter Adaptability to Experiments to Evaluate Lunar Surface Characteristics" NAS1-6216

REFERENCE: Letter to L. W. Fitchett and I. O. MacConochie from E. J. Nicholson, TBC, dated May 19, 1966, on the same subject.

Yamauchi
Montgomery

The subject report has been reviewed and the following comments are offered for the paragraphs indicated. Appendixes and paragraphs below refer to the subject document.

Appendix A, Paragraph 3 (a)

a. The nominal impact velocity is 150 ft/sec. Impact velocity should be known to ±5% below 50 ft/sec and ±10% from 50 to 250 ft/sec. An impact velocity of 200 ft/sec or less is preferred since some degradation in performance of the penetrometer occurs above this figure.

Appendix A, Paragraph 4 (b)

b. Results of Space General study should be considered as information to assist the Contractor in considering alternate modes of deployment of the penetrometers pursuant to paragraph 4.5 of the work statement. Also, pursuant to paragraph 4.5, which is not specific regarding method of deployment, spin stabilized delivery systems shall be considered as an alternate mode of deployment to determine if such a system lends itself better to the Lunar Orbiter application.

Appendix B, Paragraph 2

The flare experiment should be considered as a supplementary experiment on the spacecraft after the Block I mission objectives have been satisfied. Once selected for flight, the mission profile should be such as to satisfy the requirements of the flare first.

Appendix C, Paragraph 2, 3rd sentence in paragraph

A minimum of 15 seconds should be allowed for deployment and inflation of the balloon in the sun.

Appendix C, Paragraph 5, last two sentences in paragraph 5

The electrical interface requires 5 volts, 10 amps to fire two squibs (one is redundant). A typical rocket motor for this application

is the LRC 5-inch spherical (Cygnus 5). If it is spun before deployment a spin-up motor power requirement is 0.50 volts, .8 amps to fire the squibs. A delay of approximately 1.5 seconds is required from initiation of the spin-up motor to initiation of the deployment motor. Both the spin-up motors and the deployment motor will require arming.

As pointed out at the May 26, 27 meeting by the writer, the exact experiment configuration depends on the study results and vice versa. It is hoped that the technical information given at the meeting and the additional information supplied above will assist the Contractor in carrying out the study in accordance with the subject contract.

Ian O. MacConochie

Ian O. MacConochie
Technical Representation
of the Contracting Officer

cc:
Ken L. Wadlin
NASA Resident Engineer
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R. D. Eishenhart
Boeing Representative
1120 West Mercury Boulevard
Hampton, Virginia 23366

APPENDIX B

WORKING PAPERS ON FLARE

LUNAR ORBITER GROWTH

LUNAR FLARE EXPERIMENT

1.0 INTRODUCTION

This report is in response to a request from the NASA, Langley Research Center, to determine the adaptability of the Lunar Orbiter to a Lunar Flare Experiment. This experiment is similar in concept to the Early Lunar Flare (ELF) experiment studied by Douglas Missiles and Space Systems Division (MSSD) and described in Reference 1. The ELF modifications include using the Lunar Orbiter spacecraft to deliver the flares and to detect and transmit the flare data.

The Lunar Flare Experiment involves impacting a chemical flare on the surface and observing the resulting light flash. Various characteristics of the flash are to be recorded and analyzed to yield information on lunar surface bearing strength, chemical composition and depth of a porous layer over a solid surface. The data is transmitted to Earth by way of the existing Orbiter communications link and recorded by the existing ground reconstruction system.

2.0 LUNAR FLARE EXPERIMENT SUMMARY

The scientific objectives of the Lunar Flare Experiment are to determine lunar surface bearing strength, thickness of surface layer and surface chemical composition. Physical measurements to achieve these objectives include rise time to "first light peak," color temperature of "first peak" and spectral energy distribution of "second light peak." A parametric study of the flare experiment resulted in no final instrumentation configuration. However, a set of typical characteristics and constraints have been established, which typify the requirements for successful performance of the Lunar Flare Experiment using the Lunar Orbiter as both a transporting vehicle and an instrument platform. These characteristics are summarized in the following list for the case of a 5-pound flare and a 50-kilometer range.

Sensor Characteristics for Time to First Peak Detection

Optics	5.08 cm, f/1
Detector	(1) Photomultiplier (S-1 Response)
Field of View	20° Total (18 km on Ground)
Signal-to-Noise	220
Time Resolution	4.3 μ second
Wavelength Region	5500Å-8500Å
Data Retrieval	Real-Time - High-Gain Antenna Readout
Temperature Environment	10°-110°F
Dimensions	12" x 4" x 4" (192 cu. in.)
Input Power	2 watts (Average), 28 volts d.c.
Output Signal Characteristics	5 volts peak-to-peak, 0-230 KHz
Weight	4 pounds

Sensor Characteristics for Color Temperature Determination of First Peak

Optics	5.08 cm, f/1
Detector	6 Photomultipliers (S-1 Response)
Field of View	20° Total (18 km on Ground)
Signal-to-Noise	140
Spectral Bandpass	300Å
Wavelength Region	6 Regions (4000Å-7000Å)
Data Retrieval	Tape Storage - Sequential Readout
Temperature Environment	10°F-110°F
Dimensions	5" x 7" x 12" (420 cu. in.)
Input Power	2 watts (each tube)
	12 watts (total)
	28 vdc
Output Signal Characteristics	5 volts peak-to-peak (each tube)
	0-3 KHz
Weight	3 Pounds (each tube)
	18 Pounds (total)

Sensor Characteristics for Spectral Energy Detection

Optcis	Orbiter 24-in., f/5.6 Lens and Transmission Diffraction Grating
Detector	SO-243 Film
Field of View	12 km Cross-Range
	4.5 km Downrange
Minimum Density Level	Density of 0.2 above Gross Fog
Spectral Resolution	0.7Å
Plate Dispersion	50 Å/mm
Wavelength Region	3500Å-6500Å
Data Retrieval	Line Scan Readout - Video Link
Temperature Environment	Compatible with Existing Orbiter Photographic Subsystem
Dimensions	6" x 6" x 2" (72 cu. in.)
Power	1 Second Surge of 100-200 watts
Weight	6 Pounds

3.0 EXPERIMENT CONDITIONS AND REQUIREMENTS

The Lunar Flare Experiment discussed here is similar to the ELF concept with the following modifications:

(1) The flares will be deployed to the Lunar surface from Lunar Orbiter spacecraft, either singly or in groups of two or three at a time.

(2) The flares will be smaller than the original 100 pound size.

(3) Observation of the visible light resulting from the flare impact will be accomplished using sensors installed on the Lunar Orbiter spacecraft.

(4) Data will be transmitted to Earth via the existing Lunar Orbiter S-band link.

Figure 3.0-1 is a pictorial display of a lunar flare event adapted to the orbiter.

3.1 The Flare

The original flare design, as described in Reference 1, consists of a hollow aluminum structure filled with a mixture of chlorine trifluoride (CTF) and fine aluminum wire (Al). The passage of a high-intensity shock wave through the reagents initiates a violent exothermic reaction. A spherical configuration is assumed for the purpose of this study. The aluminum wall thickness for the sphere is expected to be approximately 0.25 inches with relationship between size and weight as shown in Figure 3.1-1. CTF is installed in the flare shell in the liquid phase (-76°C to +11.8°C). The flare is then allowed to warm to room temperature and is then capped off. Handling is similar to that for Class A explosives. The flare temperature must be maintained between 0° and 36°C during ground and flight operations.

3.1.1 Impact Conditions.- Vacuum testing of flare impacts has been conducted over a narrow range of impact velocities (2.2-2.8 km/second) and the relationship between impact velocity and light output is presently undefined. It is expected that there will be a minimum impact velocity below which the CTF-Al reaction is not initiated. At the higher velocities, the ability to test full scale models in existing facilities may be a constraint. Based on existing data, velocities between 2.2 and 2.8 kilometers per second are acceptable. Vacuum testing has included impact angles measured from the target surface as low as 42° for dust targets. The intent of the past testing was to achieve impact angles as close to 90° as possible. Most of the impacts into solid slab targets were in fact at 90°, and only the experimental difficulties involved in supporting unconsolidated surface at angles greater than 40° or so, prevented all testing from being done at impact angles of 90°. Existing test data indicates no relation between flare output and impact angle over the range of impact angles from 40° to 90°. For the purpose of this study, an impact angle of 45° or greater is assumed.

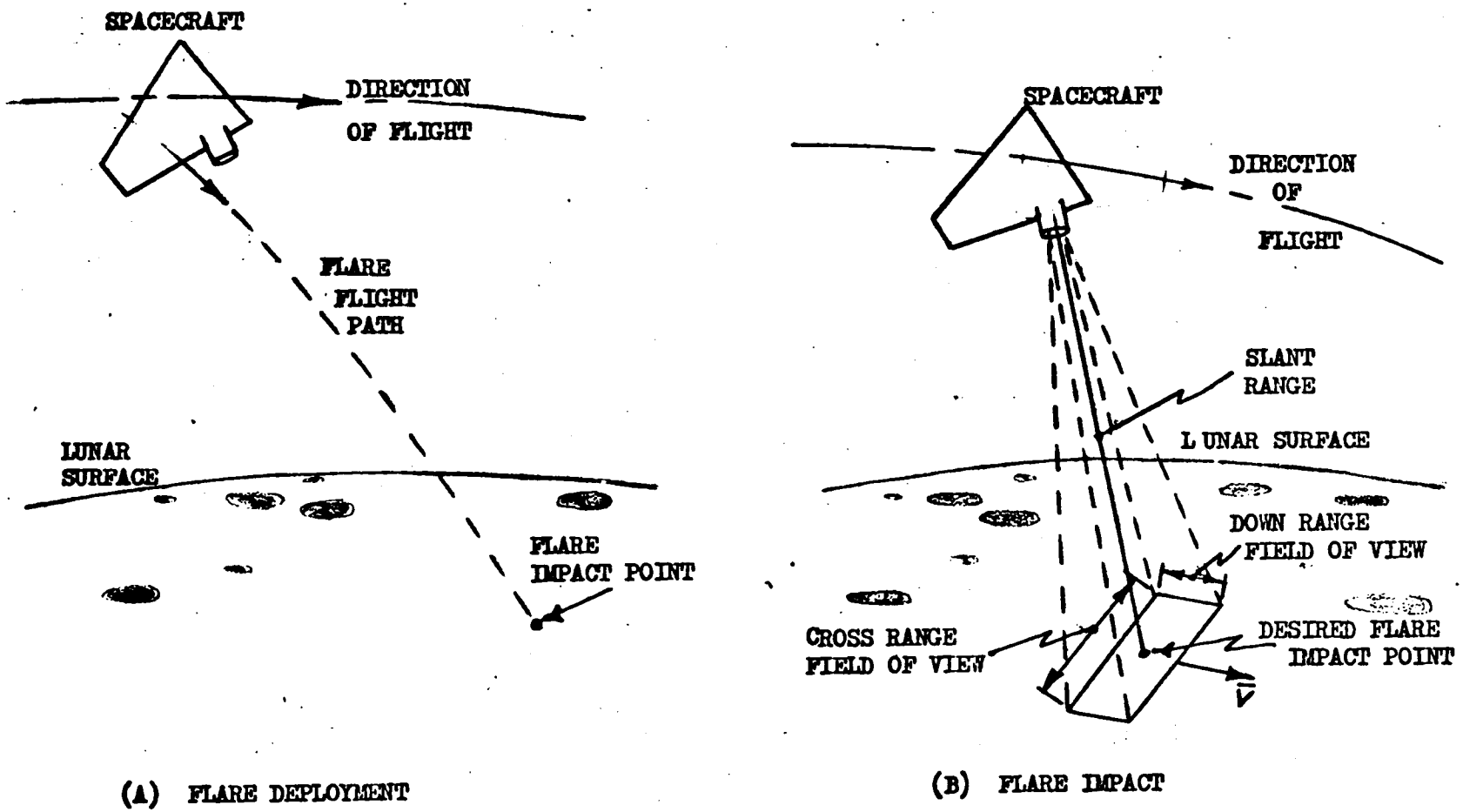


FIGURE 3.0-1 FLARE EVENT

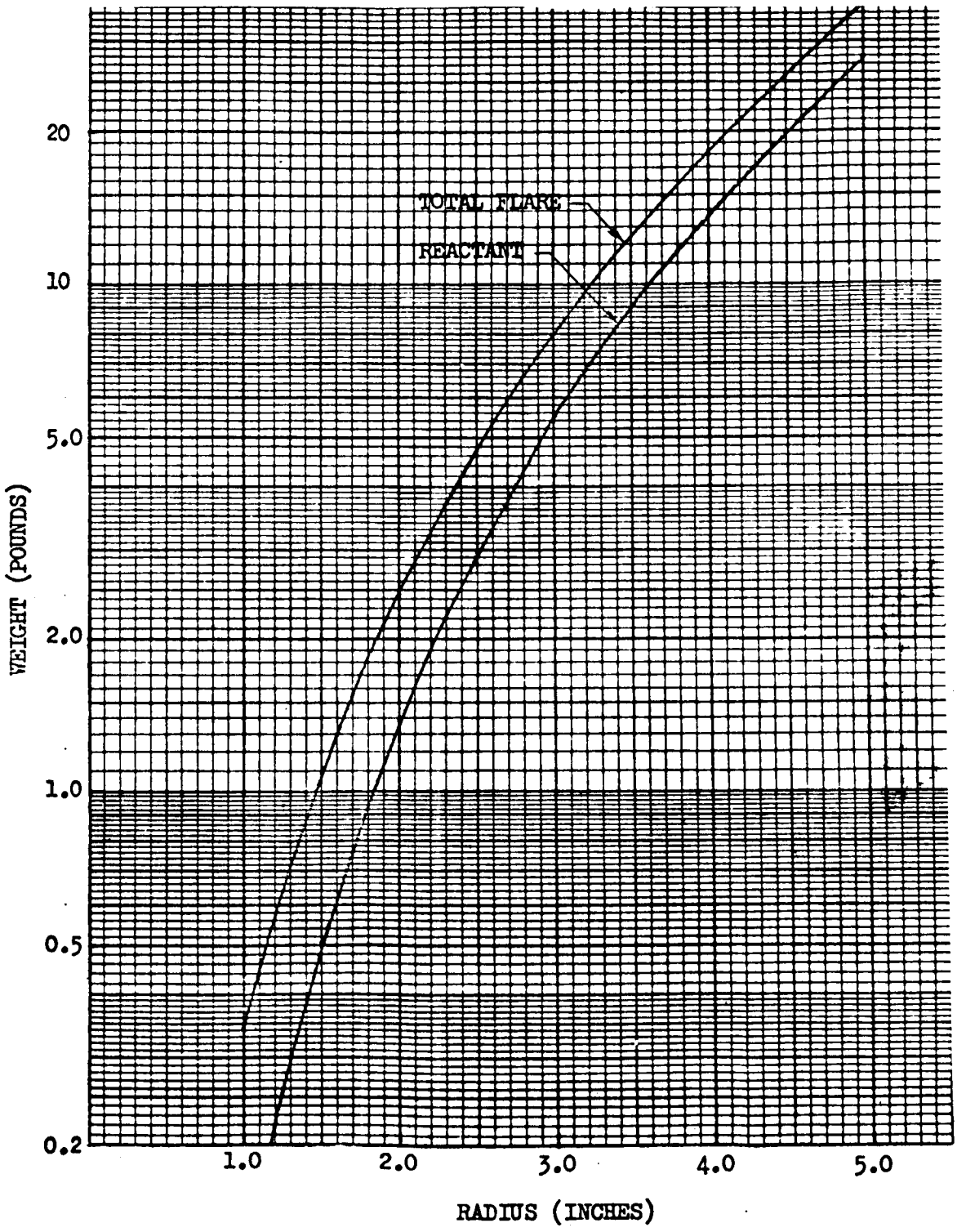


FIGURE 3.1-1 FLARE WEIGHT VS. SIZE

3.1.2 Flare radiation data.- Upon impact, a high-speed chemical reaction takes place between the chlorine trifluoride and the aluminum. This reaction is completed in a few microseconds and produces a high intensity burst of radiation with an effective black body temperature of about 4000°K. The radiation emitted during this "first peak" is centered around 7200A and reaches a maximum intensity in a time that is related to the bearing strength of the target material. Following the emission of this black body radiation, the flash intensity decreases for a short period and then increases to a "second peak," generally higher than the first. The "second peak" consists of continium radiation and spectral features characteristic of the target materials.

The observable parameters of most interest are: time to "first peak," color temperature of "first peak," and spectral content of "second peak." As mentioned previously, the time to "first Peak" shows a dependence on target bearing strength (Reference 1). Specifically, as the target is made softer, the time to "first peak" increases and times of 30 to 200 microseconds were recorded for targets of serpentinite and dust, respectively. A plot of "first peak" time versus K_ϕ of the target is shown in Figure 3.1-2. K_ϕ is a parameter which reflects the degree of immobility of particles making up the target and is thus proportional to bearing strength (Reference 1).

The apparent color temperature of the "first peak" radiation has been shown (Reference 1) to have some correlation to the thickness of the unconsolidated surface layer of the target. Testing was done by placing a layer of known thickness of dust or gravel over a solid slab and impacting a CTF-Al filled sphere into the loose material. The spectral character of the "first peak" radiation was recorded at eight wavelength bands between 4000Å and 7000Å and the data obtained compared to black-body curves for several temperatures and the closest fit determined. There is considerable scatter in the experimental data as shown by Figure 3.1-3.

The spectral content of the "second peak" radiation is indicative of the chemical composition of the target. The high temperature of the CTF-Al reaction is sufficient to excite target constituent atoms and molecules which, upon returning to a lower energy state, emit radiation at wavelengths characteristic of the materials involved. The time duration of the spectral radiation is quite long compared to the "first peak" rise time. Spectral emission can be expected to persist for approximately 10 milliseconds after flare impact (Reference 1).

3.2 Observation Requirements

The constraints imposed on the Lunar Orbiter mission for the flare experiment are briefly discussed below.

3.2.1 Surface illumination.- It is desirable to eliminate as much extraneous illumination as possible from the target area at time of impact. This reduces the background noise input to any sensor used to detect the flare radiation. For this reason, the target areas should be on the dark side of the terminator at time of impact.

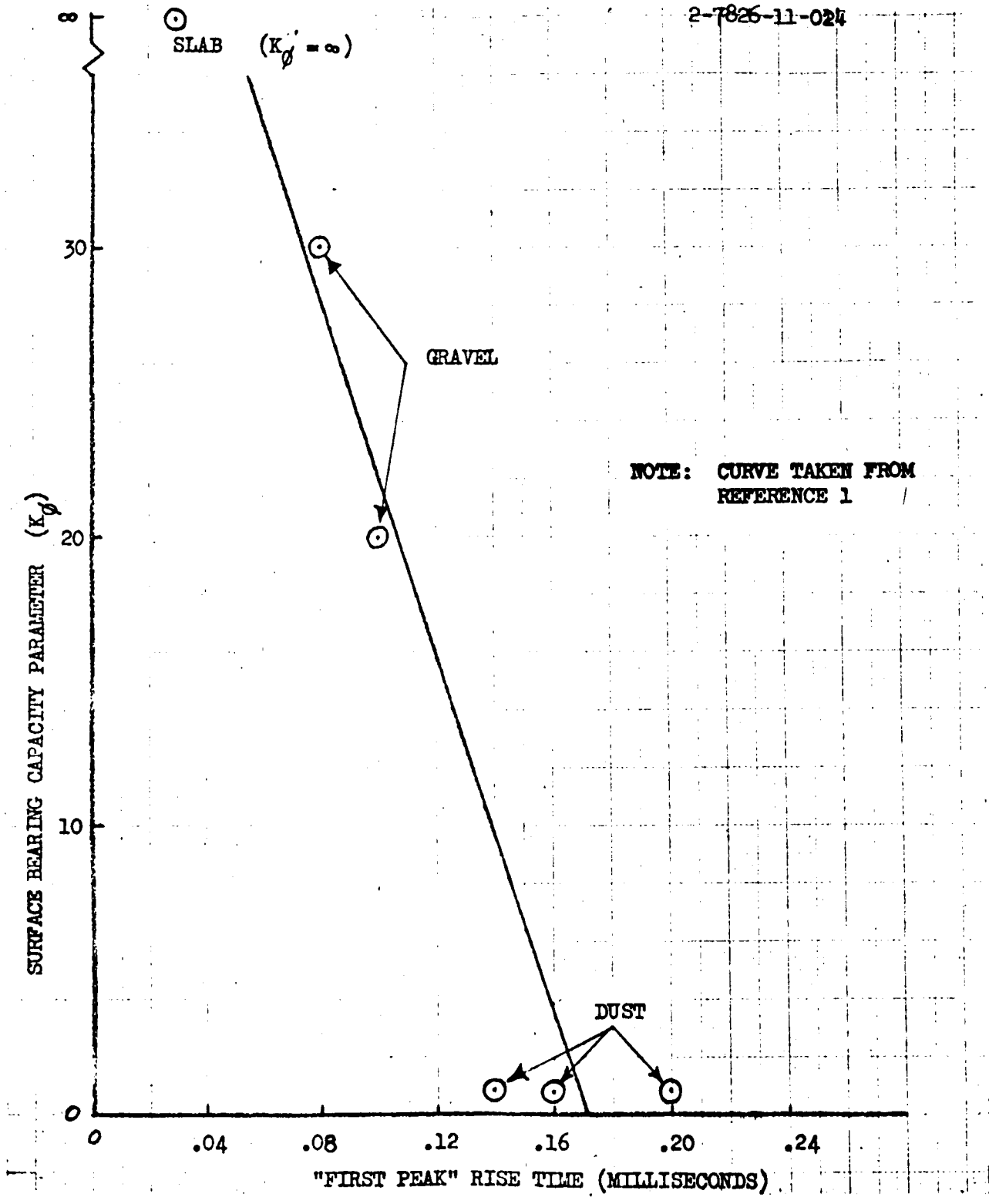


FIGURE 3.1-2 BEARING CAPACITY PARAMETER VS. FIRST PEAK RISE TIME

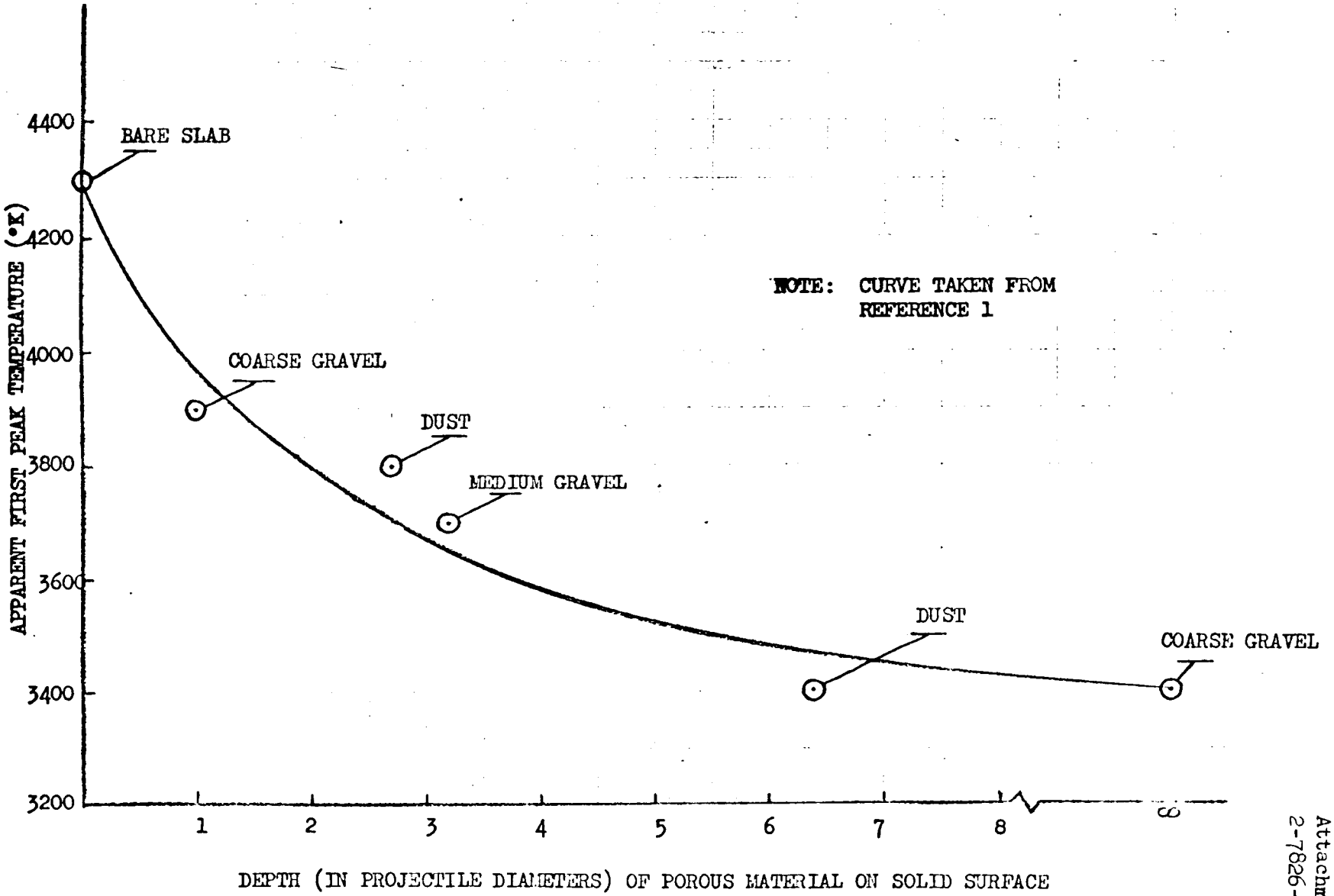


FIGURE 3.1-3 POROUS MATERIAL DEPTH VS. APPARENT FIRST PEAK TEMPERATURE

3.2.2 Slant range.- The light flux incident on a detector aboard the Orbiter is inversely proportional to the square of the Orbiter-to-flare range at impact. In the case of spectral measurements, the resolution is energy limited. This is discussed further in the section in instrumentation.

3.2.3 Viewing angle.- An obvious constraint on this parameter is that the flare must not be past the Orbiter's visual horizon. The distance (and consequently the viewing angle) to the horizon is a function of both orbiter altitude and lunar surface roughness. In a mountainous area or one in a region of high crater density, the allowable range angle is greatly reduced. The impact point must not be occulted from the Orbiter by "terrain" features. Thus, an allowable range angle will need to be determined for each prospective impact point. Multiple flare deployment will improve experimental reliability.

3.2.4 Sensor pointing.- To avoid a requirement for extremely wide field of view sensors, sensor pointing should be made compatible with Lunar Orbiter system performance. This performance is presently on the order of ± 0.5 degree and would seem adequate at this time.

3.2.5 High gain antenna pointing.- In order to achieve real-time readout of sensor outputs during flare impact, the Orbiter high gain system must be utilized. This is a necessity because of the bandwidth required to transmit a rapidly rising pulse such as the flare "first peak." Thus, the high gain antenna must be pointed at the Earth at time of impact. This restricts the positioning of sensors on-board the spacecraft. Specifically, such sensors should be located so that they look close to the spacecraft X-Z plane so that the existing high gain antenna and boom configuration may be used.

3.2.6 Attitude stability.- Vehicle rates at time of impact tend to degrade pointing accuracy. Attitude rates also will tend to degrade spectral data taken with a dispersing system such as would be required for analysis of the "second peak" flare output. Vehicle rotation in the plane of dispersion will tend to broaden the spectral lines and so reduce the spectral resolution. The effect of attitude stability on resolution is discussed in the instrumentation section.

3.2.7 Orbiter scene rate.- Scene rate (V/H) affects the energy density in the spectrograph film plane for a given spectral dispersion and is discussed in the instrument section.

3.2.8 Flare impact point.- Uncertainties in impact point due to deployment errors, trajectory variables, and terrain characteristics, all combine to increase the required sensor field of view. In the case of a photo-emissive detector, this tends to decrease the signal to noise ratio of the system. The spectral data taken will be recorded on film in the existing orbiter photo subsystem using the 24-inch lens rather than the 3.35-inch lens because of its larger collecting aperture. The lens angular field of view is approximately $5^{\circ} \times 20^{\circ}$ and the impacting flare must be well within this field to permit spectral dispersion (Figure 3.0-1).

3.2.9 Flare energy.- The flare must be sized to provide sufficient radiation flux for detection by the sensors. A flare size versus sensor sensitivity and complexity trade may be made after the type of sensors to be considered are selected.

4.0 SPACECRAFT INSTRUMENTATION

Additions and modifications to the Lunar Orbiter payload are required to permit measurement of the significant flare event parameters. The parameters considered for measurement will be time to "first peak," apparent temperature of "first peak," and distribution of spectral energy.

4.1 Time to "First Peak"

To apply the constraints pertaining to the measurement of the "first peak" rise time to the design of the instrumentation, it is necessary to derive an equation for the "first peak" signal power available at the sensor. An expression for signal-to-noise ratio is also presented.

4.1.1 Signal power.- For a flare event, the signal power may be obtained from the expression:

$$S_1 = P\,A_e \text{ lumens} \qquad (4.1\text{-}1)$$

where:

P = average power per unit area at the collector.

A_e = effective lens area.

The average power, assuming the flare radiates uniformly into a hemisphere (Reference 1), per unit area at the collector is:

$$P = \frac{E_o}{2\pi R^2 t} \text{ lumens/cm}^2 \qquad (4.1\text{-}2)$$

where:

E_o = total flare energy at visible wavelengths (scales directly with flare reactant weight) in lumen seconds.

R = spacecraft to flare range at time of impact in centimeters.

t = duration of single flare event in seconds.

The effective lens area is:

$$A_e = \pi\left(\frac{D_e}{2}\right)^2 \text{ (centimeters}^2\text{)} \qquad (4.1\text{-}3)$$

where:

D_e = optical clear aperture (centimeters)

$$D_e = \frac{f}{f/n}$$

where:

f = lens focal length (centimeters)

f/n = lens f-number

thus $$A_e = \pi \left(\frac{f}{2\ f/n}\right)^2 \text{ (centimeters}^2\text{)} \qquad (4.1\text{-}4)$$

and $$S_1 = P\pi \left(\frac{f}{2\ f/n}\right)^2 \text{ (lumens)} \qquad (4.1\text{-}5)$$

Now the required lens focal length (f) may be determined from the expression:

$$\frac{f}{D_s} = \frac{R}{L} = \frac{\cot \theta}{2} \qquad (4.1\text{-}6)$$

where:

R = range (centimeters)

L = diameter of observed area of lunar surface (centimeters)

D_s = diameter of photo cathode (centimeters)

θ = half angle of optics (degrees)

$$f = \frac{D_s}{2} \cot \theta \text{ (centimeters)} \qquad (4.1\text{-}7)$$

A graph of L versus R for $\theta = 10^\circ$ is shown as Curve 1 in Figure 4.3-9. Then:

$$A_e = \frac{\pi}{16} \frac{D_s^2}{(f/n)^2} \cot^2 \theta \text{ (centimeters}^2\text{)} \qquad (4.1\text{-}8)$$

and

$$S_1 = \frac{E_o\ D_s^2\ \cot^2 \theta}{32(f/n)^2\ R^2\ t} \text{ (lumens)} \qquad (4.1\text{-}9)$$

From this, it may be seen that the useable power may be increased by decreasing the lens f-number (f/n), the range (R), or the field angle (θ) or by increasing flare energy (E_o), or photocathode diameter (D_s). The time (t) is the duration of the flare luminous output.

4.1.2 Signal to noise.- Noise in a photomultiplier is predominately shot noise due to the random emission of electrons from the photocathode and as such is proportional to the square root of the bandwidth. Specifically, the noise current at the photocathode (I_n) is:

$$I_n = (2eI_{dc} \Delta f)^{1/2} \text{ (amperes)} \qquad (4.1\text{-}10)$$

where

e = electronic charge (1.59×10^{-19} coulomb)

Δf = bandwidth - Hz

I_{dc} = total average cathode current (amperes)

$$I_{dc} = I_d + I_s + I_b$$

where

I_d = dark current (amperes)

I_s = signal current (amperes)

I_b = background current (amperes)

thus

$$I_n = \left[2e\ (I_d + I_s + I_b)\ \Delta f\right]^{\frac{1}{2}} \text{ amperes} \qquad (4.1\text{-}11)$$

The signal current (I_s) may be found from the expression:

$$I_s = S_1\ TK_e \qquad (4.1\text{-}12)$$

where

S_1 = signal power into collector optics (lumens)

T = transmission of optics

K_e = luminous sensitivity of photocathode (amperes/lumens)

The dark current (I_d) is a function of the area and work function of the photocathode as well as its temperature. Dark current specifications for various photocathodes may be found in the manufacturer's literature.

The background current (I_b) is a function of the background brightness seen by the sensor, the f-number of the sensor optics, and the transmission of the optics. The background illumination at the detector surface may be found from the expression:

$$P_b = \frac{\pi}{4} \frac{B}{(f/n)^2} T \text{ lumens/cm}^2 \qquad (4.1\text{-}13)$$

where:

B = object brightness (candles/cm^2)

T = optics transmission

Thus

$$P_b = \frac{\pi BT D_e^2}{4(f)^2} \qquad (4.1\text{-}14)$$

The total background power at the detector is:

$$S_b = (P_b) \pi \left(\frac{D_s}{2}\right)^2$$

$$S_b = P_b \pi \left[(f) \tan \theta\right]^2 \qquad (4.1\text{-}15)$$

The current due to the background illumination is thus,

$$I_b = S_b K_e = \frac{\pi^2}{4} BD_e^2 T \tan^2 \theta K_e \qquad (4.1\text{-}16)$$

The nosie current (I_n) may then be written:

$$I_n = \left[2e \left(I_d + S_1 T K_e + \frac{\pi^2}{4} B D_e^2 T \tan^2 \theta K_e\right) \Delta f\right]^{1/2} \qquad (4.1\text{-}17)$$

and the signal-to-noise ratio is then,

$$\frac{I_s}{I_n} = \frac{S_1 T K_e}{\left[2e \left(I_d + S_1 T K_e + \frac{\pi^2}{4} B D_e^2 T \tan^2 \theta K_e\right) \Delta f\right]^{1/2}} \qquad (4.1\text{-}18)$$

Figures 4.1-1 and 4.1-2 are plots of S/N versus R for several combinations of flare weight and lens aperture. The parameters assumed are as follows:

E_o = 1.058×10^7 lumen-second/pound(reactants) (Reference 1)

θ = 10° circular (field half angle)

t = 10 milliseconds

T = 0.5

K_e = 20 μ amps/lumen (S-1 photocathode)

B = 4.28×10^{-5} candles/cm^2

I_d = 4×10^{-13} amps (S-1 photocathode)

D_e = 1.27 cm, 2.54 cm, 5.08 cm

Δf = 230 KHz (Lunar Orbiter S-band link)

4.1.3 Instrument description.- For the typical experiment just discussed, the instrumentation to be added to the Lunar Orbiter would have roughly the following characteristics:

Size	12" x 4" x 4"
Weight	4 pounds
Power	2 watts
Sensor	Photomultiplier (S-1)
Optics	5.08 cm focal length, f/1

4.1.4 Data recovery.- The output of a sensor such as that described above, when viewing a lunar flare event, will consist of a short initial pulse with a minimum rise time on the order of 30 μ seconds, followed by a second pulse with a duration of a few milliseconds. The time from initial pulse onset to peak is the primary parameter of interest and it is proposed that the sensor output be transmitted to Earth via the existing high gain video link in real-time to allow minimum distortion of the pulse prior to data analysis. The pulse rise time of 30 μ seconds or more is compatible with the video bandwidth of 230 kcps. The sensor output must not exceed 6 volts peak-to-peak to be compatible with the transmitter modulator input.

The uncertainty in the measurement of the time to "first peak" will be on the order of the minimum pulse which bandwidth limitations will allow to be passed undistorted. In the case of the 230 KHz bandwidth, this is on the order of 4.5 microseconds or 13% of the minimum "first peak" rise time from Reference 1.

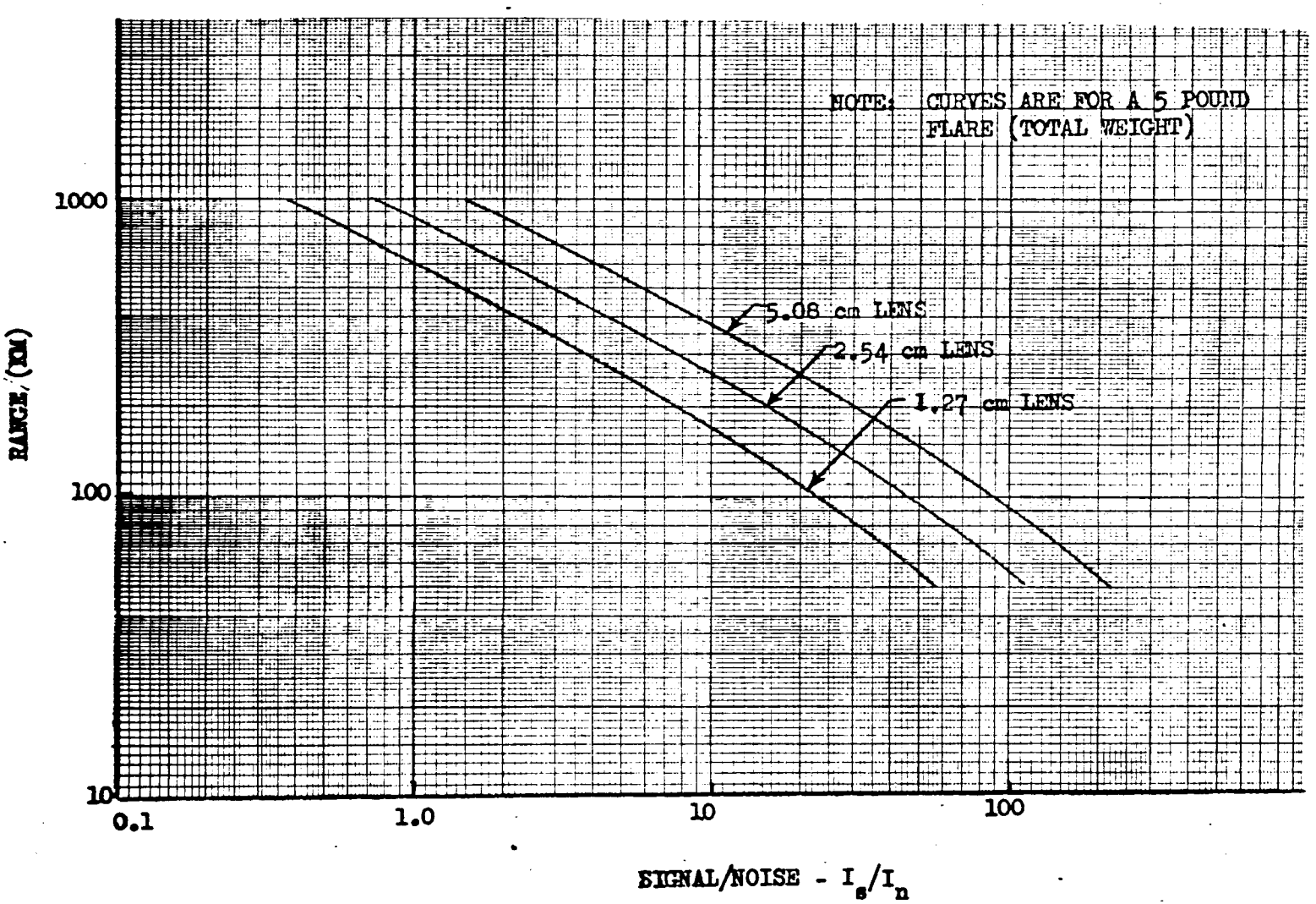


FIGURE 4.1-1 SIGNAL/NOISE FOR FIRST PEAK DETECTOR

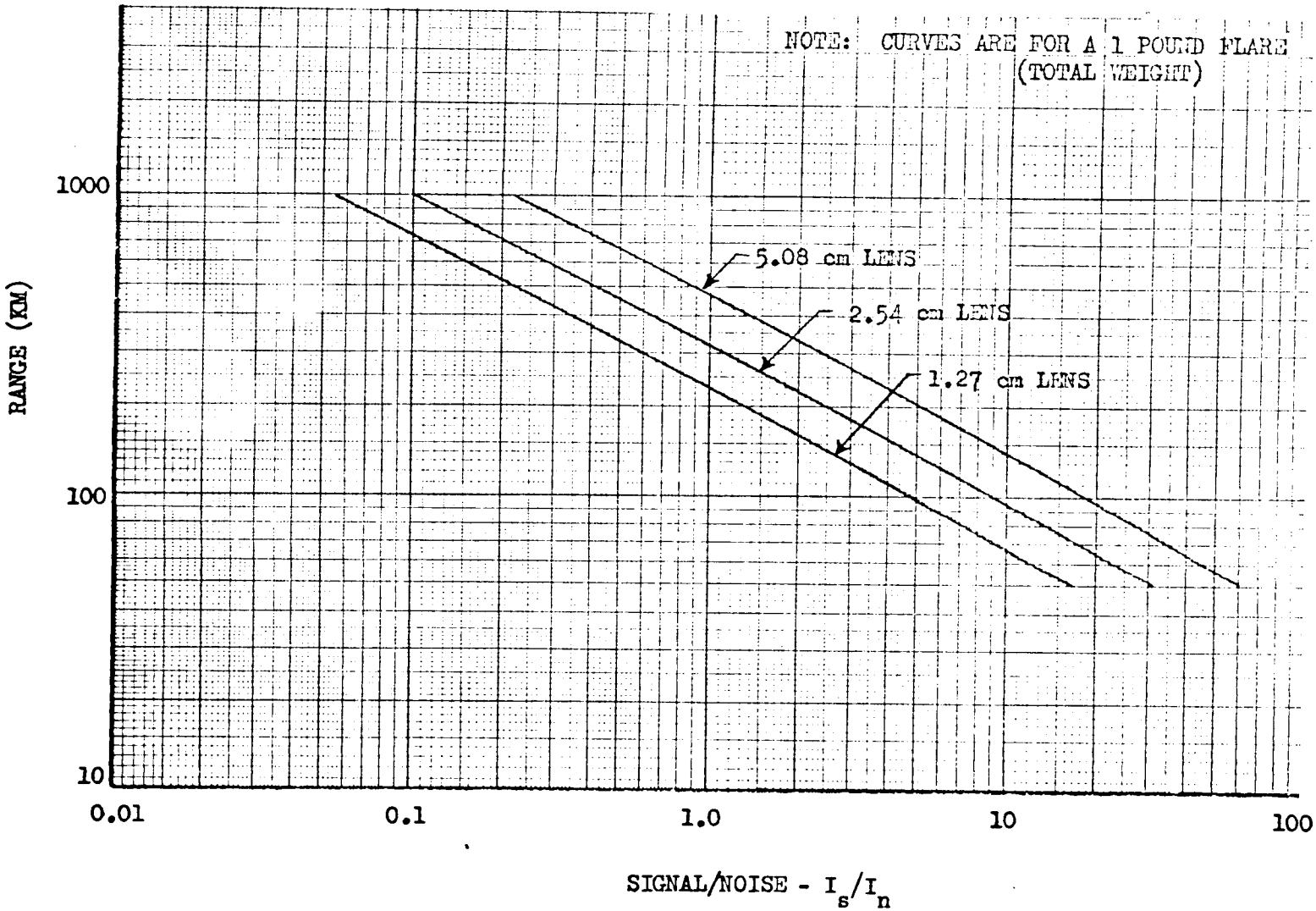


FIGURE 4.1-2 SIGNAL/NOISE FOR FIRST PEAK DETECTOR

4.1.5 Conclusion.- In conclusion, the flare event involving the impact of a 5-pound flare on the lunar surface at a velocity of 2.5 km/second can be observed from Lunar Orbiter using a commercially available photomultiplier tube and state-of-the-art optics. The spectral region from 5500-8500Å can be covered and the data thus obtained may be transmitted to Earth via the existing Lunar Orbiter video link in real time. Time resolution of 4.3 seconds and a signal-to-noise figure of about 220 can be achieved. The system weighs about 4 pounds and consumes about 2 watts of power.

4.2 First Peak Color Temperature

The determination of the apparent color temperature of the flare reaction can be accomplished by observing the flare impact with a number of photomultiplier tubes each covering a narrow region of the spectrum. The data thus obtained is used to construct an intensity versus wavelength curve for a particular time during the event. This curve is then compared with existing black-body radiation curves and a best fit determined. This establishes an apparent temperature value. Extensive trades to determine an optimum configuration have not been conducted. The Douglas data was reviewed and a similar design to that in Reference 1 has been assumed.

4.2.1 Signal current.- The expressions derived in Section 4.1.1 for the signal current apply here with the following modifications:

The average useable power (Equation 4.1-5) is reduced by the ratio of the spectral regions of interest and the reduced transmission (T_f) of the required spectral filters. The new useable power (S_f) is, therefore:

$$S_f = \frac{300\text{Å}}{3000\text{Å}} T_f S_1 \qquad (4.2\text{-}1)$$

$$= 0.1\, T_f S_1$$

Assuming $T_f = 0.2$ (Reference 1)

$$S_f = 0.02 S_1 \qquad (4.2\text{-}2)$$

Further, the expression for signal current (4.1-12) is modified due to the effective change in the photocathode efficiency brought about by reducing the spectral bandpass. For the purpose of this discussion, it will be assumed that the maximum reduction in photocathode efficiency will be by a factor of 0.75. The figure to be used with the filters is then:

$$K_{ef} = 0.25\, K_e \qquad (4.2\text{-}3)$$

Equation (4.1-12) becomes:

$$I_{sf} = S_f\, T\, K_{ef} \qquad (4.2\text{-}4)$$

$$I_{sf} = (0.02 S_1)\,(T)\,(0.25\, K_e) \qquad (4.2\text{-}5)$$

$$I_{sf} = 0.005 S_1 T K_e \tag{4.2-6}$$

Therefore:

$$I_{sf} = 0.005 I_s \tag{4.2-7}$$

4.2.2 Signal to noise.- The noise current I_n may be calculated using Equation (4.1-10)

$$I_n = (2e I_{dc} \Delta f)^{1/2} \tag{4.2-8}$$

where:

$$I_{dc} = I_{sf} + I_b + I_d \tag{4.2-9}$$

Δf = 3000 hertz (filter bandwidth)

The background current (I_b) is reduced from that of Section 4.1.2 in the same manner that I_{sf} was obtained from I_s, thus:

$$\frac{I_{sf}}{I_{nf}} = \frac{0.005 S_1 T K_e}{\sqrt{2e\left(0.005 S_1 T K_e + 0.005 \frac{\pi^2}{4} BD_e^2 T \tan^2(\theta) K_e + I_d\right) 3000}}$$

Figures 4.2-1 and 4.2-2 are plots of I_{sf}/I_{nf} versus R for several combinations of flare weight and lens aperture.

4.2.3 Experiment configuration.- The determination of apparent temperature of the flare reaction requires sufficient data to be taken to ensure that a representative curve of intensity versus wavelength may be constructed.

The requirement in existing documentation calls for three channels of spectral intensity data (Reference 2). It is felt that this number of channels may not be sufficient to adequately identify the wavelength intensity relationship. As an alternate, six channels each 300 angstroms wide evenly spaced from 4000 angstroms to 7000 angstroms are suggested.

4.2.3.1 Typical sensor characteristics

4.2.3.1.1 Independent channels.-

Size	cylindrical 10" x 2" diameter
Weight	less than 3 pounds
Power	less than 2 watts
Environment	temperature 10°F-110°F
Sensor	photomultiplier (S-1)

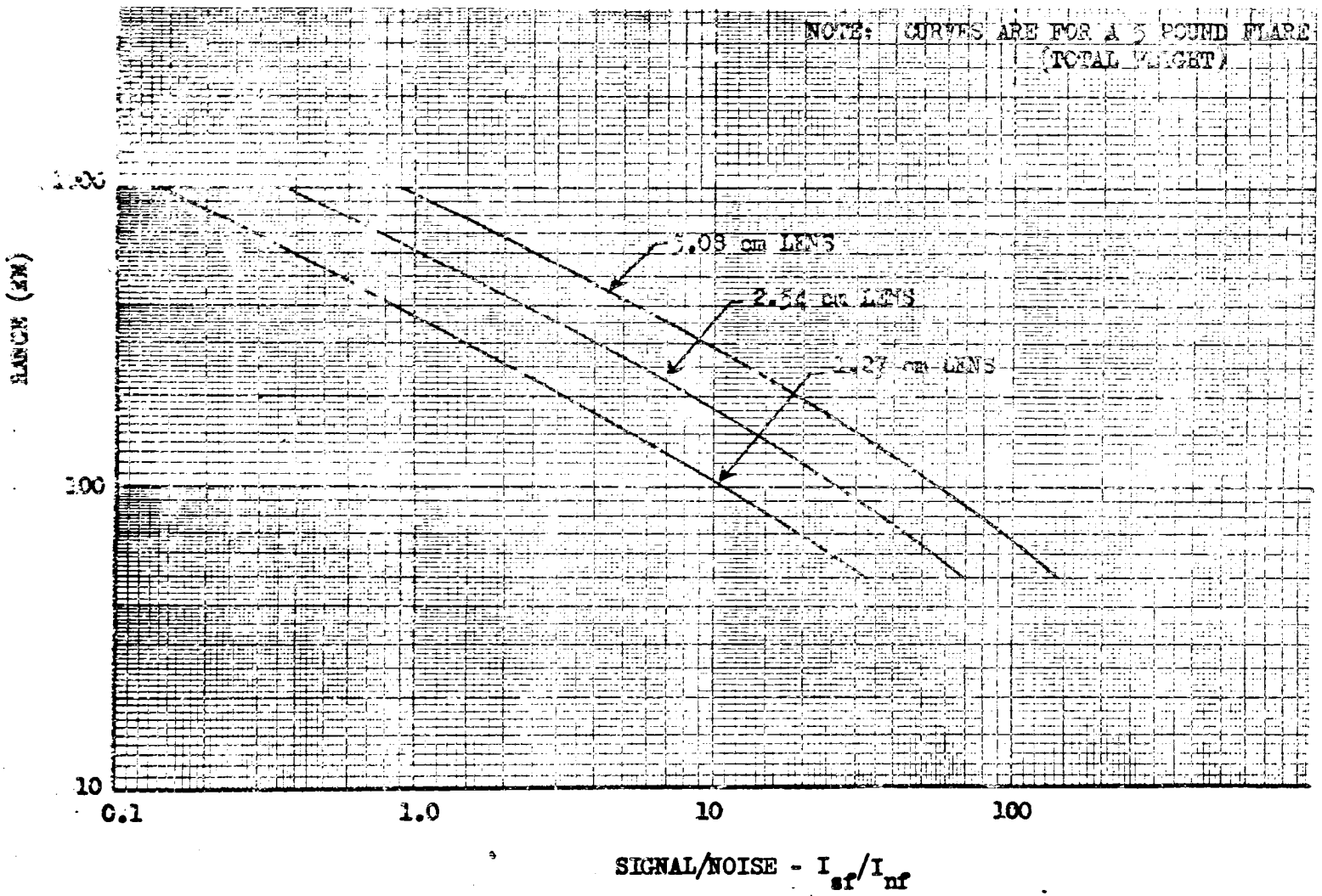


FIGURE 4.2-1 SIGNAL/NOISE FOR TEMPERATURE DETECTOR

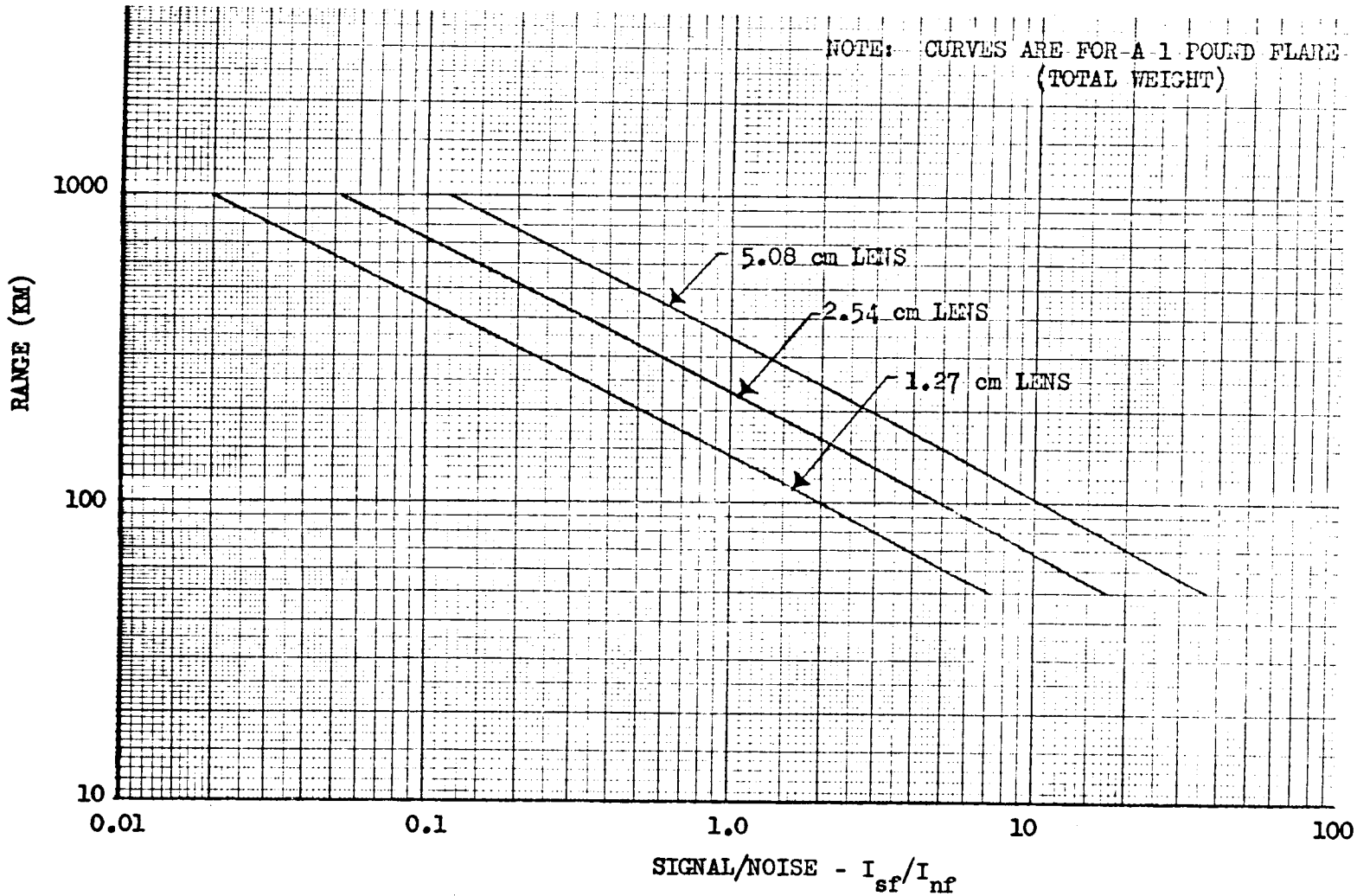


FIGURE 4.2-2 SIGNAL/NOISE FOR TEMPERATURE DETECTOR

4.2.3.1.2 Combined channels.- The above sensor description is based on the assumption that each photomultiplier channel will be electrically independent of the others. This approach obviously improves the reliability of the whole system but adds to the required power and weight. The use of a common power supply for all channels would reduce the size, weight, and power of the system. Further, a common power supply would act to eliminate one source of relative interchannel gain changes and so improve the validity of the data taken by the whole system. The weight and power requirements for such a system with some common elements would be on the order of 20% less than the aggregate weight and power of a system made up of independent channels.

4.2.4 Data recovery.- The data obtained by the filtered photomultiplier channels will be stored in some form of tape memory to allow real-time transmission of "time to first peak" data. Presently, the storage system requirements dictate a 6-channel system with at least 3 KHz bandwidth and some provision for time marking each channel if they are to be read out sequentially.

4.2.4 Conclusion.- Recording color temperature data from a flare event requires six photomultiplier tubes, covering the range from 4000 to 7000A. Using a 5-pound flare at a slant range of about 50 km, a signal-to-noise ratio of about 140 can be achieved at a cost of 15 pounds weight and 12 watts of power.

4.2.6 Problem areas.- The large amount of scatter in the experimental data, shown in Figure 3.1-3, makes the feasibility of this experiment questionable. To ensure a good correlation between color temperature and surface layer thickness, further testing is required. Also further trades should be conducted to obtain an optimum sensor design. Included in the trade study should be the possibility of integrating the system under one large set of optics and using dichroic mirrors.

4.3 Spectral Energy Distribution

The detection system for measuring the distribution of spectral energy consists of a dispersing element and recorder. The light from the flare is spread out into a spectrum and the intensity of the light in this spectrum is recorded as a function of wavelength. The recording system is assumed to be photographic film. Success of the efforts to obtain spectroscopic information from the experiment, therefore, is dependent on the ability to collect sufficient light energy from the flare to expose the film to a relative density above gross fog that will allow data reduction.

Parameters related to film exposure are: flare energy, diameter of the optics, instrument transmission, dispersion at the image plane, film sensitivity and spectrum width. These relationships are covered in the following discussion.

4.3.1 Energy required at film.- The energy required at the film plane can be determined by the following equation:

$$E_R = e\,A_f \qquad (4.3\text{-}1)$$

where:

E_R = energy required at film plane (watt-second)

e = energy density required at the film (watt-second/mm^2)

A_f = total area of film exposed (mm^2)

If the spectrum has a width w(mm) and yields a linear reciprocal dispersion K(A/mm), the energy in a 1A band is spread over an area w/K (Reference 3).

The total area (A_f), therefore, is given as:

$$A_f = \frac{w}{K} (\lambda_2 - \lambda_1) \qquad (4.3\text{-}2)$$

where:

$(\lambda_2 - \lambda_1)$ = wavelength region observed ($\mathring{A}$)

Substituting Equation (4.3-2) into (4.3-1) results in an energy expression for the film as a function of plate dispersion:

$$E_R = \frac{ew (\lambda_2 - \lambda_1)}{K} \qquad (4.3\text{-}3)$$

4.3.2 Energy available at film.- The energy per unit area at the collector for the flare event, assuming the entire radiation occurs uniformly over a hemisphere, is given in the equation:

$$H_o = \frac{E_o}{2 \pi R^2} \qquad (4.3\text{-}4)$$

where:

E_o = flare energy over spectral region of interest (watt-second)
H_o = energy per unit area at the collector $\left(\frac{\text{watt-second}}{\text{mm}^2}\right)$
R = range (mm)

The energy available at the film plane therefore depends on the area of the collector A_c and the transmission T of the optics:

$$E_A = H_o \; T \; A_c = \frac{H_o \; T \; \pi \; D_c^{\,2}}{4} \qquad (4.3\text{-}5)$$

where:

D_c = the diameter of the collecting lens (mm)

From Equations (4.3-4) and (4.3-5), the total energy available at the film plane is:

$$E_A = \frac{E_o \; T \; D_c^{\,2}}{8 \; R^2} \qquad (4.3\text{-}6)$$

The chlorine trifluoride flare has a theoretical maximum chemical energy yield of 4.24×10^6 watt-seconds per pound. The assumption is made that the integrated energy in the spectral region of interest is constant regardless of flare temperature and is equal to 1% of the total energy (Reference 1). Available energy at the image plane as a function of both weight (W_R) of reactive chemicals and range can, therefore, be determined from the following relationship:

$$E_A = \frac{(4.24 \times 10^4)\ T\ D_c^{\ 2}\ W_R}{8\ R^2} \qquad (4.3\text{-}7)$$

4.3.3 Flare weight.- The energy available at the film must be equal to the energy required, therefore from Equations (4.3-3) and 4.3-7)

$$\frac{ew\ (\lambda_2 - \lambda_1)}{K} = \frac{(4.24 \times 10^4)\ T\ D_c^{\ 2}\ W_R}{8\ R^2} \qquad (4.3\text{-}8)$$

This results in an expression for flare reactant weight as a function of range and dispersion:

$$W_R = \frac{8\ we\ (\lambda_2 - \lambda_1)\ R^2}{(4.24 \times 10^4)\ T\ D_c^{\ 2}\ K} \qquad (4.3\text{-}9)$$

From the above equation, it can be seen that flare size is influenced greatly by the range. Other parameters affecting flare weight are spectral widening and dispersion. Assuming the existing orbiter photo subsystem is used, the size of the camera lens, the transmission, and the film energy density are fixed.

Widening of spectrum.- If the image of the flare is held at one point on the film during exposure, the resultant spectrum is narrower than is desirable for photometric studies or the recognition of faint lines. The length of the line image is normally determined by the resolution of the system. In order to widen the spectrum with the necessary uniformity, the usual procedure is to allow the image to drift along the film for a distance corresponding to the required width. Widening of 0.2 to 0.3 mm has been considered desirable with an acceptable minimum width of 0.1 mm (References 1 and 3).

The width of the spectrum normal to the direction of dispersion is governed by the vehicle scene rate (V/H). For calculation purposes, a maximum velocity 1950 m/second has been assumed at an altitude of 46 kilometers. The camera focal length is 610 mm (24 inches) with a film resolution of 76 line pairs per millimeter. The duration of the flare is 10 milliseconds. This results in a "smear" of the spectrum normal to the dispersion of about 0.25 mm which should be acceptable. The spectrum width can be reduced or increased if image motion compensation is used.

The limit cycle or attitude rate of the vehicle affects spectrum widening by an insignificant amount.

Film sensitivity.- The film used in the existing orbiter photographic subsystem is E. K. Type SO-243. Figure 4.3-1 shows the spectral sensitivity curves indicating the exposure required to produce a density of 0.2 and 1.0 above gross fog. Sensitivity is defined in this case as:

$$s = \frac{1}{e} \qquad (4.3\text{-}10)$$

where:

$$e = \text{exposure in } \frac{\text{watt-seconds}}{\text{mm}^2}$$

While a density above gross fog of 0.6 is generally accepted as the desired level for accurate quantitative work, useful data can be obtained from spectrograms having much lower densities. The lowest practical value may be below 0.1, but calculations are based on densities of 0.2 to ensure an adequate signal-to-noise ratio.

Total transmission.- The overall light signal transmission due to the flare is the product of the transmission of each element in the path between the flare and the detector. The absorption of most types of glass is large in the ultraviolet, and may result in a very substantial loss of light at the shorter wavelengths. An average transmission of the 24-inch lens has been stated as 66% over the spectral region from 4000Å to 7000Å (Reference 4). The lens transmission falls off rapidly near 4500Å, however, to less than 10% at 3500Å as shown in Figure 4.3-2. A conservative value of 35% over the spectral region of interest was chosen for computation purposes. The curve in Figure 4.3-2 is axial transmission but is also valid over the entire lens field.

Other elements affecting flare energy include a transmission grating, quartz window and plane mirror. If a good quality blazed transmission grating is used, about 55% of the incident light can be diffracted into a given order. The quartz window will absorb about 5% of the near ultraviolet and visible light. An aluminum mirror will reflect about 85% at 3500A and 88% in the near infrared (Reference 3). Total transmission of the system can be determined as follows:

$$T = (T_g)\ (T_w)\ (T_c)\ (T_m) \qquad (4.3\text{-}11)$$

where:

T_g = grating transmission

T_w = window transmission

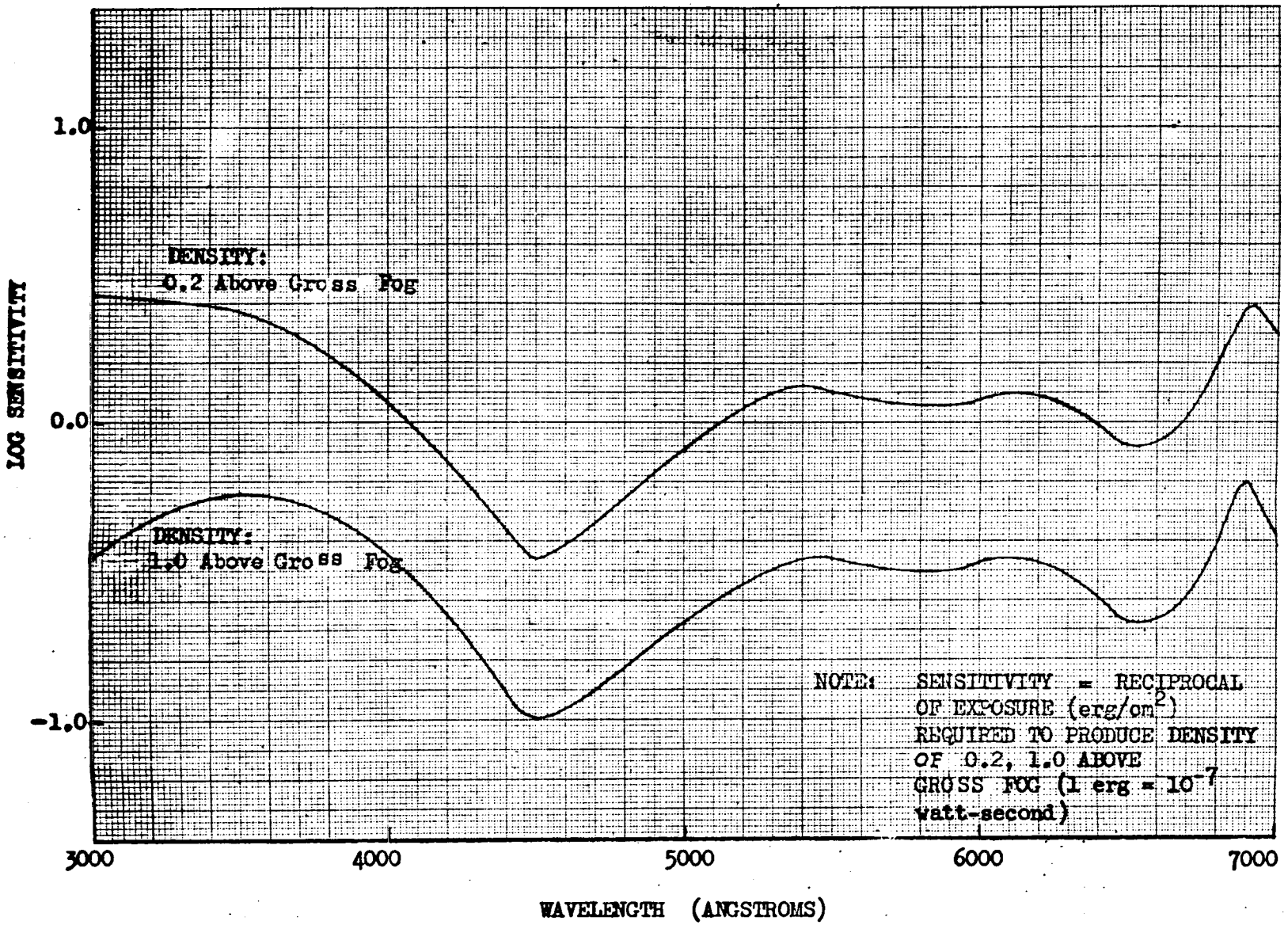


FIGURE 4.3-1 SPECTRAL SENSITIVITY SO-243 FILM

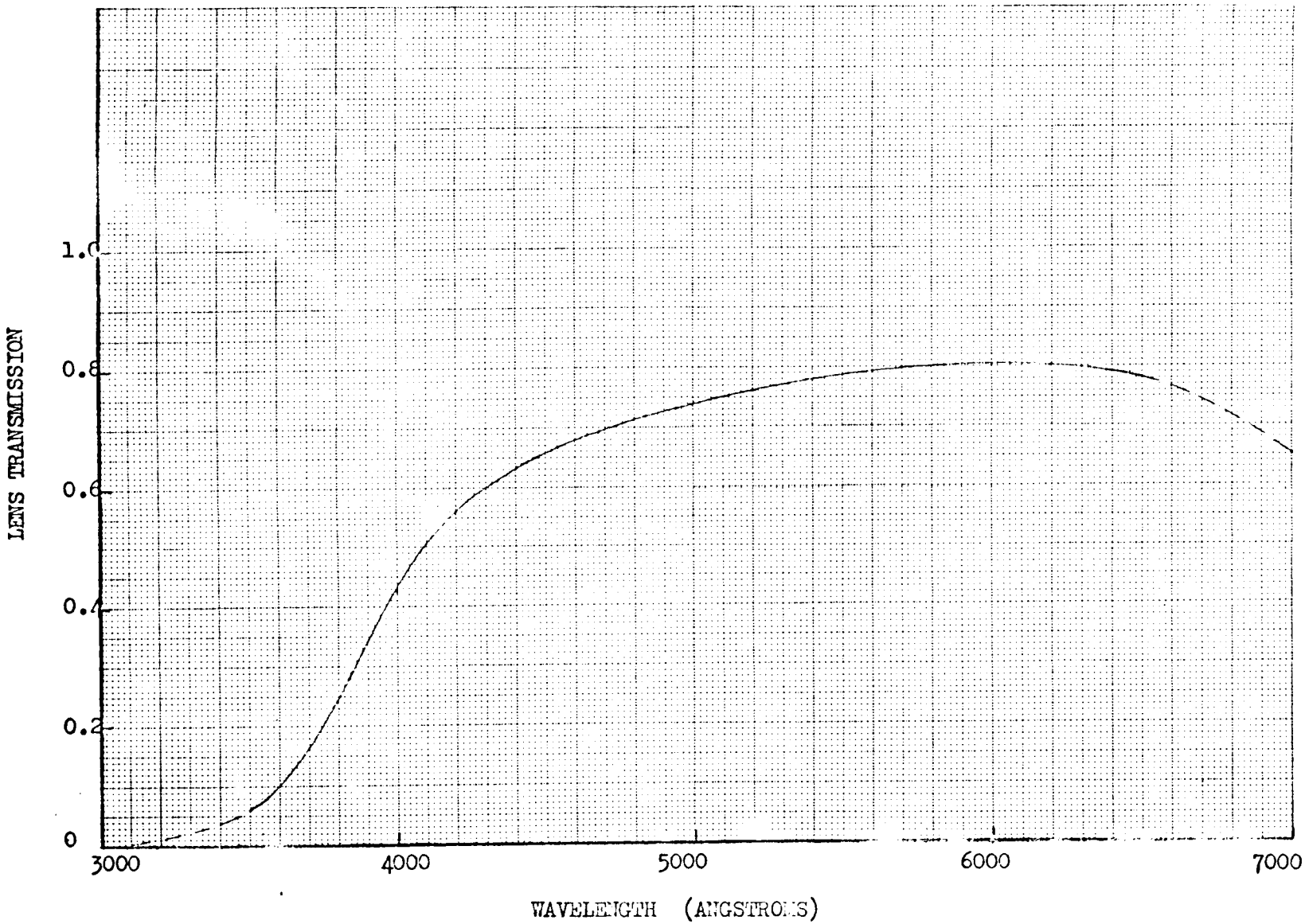


FIGURE 4.3-2 ON-AXIS TRANSMISSION, ORBITER 24-INCH, f/5.6 LENS

T_c = camera lens transmission

T_m = mirror reflection factor

Wavelength interval.- A list of atomic lines is presented in the Douglas Report (Reference 1). Spectral lines from 3581.2Å are included. Of particular interest to selenologists is the ability to predict the relative abundances of silicon and aluminum from a flare event. Unfortunately, the most easily excited atomic emission lines of silicon are below 3000Å and hence cannot be detected with the existing camera lens, which is opaque to energy below 3000A. The only readily excitable Si line above 3000Å is at 3906Å, and it requires 5.1 ev to excite it. The flare reaction can excite nearly all available lines with excitation potential less than 4 ev, and many with potentials between 4.0 and 4.6 ev. Magnesium lines are the only ones available with potentials greater than 5 ev. Thus higher reaction temperatures than have been attained experimentally, to date are required to obtain relative abundance of silicon. Molecular bands, such as silicon fluoride, could possibly yield this information.

Aluminum lines, on the other hand, can be found at 3944Å and 3961.5Å. Aluminum oxide and aluminum salt bands of interest all lie in the region below 6000Å. The wavelength interval from 3500Å to 6500Å has been used for computation purposes. It is believed that data from the flare experiment will fall in this range (Reference 5).

Plate dispersion.- An expression for the angular dispersion of a grating can be readily obtained from the classical grating equation (References 6 and 7):

$$\pm n\lambda = d(\sin\alpha + \sin\beta) \qquad (4.3\text{-}12)$$

where:

$\pm n$ = spectral order

λ = wavelength of light

d = grating constant (distance between grating rulings)

α = angle of incidence

β = angle of diffraction

Assuming α to be constant then differentiating the grating equation and rearranging, the angular dispersion can be found:

$$\frac{d\beta}{d\lambda} = \frac{n}{d\cos\beta} \qquad (4.3\text{-}13)$$

The linear dispersion $\left(\frac{d\ell}{d\lambda}\right)$ of the camera can be found using the camera focal length f_c as follows:

$$\frac{d\ell}{d\lambda} = \frac{d\beta \quad f_c}{d\lambda} = \frac{n \quad f_c}{d \cos\beta} \tag{4.3-14}$$

Again using the grating Equation (4.3-12) this becomes:

$$\frac{d\ell}{d\lambda} = \frac{f_c \quad (\sin\alpha + \sin\beta)}{\lambda \cos\beta} \tag{4.3-15}$$

The reciprocal linerar dispersion or plate dispersion as it is sometimes called is:

$$K = \frac{d\lambda}{d\ell} = \frac{\lambda \cos\beta}{f_c \quad (\sin\alpha + \sin\beta)}$$

where K is measured in Å/mm. λ is defined as the wavelength in angstroms of the center of the spectral region to be studied.

A high dispersion instrument is needed to obtain reliable data from the flare event. Dispersion on the order of 20Å/mm is desirable, but some data can be obtained with a medium dispersion instrument ($\sim$ 120Å/mm). The camera has a fixed focal length of 610 mm, so assuming a grating can be ruled with 600 lines per mm and blazed for 5000Å in the first order (n = 1), the best dispersion that can be obtained with this camera is about 25Å/mm. To observe a 3000Å band, this requires about 120 mm of the film format.

Spectral resolution.- The spectral or wavelength resolution is the minimum wavelength difference necessary for two features in the spectrum to appear separate on the record. The theoretical limit to spectral resolution is determined by the grating used. The theoretical resolving power is given as (Reference 6):

$$\frac{\lambda}{\Delta\lambda} = n\,N \tag{4.3-17}$$

where:

λ = wavelength (Å)

$\Delta\lambda$ = spectral resolution limit (Å)

n = spectral order

N = total number of lines on the grating.

For a 5-inch grating with 600 /mm blazed for 5000Å in the first order, the resolution limit due to the grating is 0.07Å.

The wavelength resolution is set by the photographic subsystem, however, rather than the grating. The spectral resolution when fixed by the camera system is given as follows:

$$\Delta\lambda = F_r K \qquad (4.3\text{-}18)$$

where:

F_r = linear resolution element (mm)

K = linear reciprocal dispersion (Å/mm)

If the camera resolution is 75 ℓp/mm and dispersion of 25Å/mm, the resulting spectral resolution is 0.3Å. Resolution capability of at least 0.5Å is desirable, but some data can be obtained from an instrument with 3Å resolution (Reference 1). High resolution is necessary to separate two adjacent lines well and to detect weak lines above the noise.

The attitude rate of the vehicle could contribute to a decrease in spectral resolution by smearing the spectra in the direction of dispersion. This contribution is small, however, amounting to less than 10% of a resolution element.

4.3.4 Summary.- The effects of linear reciprocal dispersion of the instrument on required energy at the film is shown in Figure 4.3-3 (see Equation 4.3-3) where:

e = 8×10^{-10} $\frac{\text{watt-sec.}}{\text{mm}^2}$ (average exposure required to produce density = 0.2 above gross fog)

w = 2.5×10^{-1} mm

$(\lambda_2 - \lambda_1)$ = (6500Å-3500Å)

Figure 4.3-4 shows the relation between total flare weight, range and energy reaching the film (see Equation 4.3-7). Figure 4.3-5 has been included to allow conversion from reactant weight to total flare weight including the envelope enclosing the reactants. The following parameters have been used:

T_g = 0.55

T_w = 0.95

T_c = 0.35 (average transmission over region of interest)

T_m = 0.85

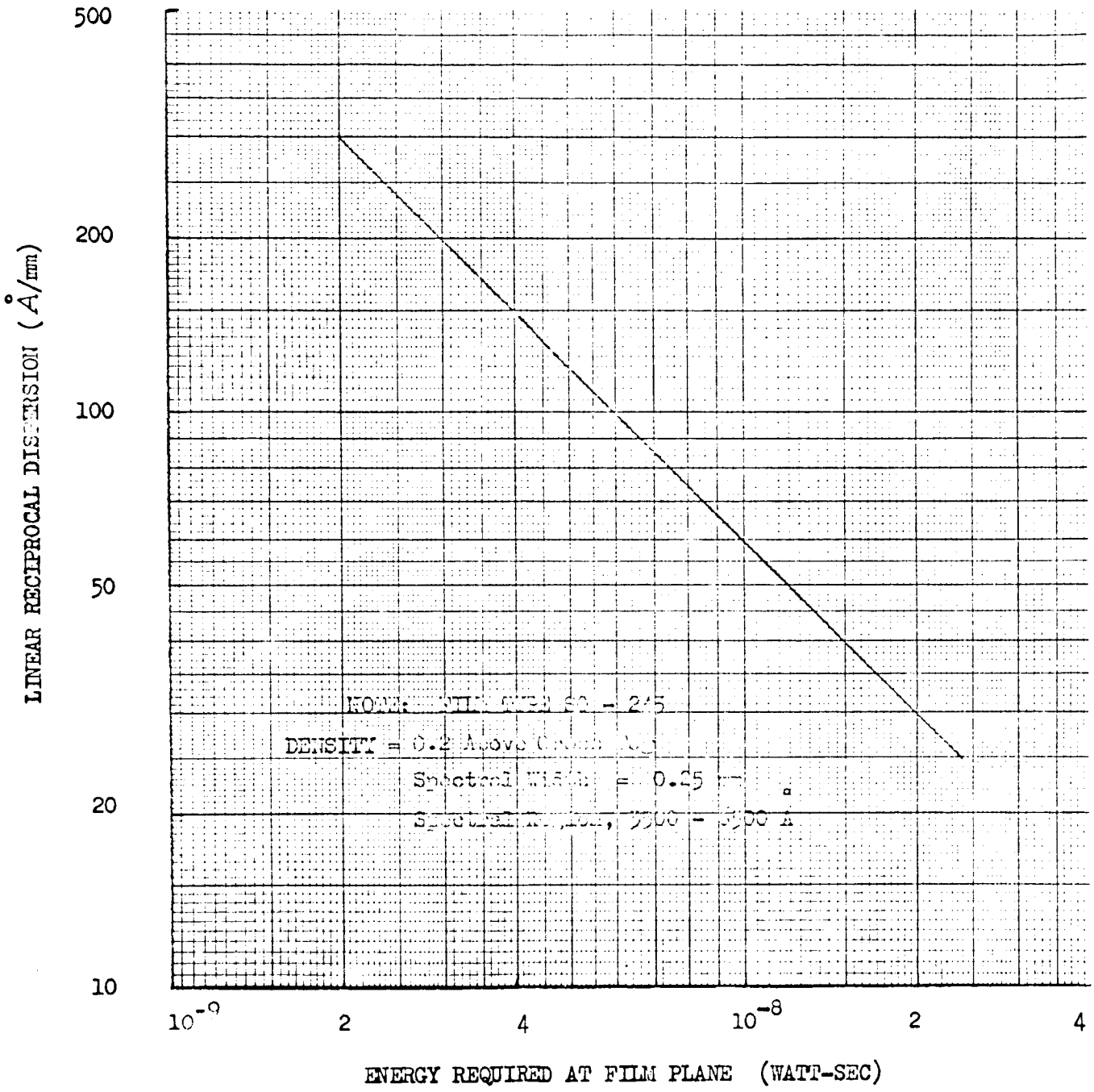


FIGURE 4.3-3 REQUIRED ENERGY AT FILM

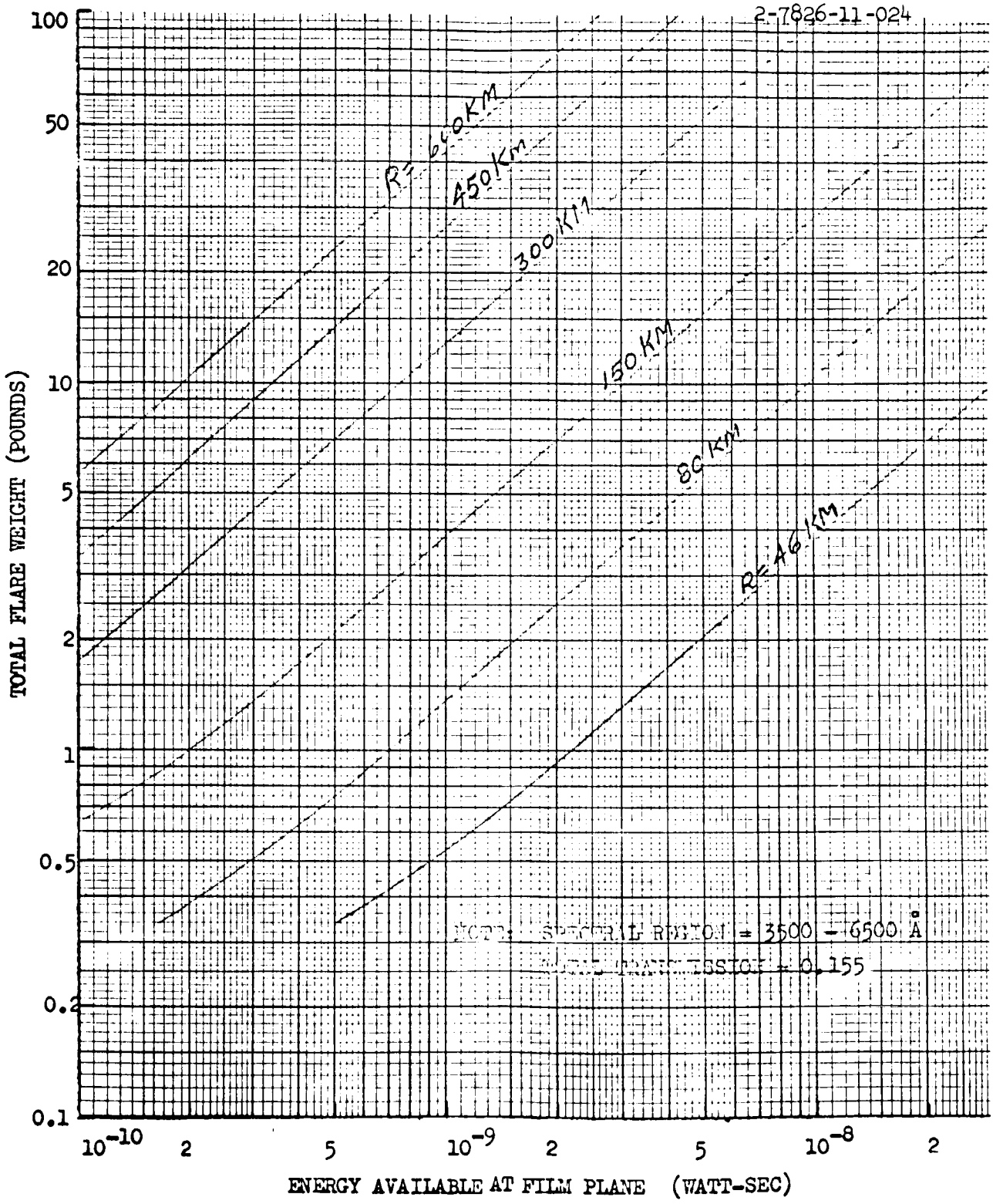


FIGURE 4.3-4 AVAILABLE ENERGY AT FILM

The result is a total transmission factor of 0.155 for the entire system. The diameter of the camera lens D_c is 109 mm.

The result of the study has been summarized in Figure 4.3-6 where the relation between total flare weight, range, and dispersion or resolution is shown. These curves show that to detect a flare using the camera at maximum dispersion of 25Å/mm and a range of 46 km, a 7.7-pound flare is required. If range is increased to 80 km, flare weight is increased to about 20 pounds.

Lower dispersion is achieved using a grating with a smaller number of rulings. Some useful data can be obtained with a 225Å/mm dispersion. A 1.25-pound flare will radiate sufficient energy if observed at a 46 km range. If the range is increased to 80 km, the flare weight must be 3 pounds. It is assumed that up to 4 flares are required to yield sufficient information about surface composition.

4.3.5 Instrumentation.- The instrumentation to obtain spectral data from the flare experiment consists of the existing Orbiter photographic subsystem with the addition of a transmission diffraction grating. Two possible positions exist for the grating: in front of the camera lens and behind it. Two problems arise when the grating is placed between the lens and film. One, possibly the most stringent, is the lack of room. It is certain that the photo subsystem must be redesigned to allow sufficient area for the grating. The fact that the grating is placed in a diverging beam brings about the second problem. The phenomena is no longer Fraunhofer diffraction because the light incident on the dispersing element is not parallel. Focusing and calibration could, therefore, be a problem (References 8 and 9).

Because of the problems inherent in placing the dispersing element behind the lens, the position before the lens has been chosen. Light incident on the grating is now approximately parallel due to the relatively great distance from the source. Figure 4.3-7 shows an optical schematic of this arrangement.

The position of the grating with respect to the camera lens depends on the dispersion and, therefore, on the particular grating. If 25Å/mm dispersion is to be obtained at normal incidence, the grating must be tilted 17.5° from the camera optical axis. Ideally, the grating is placed between the window and camera lens inside the photographic subsystem.

To ensure that all available energy is collected and recorded, the angle of incidence must not vary more than ±4.6 degrees from normal incidence (= 0°). The flare impact point must, therefore, lie within ±4.6 degrees of the orbital plane and ±2.5 degrees along the flight path (±3.7 km cross-range and ±2.1 km downrange at 46 km slant range). These values are for the case of 25Å/mm dispersion. Required impact areas can be found in Table 4.3-1 and Figure 4.3-8 for various range and dispersion values. Figure 4.3-9 has been included to compare the field of view for the 24-inch camera with and without the diffraction grating.

TABLE 4.3-1

IMPACT AREA REQUIRED VS. RANGE AND RECIPROCAL LINEAR DISPERSION

SLANT RANGE (KM)	IMPACT AREA (KM)				
	25 Å/mm	50 Å/mm	75 Å/mm	150 Å/mm	225 Å/mm
20	± 0.90 DR*	± 0.9	± 0.9	± 0.9	± 0.9 DR
	± 1.6 CR	± 2.44	± 2.88	± 3.18	± 3.34 CR
46	± 2.07 DR	± 2.07	± 2.07	± 2.07	± 2.07 DR
	± 3.7 CR	± 5.61	± 6.62	± 7.32	± 7.69 CR
80	± 3.6 DR	± 3.6	± 3.6	± 3.6	± 3.6 DR
	± 6.5 CR	± 9.75	± 11.52	± 12.72	± 13.38 CR
150	± 6.8 DR	± 6.8	± 6.8	± 6.8	± 6.8 DR
	± 12.1 CR	± 18.29	± 21.6	± 23.82	± 25.07 CR
300	± 13.5 DR	± 13.5	± 13.5	± 13.5	± 13.5 DR
	± 24.2 CR	± 36.58	± 43.2	± 47.7	± 50.2 CR
400	± 20.3 DR	± 20.3	± 20.3	± 20.3	± 20.3 DR
	± 36.3 CR	± 48.78	± 57.6	± 63.6	± 66.8 CR
600	± 27.0 DR	± 27.0	± 27.0	± 27.0	± 27.0 DR
	± 48.4 CR	± 73.16	± 86.4	± 95.4	± 100.3 CR

*DR - Downrange
CR - Crossrange

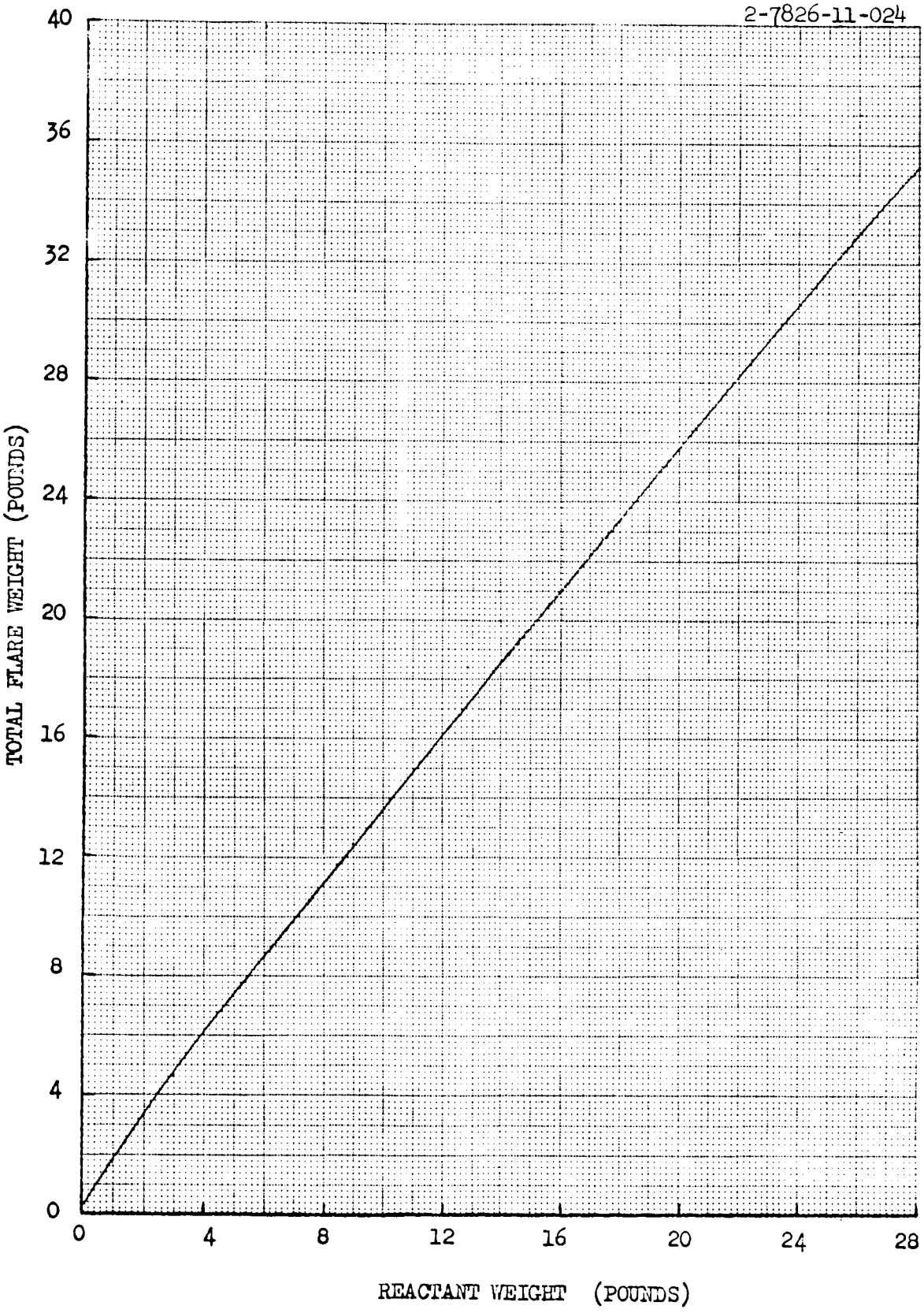


FIGURE 4.3-5 REACTANT WEIGHT - FLARE WEIGHT CONVERSION

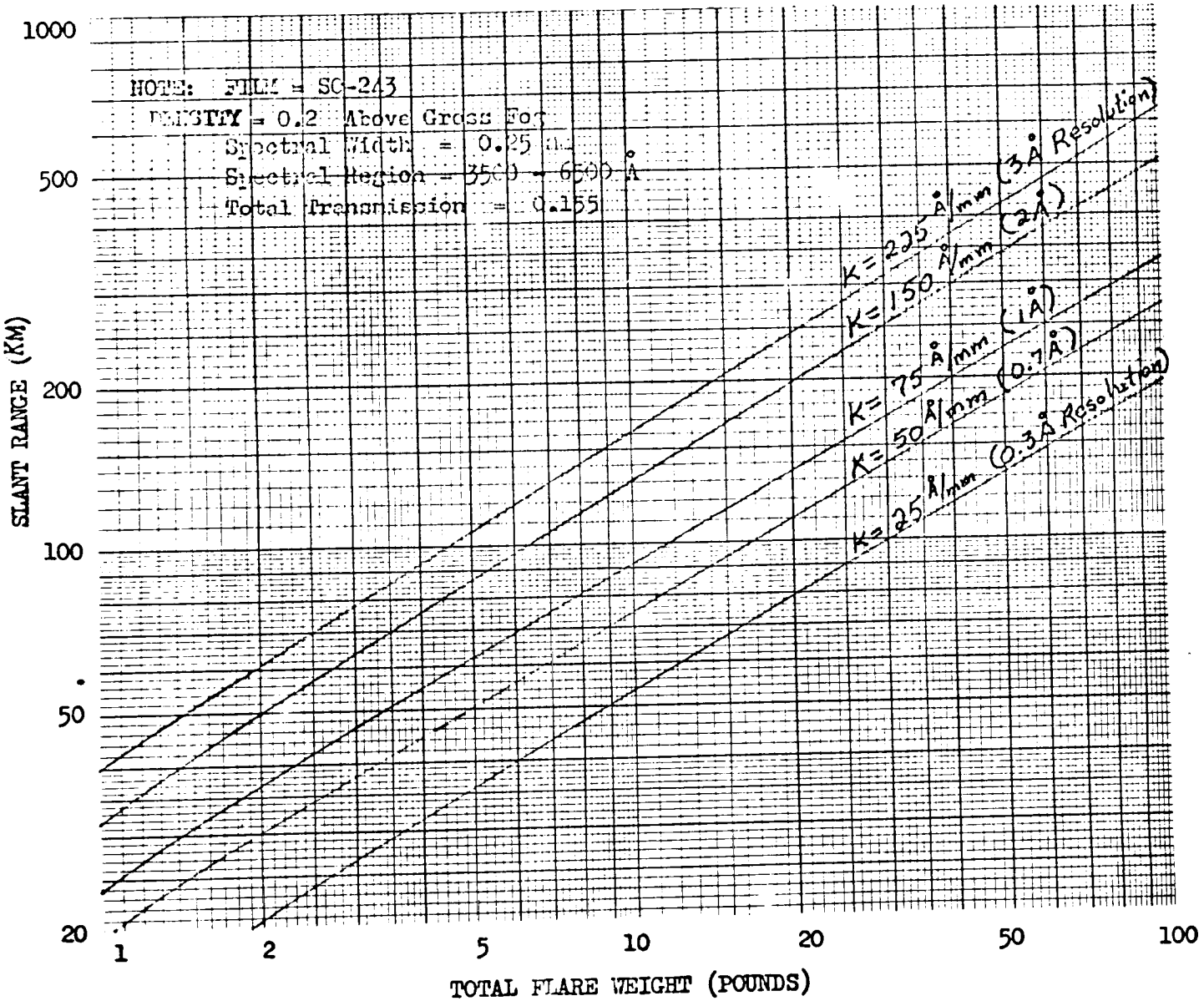
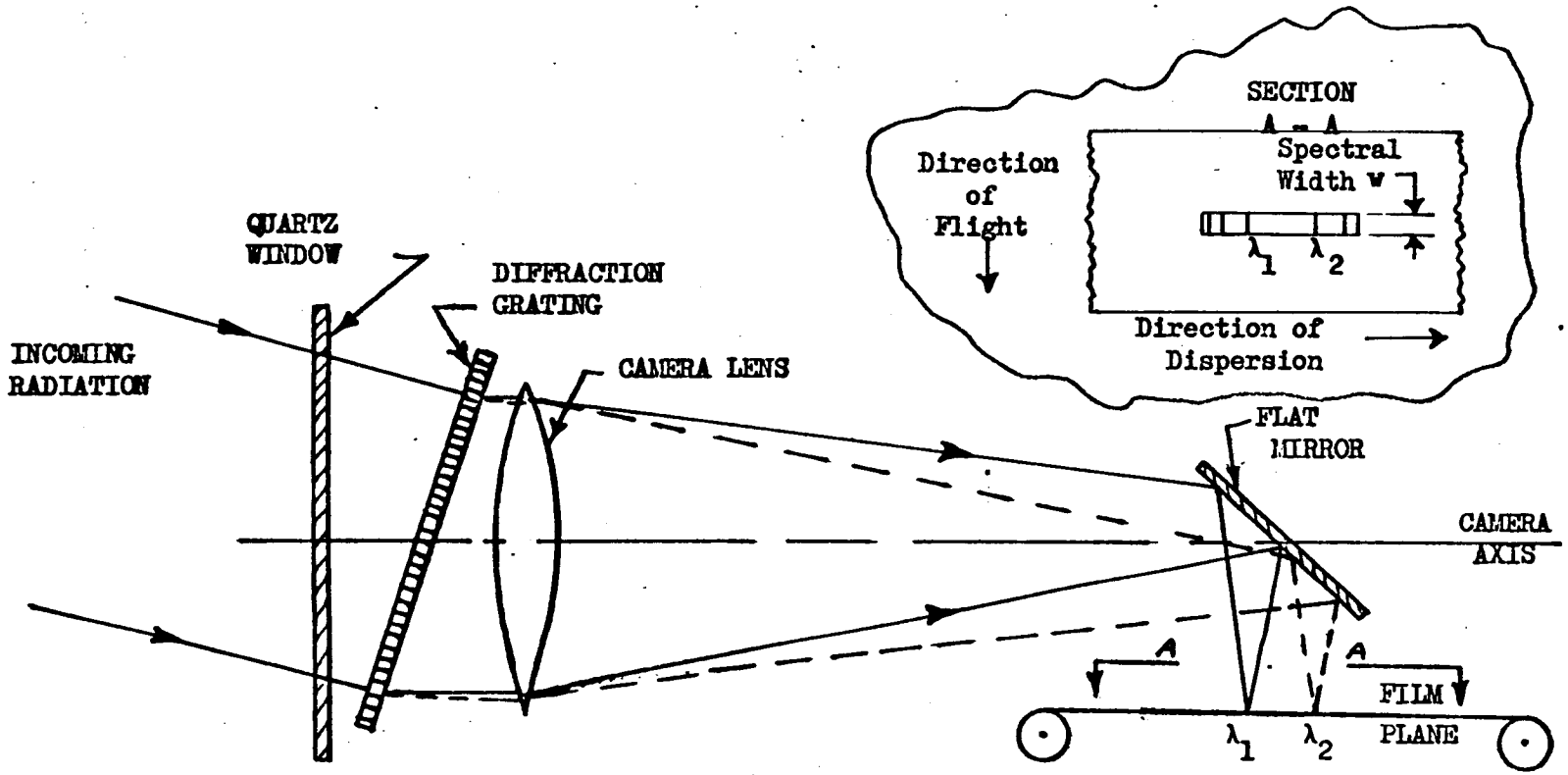
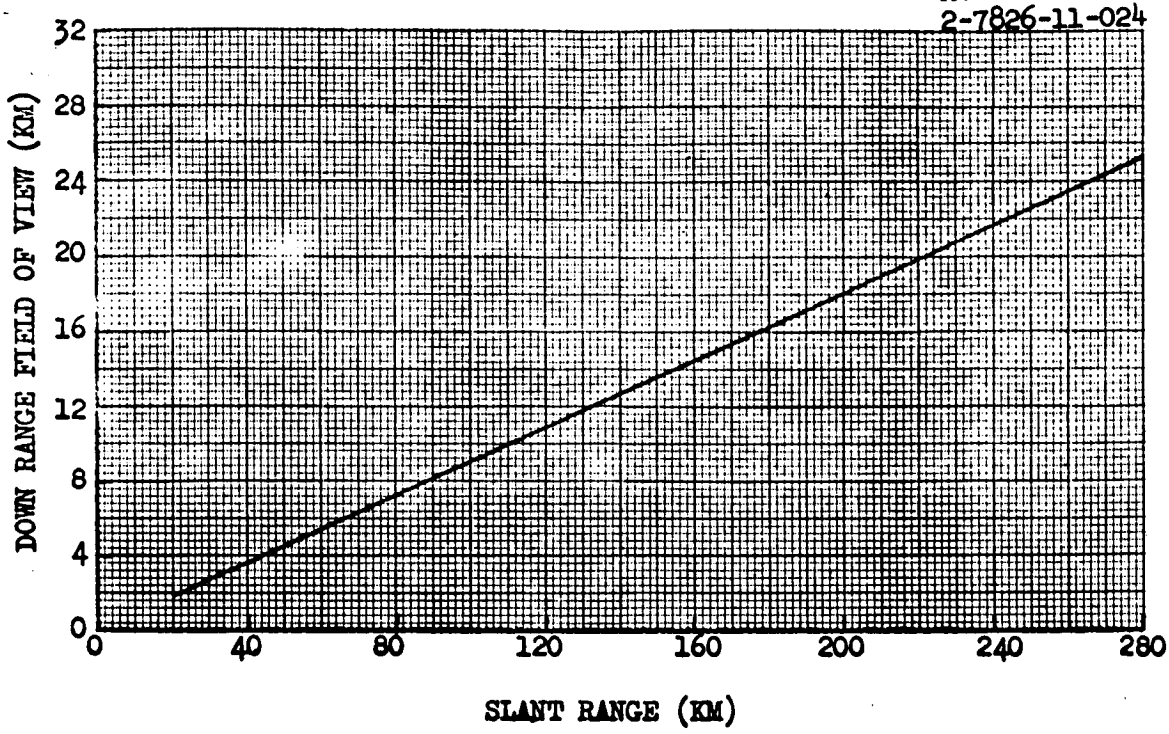


FIGURE 4.3-6 FLARE WEIGHT



NOTE: VEHICLE MOTION IS PERPENDICULAR TO DIRECTION OF DISPERSION

FIGURE 4.3-7 SPECTROGRAPH OPTICAL SCHEMATIC



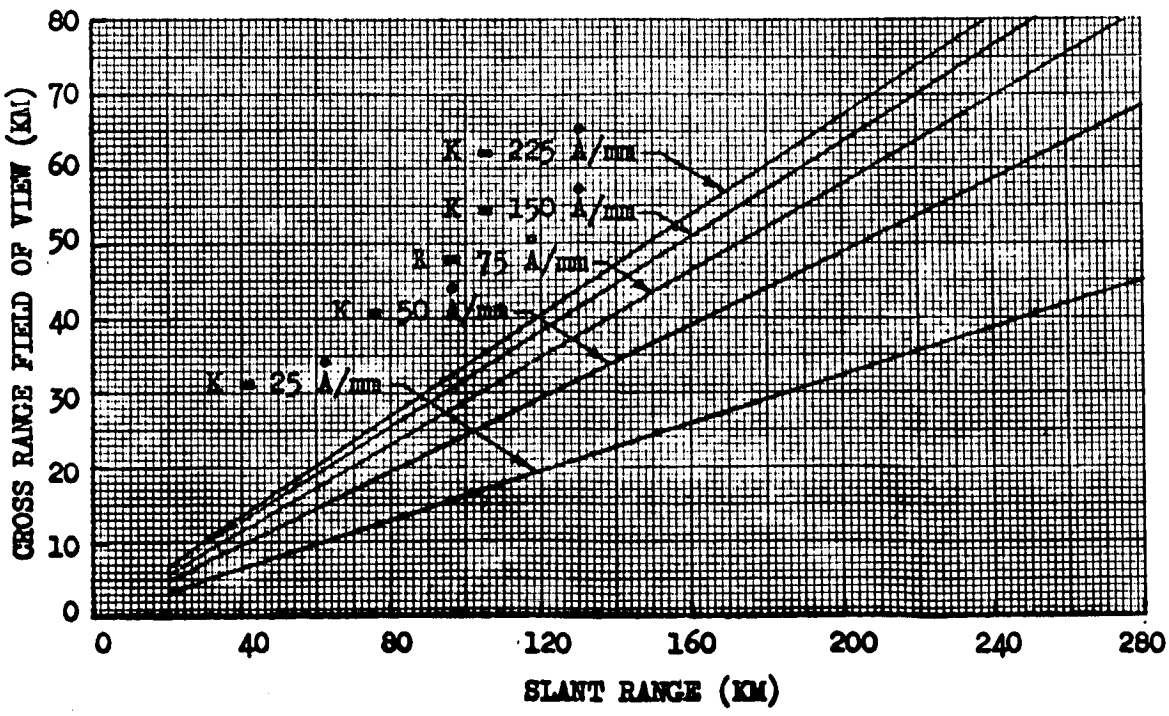


FIGURE 4.3-8 SPECTRAL ENERGY DETECTOR FIELD OF VIEW

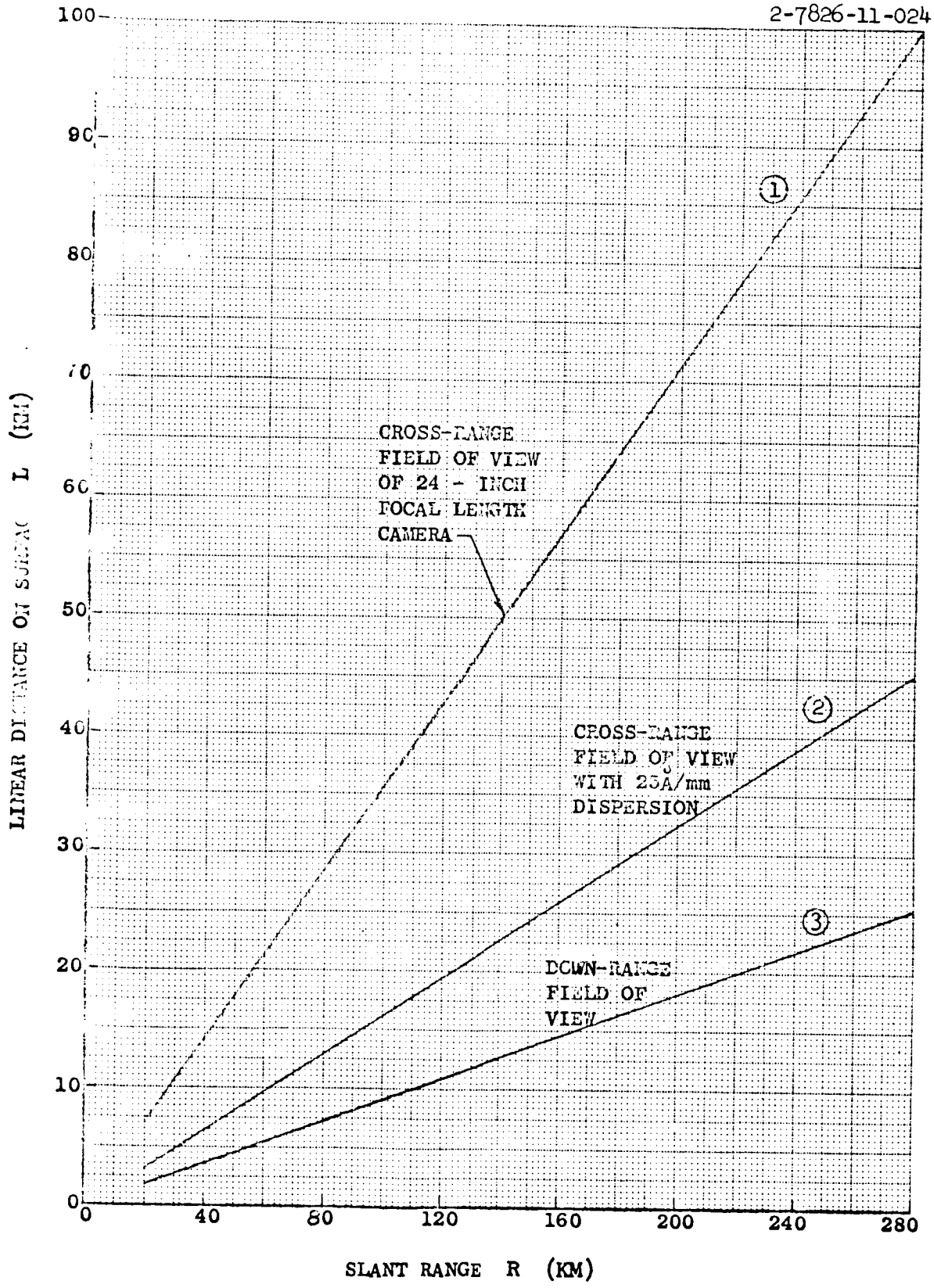


FIGURE 4.3-9 SPECTRAL ENERGY DETECTOR FIELD OF VIEW

The size of the grating is determined by the angle of diffraction, size of the camera lens and allowable deviation of angle of incidence. A ruled area of 5-1/4 inches on a side has been calculated. The grating blank is assumed to be quartz and will weigh 2 pounds. The associated mounting hardware required is approximately 4 pounds.

4.3.6 Conclusions.- Obtaining spectral data from a flare event is difficult with the existing orbiter photo subsystem. Factors affecting the success of this experiment are low sensitivity of the emulsion and short wavelength limit of the optics. To obtain spectral information at relatively low dispersion (225Å/mm) and wavelength resolution of 3Å, up to 4 flares, totaling 5 pounds will be required assuming a slant range of 46 km. In addition to this, the grating and mount weighs 6 pounds. This is the minimum cost to the vehicle not including additional equipment necessary to store and fire the flares.

4.3.7 Problem areas.- Several problems have been generated during this study. Among the problems affecting the design is the grating mounting position. If the grating cannot be located inside the photo subsystem, it must be mounted outside in front of the window. This means it must be provided with radiation and micrometeoroid protection. Since the grating is now not an integral part of the camera, it may be more difficult to maintain a rigid mount with respect to the camera lens. Therefore, grating mounting tolerances with respect to the camera and spacecraft axes must be determined.

The existing camera is equipped with a constant-velocity variable-slit focal-plane shutter. For this shutter, slit widths of 2, 1 and 1/2 inches can be selected by command. These slit widths result in exposure times of 1/25, 1/50, and 1/100 second, respectively. The shutter is capable of completing each cycle (forward motion and return) in 0.1 second and cycles at a maximum rate of one operation per 1.6 seconds (Reference 4). For the flare experiment, the error in time of impact due to injection dictates the required shutter time. If for instance this error is ± 5 seconds, the 2-inch slit width is required at a shutter speed of about 10 seconds. This relatively long exposure time results in a possible redesign of the shutter mechanism. Exposure times of this order also means that reciprocity affects in film cannot be ignored.

An additional problem exists if the flares are deployed in groups to achieve a "shotgun" effect. To be detected by the film, each flare in the group must be sized from Figure 4.3-6 as for a single flare. The flares must be sufficient distance apart on the surface not to cause overlapping spectral data on the film. Spacecraft velocity, slant range and required spectral widening dictate the required flare dispersion pattern on the surface. The dispersion pattern requirements have not yet been established.

4.3.8 Recommendations.- The problem of detecting faint objects with a photographic film involves choosing that exposure point on the film which yields the greatest output signal-to-noise ratio. In selecting the exposure point to produce a density of 0.2 above gross fog, it was assumed that sufficient signal-to-noise would be obtained. A detailed theoretical signal-to-noise calculation should be performed, including noise due to background light from the Earth. This noise has been ignored in this study because it changes the fog level by

0.02 above the original base plus chemical fog. Therefore, the original exposure point of 0.2 above gross fog is still conservative. A detailed calculation, however, would determine how conservative.

Further study is also necessary on an experimental level. SO-243 should be tested for its low-light level response over the entire spectral region from at least 3000A to about 7000A. The grating-lens-film combination should also be tested with short duration, low-energy sources.

5.0 REFERENCES

1. Douglas Aircraft MSSD, Final Report for Continuing Study for Early Lunar Flare, Vol. II, No. SM-49259, December 1965

2. Boeing Letter to NASA-Langley, 2-1590-00-049, Appendix B, May 19, 1966

3. Bowen, I. S., 1962, Astronomical Techniques, Vol. II, p. 34

4. Eastman Kodak Company, Photographic Subsystem Reference Handbook for the Lunar Orbiter Program, No. L-018375-RU, March 1966

5. Douglas Aircraft MSSD, Final Report: Study of the Feasibility of an Early Lunar Flare, No. SM-47954, July 1964

6. Sawyer, R. A., Experimental Spectroscopy, 1963

7. Diffraction Gratings, Bausch and Lomb, Rochester 2, New York, 1957

8. Born, M. and Wolf, E., Principles of Optics, 1959

9. Jenkins, F. A., and White, H. E., Fundamentals of Optics, 1950

LUNAR ORBITER COMMUNICATION MODIFICATIONS FOR THE LUNAR FLARE EXPERIMENT

INTRODUCTION

The referenced correspondence establishes the requirement to record and playback data from the flare experiment (paragraph B-4). The transmitted data is divided into realtime and playback modes. The "time to first peak" is realtime data and the "temperature of first peak" is recorded data.

The discussion is based on the following requirements :

1) The time to first peak data (realtime) requires one channel of the communication subsystem to pass a signal with a minimum rise time of 4.5 microseconds. In bandwidth (BW) terms;

$$BW = 1/tr$$
$$BW = 222 \text{ KHz}$$

The existing Lunar Orbiter communication subsystem has a bandwidth of 230 KHz.

2) The temperature of first peak data requires three or six channels (recorded) with a minimum response of 0.33 milliseconds. In bandwidth terms;

$$BW = 3 \text{ KHz}.$$

3) A maximum recording time of sixty seconds is required to allow for flare impact time dispersions and data acquisition.

4) Simultaneous pointing is required for Spacecraft to Earth high-gain S-band antenna and the camera to flare impact area.

5) The Lunar Orbiter camera subsystem data is transmitted as in the existing communication subsystem.

SUMMARY

A summary of several tape recorder-reproducers, available for space applications is given in Table I. Core memory and wire recorders have been eliminated because of better performance, weight and volume of tape recorders for this application.

A seven track loop type recorder-reproducer with 90 seconds of recording time capability and 200 Hz to 3000 Hz response capability is available from existing hardware. Additional ground commands and subsequent Lunar Orbiter programming revisions are required to allow record and playback modes as well as selection of real-time or recorded data (Figure 1) for transmission. Recorded data is presented to the modulation selector sequentially (channel by channel basis) so that only one subcarrier channel is required in the RF portion of the system. The time to first peak data will be presented directly to the video channel through appropriate interface equipment. Total weight, including interface equipment, is 5.0 pounds.

TECHNICAL DISCUSSION

The relatively narrow-band response requirements (3000 Hz) allow a choice of several manufacturers for tape recorder-reproducers. Core memory storage has been eliminated due to the additional equipment necessary to perform analog to digital conversion prior to storage, and additional changes to the existing Lunar Orbiter communication subsystem. Wire recorders have also been eliminated due to limited capabilities and subsequent manufacturer de-emphasis in recent years.

No severe weight, power or volume constraints are necessary for the addition of a tape recorder-reproducer. The two types considered are the (1) reel-to-reel; and (2) endless loop. The reel-to-reel machine requires programming to allow for record, rewind and reproduce modes. The loop machine may run in one direction with end of loop sensing to stop the tape so as not to erase data already recorded. For short term recording of data the loop machine provides the most reliable operation.

Recorder Characteristics

The following characteristics summarize a typical (Raymond Engineering Model 1849) tape recorder-reproducer for space applications :

Wow and Flutter	
Bench Conditions	2% peak-to-peak
Extreme Environment	5% peak-to-peak
Tape Speed	30, 15, 7-1/2, 3-3/4 ips.
Tape Speed Accuracy	± 0.5%
Tape Capacity	100 ft.
Tape	1/2 or 1 inch 1 mil Mylar base.
Recording Time (Estimated 3 db response)	40 seconds at 30 ips. (0.2 to 45 KHz)
	90 seconds at 15 ips. (0.2 to 22.5 KHz)
	180 seconds at 7-1/2 ips. (0.2 to 12 KHz)
	360 seconds at 3-3/4 ips. (0.2 to 6 KHz)

TABLE 1. RECORDER - REPRODUCERS FOR SPACE APPLICATION

MANUFACTURER	APPLI-CATION	TYPE	DATA FREQUENCY RESPONSE (Hz)	# OF CHANNELS	TAPE SPEED (IN. / SEC.)	POWER (WATTS)	SIZE (INCHES)	WEIGHT (POUNDS)
Borg Warner	Re-entry	Loop	1000 to 100,000 ±2db	2	30.	22 Max.	7.5 in Dia. x 7.5 in.	10.5
Electronic Spec.	---	Reel	---	2	7.5	30 Ave.	13.3 in. x 6.5 in. x 3.13 in.	12.
Lockheed	---	Reel	10 to 5000	7	---	5	5 in. x 5 in. x 10 in.	5.
Raymond Engr.	Mariner, NASA Rocket & Balloon Tests	Loop	Greater than 5000	7	15.	9 Max.	4 in. Dia. x 3.625 in.	2.5
Telectro Ind.	---	Reel	200 to 10,000	4	3.75	Less than 60.	11 in. x 11 in. x 6.5 in.	13.4
Westrex	Missile	Loop	Direct--100 to 50,000 Carrier-- 0 to 2000	14	15.	25	4 in. Dia. x 3 in.	5.

Start Time	From rest to 15 ips -0.5 sec.
Stop Time	From 15 ips to rest -0.5 sec.
End of Tape Sensing	Stop or reversal at either end of tape.
Power Requirements	9 watts maximum
Weight	3 pounds maximum
Volume	Less than 150 cu. in.

Programming Characteristics

The loop machine requires programming for record and reproduce modes. In addition, programming is required to switch from real time to recorder data, and select one recorder channel for transfer to the RF subsystem (Figure 1). The estimated weight for additional related interface equipment is 2 pounds.

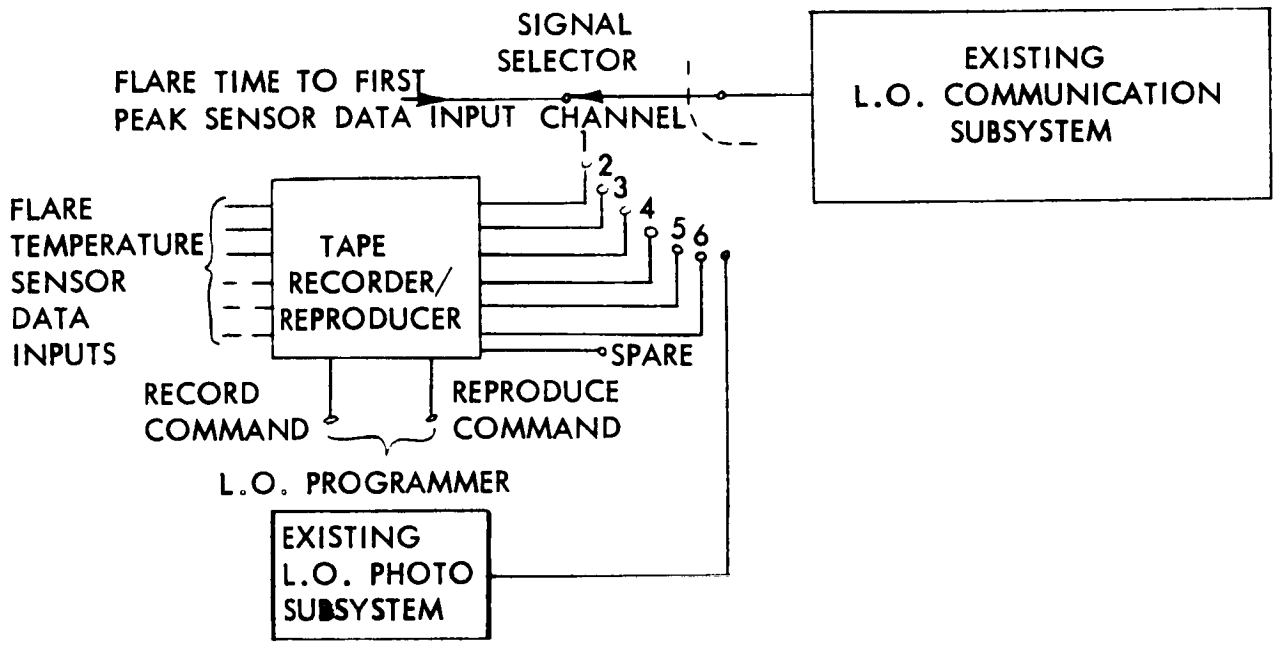


FIGURE 1. BLOCK DIAGRAM OF TAPE RECORDER / REPRODUCER MODIFICATION FOR LUNAR FLARE EXPERIMENT.

REFERENCE: Memorandum 2-1590-00-049, plus Attachment, dated May 19, 1966; E. J. Nicholson to the NASA Contract, "Contract NASA 6216 Report of Meetings between NASA and Contractor Personnel held 26-27 April 1966, Concerning Study Contract of Lunar Orbiter Adaptability to Experiments to Evaluate Lunar Surface Characteristics"

APPENDIX C

WORKING PAPERS ON PENETROMETERS

SPACE-GENERAL CORPORATION, LUNAR SURVEY PROBE DELIVERY VEHICLE

Introduction

In the accelerometer-penetrometer type of lunar surface probe a so-called delivery vehicle is required to perform the following general tasks:

- Transport a payload of several penetrometers from an orbiting parent vehicle to the area of the surface to be investigated.
- Release the penetrometers with a low (or zero) vertical velocity from a predetermined altitude so the impact velocity will approximate, within narrow limits, the desired value. In effect, to provide a semi-soft landing for the penetrometers.
- In some concepts, the delivery vehicle must hover for a brief additional time in order to provide a communication link for relaying the impact acceleration data from the penetrometers.

Before proceeding with a design concept for a delivery vehicle for use with the Lunar Orbiter, another delivery vehicle having similar requirements was reviewed. This vehicle, which is the result of a study by Space-General Corporation, was intended for use with the Apollo CSM. The only available information on this vehicle is contained in the referenced report, which is a copy of the figures which accompanied the final oral report. The following description is our interpretation of the control concepts and flight sequence employed in this delivery system.

System Description and Flight Sequence

Prior to separation from the Apollo CSM the gyros in the "strapped-down" inertial reference unit (IRU) are started, the flight program is stored. The cold gas reaction control system activated upon separation and a programmed attitude maneuver is accomplished (by torquing the gyros) to orient the vehicle for first retro firing. The first retro motor is then fired by a programmer signal. During the firing, pitch and yaw attitude is maintained and thrust misalignments compensated by a high-level, hydrazine, reaction control system which responds to error signals from both the gyros and a pendulum-type sensor in the IRU. Roll control is by the cold gas reaction jet system. During the firing, the velocity change, as sensed by an integrating accelerometer, is monitored by the programmer. After burn-out the velocity increment is trimmed to the desired value by four, 75 lb. thrust, hydrazine, vernier retro nozzles. Two seconds later a 180° attitude maneuver is accomplished as the vehicle coasts for approximately 625 seconds in a descent transfer trajectory. The second, and much larger, velocity change is then accomplished in exactly the same manner. The first stage motor case is separated prior to second stage ignition since it is mounted inside the second stage nozzle. This second retro firing removes the total horizontal component of velocity and, since the vertical component is negligible, the vehicle starts a free-fall from this point about 13,000 feet above the desired impact area. A 90 degree pitch over is programmed at this time to align the vehicle vertically. An altitude marking unit (AMU) senses when two pre-set values of true altitude are reached (9000 ft. and 5200 ft.). A descent timing unit (DTU) uses the elapsed time between altitude marks to compute the velocity change (ΔV_3) required to reduce the vertical velocity to zero at the penetrometer release altitude of 270 feet. The proper value of ΔV_3 can be found from a solution of the following

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System Description and Flight Sequence (Continued)

equation, the derivation of which is given in the appendix:

$$\Delta V_3 = \frac{h_1 - h_2}{\Delta t} + \frac{1}{2} g' \Delta t + \sqrt{2 g'(h_2 - h_3)}$$

where:

h_1 = first altitude mark

h_2 = second altitude mark

h_3 = penetrometer-release altitude

g' = lunar gravitational acceleration

Δt = time interval between first and second altitude marks

For a constant thrust system, the third retro ignition must take place at precisely the right time if the desired combination of zero velocity at 270 feet altitude is to be achieved. The ignition delay time (t_s), after the second altitude mark, can be computed from the following equation also derived in the appendix:

$$t_s = g' \sqrt{2 (h_2 - h_3)} - \frac{W_{SB} I_S}{F}\left(1 - e^{-\frac{\Delta V_3}{g I_s}}\right)$$

where: W_{SB} = vehicle weight at start burn

I_S = propellant specific impulse

F = retro thrust

g = Earth gravitational acceleration

(other symbols previously defined)

After the time delay, t_s, the third retro firing is initiated. The propulsion system in this case is the same hydrazine system (four 75 lb. nozzles) used for velocity trim in the first and second retro firings. The penetrometers are released when the desired value of ΔV_3 is attained as indicated by the integrating accelerometer. The thrusting continues for seven seconds to ensure that the penetrometers impact before the next operation in the sequence. After cut-off an undefined lunar surface probe capsule payload is released. One more firing is then initiated, accompanied by a pitch-over maneuver , presumably to prevent the delivery vehicle from impacting on or near the capsule.

To gain some idea of the data handling and storage requirements, Table I was made. This shows the magnitudes of the various quantities mentioned in the sequence description together with bit requirements for the estimated required accuracies shown.

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Reference: Final Report Oral Briefing, 29 November 1965; Lunar Survey Probe Delivery Vehicle Study; Contract NAS 9-4897; Space General Corporation.

Prepared by: D. F. Sears, May 19, 1966

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TABLE I

DELIVERY VEHICLE DATA HANDLING & STORAGE REQUIREMENTS

QUANTITY	UNITS	APPROX. MAGNITUDE	WORD LENGTH (BITS)	RESOLUTION
Reference or separation attitude (3-axis)	deg.	0-360	15	.01*
First retro orientation attitude maneuver (2-axis)	deg.	- 1	15	.01
Time of first retro firing	sec.	10 2	9	.08**
ΔV_1 magnitude	fps	414	14	.1
Time from first retro burnout to midcourse maneuver	sec.	5	7	.08
Attitude maneuver magnitude (1-axis)	deg.	180	15	.01
Coast time from first to second retro firing	sec.	625	14	.08
ΔV_2 magnitude	fps	5500	16	.1
Time from ΔV_2 cut-off to 90° pitch maneuver	sec.	2	7	.08
Pitch-over attitude maneuver magnitude	deg.	90	14	.01
Distance between altitude marks	ft.	3800	13	1
Vehicle "gravity loss" from 2nd. alt. mark to drop alt.	fps	232	13	.1
Lunar gravitational acceleration	fps^2	5.45	11	.01
$W_{SB}(I_s)/F$ (depends on vehicle design)	sec.	-	-	-
gI_s (depends on propellant used)	fps	6000	14	1

TABLE I

DELIVERY VEHICLE DATA HANDLING & STORAGE REQUIREMENTS (CONTINUED)

QUANTITY	UNITS	APPROX. MAGNITUDE	WORD LENGTH (BITS)	RESOLUTION
ΔV_3 magnitude (computed from Δt)	fps	400	13	.1
Time interval between altitude marks (Δt)	sec.	20	9	.08
Third retro firing delay (t_s)	sec.	5	7	.08
$\sqrt{2g'(h_2 - h_3)}/g'$	sec.	42.5	11	.05
$e^{-\frac{\Delta V_3}{g I_s}}$ (computed from ΔV_3)	-	1	11	.001

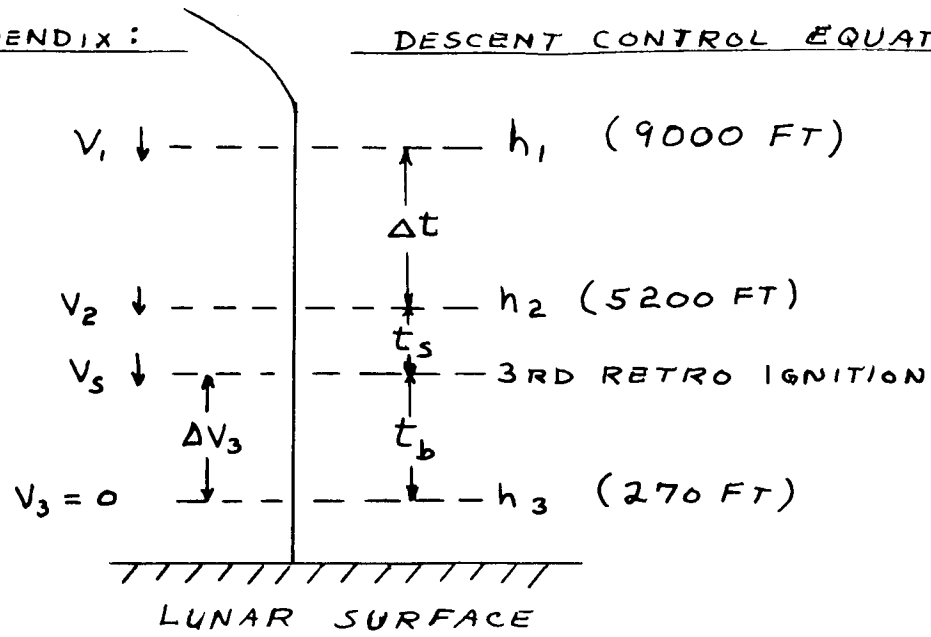
1 Not required if proper orientation is supplied by parent vehicle.

2 Not required if ΔV_1 is part of separation scheme.

* Based on Lunar Orbiter capability

** Based on minimum frequency from countdown chain (12.5 cps)

APPENDIX: DESCENT CONTROL EQUATIONS



PROBLEM: GIVEN h_1, h_2, h_3, AND Δt

SOLVE FOR ΔV_3 AND t_s FOR $V_3 = 0$

$$V_2 = \frac{h_1 - h_2}{\Delta t} + \frac{1}{2} g' \Delta t \quad \text{(FREE FALL)} \qquad (1)$$

WHERE: g' = LUNAR GRAVITATIONAL ACCELERATION

ΔV_3 MUST REMOVE V_2 AND THE VELOCITY INCREMENT EQUIVALENT TO FREE FALL FROM h_2 TO h_3

$$\Delta V_3 = V_2 + \sqrt{2 g' (h_2 - h_3)} \qquad (1a)$$

$$\boxed{\Delta V_3 = \frac{h_1 - h_2}{\Delta t} + \frac{1}{2} g' \Delta t + \sqrt{2 g' (h_2 - h_3)}} \qquad (2)$$

NEXT, GIVEN:

F, W_{SB}, ΔV_3 (from (2)), Δt AND $\Delta V = g I_s \ln \frac{W_{SB}}{W_{SB} - W_P}$

SOLVE FOR t_s

WHERE: W_{SB} = VEHICLE WEIGHT AT START BURN

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$$W_p = \frac{t_b F}{I_s} \qquad (4)$$

WHERE: t_b = BURN TIME

F = THRUST

I_s = SPECIFIC IMPULSE

W_p = PROPELLANT WEIGHT

SUBSTITUTE (4) IN (3) AND SOLVE FOR t_b

$$t_b = \frac{W_{SB} I_s}{F}\left(1 - e^{-\frac{\Delta V}{g I_s}}\right) \qquad (5)$$

TO SATISFY THE REQUIREMENT ΔV_3 MUST ALSO BE EQUAL TO THE VELOCITY AT START BURN (V_S) PLUS THE GRAVITY INCREMENT DURING THE BURN TIME.

$$\Delta V_3 = V_s + g' t_b \qquad (6)$$

BUT:

$$V_s = V_2 + g' t_s \qquad (7)$$

EQUATE (1a) AND (6), SUBSTITUTE (7) IN (6) AND SOLVE FOR t_s:

$$t_s = g' \sqrt{2(h_2 - h_3)} - t_b \qquad (8)$$

SUBSTITUTE (5) IN (8)

$$\boxed{t_s = g' \sqrt{2(h_2 - h_3)} - \frac{W_{SB} I_s}{F}\left(1 - e^{-\frac{\Delta V_3}{g I_s}}\right)}$$

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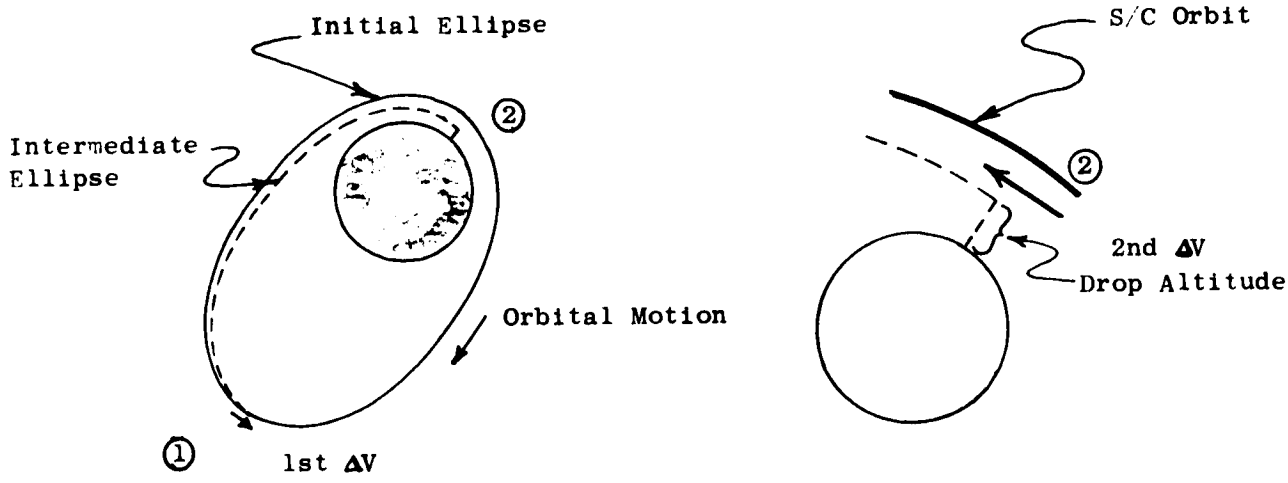
BASIC DATA FOR PENETROMETER DROP DELIVERY

SUMMARY

Basic data pertaining to the "drop delivery" considered for the penetrometer are presented herein. In this type of delivery, the spacecraft and the penetrometer are initially in the same orbit. The nominal Lunar Orbiter photographic orbit, 1850 - 46 km, was chosen for this study. A small retro ΔV is applied to the penetrometer at apolune of the initial orbit, placing it in an intermediate ellipse with an apolune altitude of 1850 km. At perilune of the intermediate orbit a large retro ΔV, equal to the orbital velocity, is applied. Since the resultant penetrometer velocity is zero (nominally), the penetrometer drops to the lunar surface. The perilune altitude of the intermediate orbit is referred to as the "drop altitude" in the following discussion. The sketch below illustrates the sequence of events in this "drop delivery" scheme.



PENETROMETER DROP DELIVERY

DISCUSSION

Impact Velocity:

The impact velocity depends only upon the altitude from which it is dropped. In this study, the penetrometer is dropped from the perilune of the intermediate orbit. Figure 1, gives the impact velocity for drop altitude from 0 to 50 kms.

Impact Velocity Control:

Since the impact velocity depends only upon the drop altitude, control of the impact velocity is equivalent to controlling or accurately predicting the drop altitude. The sensitivity of impact to an error in drop altitude is also given in Figure 1. The sensitivity increases with decreasing drop altitude; for example, at a nominal drop altitude of 2.5 km, a 100-meter altitude error causes a 1.95 m s error in impact velocity. At a nominal drop altitude of 40 km, a 100-meter altitude error produces a .45 m/s error. The acceptable uncertainty in drop altitude then depends upon the nominal drop altitude and the maximum acceptable uncertainty in impact velocity.

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Drop Altitude Control:

The velocity required at apolune to acquire a specified drop altitude is illustrated in Figure 2. As can be seen, this impulse is small. However, the error in drop altitude is quite sensitive to errors in the magnitude of the velocity increment. Figure 2 also gives this sensitivity for nominal drop altitudes from 0 to 50 km. For this range of drop altitudes shown, an error of ±1 m/s in the ΔV at apolune results in about a ±5.5 km error in the drop altitude. Errors in the direction of the ΔV at apolune have a negligible effect on the drop altitude. Once the uncertainty in drop altitude is known, the uncertainty in impact velocity can be found from Figure 1, top.

Variations in Drop Altitude Due to Orbital Perturbations:

The non-spherical lunar gravitational and third body effects of the Earth cause a continuous fluctuation in the semi-major axis and the eccentricity. This results in a continuous variation in the unperturbed perilune altitude. Figure 3 shows a typical time variation due to the lunar gravitational potential (Clarke's Model) and Earth effects. The vehicle would reach the unperturbed perilune altitude shown for a given time, if the perturbations vanished at that time. For example, in the case shown in Figure 3, the vehicle is injected into orbit at apolune: altitude - 1850 km. The initial unperturbed perilune altitude is 46 km. However, as the vehicle progresses around the orbit, the unperturbed perilune altitude decreases. By the time the vehicle reaches perilune, the perilune altitude is .68 km less than the initial unperturbed altitude.

Since this effect is significant, and can be predicted, it should be included in mission planning. Also, by careful selection of the initial orbit orientation, this effect can be minimized.

Total Velocity Required:

Figure 3, bottom, gives the total velocity required to complete the penetrometer "drop delivery" for drop altitudes ranging from 0 to 50 km. This requirement is the sum of the apolune maneuver requirements shown in Figure 2, and the velocity increment required to completely cancel the vehicle's orbital velocity at perilune. Although the perilune impulse is very large, the effect of errors in the magnitude and direction on impact velocity is negligible. (Velocity requirements for maneuvers performed off-apolune are included in the Reference 1.)

CONCLUSION

Once the nominal impact velocity and the allowable uncertainty in velocity are known, the drop altitude and the allowable uncertainty in altitude can be determined. Prediction of the drop altitude should include the effect of the non-spherical gravitational potential, and the Earth. This effect can be minimized by careful selection of the initial orbit parameters. Uncertainties in the size of the impulse at apolune cause significant uncertainties in the "drop altitude." If this velocity increment cannot be controlled with sufficient precision, ways of reducing this sensitivity must be investigated.

REFERENCE

1. Coordination Sheet 2-5763-1-CS098, "Use of Lunar Orbiter for Lunar Surface Penetration Experiment," August 17, 1965

Prepared by T. J. Miller, 2/17/66

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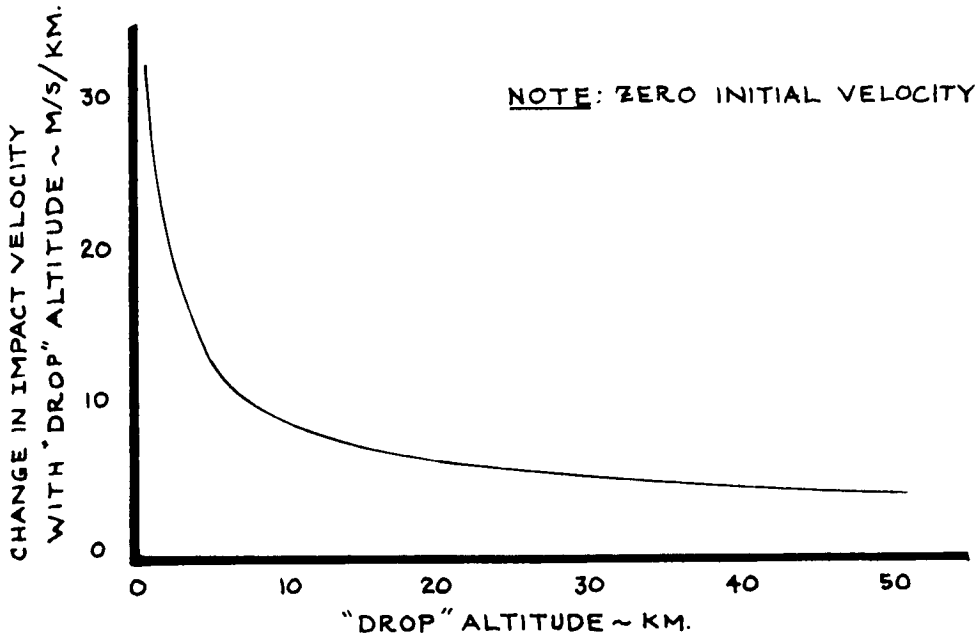
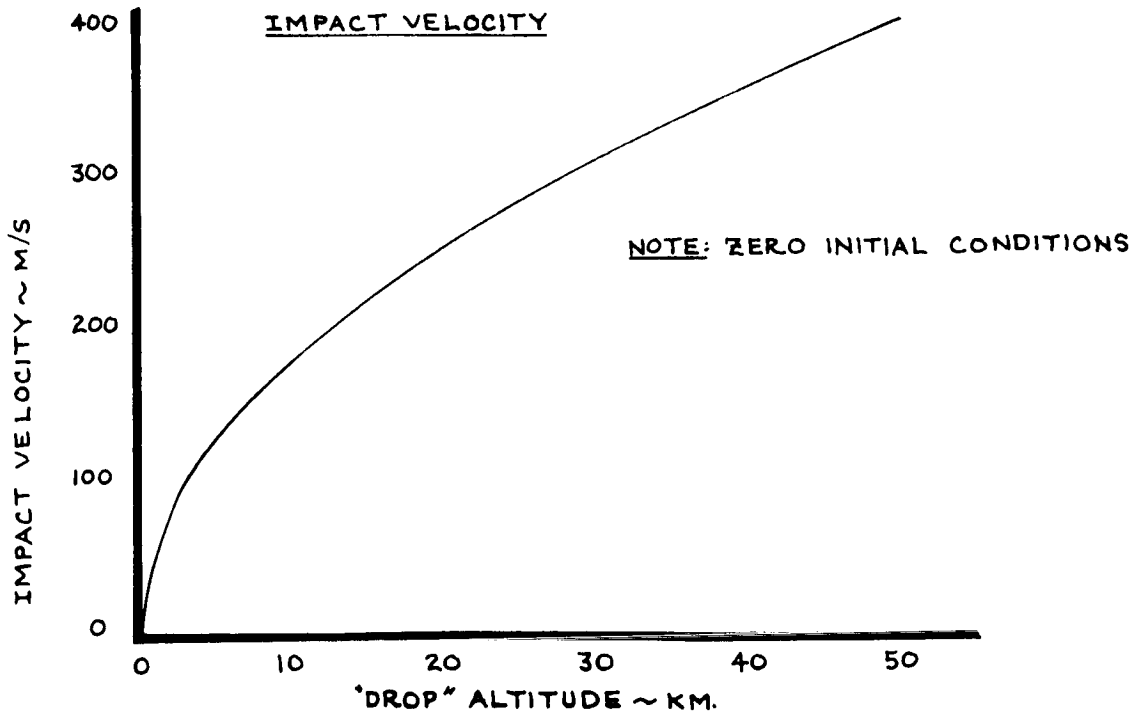
PENETROMETER STUDIES

IMPACT VELOCITY SENSITIVITY TO "DROP" ALTITUDE





	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
CALC	T. MILLER	2/1/66			PENETROMETER STUDIES IMPACT VELOCITY AND SENSITIVITY TO DROP ALTITUDE	FIG. 1
CHECK						
APPD						
APPD						

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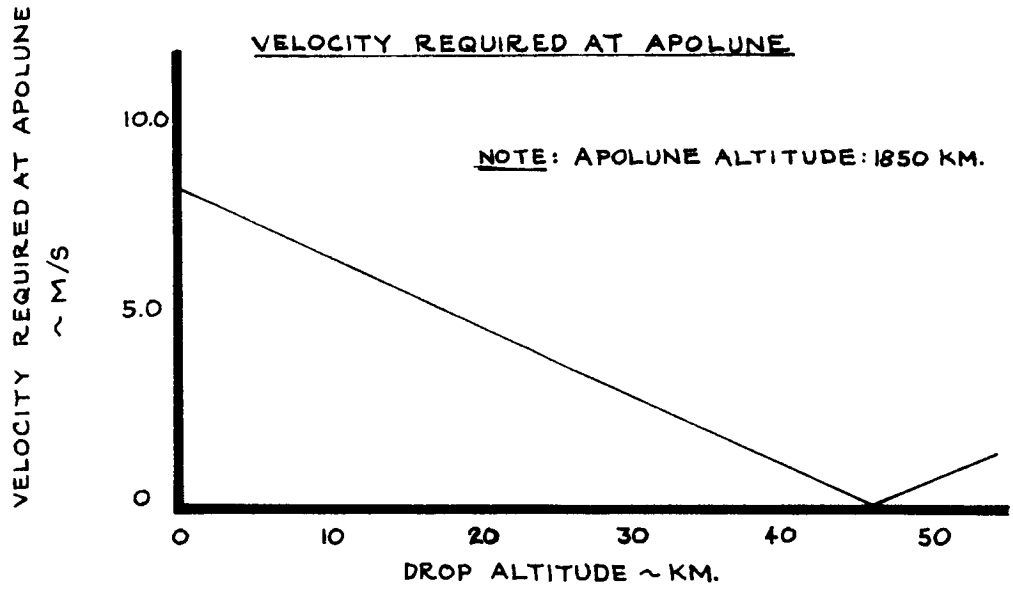
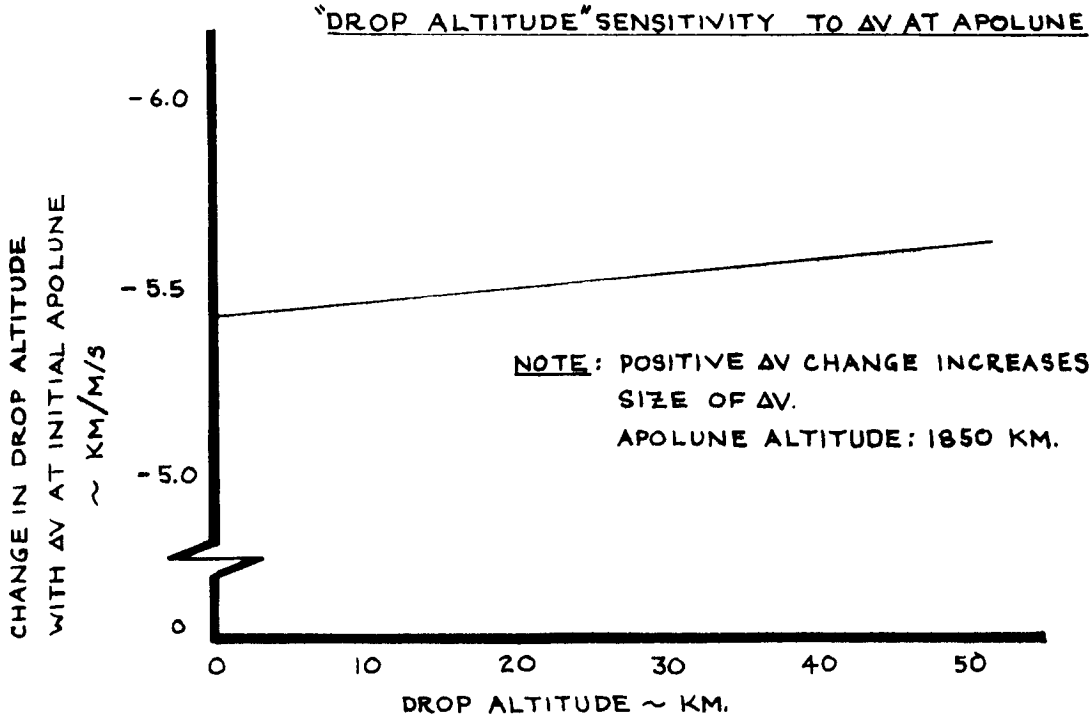
REV LTR

BOEING NO D2-100690-2

SH 228

PENETROMETER STUDIES





	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
CALC	T. MILLER	2/1/66			PENETROMETER STUDIES ΔV REQUIRED AT APOLUNE AND DROP ALTITUDE SENSITIVITY	FIG. 2
CHECK						
APPD						
APPD						

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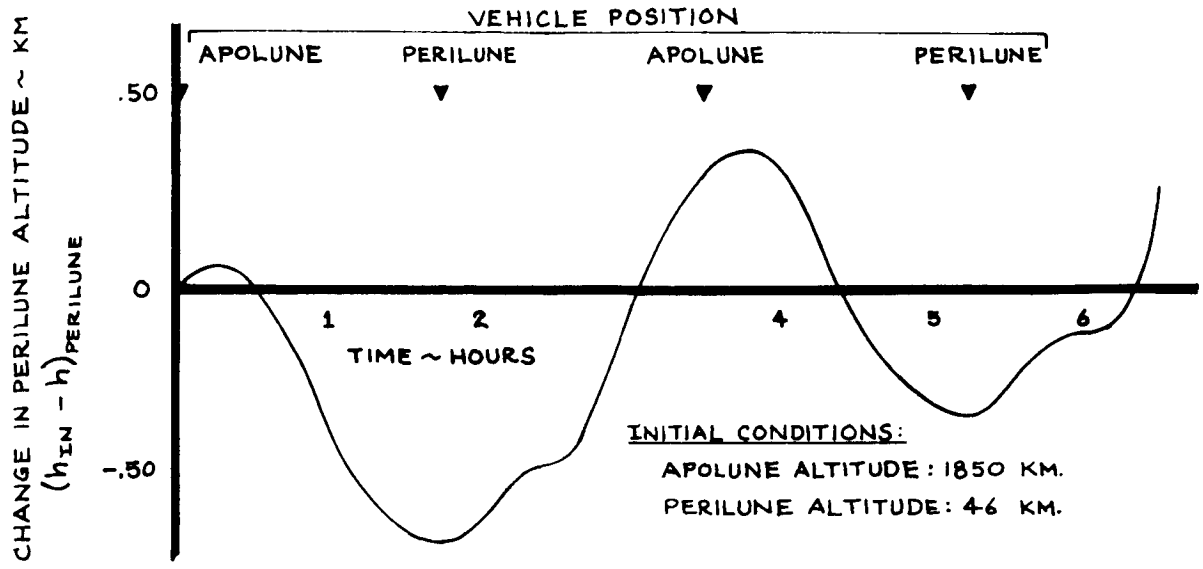
REV LTR

BOEING NO D2-100690-2
SH 229

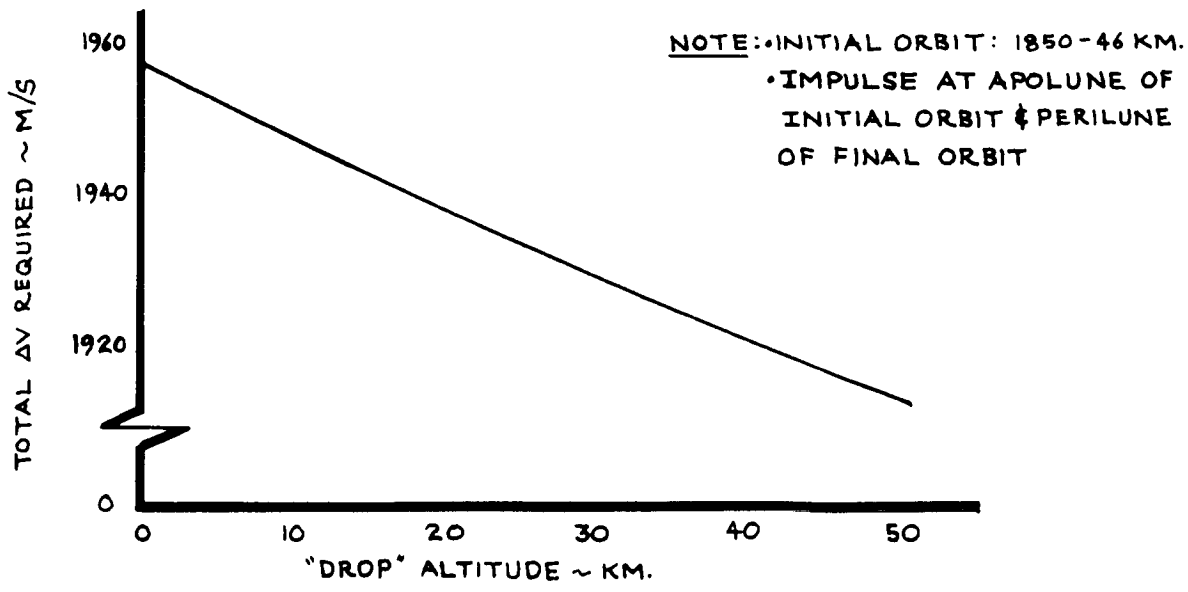
PENETROMETER STUDIES

SHORT-SHORT PERIOD VARIATION IN PERILUNE ALTITUDE

NOTE: RESULTS BASED ON CLARKE'S MODEL
EARTH EFFECTS INCLUDED



TOTAL ΔV REQUIRED



	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
CALC	T. MILLER	2/1/66			PENETROMETER STUDIES TOTAL ΔV REQUIRED AND VARIATION IN PERILUNE ALT.	FIG. 3
CHECK						
APPD						
APPD						

U3 4013 8000 REV. 12-64

REV LTR

BOEING NO D2-100690-2
SH 230

U.H.F. ANTENNA COVERAGE FOR PENETROMETER-TO-LUNAR ORBITER COMMUNICATION LINK

INTRODUCTION

The referenced letter establishes the requirement to determine a suitable L.O. UHF antenna configuration to receive the Penetrometer-to-Lunar Orbiter telemetry data signals from the accelerometer waveforms at time of impact (paragraph A-7 of reference). The antenna will provide dual polarization for reception of the penetrometer signals for diversity combining.

The antennas shall provide a load impedance of 50 ohms with an allowable VSWR at the receiver of 2:1 over the frequency range of 430 MHz to 460 MHz. The purpose of this discussion is to identify the UHF antenna requirements which impose constraints on the spacecraft and penetrometer trajectories and the spacecraft attitude. The discussion is based on the consideration of the following three configurations:

Configuration 1: Fixed-Antenna Pointing Normal to the Lunar Surface (Figure 1)

Configuration 2: Fixed-Antenna Pointing 30° from the normal to the Lunar Surface (Figure 2), and

Configuration 3: Sweeping-Antenna Pointing at Preselected Angles as a Function of Time (Figure 3) (Angular rotation limited to ± 30° from normal to the Lunar surface.)

These configurations have been considered on a relative basis to allow comparison and visibility in selecting a final configuration. This approach was based on the following assumptions to provide a realistic comparison:

1) The altitude of the L.O. spacecraft trajectory is constant in the vicinity of the penetrometer impact area.

2) The perilune velocity of the L.O. spacecraft in the vicinity of the penetrometer impact area is a constant 2 KM/sec.

3) The penetrometers will fall within a circle one kilometer in diameter.

4) The UHF antenna coverage area must be sufficient to account for the area and time dispersions of the impacting penetrometers, spacecraft position and antenna pointing errors relative to the impact area at time of impact.

5) The antenna pattern is conical with a circular cross-section. The maximum UHF antenna gain is limited to + 16 db (26 degrees beamwidth) to provide realistic limitations of size and weight to be added to the L.O. spacecraft. UHF antennas with a + 10 db (52 degrees beamwidth) gain have been designed for bandwidths of greater than 500 MHz with a weight of 4.3 pounds.

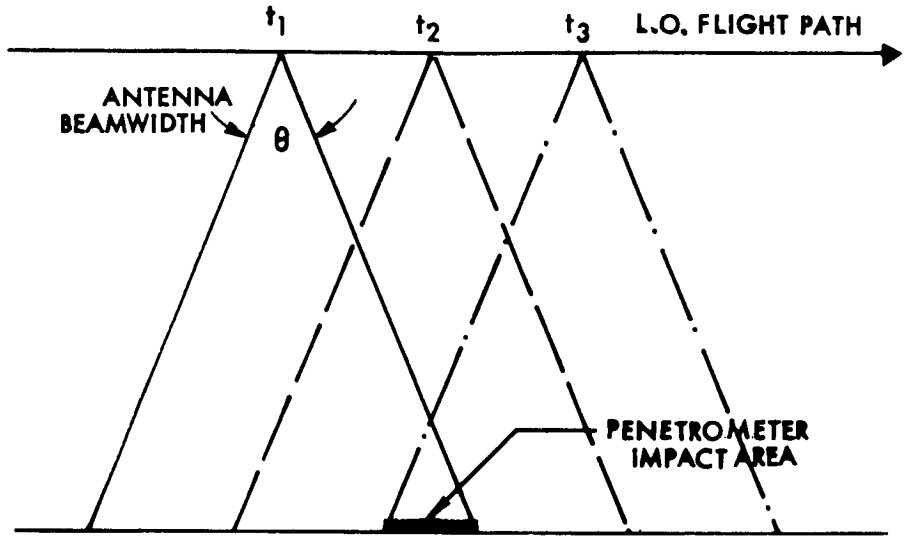


FIGURE 1. CONFIGURATION 1 -- FIXED ANTENNA, NORMAL POINTING

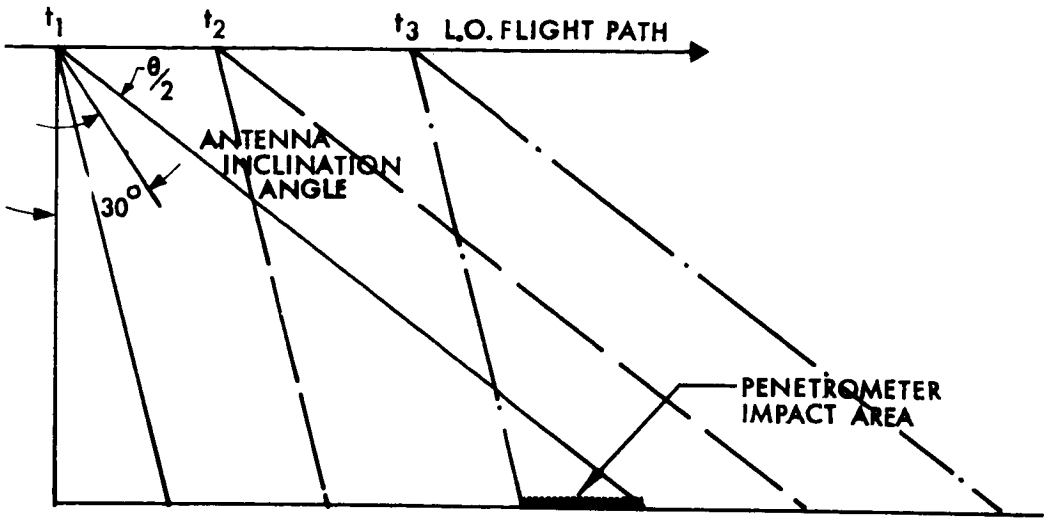


FIGURE 2. CONFIGURATION 2 -- FIXED ANTENNA INCLINED POINTING

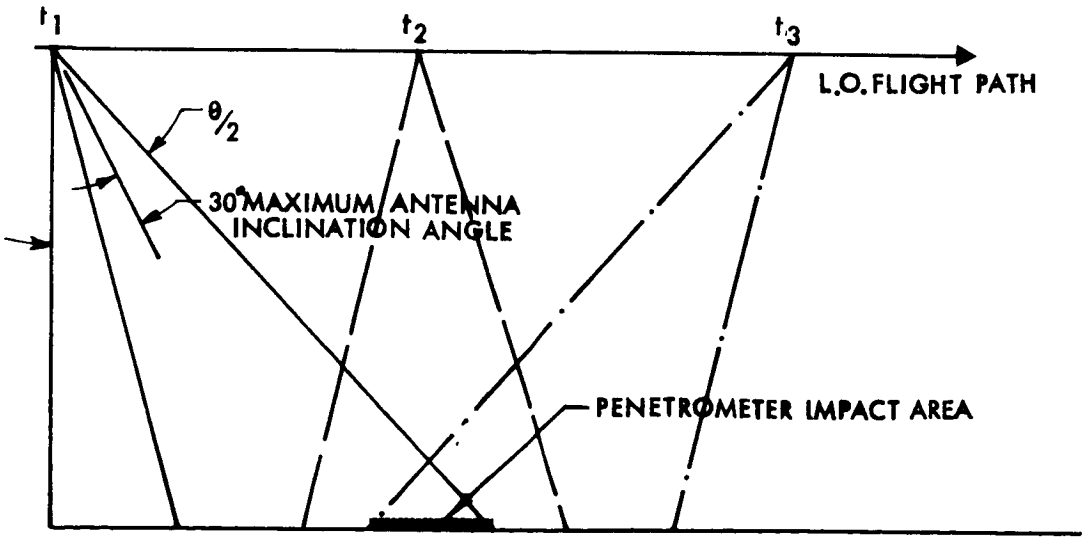


FIGURE 3. CONFIGURATION 3 -- SWEEPING ANTENNA

6) The link margin of + 13 db includes allowances for a -3 db variation of UHF antenna gain, attenuation of penetrometer signal due to burial and other system tolerances. The acceptance of a + 13 db link margin is contingent on the approval of LRC as requested in the reference memorandum. However, unless otherwise notified the +13 db will be used as a basis for design. Since a + 18 db link margin was also mentioned by NASA, it has been included.

7) The downlink transmission of telemetry data to Earth via the high-gain S-band antenna may be accomplished by either one of the following two methods.

- Real Time Transmission

  The S-band antenna must be pointed to Earth and the UHF antenna must cover the impact area at time of impact simultaneously

- Record-Playback Transmission

  The S-band antenna may be pointed at Earth after data acquisition to avoid the simultaneous antenna pointing requirements by use of an on-board recorder. The information would be recorded for data acquisition followed by playback at a later time.

SUMMARY

The detailed approach and results are described in the technical discussion and may be summarized as follows:

1) The trade between antenna gain, which determines maximum communication range, and antenna beamwidth results in approximately a constant ground coverage capability as altitude is varied (Figure 4, typical). However, higher gain antennas such as + 16 db do result in higher altitude or range capability.

   The inclined pointing antenna does not allow a greater ground path coverage than the fixed normal pointing, but does require a lower altitude for the same beamwidth (Figure 6, typical).

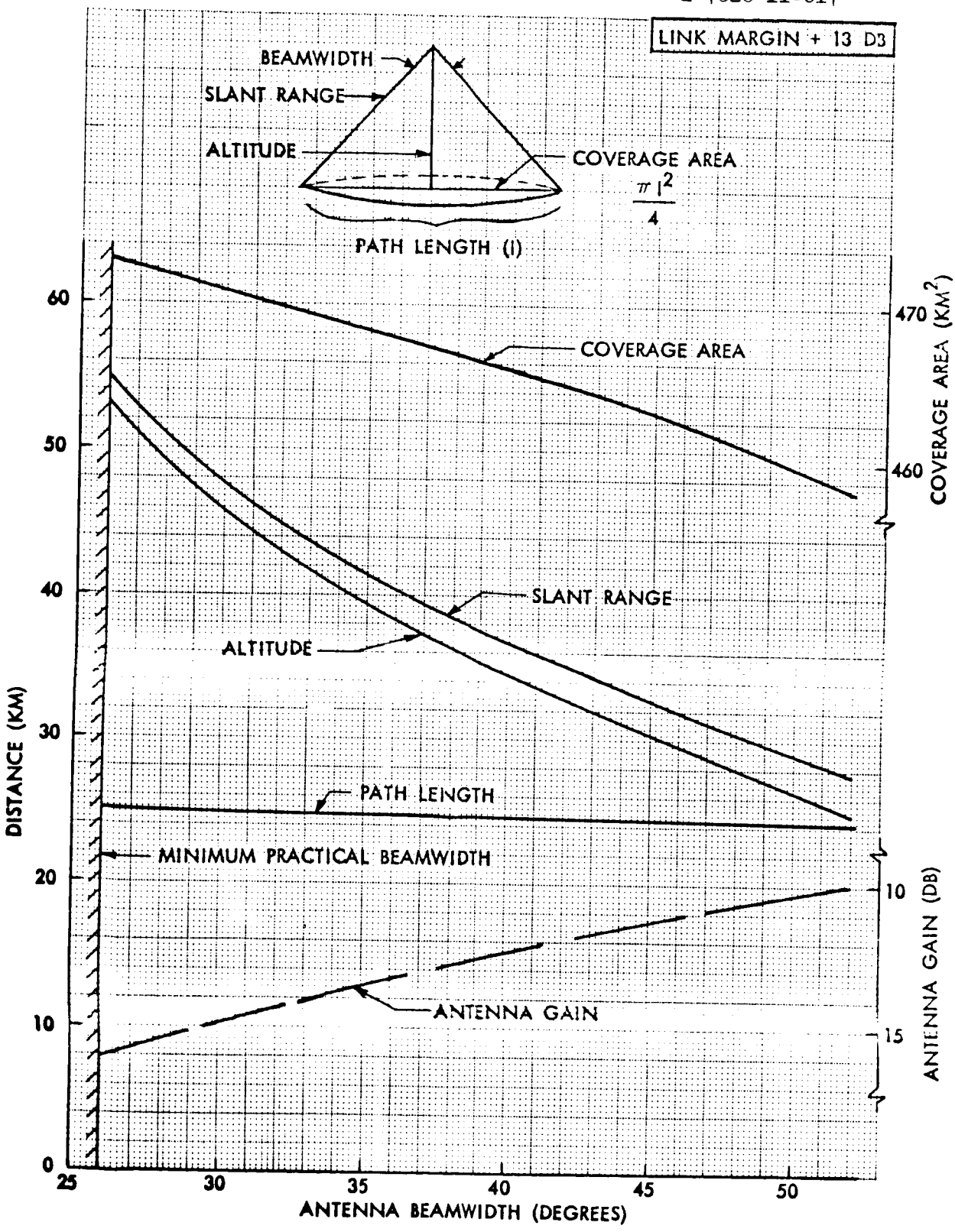


FIGURE 4 PENETROMETER - LUNAR ORBITER UHF ANTENNA CONSTRAINTS

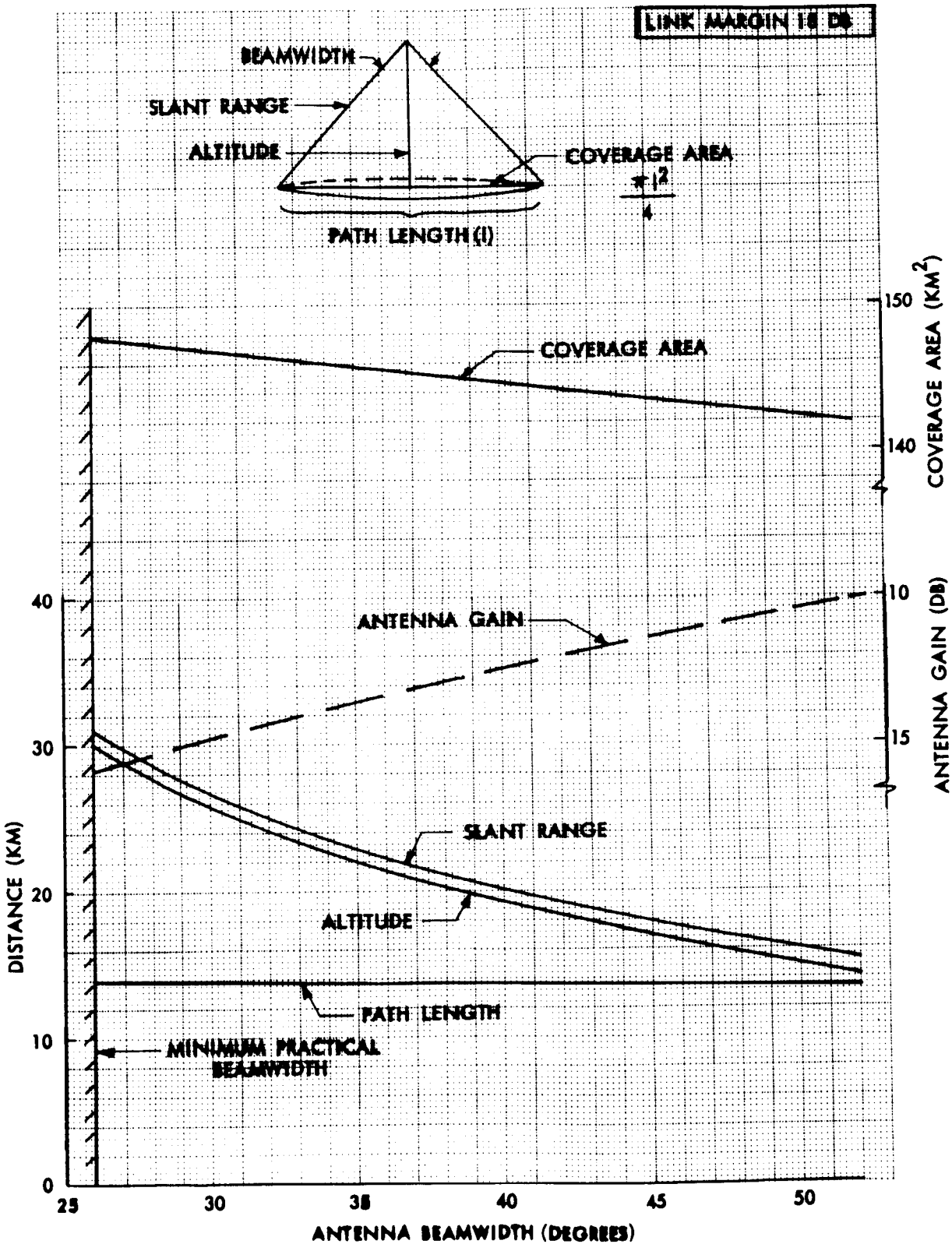


FIGURE 5. PENETROMETER - LUNAR ORBITER UHF ANTENNA CONSTRAINTS

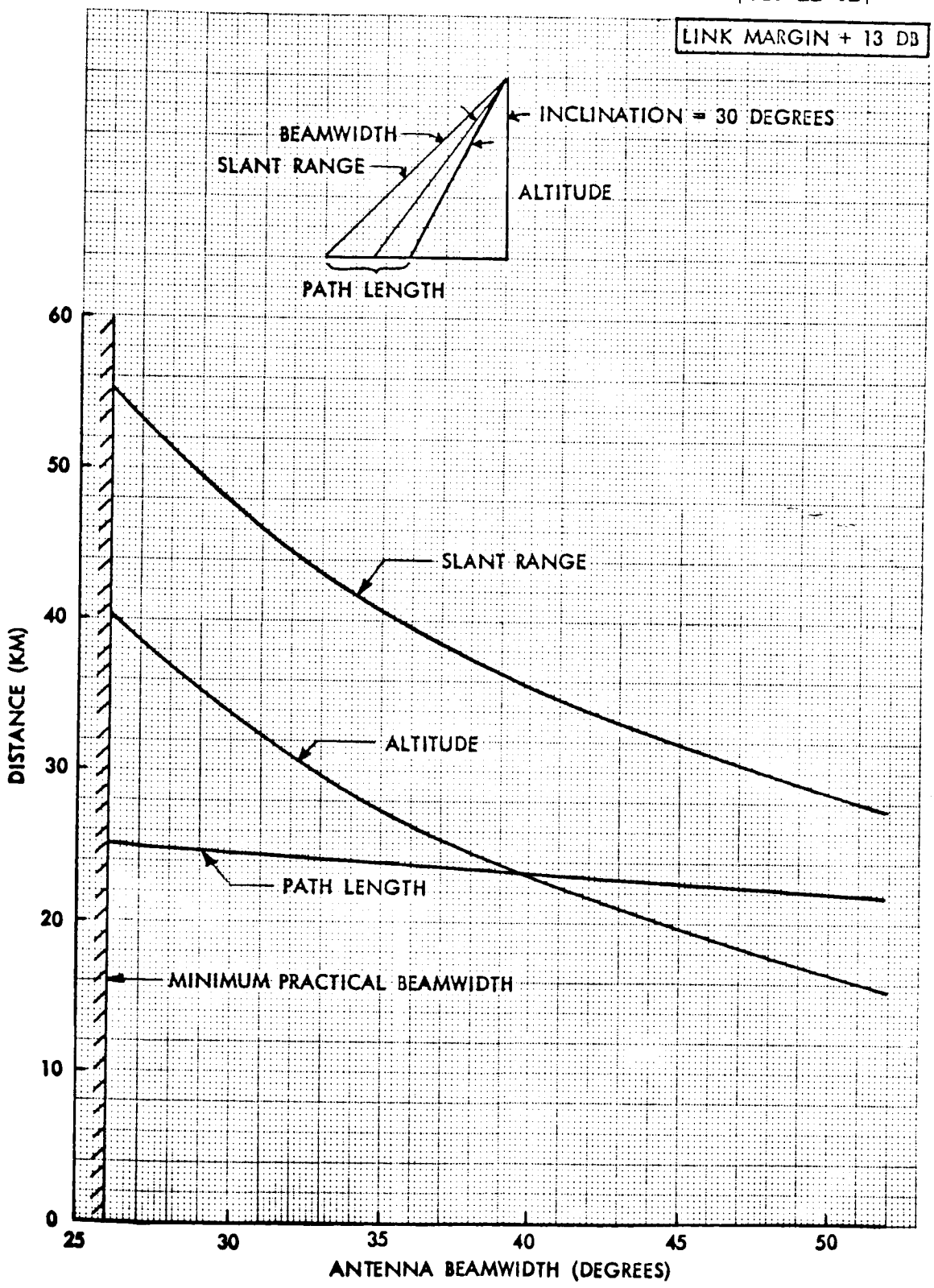


FIGURE 6 PENETROMETER - LUNAR ORBITER UHF ANTENNA CONSTRAINTS

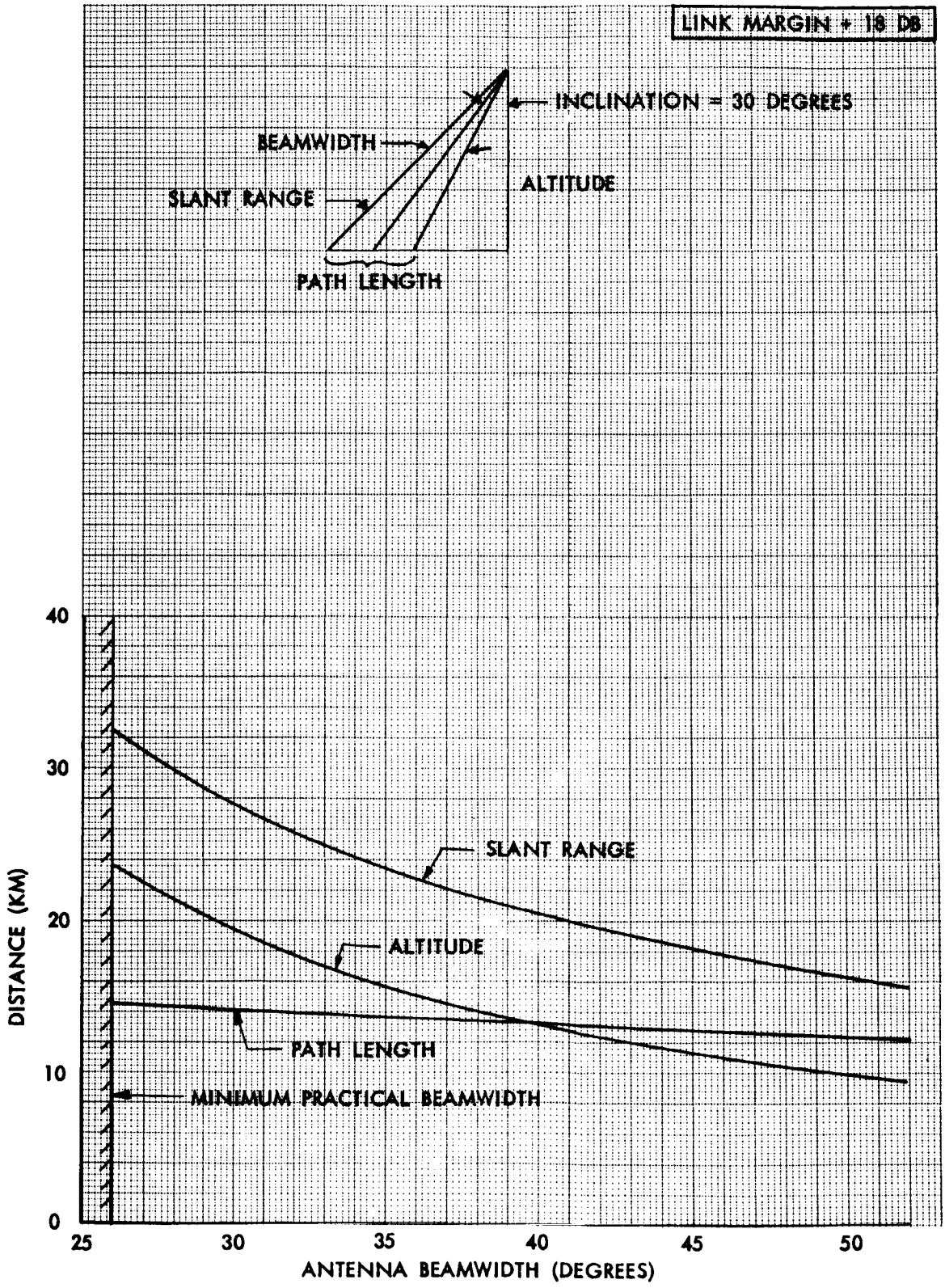


FIGURE 7. PENETROMETER - LUNAR ORBITER UHF ANTENNA CONSTRAINTS

2) The viewing time of the impact area is approximately 12 seconds for fixed antenna pointing whether the beamwidth is 26 degrees or 52 degrees, and regardless of normal or inclined antenna pointing (Figure 8). However, the viewing time for a sweeping antenna is 35 seconds for a beamwidth of 26° and 19 seconds for a beamwidth of 52° (+ 13 db link margin). It should be noted that although an increase in viewing time results with the use of a sweeping antenna, there may be a significant weight penalty compared with the fixed pointing cases. Preliminary information indicates that a 52° beamwidth antenna would have minimum dimensions of 0.75 feet diameter by 2.2 ft. in length and a 26° beamwidth antenna would have minimum dimensions of 0.75 feet diameter by 9 ft. in length.

A summary of the results are shown in Table 1 for the three configurations which were considered. Additional work on the antenna configuration will be suspended until the following information is available:

1) LO spacecraft postion uncertainties relative to penetrometer impact area during data acquisition time,

2) LO spacecraft UHF antenna pointing uncertainties relative to impact area during data acquisition time,

3) LO spacecraft nominal trajectory during data acquisition time,

4) Penetrometer area and time dispersion (3σ value) from nominal target area and impact time, and

5) The desired configuration for transmitting the data to Earth

   a) real-time transmission, or
   b) record-playback transmission.

TABLE 1

SUMMARY COMPARISON OF THREE ANTENNA CONFIGURATIONS

CONFIGURATION	DESCRIPTION	ANTENNA GAIN (db)	ANTENNA BEAMWIDTH (Degrees)	MAXIMUM ALTITUDE (KM)	MAXIMUM SLANT RANGE (KM)	MAXIMUM ANTENNA VIEWING* TIME/Sec.	MAXIMUM GROUND PATH LENGTH (KM)
	For + 13 db Link Margin						
NO. 1	Fixed antenna pointed normal to Lunar surface	+ 16	26	53.5	55.0	12.0	24.8
		+ 10	52	24.5	27.5	11.6	24.2
NO. 2	Fixed antenna pointed 30 degrees from the normal to the Lunar surface	+ 16	26	40.2	55	12.0	24.2
		+ 10	52	15.3	27.5	10.4	21.6
NO. 3	Sweeping antenna from ± 30 degrees from normal to the Lunar surface	+ 16	26	40.2	55	37.1	24.2
		+ 10	52	15.3	27.5	22.8	21.6
	For + 18 db Link Margin						
NO. 1	Fixed antenna pointed normal to Lunar surface	+ 16	26	30.0	31.0	6.5	13.7
		+ 10	52	24.0	25.3	6.2	13.5
NO. 2	Fixed antenna pointed 30 degrees from the normal to the Lunar surface	+ 16	26	22.5	31.0	6.5	14.0
		+ 10	52	9.5	15.3	10.4	12.2
NO. 3	Sweeping antenna from ± 30 degrees from normal to the Lunar surface	+ 16	26	22.5	31.0	20.6	14.0
		+ 10	52	9.5	15.3	12.3	12.2

* Based on 1 KM Path Length for Impact Area

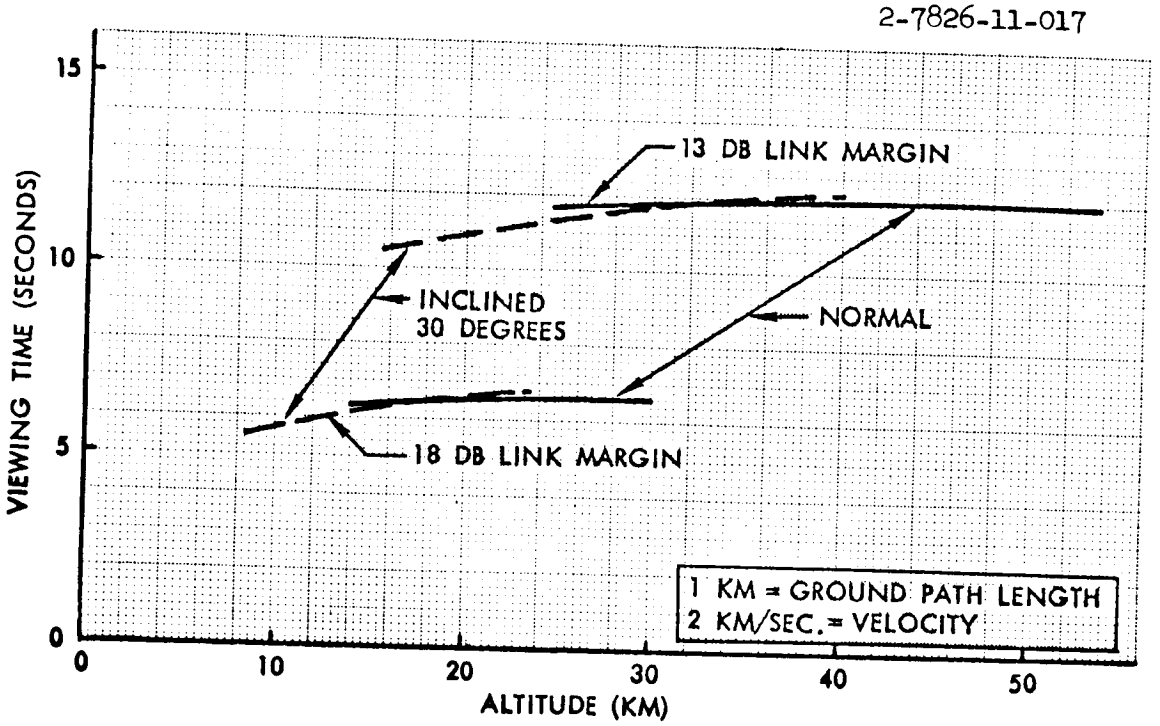


FIGURE 8 VIEWING TIME CAPABILITY - REALIZABLE, FIXED ANTENNAS

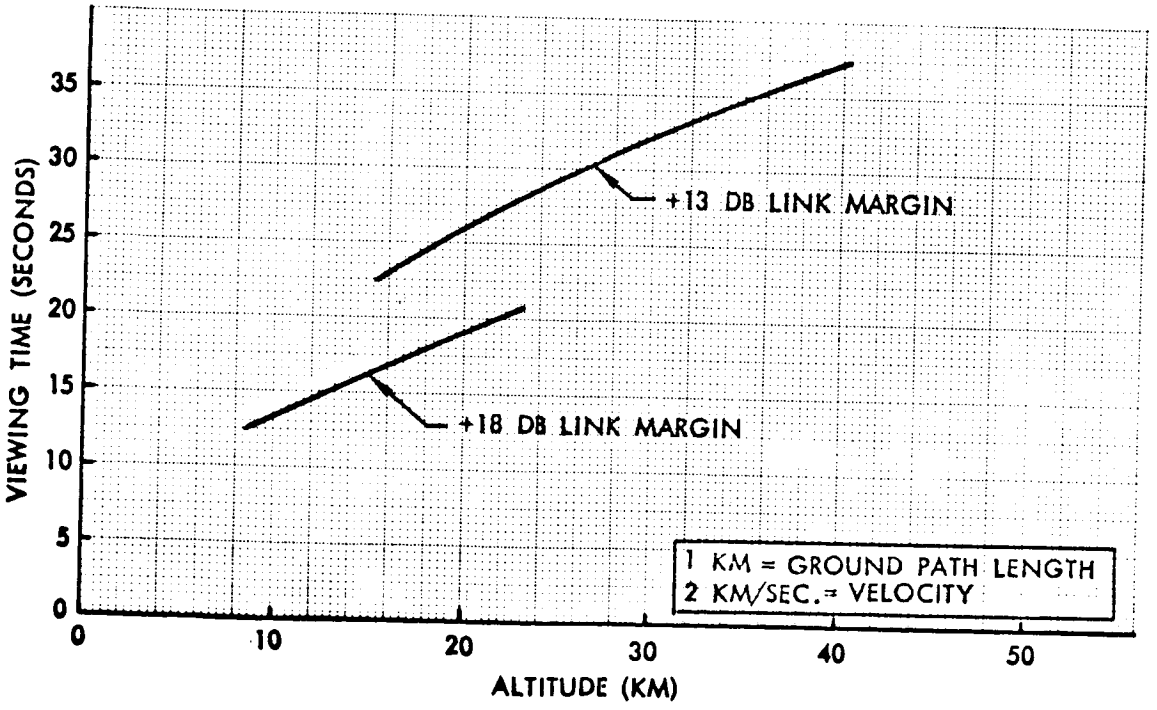


FIGURE 9 VIEWING TIME CAPABILITY - REALIZABLE, SWEEP ANTENNAS

TECHNICAL DISCUSSION

1.0 Fixed Antenna Pointing Normal to the Lunar Surface

Figures 4 and 5 show the constraints on spacecraft altitude, range, and ground path length for various beamwidth (gain) antennas. The following definitions apply:

Altitude (h)	-	Normal distance of spacecraft from Lunar surface, in kilometers
Slant Range (R)	-	Distance measured from spacecraft where antenna gain is 3 db down, in Kilometers.
Ground Path Length (ℓ)	-	Diameter of cone shaped beam as projected on Lunar surface, in Kilometers
Coverage Area (A)	-	Area of beam as projected on Lunar surface, in square Kilometers for - 3 db beamwidth
θ	=	- 3 db Beamwidth in degrees

From Appendix:

$$*R_{13} = \frac{1430}{\theta} \quad (1)$$

$$*R_{18} = \frac{805}{\theta} \quad (2)$$

$$h = R \cos \frac{\theta}{2} \quad (3)$$

$$\ell = 2R \sin \frac{\theta}{2} \quad (4)$$

$$A = \pi R^2 \sin^2 \frac{\theta}{2} \quad (5)$$

From equations (1, 4) it is noted that ℓ varies as $\frac{\sin \theta/2}{\theta}$ and for θ between 26° and 52°, this variation is small. For this reason the ground path length and coverage area do not increase significantly with altitude.

* Subscripts refer to link margin in db.

2.0 Fixed Antenna - Pointing 30° from Normal

Figures 6 and 7 show constraints on spacecraft altitude, range and ground path length for various beamwidth (gain) antennas. The same definitions apply as for the normal pointing case with the addition of:

σ = Inclination angle from normal to Lunar surface in degrees (30 degrees).

From Appendix:

$$h = R \cos\left(\sigma + \frac{\theta}{2}\right) \tag{6}$$

$$\ell = R \cos\left(\sigma + \frac{\theta}{2}\right)\left[\tan\left(\sigma + \frac{\theta}{2}\right) - \tan\left(\sigma - \frac{\theta}{2}\right)\right] \tag{7}$$

From equation (6) it is noted that due to the fact that R_{13} and R_{18} are constant for a given beamwidth, h must decrease when an inclined beam is considered. This fact indicates why inclined antenna pointing requires lower altitudes than normal pointing. The inclined antenna has the advantage of allowing the penetrometer impact area to lie ahead or behind the L.O. spacecraft at time of impact.

3.0 Sweeping Antenna ± 30° from Normal to Lunar Surface

Figures 4 through 7 show only constraints due to fixed antenna geometry. For the sweeping antenna case the geometry is defined by these constraints, i.e., if the antenna starts its sweep from the 30° inclined position, altitude, range and ground path length are constrained to be those of the fixed inclined case (Figures 6 and 7). Of course, this is true for only one instant in time. However, due to the previous assumption of constant altitude flight path at impact, the altitude for any time during the pass is equal to that required at the ± 30° inclination angles.

The ground path length will vary as the spacecraft passes over a target due to the antenna sweeping (Figure 3). In order to determine the advantage of the sweeping antenna over the fixed cases, it is necessary to assume a coverage area or path length and a spacecraft velocity. These assumptions allow a comparison of the viewing time for each case.

The total viewing time is considered to be the time period when the penetrometer impact area is entirely within the antenna beamwidth. From the Appendix (Section F), the total viewing time is:

$$T = \frac{2\ R\ \mathrm{Sin}\ (\sigma + \theta/2) - \ell c}{v}$$

where $\ell c = 1$ km

$v = 2$ km/second

Figures 8 and 9 give a comparison of viewing times for the fixed and sweeping antenna.

APPENDIX: Fixed Antenna Pointing - Link Calculations for Figures 4, 5, 6 and 7

A) Link Analysis

PENETROMETER - LUNAR ORBITER (f = 450 MHz)

Transmitter Power	P_T	- 3.0 dbw
Transmitter Circuit Loss	L_{TR}	- 1.0 db
Transmitting Antenna Gain	G_{TR}	- 3.0 db
Polarization Loss	L_{POL}	- 3.0 db
Free Space Loss	L_{FS}	-85.7 - 20 log R(Rin KM)
Receiving Antenna Gain	G_R	G_R
Received Signal Power	P_{REC}	-95.7 - 20 log R + G_R (dbw)
Noise Spectral Density	kt	- 204 dbw/cps
System Noise Figure	F	6 db
IF Bandwidth (700K H_z)	B_{IF}	58.5 db
Signal-to-Noise (IF)	$(C/N)_{IF}$	12.0 db
Margin	M	M
Received Noise Power at Threshold	P_{THR} =	- 127.5 + M (dbw)

For Maximum Range $P_{REC} = P_{THR}$

$$-95.7 - 20 \log R + G_R = -127.5 + M$$

for M = 13 db

$$20 \log R = -95.7 + 127.5 - 13 + G_R = 18.8 + G_R$$

for M = 18 db

$$20 \log R = 13.8 + G_R \quad \text{(in db)}$$

B)

for + 13 db Margin

$$20 \log R = 18.8 + G_R$$

$$\log R = .94 + 0.5 \log gr \quad (10 \log gr = G_r)$$

$$R = 8.7 \sqrt{gr}$$

$$\theta^2 = \frac{27000}{gr}$$

$$\theta = \frac{164}{\sqrt{gr}}$$

$$R = 164 \left[\frac{8.7}{\theta}\right] = \frac{1430}{\theta}$$

C) For 18 db Margin

$$\log R = 0.69 + 0.5 \log gr$$

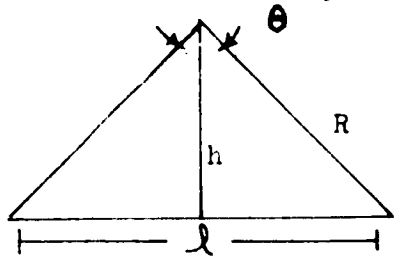
$$R = 4.9 \sqrt{gr}$$

θ = Beamwidth

$$\theta^2 = \frac{27000}{gr} \quad \text{(ITT Handbook)}$$

$$\theta = \frac{164}{\sqrt{gr}}; \quad \frac{\theta}{2} = \frac{82}{\sqrt{gr}}$$

$$R = \frac{805}{\theta}$$



R = Slant range

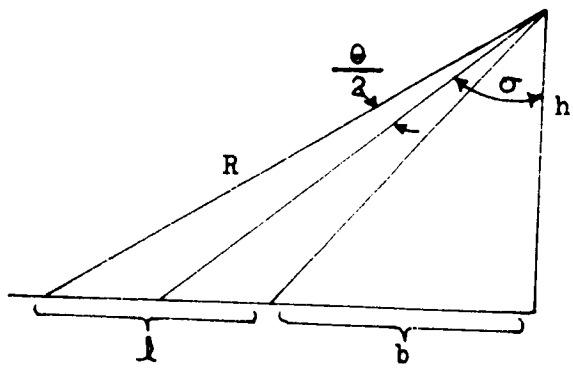
ℓ = Diameter of cone base projected on lunar surface (ground path length)

h = Orbiter altitude

Path length $\ell = 2 R \sin \frac{\theta}{2}$; Altitude $h = R \cos \frac{\theta}{2}$

Area $= A = \frac{\pi \ell^2}{4} = \pi R^2 \sin^2 \frac{\theta}{2}$

E) VHF Antenna Pointing Other Than 90° to Lunar Surface for Figures 5 and 7



For + 18 db Margin; For + 13 db Margin

$$R = \frac{805}{\theta} \qquad R = \frac{1430}{\theta}$$

$$h = R \cos\left(\sigma + \frac{\theta}{2}\right)$$

$$\frac{\ell + b}{h} = \tan\left(\sigma + \frac{\theta}{2}\right)$$

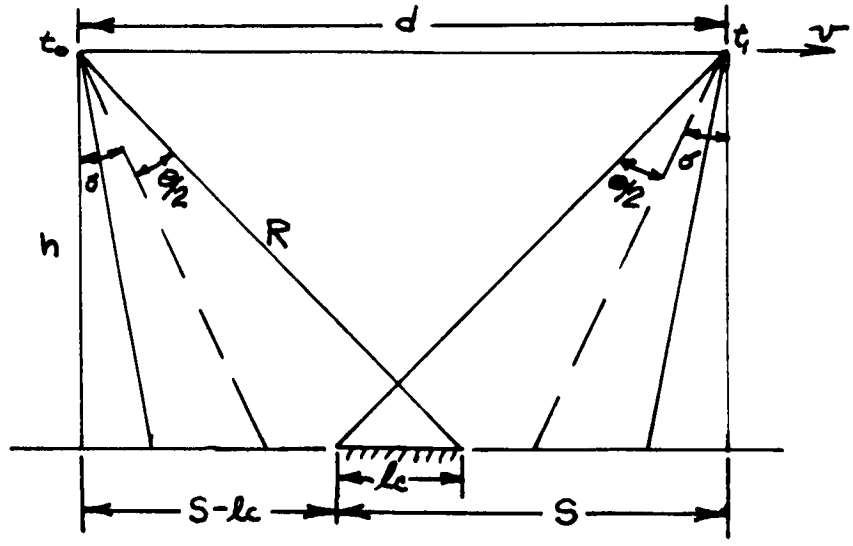
$$\frac{b}{h} = \tan\left(\sigma - \frac{\theta}{2}\right)$$

$$\ell + h \tan\left(\sigma - \frac{\theta}{2}\right) = h \tan\left(\sigma + \frac{\theta}{2}\right)$$

$$\ell = h\left[\tan\left(\sigma + \frac{\theta}{2}\right) - \tan\left(\sigma - \frac{\theta}{2}\right)\right]$$

$$\ell = R \cos\left(\sigma + \frac{\theta}{2}\right) \quad \tan\left(\sigma + \frac{\theta}{2}\right) - \tan\left(\sigma - \frac{\theta}{2}\right)$$

F) Viewing Time for Sweeping Antenna



d = distance traveled from beginning to end of sweep (km).

ℓ_c = ground path length (km).

v = velocity of spacecraft (km/sec).

$$d = vT = v(t_1 - t_0)$$

$$\cos\left(\sigma + \frac{\theta}{2}\right) = \frac{h}{R}$$

at t_0, t_1 $\sin\left(\sigma + \frac{\theta}{2}\right) = \frac{S}{R}$

$$d = 2S - \ell_c$$

$$= 2R \sin\left(\sigma + \frac{\theta}{2}\right) - \ell_c$$

$$T = \frac{2R \sin\left(\sigma + \frac{\theta}{2}\right) - \ell_c}{v}$$

REFERENCE: Memo 2-1590-00-049 and attachment dated May 19, 1966, E. J. Nicholson to the NASA, "Contract NASA 6216, Report of Meetings between NASA and Contractor Personnel Held 26-27 April 1966, Concerning Study Contract of Lunar Orbiter Adaptability to Experiments to Evaluate Lunar Surface Characteristics"

VHF ANTENNA COVERAGE FOR RELAY-TO-LUNAR ORBITER COMMUNICATION LINK

INTRODUCTION

The referenced correspondence establishes the requirement to determine a suitable Lunar Orbiter VHF antenna configuration to receive the Relay-to-Lunar Orbiter telemetry signals originating from the penetrometer(s) (Paragraph A-9 of the reference). The purpose of this discussion is to identify the spacecraft and relay communication characteristics and trajectories which influence the VHF antenna design. The Lunar Orbiter VHF antenna will provide for right-hand circular polarization and a nominal load impedance of 50 ohms with an allowable VSWR of 2:1 over the frequency range of 259.5 MHz to 260.5 MHz.

The discussion is based on the following assumptions:

1. The VHF antenna coverage must be sufficient to account for the position and time dispersions of the Relay and Lunar Orbiter spacecraft.

2. The VHF antenna pattern is omni-directional with a minimum of zero db gain over 90% of the radiation sphere.

3. The downlink transmission of telemetry to Earth via the high gain S-band antenna is in real-time with respect to Relay transmission.

4. The distance between the Relay and Lunar Orbiter spacecraft is 185 km or less during penetrometer impact.

5. The Lunar Orbiter spacecraft velocity in the vicinity of the penetrometer impact area is 2 km/second.

SUMMARY

The high power Relay transmitter (40 watts) allows the use of a 0.0 db gain VHF receiving antenna on the Lunar Orbiter spacecraft.

To minimize the Lunar Orbiter attitude constraints, it is desirable to design the receiving antenna for close to spherical (omni-directional) coverage. Due to the right-hand circular polarized signal transmitted from the Relay, it is necessary for the Lunar Orbiter receiving antenna to have right-hand circular polarization. Effective circular polarization may be obtained with a crossed dipole or canted turnstile configurations. A crossed dipole configuration (Figure 1) can be mounted on the existing Lunar Orbiter low gain antenna boom. The S-band and VHF frequencies are sufficiently different that no interference problems result. Each dipole would be approximately 12 inches long with the assembly, mounted about 40 inches above the existing Lunar Orbiter low gain S-band antenna. The additional weight would be less than 1 pound. A canted turnstile configuration (Figure 2) would consist of four rods mounted at angles of 45 degrees from the solar panel plane and 90 degrees

from each other in the normal axis. The rods would be stored during transit to the Moon and extended prior to experiment initiation. The turnstile weight would be less than 1 pound. Each rod would be less than 12 inches in length. Due to the short length of each rod, this configuration could also be mounted on the existing low gain S-band antenna boom. The possibility of mounting either of these configurations on the spacecraft proper requires investigation to insure that attitude and existing structure do not result in blocked view of the Relay at the time of penetrometer impact. To allow for the lack of perfect circular polarization with the above linear elements, a 3 db polarization loss has been included in the link calculations (Appendix A).

A potential problem area is the effect of the lunar ground reflected signal from the Relay on the direct Relay to Lunar Orbiter signal. No estimates are made in the Philco study. The margin (carrier) for Relay-CSM link is stated as +19.1 db in Reference 2, Table 6.1. This number includes -9.0 db tolerance for the CSM receiving antenna, and -10.1 db for other system tolerances. For the Lunar Orbiter VHF antenna, the negative tolerance is assumed to be 0.0 db. For the unknown effect of the ground reflected signal, -9.0 db is added giving a total negative tolerance (required margin) of 19.1 db.

The Relay path length (diameter of circle projected on the lunar surface by omni-antenna pattern) as a function of Lunar Orbiter altitude is shown in Figure 3.

REFERENCE: Memo 2-1590-00-049 plus Attachment, dated May 19, 1966; E. J.Nicholson to the NASA, "Contract NAS 6216, Report of Meetings between NASA and Contractor Personnel Held 26-27 April 1966, Concerning Study Contract of Lunar Orbiter Adaptability to Experiments to Evaluate Lunar Surface Characteristics"

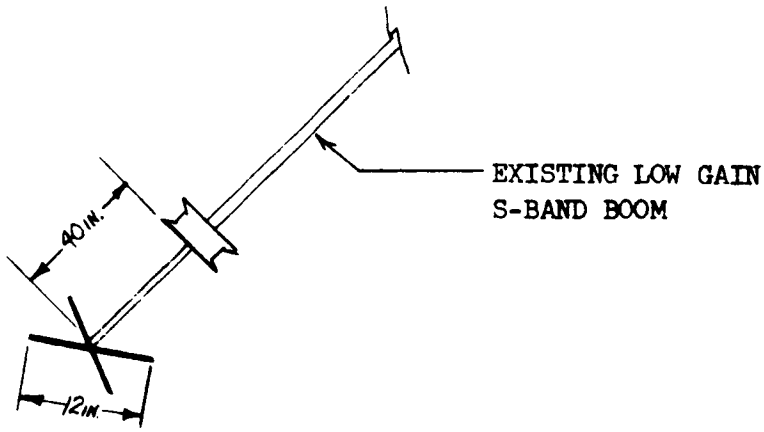


FIGURE 1 L.O. VHF RECEIVING ANTENNA-CROSSED DIPOLE

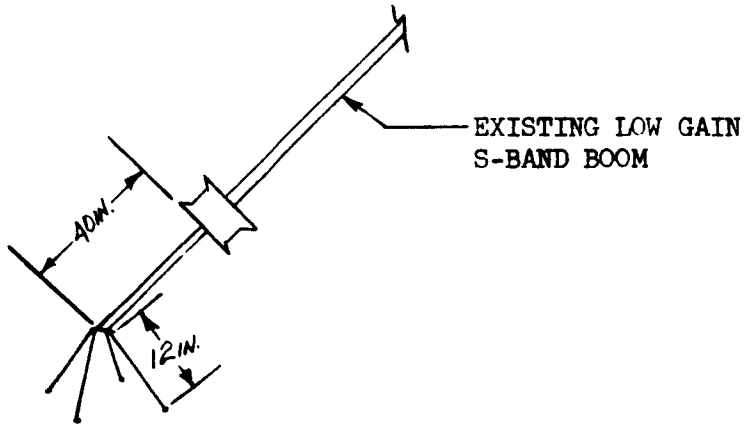


FIGURE 2 L.O. VHF RECEIVING ANTENNA-CANTED TURNSTILE

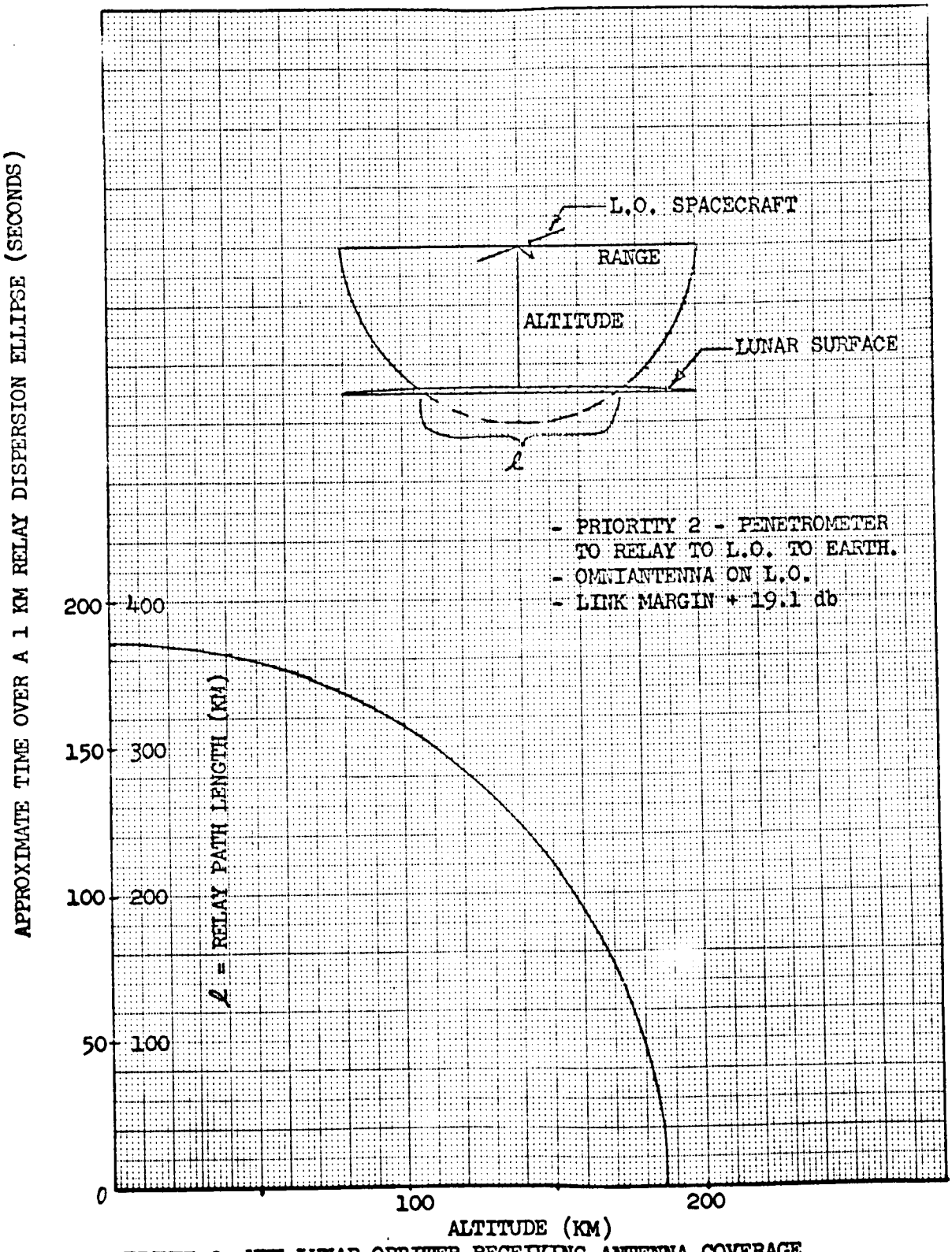


FIGURE 3 VHF LUNAR ORBITER RECEIVING ANTENNA COVERAGE

APPENDIX A. RELAY TO LUNAR ORBITER LINK ANALYSIS (f = 260 ± 0.5 MHz)

Transmitter Power (40 watts)	P_T	+	16. dbw
Transmission Circuit Loss	L_{TR}	-	1.0 db
Transmitting Antenna Gain	G_T		0.0 db
Polarization Loss	L_{POL}	-	3.0 db
Free Space Loss	L_{FS}	-	80.8 db -20 logR (R→Km)
Receiving Antenna Gain	G_R		G_R
Receiving Circuit Loss	L_{RC}	-	1.0 db
Received Signal Power	P_{REC}	-	69.8 -20 Log R + G_R (dbw)
Noise Spectral Density	Kt	-	204.0 dbw / Hz
Receiver Bandwidth (300 KHz)	B_{IF}	+	54.8 db
System Noise Figure	F	+	6.0 db
Signal-to-Noise (IF)	$(C/N)_{IF}$	+	9.0 db
Margin	M	+	19.1 db
Received Noise Power at Threshold	P_{THR}	-	115.1 dbw

FOR MAXIMUM RANGE $P_{REC} = P_{THR}$

$$-69.8 - 20 \text{ Log } R + G_R = -115.1$$

$$20 \text{ Log } R = 45.3 + G_R$$

$$\text{Log } R = 2.27 + 0.5 \text{ Log } g_r \quad (G_R = 10 \text{ Log } g_r)$$

$$R = 186 \sqrt{g_r}$$

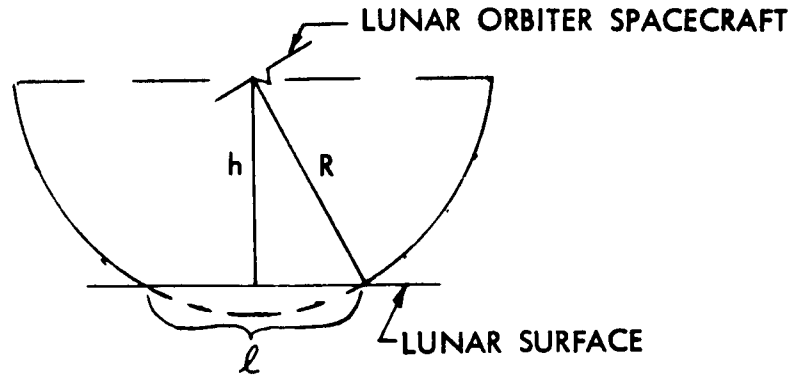
Example:

$$g_r = 1.0$$

$$G_R = 0.0 \text{ db}$$

$$R = 186 \text{ (KM)} \sim 100 \text{ Nautical Miles}$$

APPENDIX B. GEOMETRY CONSTRAINTS FOR VHF LUNAR ORBITER RECEIVING ANTENNA



R = Maximum Range = 186 Km

h = Spacecraft Altitude (Km)

ℓ = Relay Path Length

$$\ell = 2\,(R^2 - h^2)^{1/2}$$