3.2.2 (Continued)

and is very sensitive to small changes in the orbit period. Consequently, it is necessary to consider carefully perturbation effects on orbit period such as that associated with earth oblateness. The nodal period (including oblateness effects) for two different inclinations (35° and 45°) and altitudes at the equator ranging between 80 and 180 n.m. is shown in Figure 3.2-3.

A typical mapping pattern for an inclination of 35 and an altitude of 120 n.m., is shown in Figure 3.2-4. This pattern, however, is not very efficient in gathering information because of an overlap of about 50%. To reduce this redundant coverage, the information contained in Figures 3.2-5 (Percent Overlap vs. Altitude) and 3.2-6 (Mission Duration vs. Altitude) can be used to select orbit characteristics which provide for a better combination of inclination and altitude.

To illustrate this selection procedure, assume a design overlap of 10%. From Figure 3.2-5, orbital altitudes between 148.2 and 142.5 n.m. can be used with inclinations between 35° and 45°. The total mission time required to obtain full coverage is obtained from Figure 3.2-6 and ranges from 3 1/2 to 4 days. The actual choice of inclination should be based upon the desired latitude band to be mapped and consideration of propulsion requirements to obtain that inclination with an altitude between 148.2 and 142.5 n.m. With the inclination specified by this process, the altitude design point can be determined.

3.2.3 <u>Spacecraft Characteristics</u> - The Gemini with modification to a one man vehicle is quite suitable for the mission. As shown in Table 3.2-2 the launch weight with a 300 lb. mapping system payload and designed for a seven-day mission duration is 7200 lbs., including circularization propellant, which is well within GLV capability. Sufficient margin exists between launch weight and GLV capability, Figure 3.2-7, to provide versatility in selection of orbital altitude, inclination, or in payload, such as providing additional experimental packages.

ORBITAL PERIOD vs. ALTITUDE
(INCLUDING OBLATENESS EFFECTS)

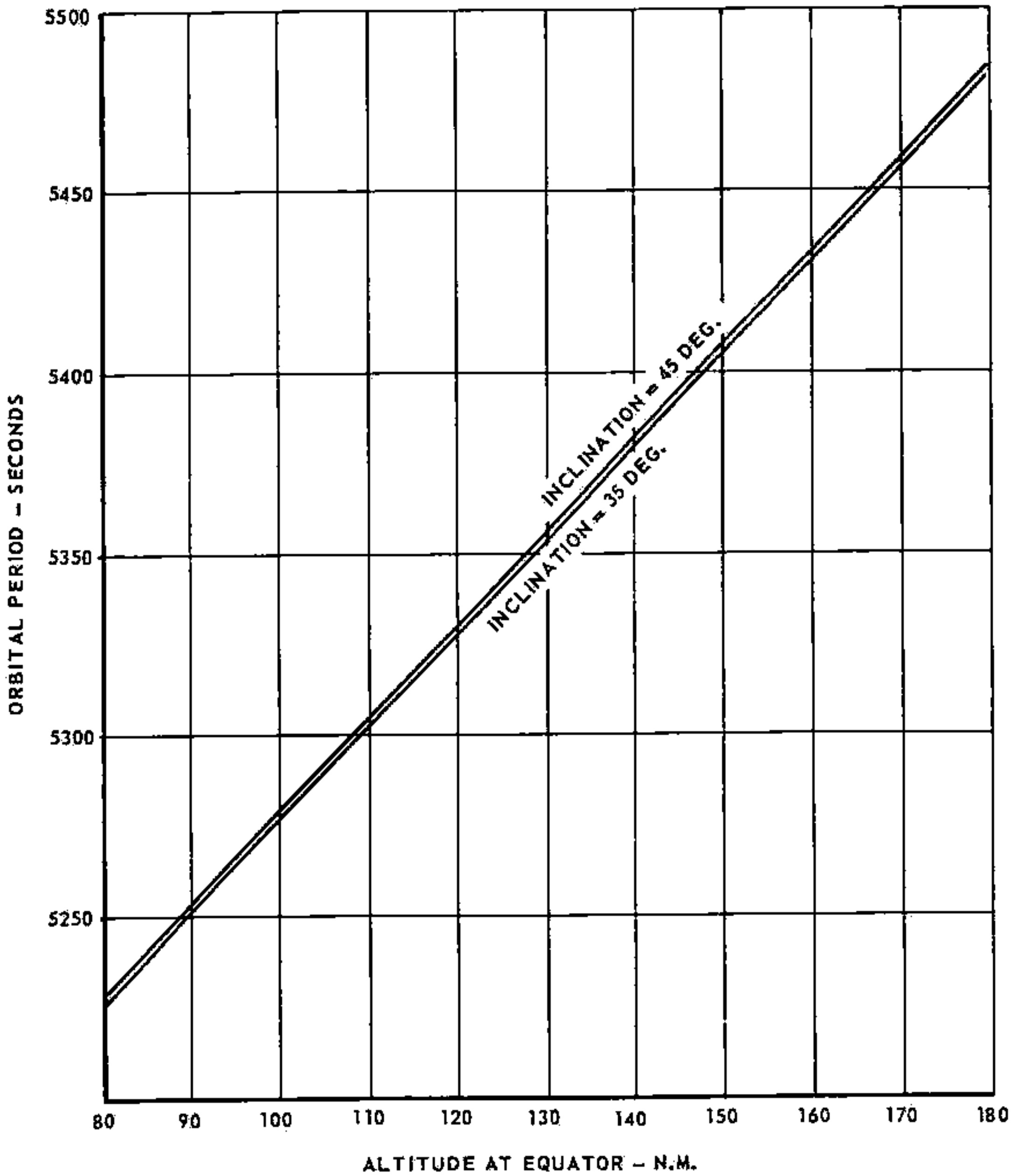


FIGURE 3.2-3

PERCENT OVERLAP vs ALTITUDE

CAMERA FIELD = ±45 DEGREE CROSSTRACK
ZERO DEGREE LATITUDE NO ORBIT DECAY

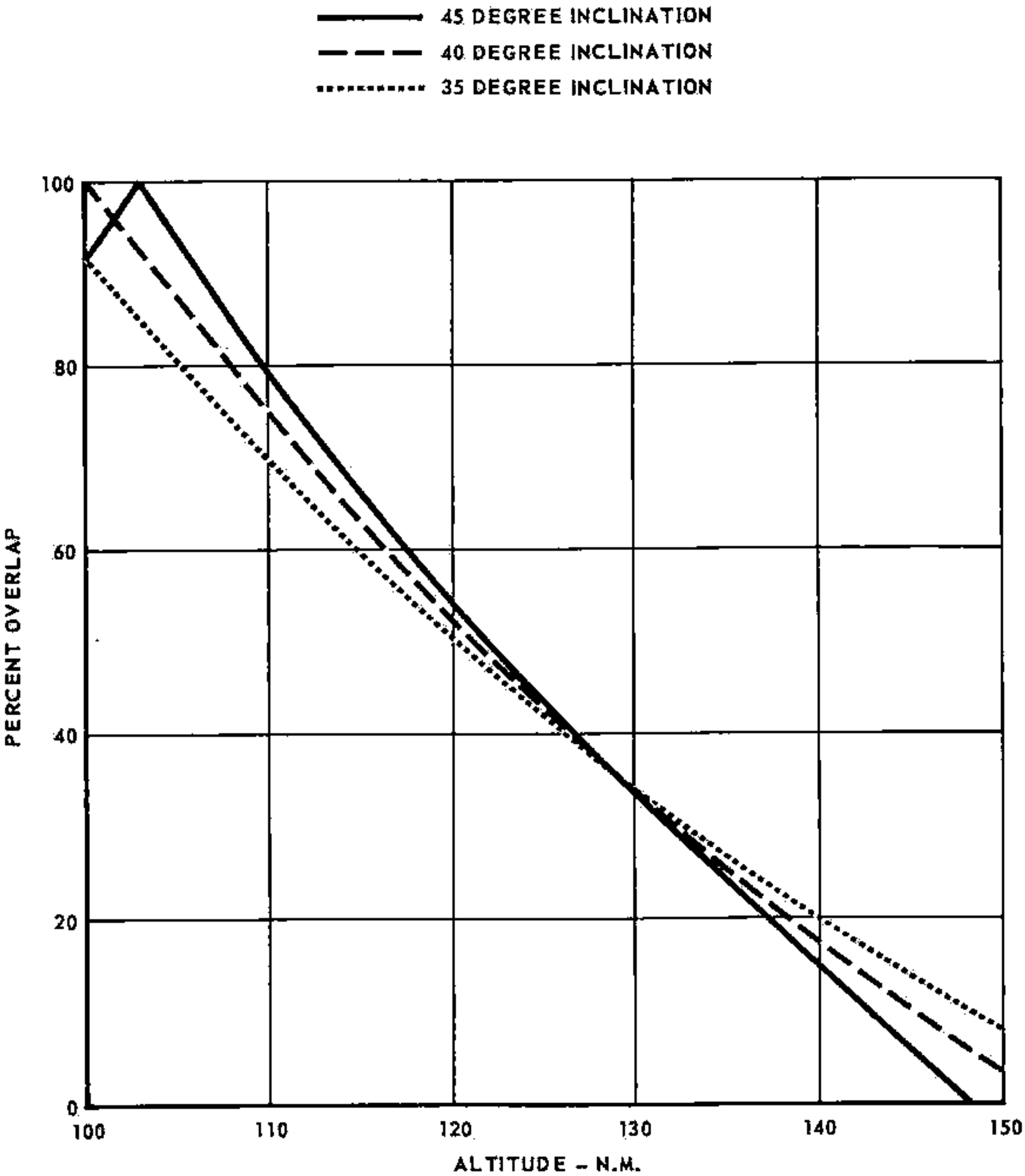


FIGURE 3.2-5

MISSION DURATION vs ALTITUDE

FULL COVERAGE AT ZERO DEGREE LATITUDE
NO ORBIT DECAY
CAMERA FIELD = ±45 DEGREE CROSSTRACK

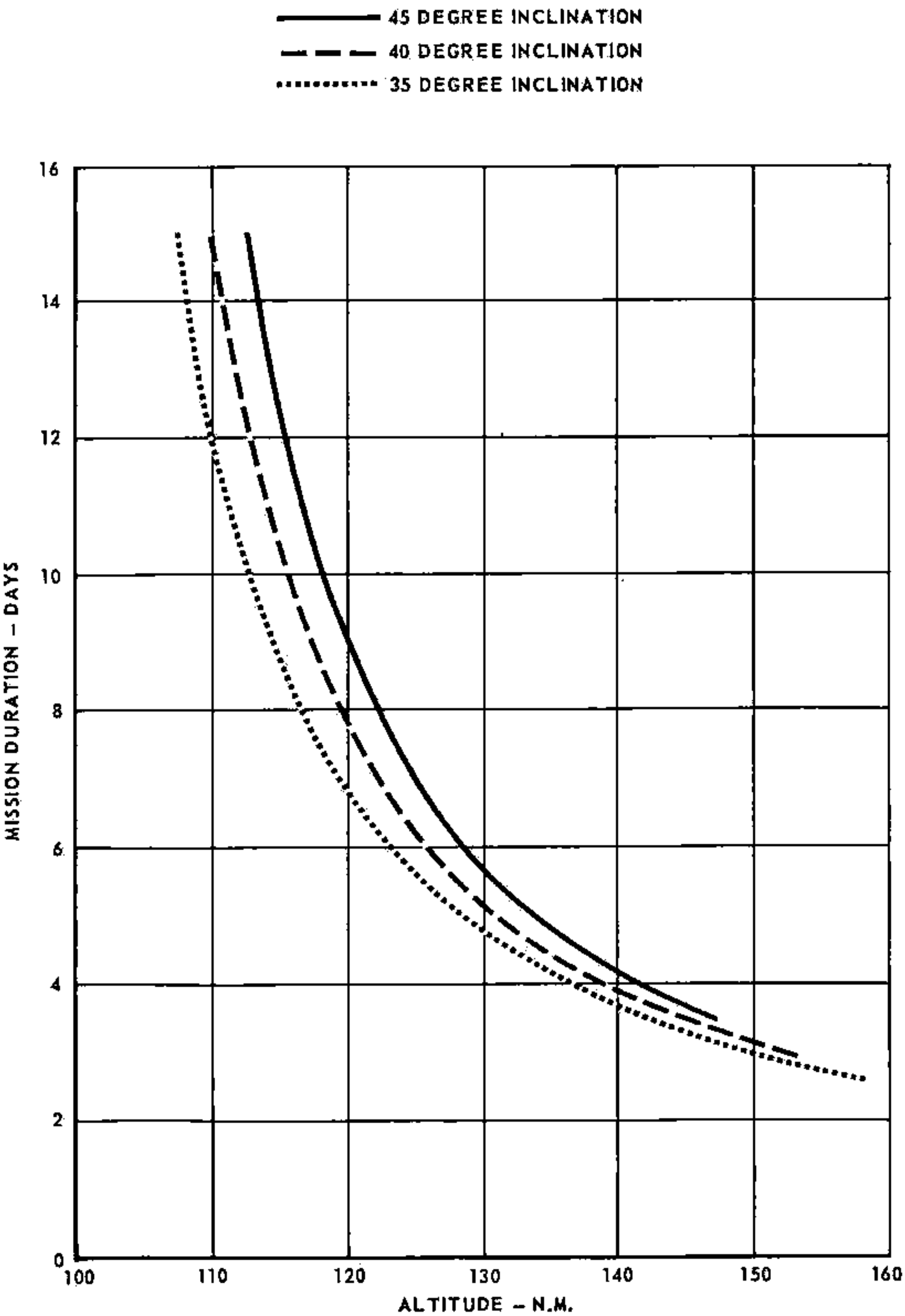


FIGURE 3.2-6

TABLE 3.2-2

WEIGHT STATEMENT

ONE MAN GEMINI EARTH SURFACE MAPPING
7 DAY MISSION

RE-ENTRY MODULE (1 MAN-5 DAYS)	4,169
Δ 2 DAYS FOOD	3
Δ 2 DAYS H_2O	32
BULKHEAD	37
CAMERA SYSTEMS MOUNTING	300
MOUNTING	70
	4,611
ADAPTER	2636
ADD: Δ 2 DAYS O_2	7
REMOVE: RSS PROPELLANT	-54
	2,589
TOTAL LAUNCH WT. =	7,200 LB.

GLV WEIGHT-IN-ORBIT CAPABILITY

INSERTION AT 87 N.M. PERIGEE
-3σ PERFORMANCE

NOTE: ONE-MAN GEMINI EARTH SURFACE MAPPING VEHICLE LAUNCH WEIGHT IS 7200 LB.

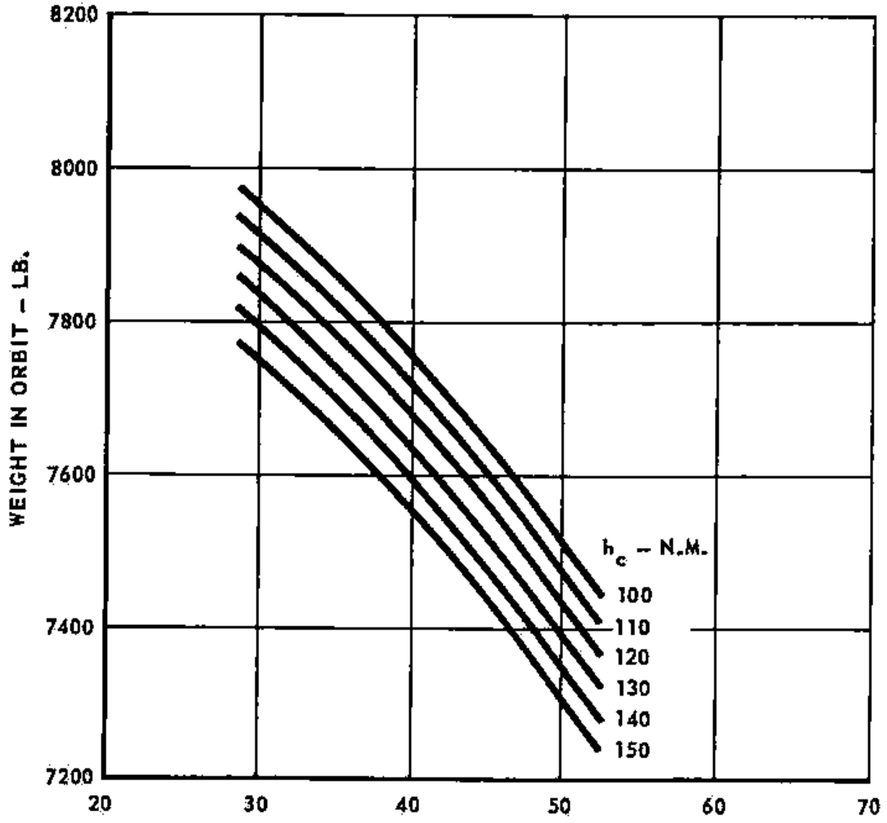


FIGURE 3.2-7

3.2.3 (Continued)

Two basic installation areas for the mapping system are shown in Table 3.2-3. The first allows installation of the two 13 inch focal length cameras in the R.H. crewman's compartment. However, a maximum reel diameter of 18 inches limits the film carried in a single load to that indicated in the illustrative system example. The second installation shown, wherein the camera system is mounted in the adapter, permits an increase in film load to a maximum reel diameter of 24 inches. This reel is retrievable by EVA. The spacecraft weight for the adapter mounted configuration has not been calculated in detail. However, it is estimated to be well within the capability of the GLV.

During mapping, attitude excursions in pitch and roll as sensed by a horizon sensor, are $\pm 1^{\circ}$ while attitude rate is maintained within 0.1°/second. Yaw attitude is sensed visually through the viewfinder and manually controlled to within $\pm 2^{\circ}$ and 0.1°/second.

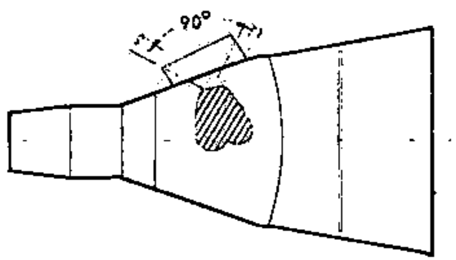
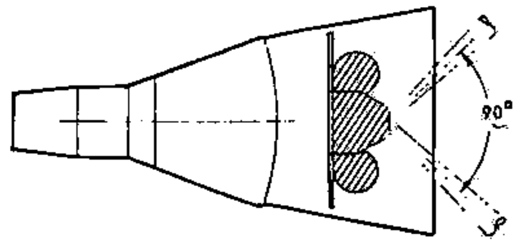
A horizon scanner mounted on the horizon camera head is utilized to maintain pitch and roll attitude. The orientation of the spacecraft during mapping operations precludes the use of the Gemini horizon sensors as presently mounted.

3.2.4 Additional Data - A lens data sheet for the proposed 13 inch panoramic camera design, reproduced with the permission of Itek Corporation, is given in Figure 3.2-2. The lens is in production as a member of a family of near diffraction limited designs. Of particular interest are the outstanding resolution and negligible distortion.

The theoretical resolution loss of a diffraction limited camera at f/3.5 due to linear image motion (rate) during the exposure is shown in Figure 3.2-8. At a 120 n.m. altitude, nadir V/h is approximately 2°/second. A 10% IMC error reduces system resolutions at the stated contrasts by a negligible amount.

TABLE 3.2–3

EARTH SURFACE MAPPING
ONE MAN GEMINI

		
MISSION DURATION (DAYS)	7	7
CAMERAS INSTALLED	13 IN. FOCAL LENGTH f/3.5 PANORAMIC (2) 52 MM FOCAL LENGTH f/3.5 COLOR (2) 10 IN. FOCAL LENGTH f/10 HORIZON RECORDING	13 IN. FOCAL LENGTH f/3.5 PANORAMIC (2) 52 MM FOCAL LENGTH f/3.5 COLOR (2) 10 IN. FOCAL LENGTH f/10 HORIZON RECORDING
ALTITUDE (N.M.)	120	120
VEHICLE WEIGHT (LB.)	7200	–
FILM CARRIED (FT.) (ALL 70 MM)	12,500 – (PANORAMIC CAMERAS) 850 – (COLOR CAMERAS) 1550 – (HORIZON CAMERA)	16,500 – (PANORAMIC CAMERAS) 1100 – (COLOR CAMERAS) 2050 – (HORIZON CAMERA)
CAPSULE MODIFICATIONS	REMOVAL OF RIGHT HAND SEAT REMOVAL OF RIGHT HAND WING INSTRUMENT PANEL ADDITION OF MISSION EQUIPMENT ADDITION OF PRESSURE WALL RIGHT HAND HATCH MODIFIED FOR AUTOMATIC CONTROL	REMOVAL OF RIGHT HAND SEAT MODIFIED PRESSURE BULKHEAD STRUCTURE
ADAPTER MODIFICATIONS	NONE	ADDITION OF MISSION EQUIPMENT EQUIPMENT RE-ARRANGED
MAJOR TECHNICAL PROBLEMS	NONE	NONE

SENSITIVITY TO IMAGE MOTION SMEAR
13 INCH PANORAMIC MAPPING CAMERA

- SO-206 FILM
- 1/1000 SECOND EXPOSURE DURATION
- 120 N.M. ALTITUDE

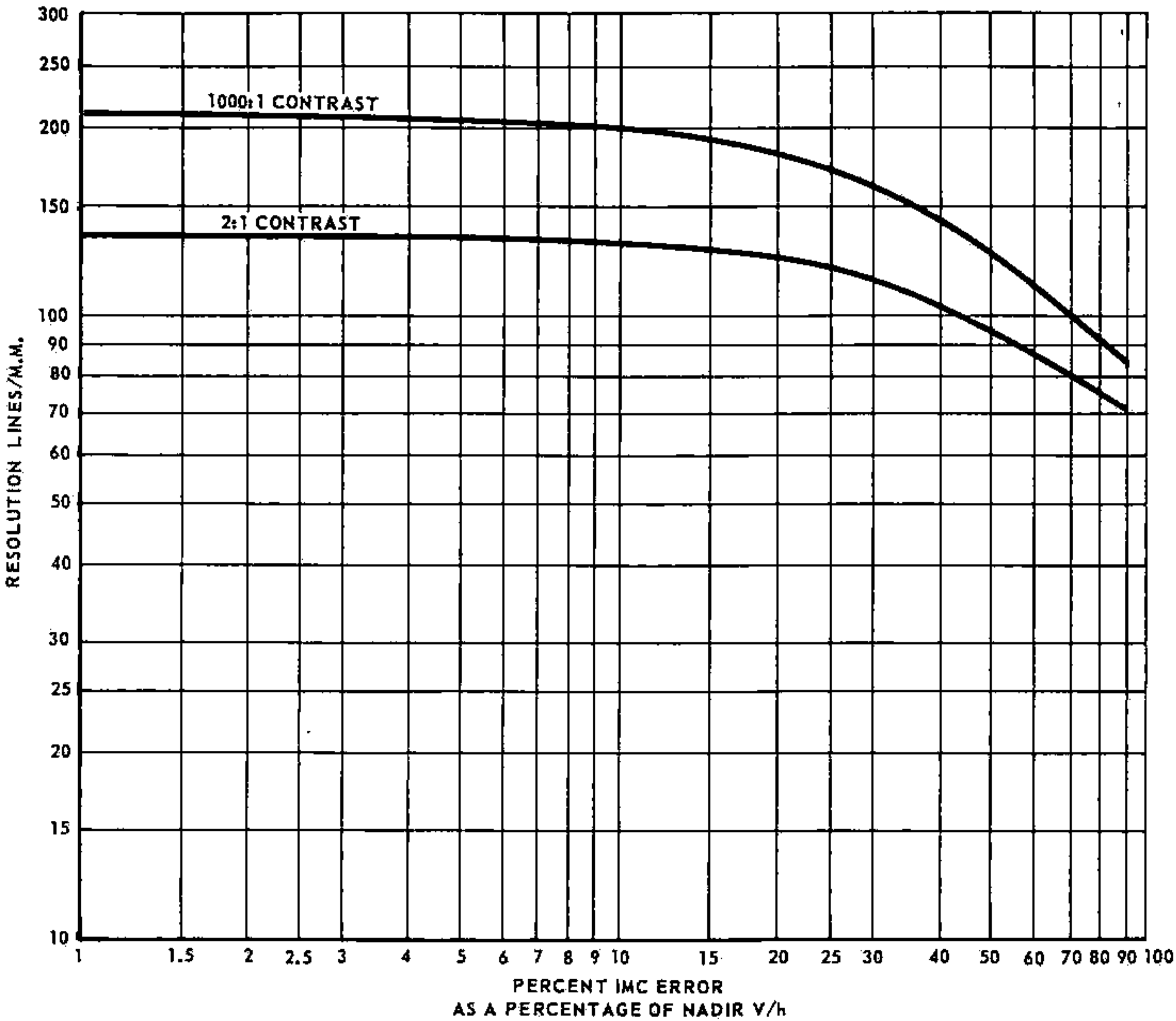


FIGURE 3.2-8

3.2.4 (Continued)

The earth area mapped as a function of altitude with the two stated film quantities and a constant 10% overlap, is shown in Figure 3.2-9.

A power system design summary if contained in Table 3.2-4. Two fuel cell sections are used because the requirements for the mission load are slightly greater than single cell capability. The summary is based on a camera running load duration of 4.5 hours.

EARTH AREA MAPPED AS A FUNCTION OF ALTITUDE

OVERLAP FORWARD AND LATERAL OF 10%

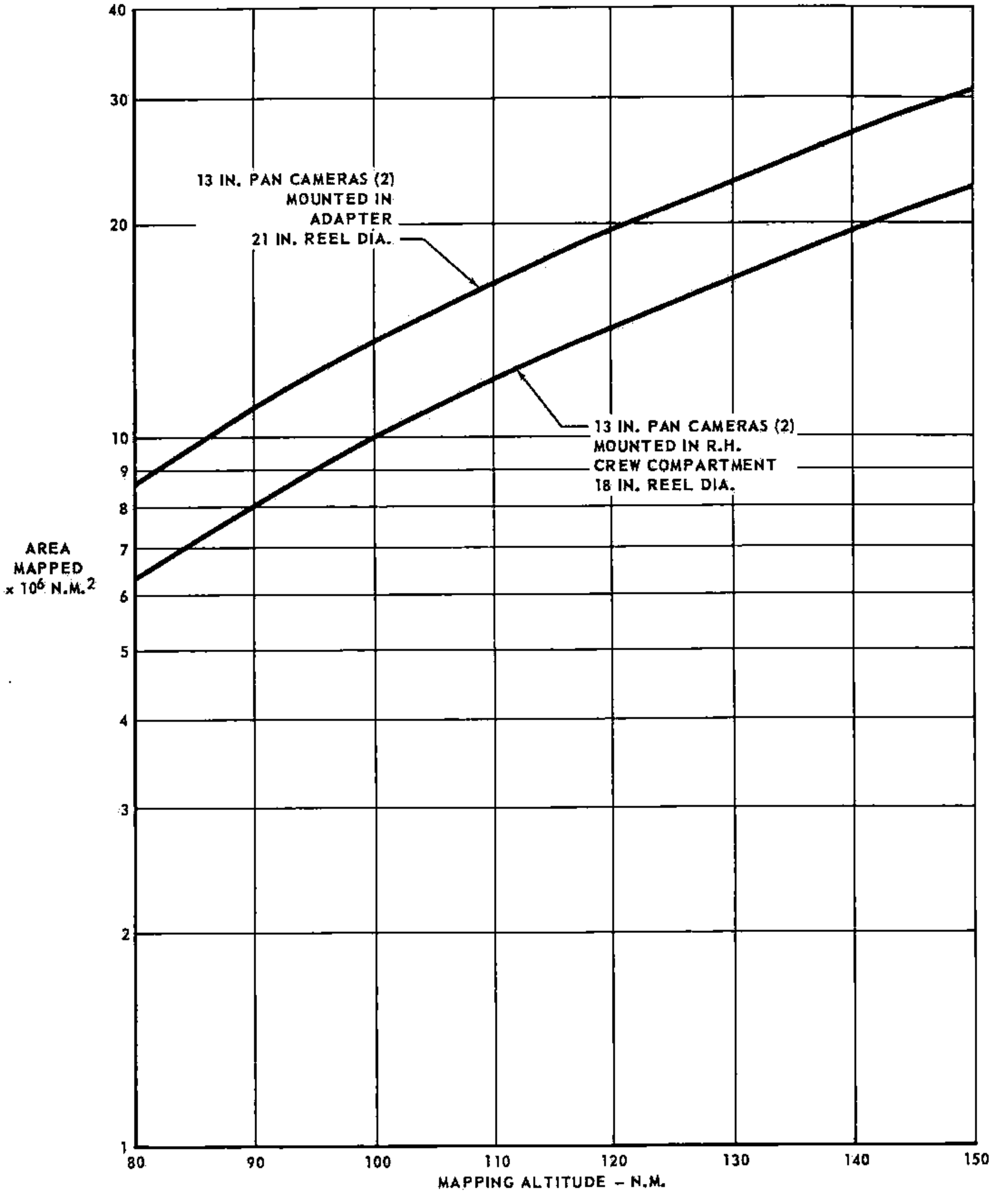


FIGURE 3.2-9

TABLE 3.2-4

ELECTRICAL POWER SYSTEM SUMMARY
ONE-MAN GEMINI EARTH SURFACE MAPPING

PEAK POWER SUMMARY	WATTS
GEMINI EQUIPMENT STEADY LOADS	382
GEMINI EQUIPMENT INTERMITTANT LOADS	387
MAPPING CAMERAS	250
PEAK LOAD POWER	1,019
DIODE AND DISTRIBUTION LOSS	81
PEAK FUEL CELL OUTPUT POWER	1,100 WATTS

7-DAY MISSION ENERGY SUMMARY	WATT-HOURS
GEMINI EQUIPMENT	73,000
MAPPING CAMERAS – 4.5 HR. USAGE (BASED ON 120 N.M.)	9,300
TOTAL LOAD ENERGY	82,300
DIODE AND DISTRIBUTION LOSSES	6,600
TOTAL MISSION FUEL CELL OUTPUT ENERGY	88,900 WATT-HOURS

7-DAY MISSION SUMMARY

FUEL CELL SECTIONS – GENERAL ELECTRIC: 2 SECTIONS
FUEL CELL REACTANTS – H_2 AND O_2: 110 LB.
REACTANTS TANKAGE: 14-DAY GEMINI TANKS
RE-ENTRY BATTERIES: FOUR – SAME AS 14-DAY GEMINI

3.3 One Man Gemini with Astronomical Telescope - The material in this section supports the choice of orbit altitude, attitude, telescope size, and spacecraft configuration for carrying out astronomical measurements and for developing systems and techniques for manned telescope operation.

3.3.1 Telescope Weight - The weight analysis which shows that 560 lbs. are available for a telescope in a one-man Gemini with a five day mission at 200 nautical miles is summarized in Tables 3.3-1 and 3.3-2. The GLV payload performance, shown in Figure 3.3-1, is used to determine the weight available for the telescope system at other altitudes, as shown in Figure 3.3-2. The use of a 32.5 degree inclination orbit instead of 28.5 degrees will reduce the payload by about 70 lbs.

The dependence of the telescope system weight on aperture diameter is shown in Figure 3.3-3. The range in weight for a fixed aperture diameter corresponds to the us of different spectrometers or accessories with the telescope. The maximum aperture di meter curve, Figure 3.3-4, was determined by using the minimum weight curve in Figure 3.3-3 and the available weight curve in Figure 3.3-2. For altitudes of 180 to 200 nau tical miles, which are used to obtain low aerodynamic torques to permit precise pointing, the maximum telescope aperture is approximately 26 inches.

3.3.2 Telescope Installation - The GLV launched one-man Gemini offers many possi bilities for installation of an astronomical telescope. The primary possibilities are: (1) a gimbaled 26 inch diameter telescope in the adapter for a mission with an orbit altitude of 180 to 200 na. mi., (2) a 16 inch diameter telescope (with pointing mirror) in the right hand side of the re-entry module for a mission with an orbit altitude of about 200 to 255 na. mi., and (3) a body fixed 40 inch diameter by 93.5 inch long telescope in an extended adapter for a mission with a 100 to 120 na. mi. altitude. The pointing stability of the telescope is greatly improved for the 180 na. mi. to 200 na. mi. altitude in comparison to the 100 na. mi.

TABLE 3.3-1

WEIGHT OF ONE-MAN GEMINI WITH ASTRONOMICAL TELESCOPE
ALTITUDE: 200 N.M.

	WEIGHT-LB.
RE-ENTRY MODULE MODIFICATIONS	(46)
ADD:	
HEAT SHIELD HATCH	36
HORIZON SENSOR	10
ADAPTER MODULE MODIFICATIONS	
ADD:	(898)
EXTENSION SECTION	67
TUNNEL	29
FUEL CELL (MINUS TOTAL FUEL REDUCTION)	4
RETROROCKET BEAM REDESIGN	5
TELESCOPE	560
TELESCOPE MOUNT	43
FINE ATTITUDE CONTROL SYSTEM	190
REMOVE	
EXCESS OAMS PROPELLANT	(-53)
TOTAL MODIFICATIONS	891
ONE-MAN GEMINI SPACECRAFT WITHOUT TELESCOPE	6,805
ONE-MAN GEMINI LAUNCH WEIGHT WITH TELESCOPE	7,696
OAMS FUEL FOR CIRCULARIZATION	-185
ONE-MAN GEMINI WEIGHT IN ORBIT	7,511
GLV CAPABILITY	7,570
MARGIN	59

TABLE 3.3-2

WEIGHT COMPARISON OF TWO-MAN AND ONE-MAN GEMINI WITHOUT TELESCOPE

	WEIGHT OF TWO-MAN GEMINI (14 DAYS) LB.	WEIGHT REMOVED OR ADDED LB.	WEIGHT OF ONE-MAN GEMINI (5 DAYS) LB.
RE-ENTRY MODULE			
STRUCTURE	1,473	(−22)	1,451
HATCH ACTUATOR		−22	
HEAT SHIELD	349		349
RE-ENTRY CONTROL SYSTEM	542		542
RETROGRADE SYSTEM	7		7
LANDING SYSTEM	213		213
INSTRUMENTATION AND NAVIGATION EQUIPMENT	130	(−12)	118
ATTITUDE DIRECTION INDICATOR		−8	
FLIGHT DIRECTION INDICATOR		−2	
MISCELLANEOUS		−2	
ELECTRICAL POWER	264		264
COMMUNICATIONS	62		62
ENVIRONMENTAL CONTROL	390	(−72)	318
SECONDARY O_2		−7	
SECONDARY O_2 PACKAGE		−20	
SECONDARY O_2 MOUNTS		−1	
LiOH		−44	
TELE-INSTRUMENTATION	191	(−3)	188
BIO-MED TAPE RECORDER		−3	
RECOVERY SYSTEM	30		30
RENDEZVOUS SYSTEM	27		27
CREW AND SURVIVAL	989	(−465)	524
CREWMAN AND SUIT		−190	
SEAT AND PYRO		−161	
CATAPULT		−28	
EGRESS KIT		−26	
SEAT BACK-UP		−23	
CIRCUITRY		−2	
FOOD		−30	
FOOD STORAGE		−5	
EXPERIMENTS	66	(−66)	
BALLAST ADJUSTMENT	18		18
WATER MANAGEMENT SYSTEM FROM ADAPTER		(+58)	58
TOTAL	4,751	−582	4,169
ADAPTER MODULE			
STRUCTURE	439		439
RETROGRADE SYSTEM	383		383
ELECTRICAL POWER SYSTEM	565		565
COMMUNICATIONS SYSTEM	20		20
ENVIRONMENTAL CONTROL SYSTEM	421	(−78)	343
PRIMARY O_2		−78	
TELE-INSTRUMENTATION	109		109
ORBIT ATTITUDE AND MANEUVER SYSTEM	777		777
CREW AND SURVIVAL	239	(−239)	
DRINKING WATER			
WATER TANK			
EXPERIMENTS	175	(−175)	
ADAPTER MODULE TOTAL	3,128	−492	2,636
RE-ENTRY MODULE TOTAL	4,751	−582	4,169
SPACECRAFT TOTAL	7,879	−1,074	6,805

GEMINI LAUNCH VEHICLE PERFORMANCE

- INSERTION AT 87 N.M. PERIGEE
- PERFORMANCE: -3σ
- ORBIT INCLINATION: 28.5 DEG.

NOTE: WEIGHT IN CIRCULAR ORBIT = ALLOWABLE SPACECRAFT WEIGHT AFTER CIRCULARIZATION.

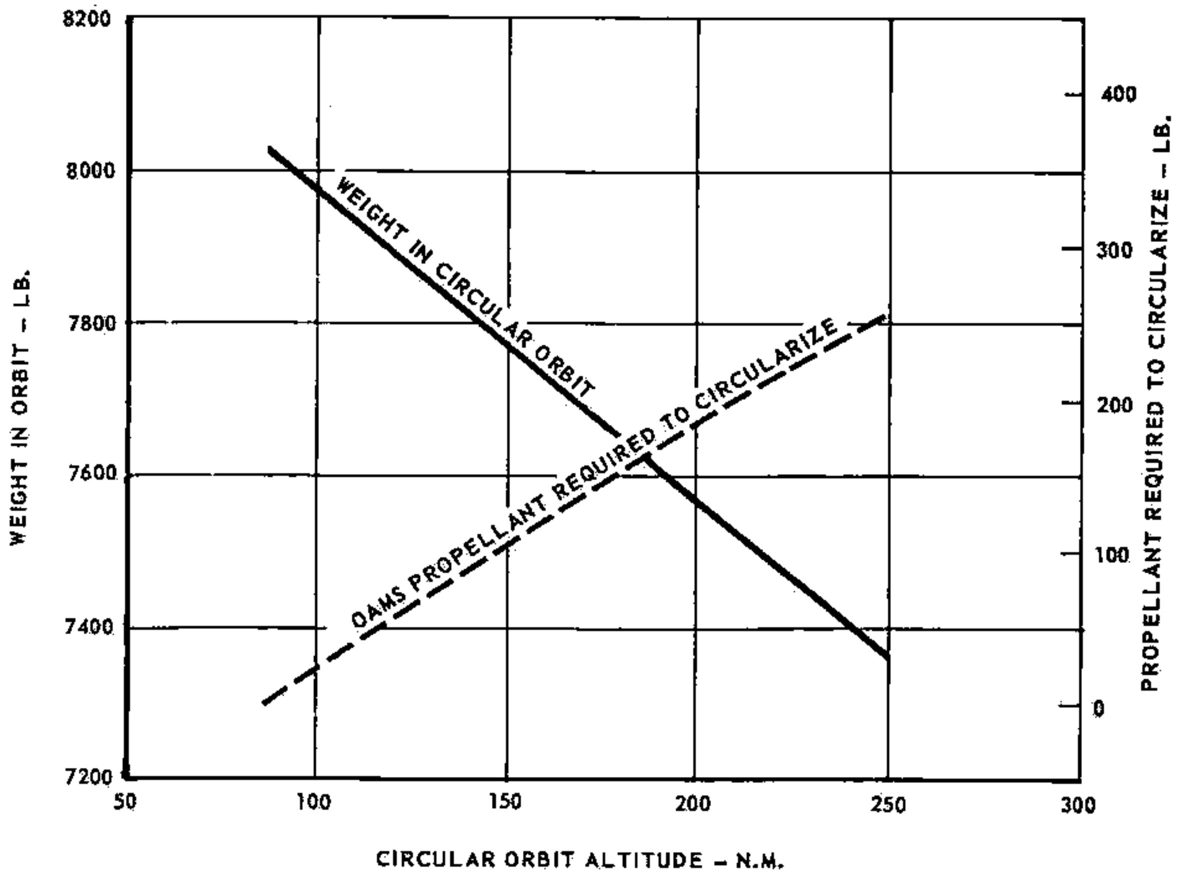


FIGURE 3.3-1

PAYLOAD WEIGHT AVAILABLE FOR THE ASTRONOMICAL TELESCOPE SYSTEM

THE ASTRONOMICAL TELESCOPE SYSTEM WEIGHT INCLUDES THE TELESCOPE STRUCTURE, OPTICS, AND INSTRUMENTATION BUT DOES NOT INCLUDE THE 190 LB. ALLOTTED TO THE FINE ATTITUDE CONTROL SYSTEM.

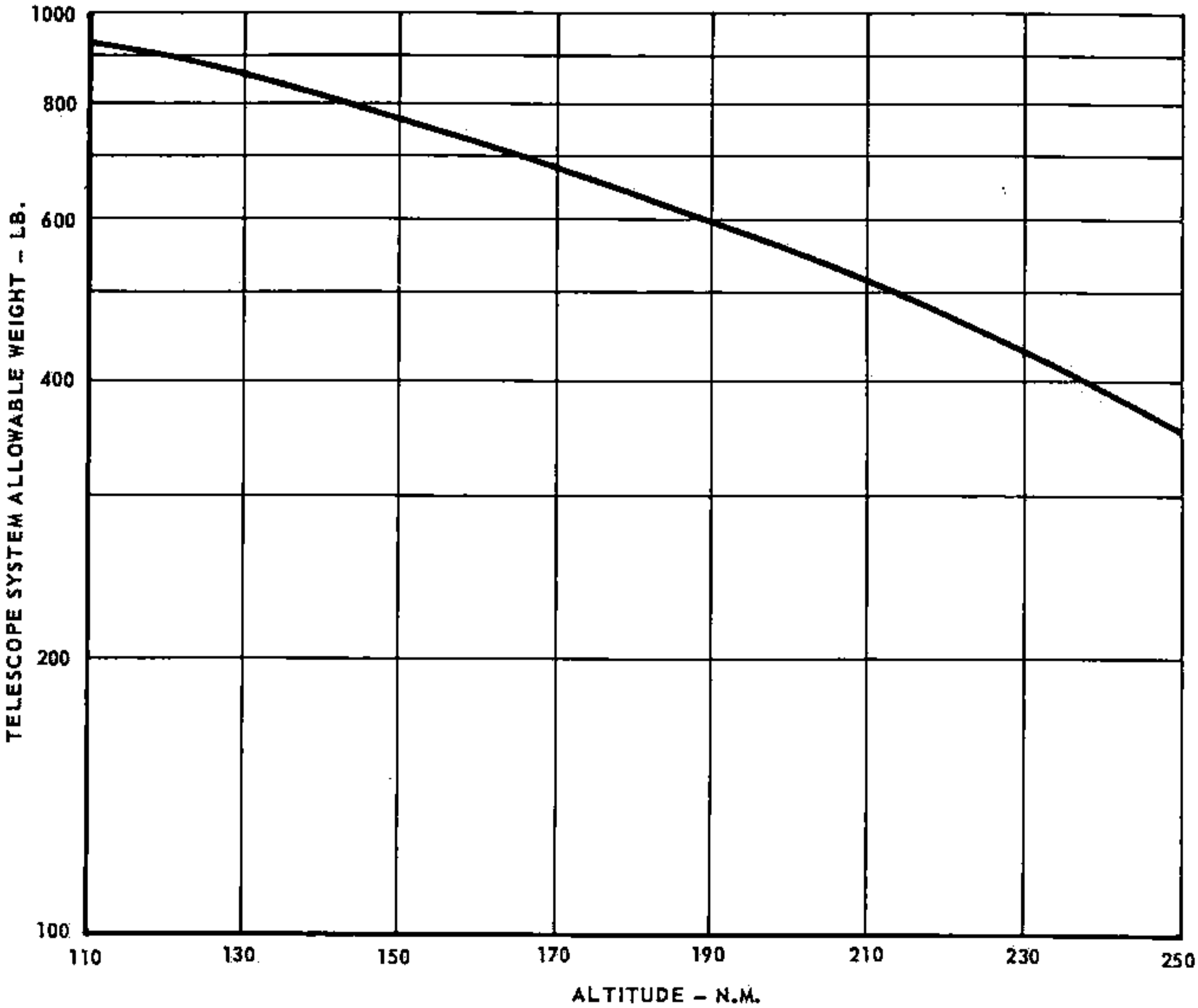


FIGURE 3.3-2

ASTRONOMICAL TELESCOPE WEIGHT

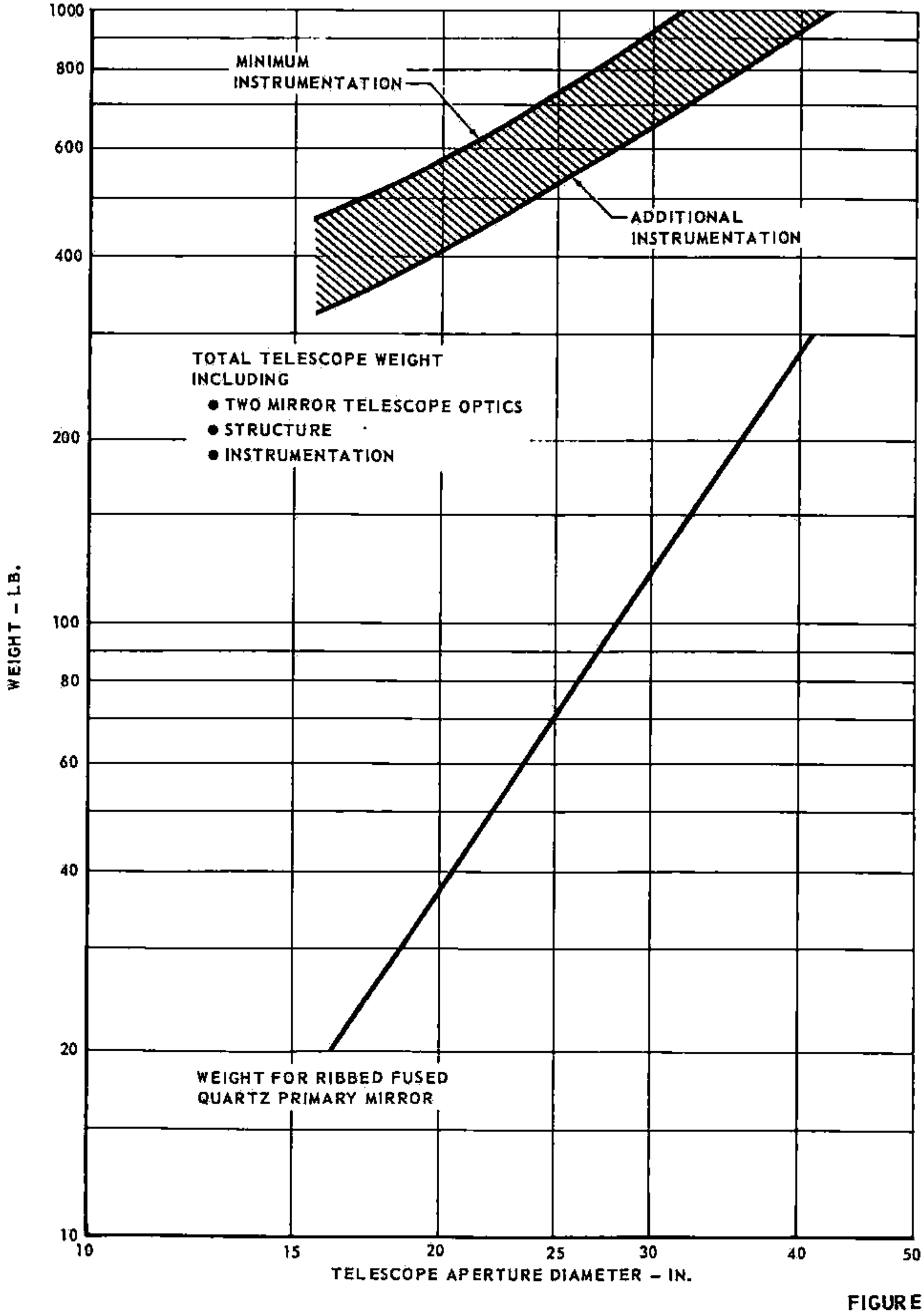


FIGURE 3.3-3

MAXIMUM APERTURE DIAMETER
OF THE TELESCOPE

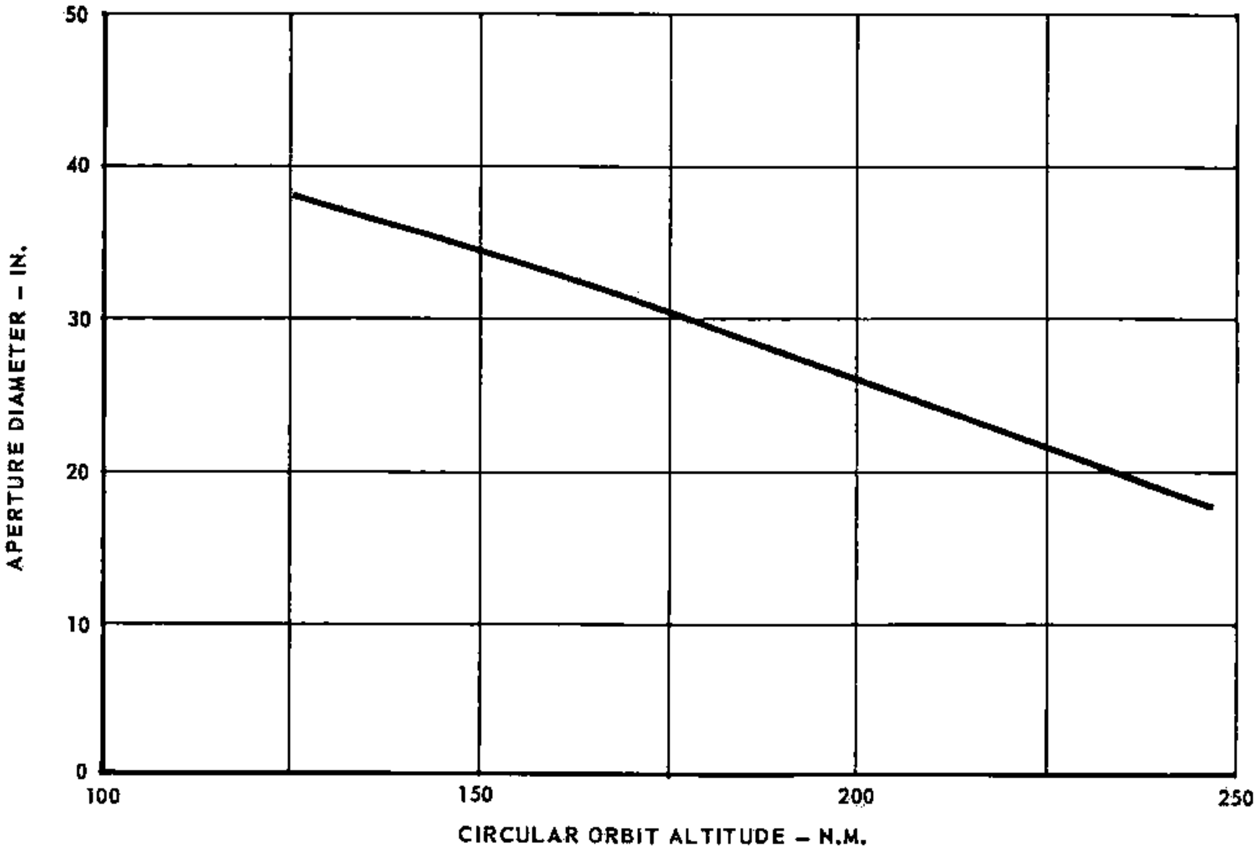


FIGURE 3.3-4

3.3.2 (Continued)

to 120 na. mi. altitude because of the smaller aerodynamic disturbance torques. The 40 inch telescope is thus limited to measurements not requiring precise pointing. The 26 inch diameter telescope has a better light gathering power and better resolution than the 16 inch telescope. However, the spacecraft modification required for the 26 inch telescope are more extensive. With an adequate schedule for development, the 26 inch diameter telescope is thus the choice for best performance. The installation of the 26 inch diameter telescope in the adapter is shown in Figure 3.3-5.

Five telescope installation approaches, the equipment limitations associated with each, and an indication of the degree of complication are shown in Figure 3.3-6. The heat shield hatch used for the first four approaches is included since it should be qualified by 1967. The telescope size limit is primarily due to the length. The telescope shown in Figure 3.3-6 is the OAO Goddard Experimental Package type.

The telesceope diameter can be increased in each case if an instrumentation section, shorter than the Goddard Experimental Package spectrometer, is used. The inclusion of a blow out door in the adapter side wall structure, which is necessary for two of the telescope installations, requires a re-routing of the radiator tubes.

A telescope with an aperture diameter of 12 to 16 inches can be mounted in the right hand side of the re-entry module, as shown in Figures 3.3-6 and 3.3-7. A bulkhead is added so that the right hand hatch can be opened without depressurization of the left hand astronaut section. The space available in the right hand side of the re-entry module, for telescope installations other than that shown in

ONE MAN GEMINI WITH ASTRONOMICAL TELESCOPE

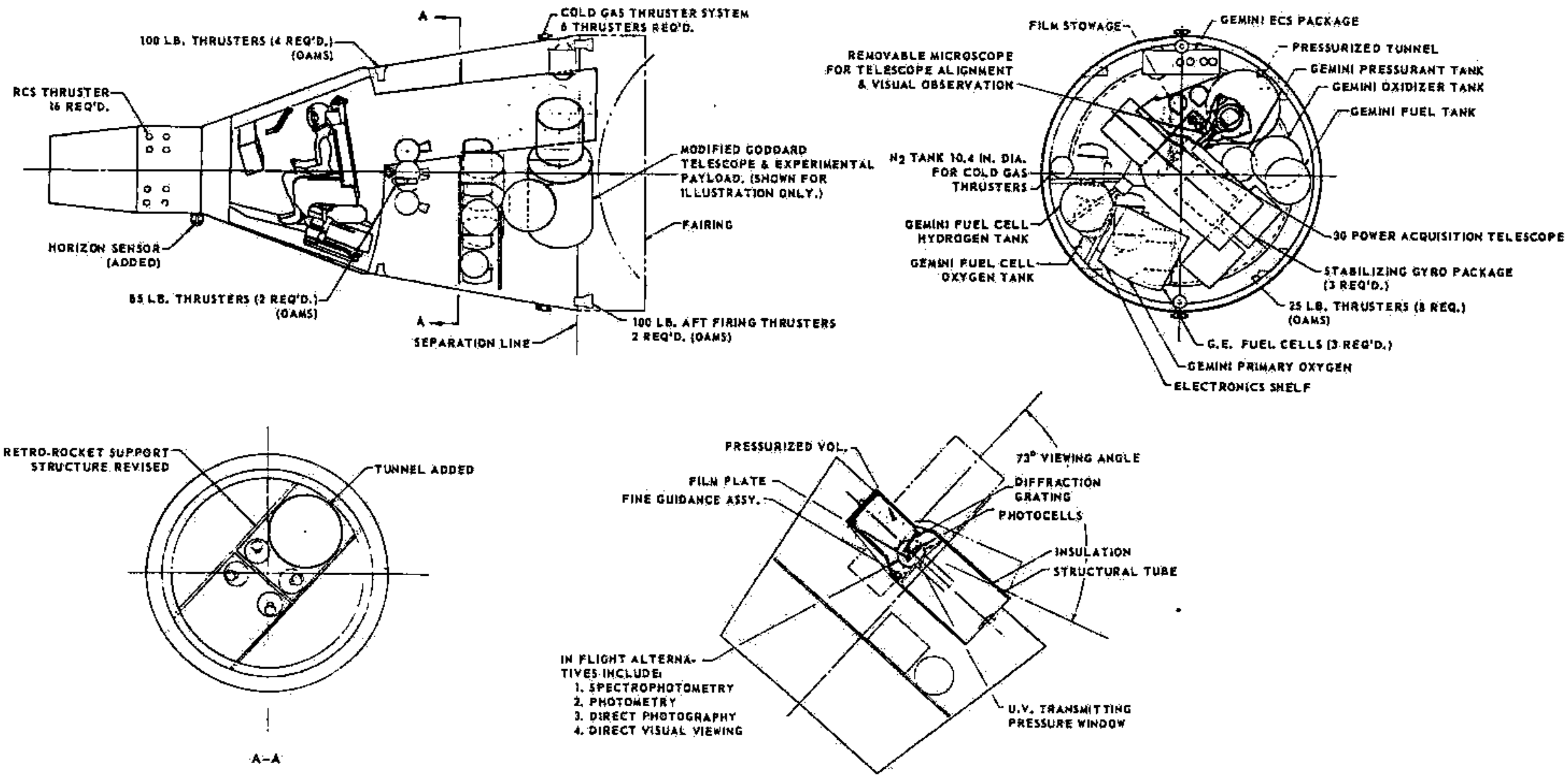


FIGURE 3.3-5

ONE MAN GEMINI WITH ASTRONOMICAL TELESCOPE

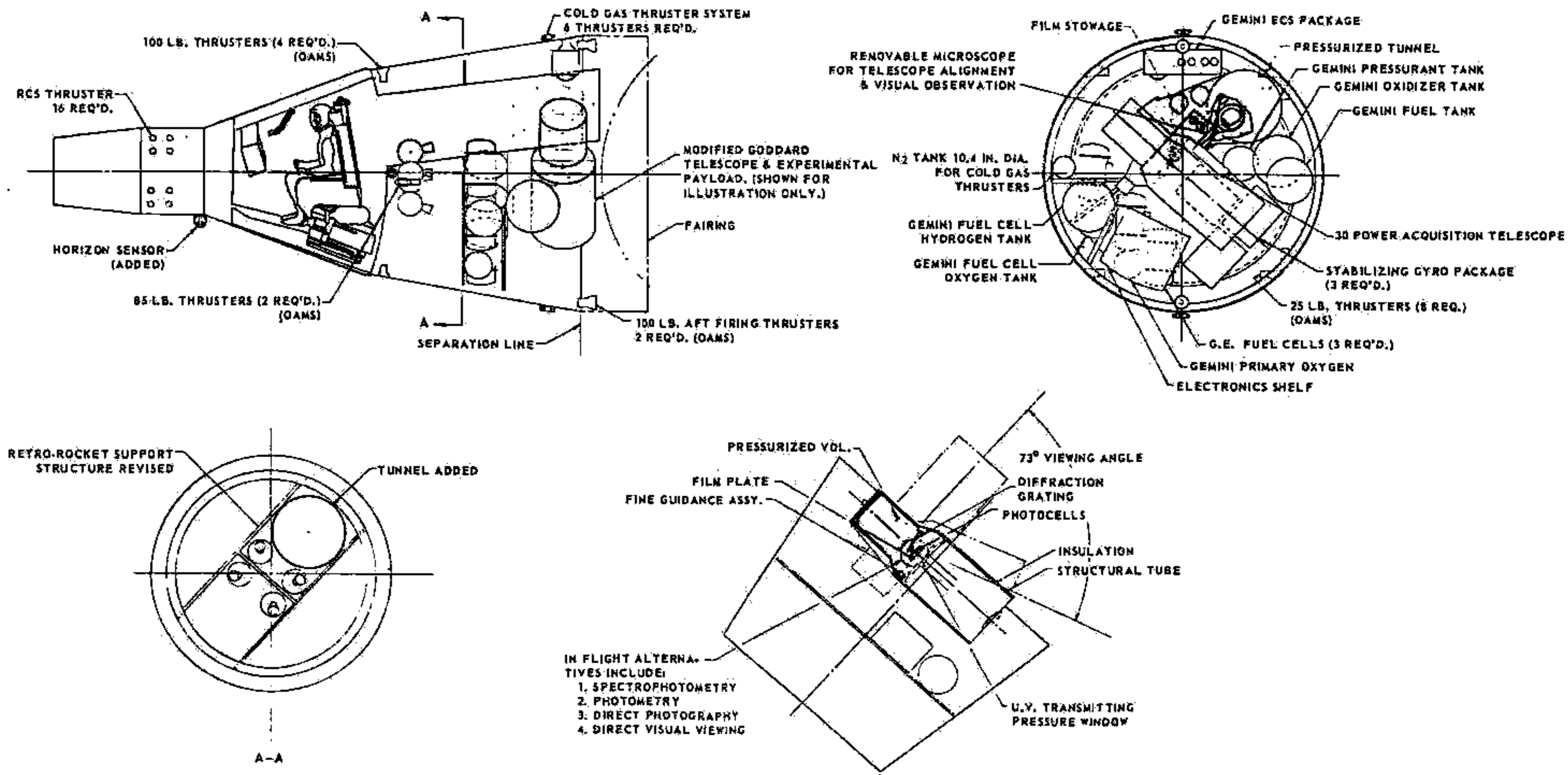
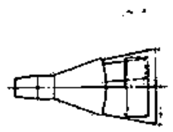
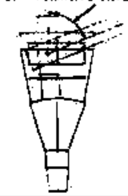
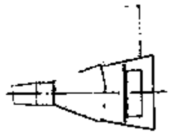
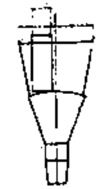
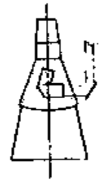


FIGURE 3.3-5

ASTRONOMICAL TELESCOPE INSTALLATION ALTERNATIVES

ONE MAN GEMINI WITH ASTRONOMICAL TELESCOPE		67° VIEWING ANGLE 			
DESCRIPTION OF TELESCOPE INSTALLATION	SIDE LOOKING BODY FIXED 8 IN. SKIRT ADDED OR 14 IN. SKIRT ADDED**	END POINTING 28 IN. SKIRT ADDED VARIABLE ANGLE	BODY FIXED SKIRT NOT ADDED	END LOOKING-END FIXED 28 IN. SKIRT ADDED	IN RE-ENTRY VEHICLE
ACCESS TO TELESCOPE	HATCH AND STUB TUNNEL	HATCH AND STUB TUNNEL	HATCH AND STUB TUNNEL	HATCH AND STUB TUNNEL	ADJACENT TO CREWMAN
DURATION (DAYS) MAXIMUM	5	5	5	5	5
EQUIPMENT – VOLUME LIMIT	34 DIA. x 94.5 OR 40 DIA. x 94.5**	34 DIA. x 94.5 26 DIA. x 73.5***	26 DIA. x 73.5	32 DIA. x 89*	18 IN. DIA. – 1/4
HATCH IN HEAT SHIELD	YES	YES	YES	YES	NO
TELESCOPE POINTING	MOVE AND HOLD ENTIRE SPACECRAFT	SPACECRAFT IN ROLL ONLY TELESCOPE IN PITCH	MOVE AND HOLD ENTIRE SPACECRAFT	MOVE AND HOLD ENTIRE SPACECRAFT	MIRROR
ADDITIONS	SKIRT AND BLOW OUT DOOR	SKIRT	BLOW OUT DOOR	SKIRT	AUTOMATIC HATCH
REARRANGEMENT OF ADAPTER EQUIPMENT	EXTENSIVE	EXTENSIVE	EXTENSIVE	MODERATE	NONE
GENERAL COMMENTS ON INSTALLATION AND OPERATION	SIMPLE INSTALLATION	HAS PROBLEM OF TUNNEL TO TELESCOPE CONNEC-TION	MINIMUM CHANGE	SIMPLE INSTALLATION	MINIMUM CHANGE LOWEST COST SMALLEST TELESCOPE

*BASED ON INTERFERENCE WITH RETROGRADE SUPPORTS
**WITH SHORT SPECTROMETER
***WEIGHT LIMITED DUE TO ALTITUDE REQUIREMENT FOR PRECISE POINTING & GLV LAUNCH

FIGURE 3.3-6

RE-ENTRY VEHICLE TELESCOPE INSTALLATION

- WIDE FIELD OF VIEW PHOTOGRAPHY
- DIRECT VISUAL VIEWING
- VISUAL SPECTRUM PHOTOMETRY

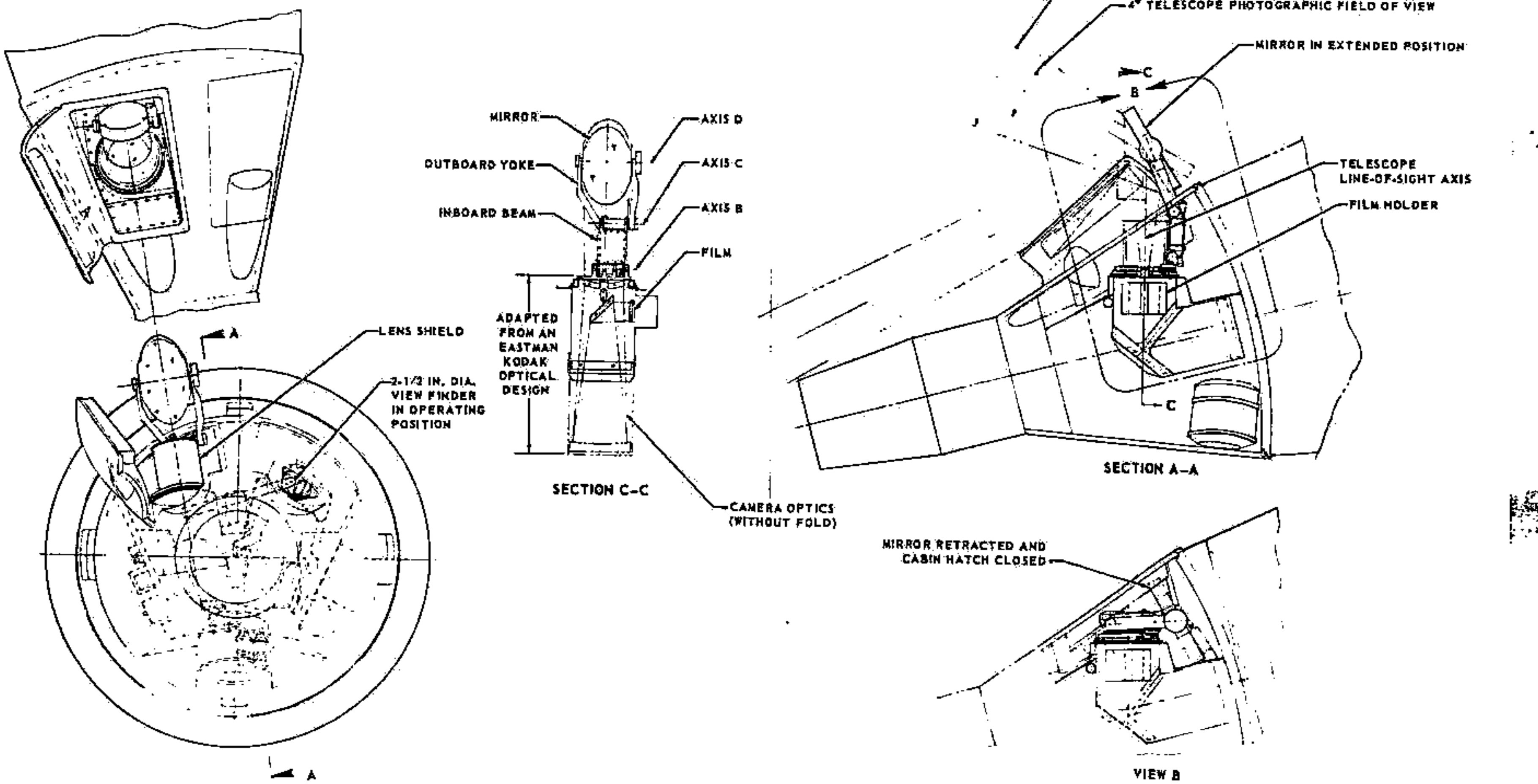


FIGURE 3.3-7

3.3.2 (Continued)

Figure 3.3-7 is shown in Figure 3.3-8.

3.3.3 Attitude Control System - The attitude control system consists of the Gemini OAMS attitude control for slewing to acquire stars, control gyros or inertial wheels for fine attitude control, a pointing angle error detector in the astronomical telescope, and a nitrogen cold gas momentum desaturation system. The purpose of the fine attitude control system is to precisely point the telescope during photographic exposures or photometric measurements. It is necessary to keep the disturbance torques on the spacecraft low and the control gain (ft lbs. per arc second pointing error) high for precise pointing.

3.3.3.1 Disturbance Torques - The primary disturbance torques on the spacecraft shown in Table 3.3-3.

TABLE 3.3-3

DISTURBANCE TORQUES

GRAVITY GRADIENT TORQUE	
MAXIMUM	6.9×10^{-3} FT. LBS.
VEHICLE ONE DEGREE FROM ZERO TORQUE POSITION	2×10^{-4} FT. LBS.
GAS LEAKAGE	6×10^{-5} FT. LBS.
AERODYNAMIC TORQUE	
100 NA.MI. ALTITUDE	2.5×10^{-2} FT. LBS.
200 NA.MI. ALTITUDE	8×10^{-4} FT. LBS.
(SEPARATION OF CENTER OF PRESSURE AND CENTER OF GRAVITY –0.5 FT.)	
MAN BREATHING	6×10^{-4} FT. LBS.
HEART BEAT	6×10^{-4} FT. LBS.

NOTE:
THE LAST TWO FACTORS ASSUME THE ASTRONAUT DISTURBANCES ARE ATTENUATED BY A FACTOR OF TWO.

With a circular orbit altitude of 200 na. mi. and an attitude with the spacecraft longitudinal inertial axis within one degree of the normal to the orbit

CLEARANCE ENVELOPE
ONE-MAN GEMINI

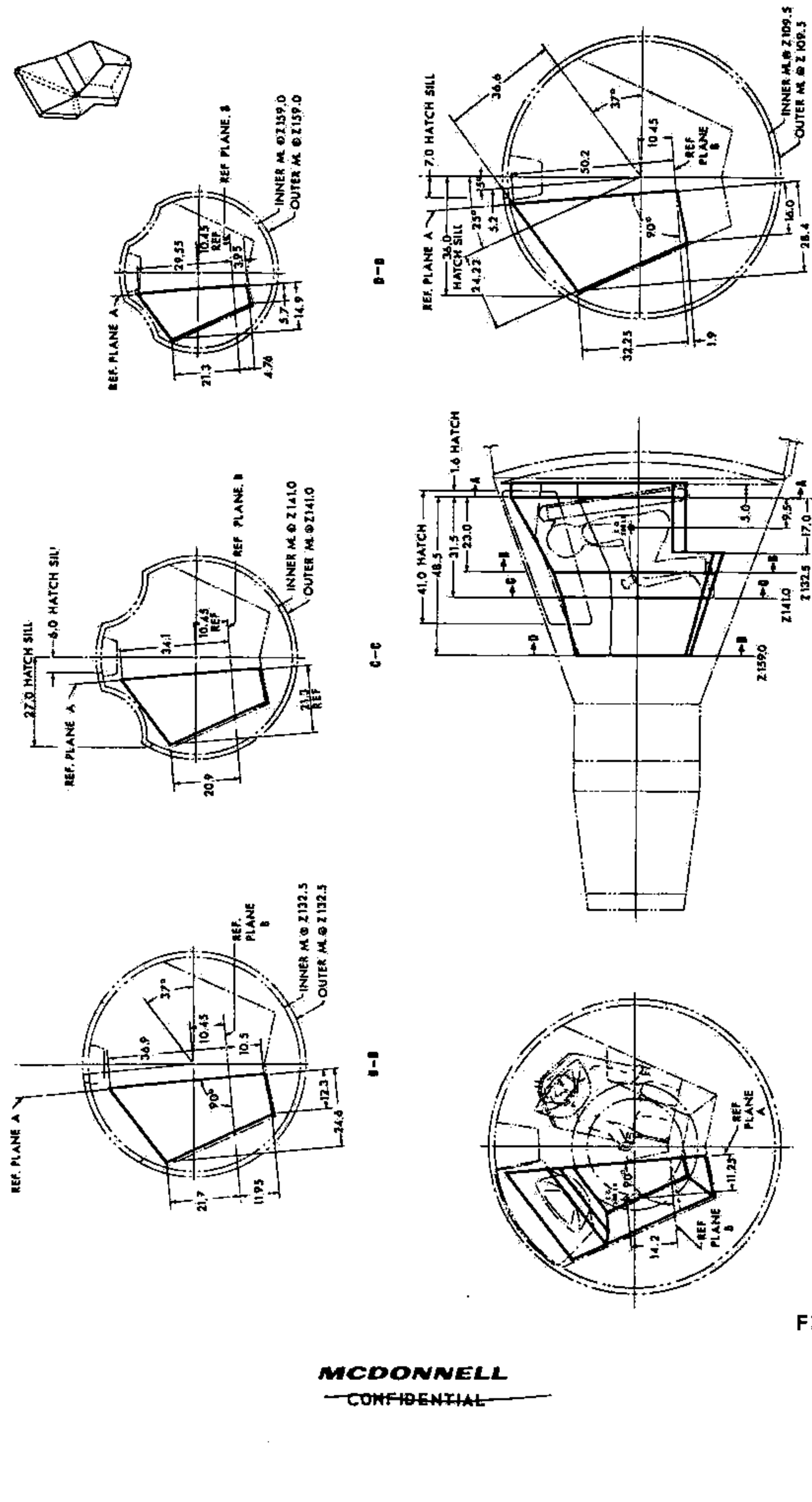


FIGURE 3.3-8

3.3.3.1 (Continued)

plane, the torques from the disturbances are each less than 10^{-3} ft.-lbs. An altitude much higher than 200 na. mi. is undesirable for a GLV-launched one-man Gemini since the size telescope which can be carried decreases greatly, as discussed in Section 3.3.1.

The gravity gradient torques are periodic with a period of half the orbit period, while the aerodynamic torques have a period equal to the orbital period. The disturbance torques along each spacecraft axis for the fixed roll attitude suitable for stellar measurements are shown in Figures 3.3-9 and 3.3-10, while the peak torques at other attitudes and altitudes are given in Figures 3.3-11 and 3.3-12.

A long-time, constant attitude control system can be used to compensate for pointing angle errors due to these slowly varying torques. The largest disturbance, aerodynamic torque, is shown in Figure 3.3-12. The separation of the center of gravity and center of pressure for the spacecraft with a 26 inch diameter telescope is less than three inches when the astronaut is in the viewing position at the telescope. The nighttime aerodynamic torques are thus approximated by the 3 inch center of gravity-center of pressure line in Figure 3.3-12. The variation in the peak gravity gradient torque with spacecraft pitch or yaw attitude misalignment, shown in Figure 3.3-11, demonstrates the advantage of alignment within 0.5 to 1 degree. The addition of the third horizon scanner, 90 degrees from the other two, permits an initial alignment and periodic adjustments with a 0.5 degree accuracy. The disturbance torque corresponding to a 0.5 degree misaligmment is 1.2×10^{-4} ft. lbs.

The attitude disturbances due to astronaut arm motions can be quite large. A typical arm motion has two 0.16 ft-lbs sec. impulses of opposite sign separated by approximately one second. The corresponding spacecraft attitude disturbance is

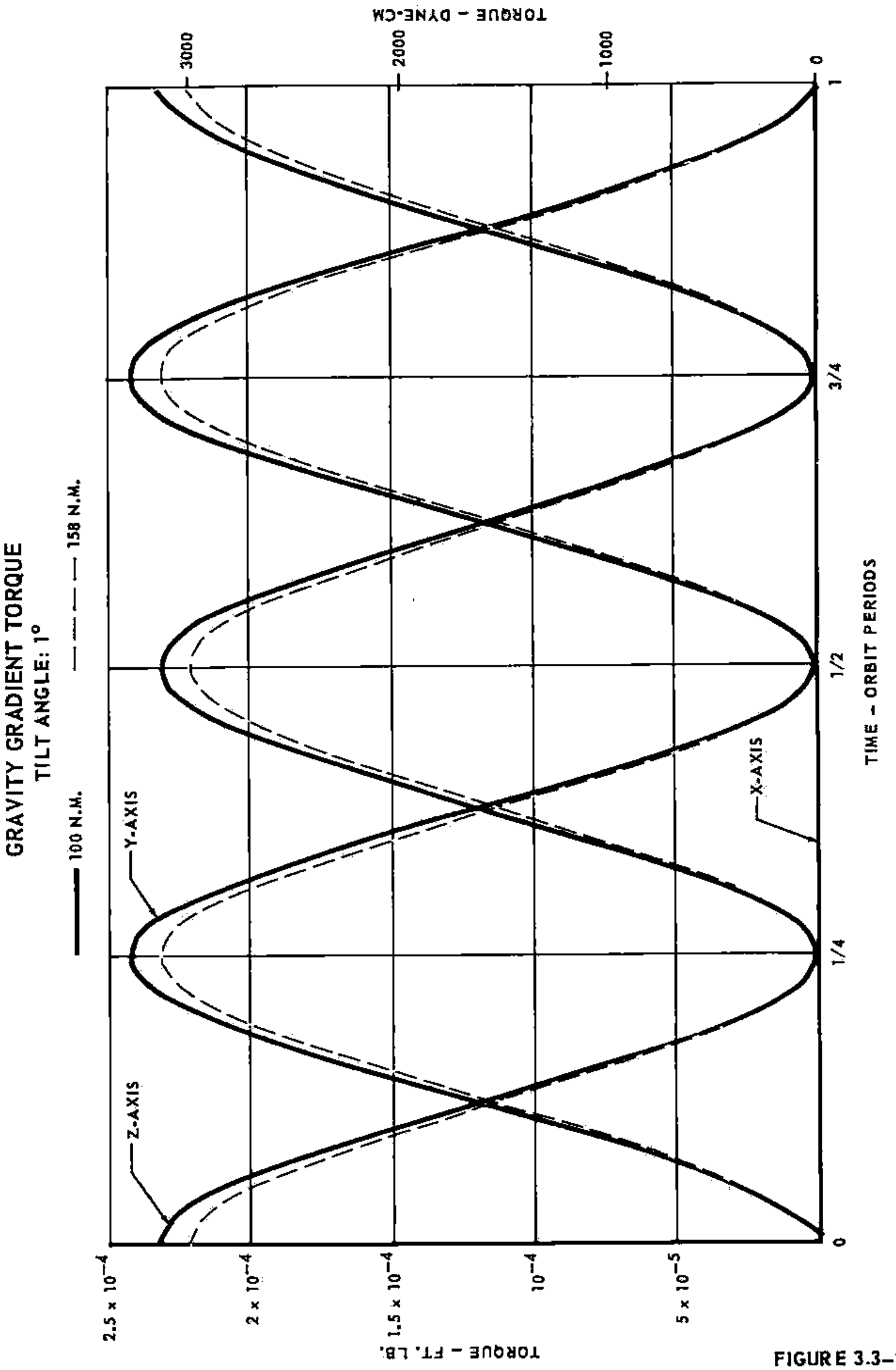

GRAVITY GRADIENT TORQUE
TILT ANGLE: 1°
100 N.M.
158 N.M.
TORQUE – DYNE-CM
3000
2000
1000
0
Z-AXIS
Y-AXIS
X-AXIS
TIME – ORBIT PERIODS
0
1/4
1/2
3/4
1
TORQUE – FT. LB.
2.5 x 10^-4
2 x 10^-4
1.5 x 10^-4
10^-4
5 x 10^-5

FIGURE 3.3-9

AERODYNAMIC TORQUE
ALTITUDE = 158 N.M.

$C_D = 2.2$
CROSS SECTIONAL AREA = 112 FT.^{2}

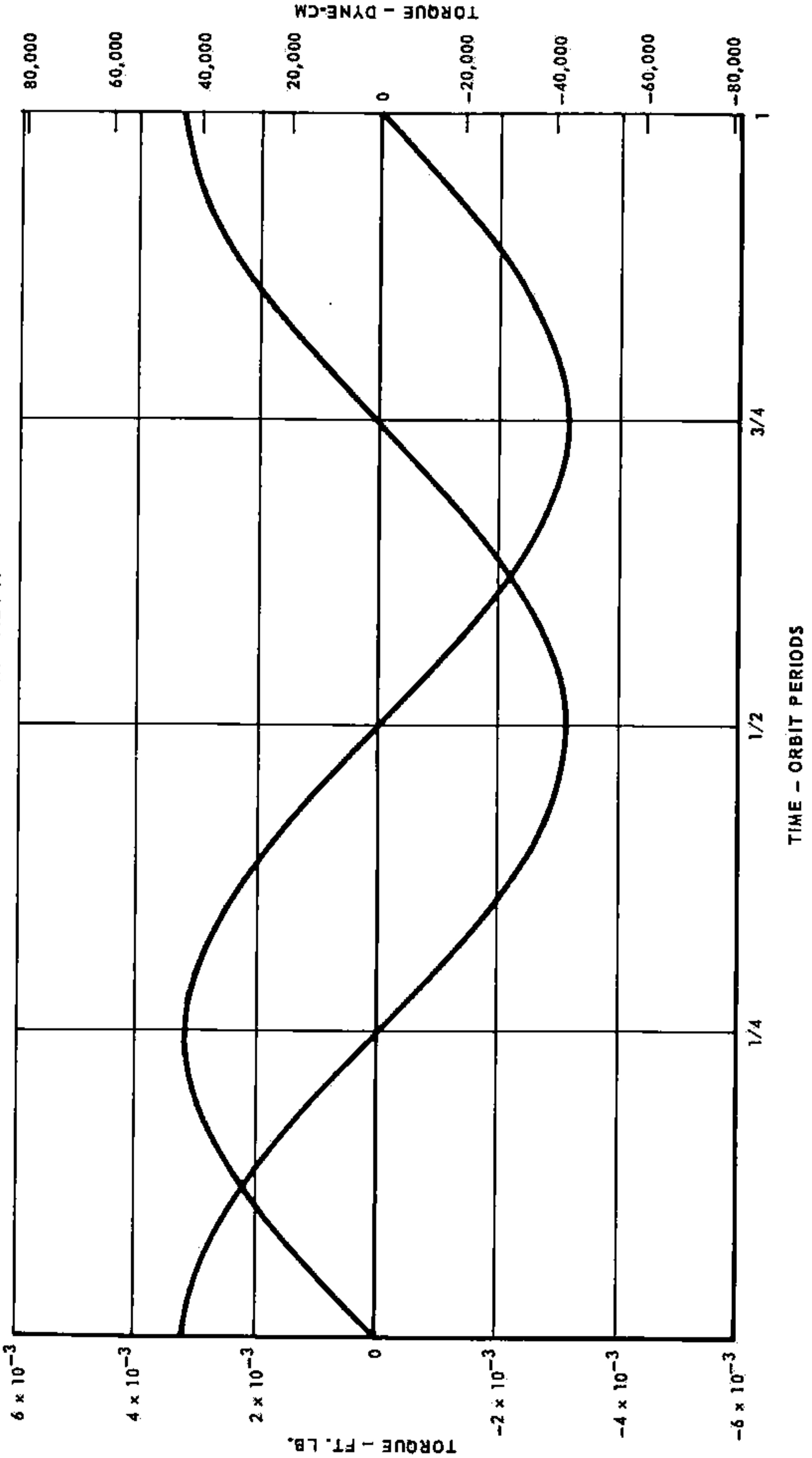


FIGURE 3.3-10

GRAVITY GRADIENT TORQUE
ON THE SPACECRAFT

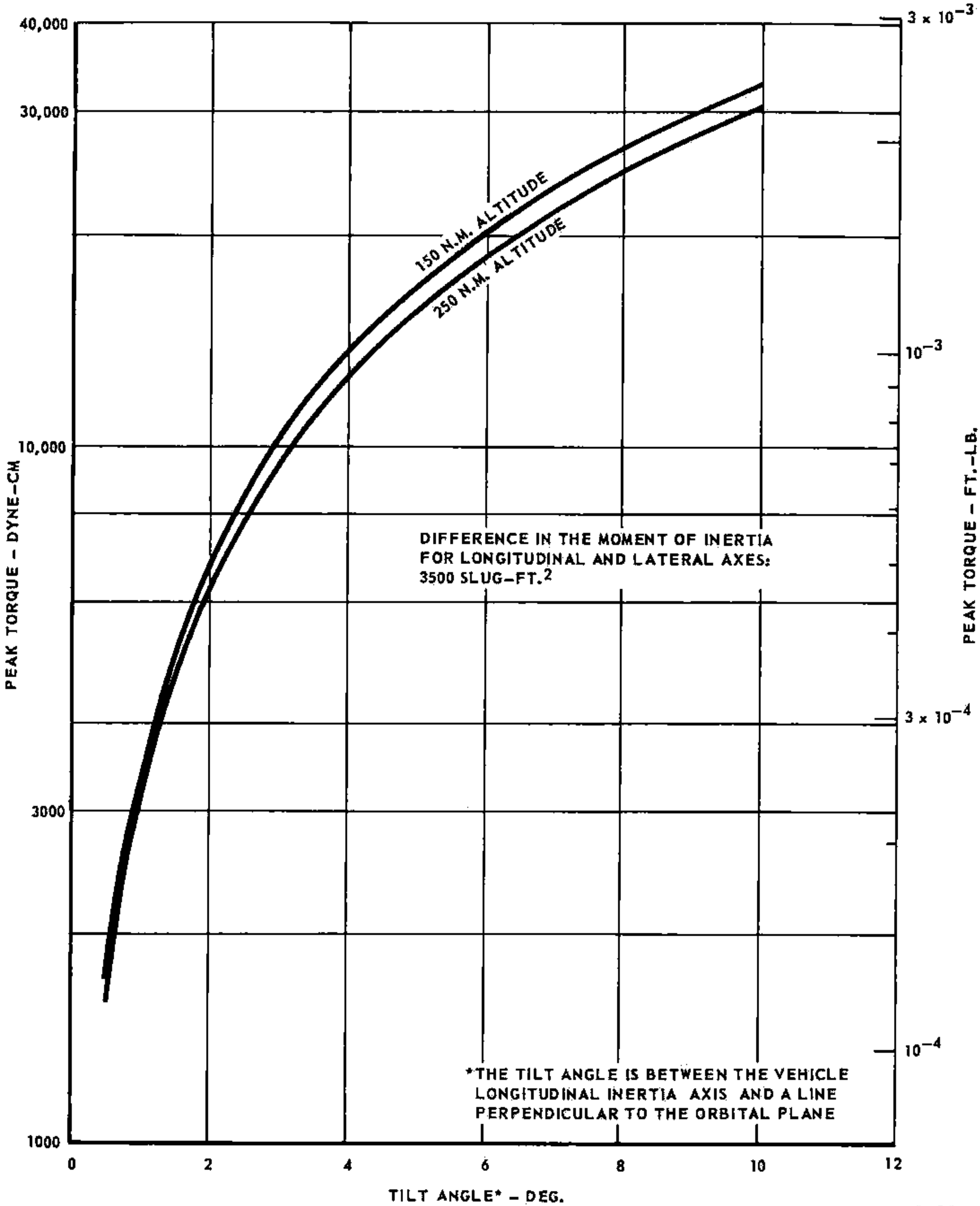


FIGURE 3.3-11

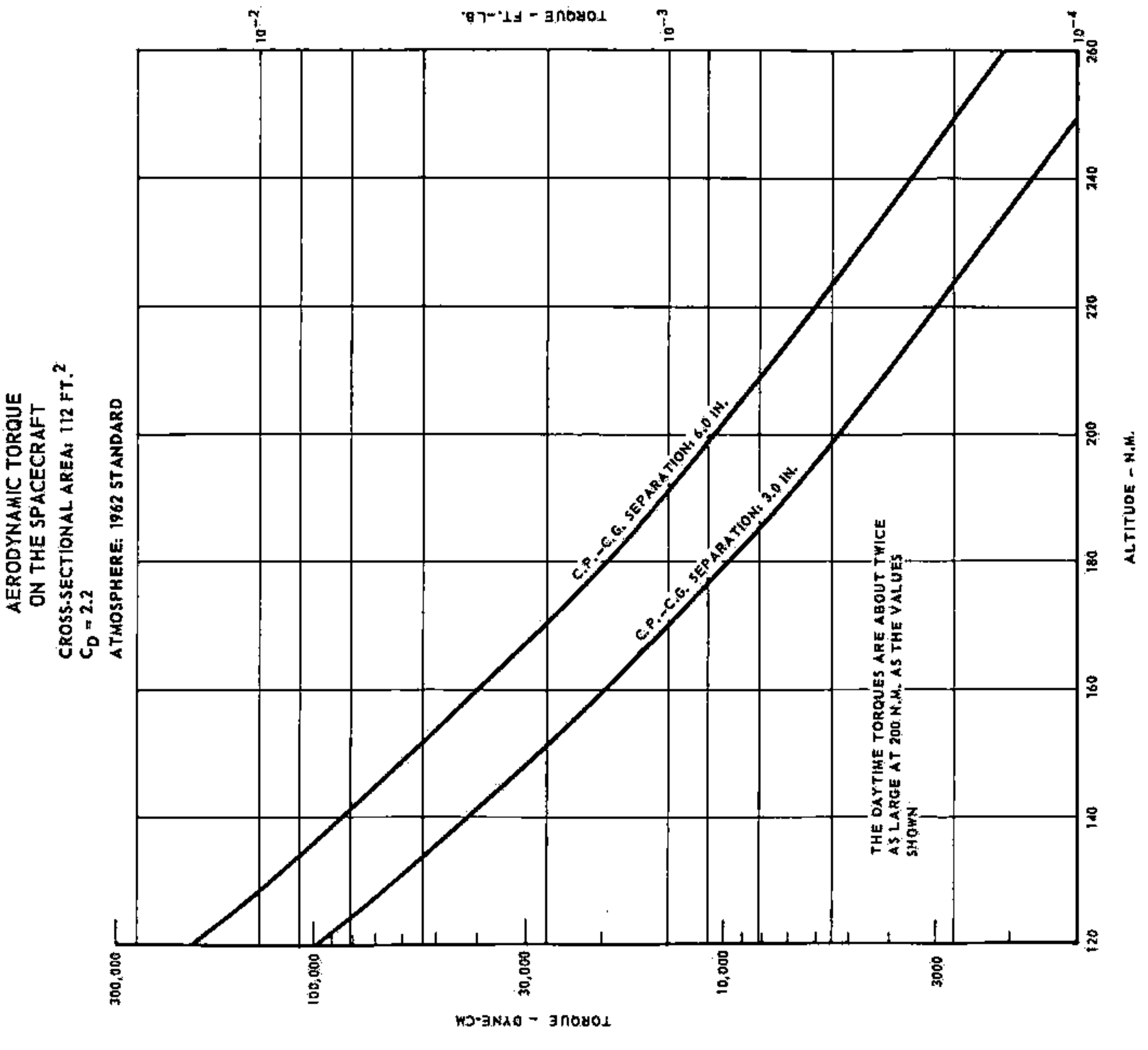

AERODYNAMIC TORQUE
ON THE SPACECRAFT
CROSS-SECTIONAL AREA: 112 FT.^{2}
C_D = 2.2
ATMOSPHERE: 1962 STANDARD
TORQUE – FT.-LB.
10^{-2}
10^{-3}
10^{-4}
C.P.–C.G. SEPARATION: 6.0 IN.
C.P.–C.G. SEPARATION: 3.0 IN.
THE DAYTIME TORQUES ARE ABOUT TWICE AS LARGE AT 200 N.M. AS THE VALUES SHOWN
300,000
100,000
30,000
10,000
3000
TORQUE – DYNE-CM
120
140
160
180
200
220
240
260
ALTITUDE – N.M.

FIGURE 3.3–12

3.3.3.1 (Continued)

seven arc seconds in the absence of fine attitude control.

3.3.3.2 Fine Attitude Control - The fine attitude control can use either control gyros or inertia wheels with a momentum storage capacity of 4 to 7 ft-lb. sec. for each control axis. A cold gas system for desaturation is utilized. Either the gyro or the inertia wheel control system can have the capability for stabilizing the spacecraft to within two arc. sec. in the absence of astronaut body or limb motions.

The control gyro is a better choice than the inertia wheel for compensating for astronaut body and limb motions since the gyro can have a closed loop time constant of the order of one second. Fractional arc second pointing stability can be achieved by use of a fine inertia wheel control or image motion compensation near the focal plane of the telescope.

Three things can be done to compensate for high frequency disturbances due to the astronaut which are too fast for compensation by the momentum wheels. First, the astronaut can be isolated; second, the telescope can be isolated magnetically or pneumatically, and, third, an image motion compensation control can be added to the telescope. The third device would be similar to shimmer compensation devices in ground observatories.

The low impulse cold gas system is required with the control momentum system since the minimum impulse of the Gemini OAMS (0.25 lbs. sec./thruster) corresponds to an angular momentum of 2 (0.25 lbs. sec.) (7.5 ft) = 3.75 ft. lbs. sec. which is comparable to the angular momentum to be stored.

The total fine attitude control system weight is estimated to be 190 lbs., including the momentum exchange system, electronics, fine pointing error detector, cold gas reaction system, and displays and controls for the astronaut.

3.3.3.3 Attitude Control Propulsion System - The attitude control propulsion system for the one-man Gemini consists of:

A. Re-entry control system - Same as Gemini

B. Retrograde system - Same as Gemini

C. OAMS - Same as Gemini with one 22 inch fuel tank, one 22 inch oxidizer tank and one pressurant tank. The OAMS weight is summarized in Table 3.3-4

TABLE 3.3-4

OAMS PROPELLANT REQUIREMENTS

ATTITUDE HOLD:	
DUTY CYCLE: LESS THAN 67%	60 LBS.
LIMIT CYCLE: $\theta = \pm$ 0.2 DEG.	
$\dot{\theta} = \pm$ 0.03 DEG./SEC.	
ATTITUDE SLEW:	40 LBS.
NORMAL SLEW RATE: $\dot{\theta}$ 1 DEG./SEC.	
ORBIT CIRCULARIZATION:	185 LBS.
CONTINGENCY (16%):	47 LBS.
TRAPPED PROPELLANT:	16 LBS.
TOTAL PROPELLANT:	348 LBS.
TANKAGE, LINES, AND VALVES	376 LBS.
TOTAL OAMS WEIGHT	724 LBS.

D. Cold gas system - The cold gas system to be added to the Gemini adapter consists of a nitrogen storage tank, regulator, relief valve, and two modules of three thrusters each.

A preliminary estimate of system characteristics is:

Storage tank diameter - 10.4 in.

Loaded weight of nitrogen - 5.5 lbs.

Thrust for each chamber - 0.25 to 0.5 lbs.

3.3.4 (Continued)

Electrical signal width - 20 to 10 milli-sec.

Impulse per pulse - 0.005 lb. sec.

3.3.3.4 Star Position Detector for Fine Attitude Control System - The purpose of the star position detector is to generate a signal with an accuracy better than 0.1 arc second for use as an error signal in the fine attitude control system. If the light from a guide star is separated into four beams by a prism, or slit jaws, near the focal plane of the telescope and directed into four photomultipliers, the difference of the signals for each pair of phototubes can be used for the control signal for each of two axes.

The detector can detect angular displacements of 0.1 arc seconds for eighth magnitude stars with an integration time of one second and aperture diameter of 16 inches since the signal is sufficiently larger than the background and noise.

The error signal is the change in the photomultiplier output when the star image is displaced. The photomultiplier output for an eighth magnitude star, shown in Figure 3.3-13, is 10^4 counts/second and the corresponding fluctuation is 100 counts/second for a one second integration time. If the edge splitting the star image is positioned so that the photomultiplier output is nearly linear up to a peak 10^4 count/second change for a one arc second displacement and 1000 count/second change for a 0.1 arc second displacement, the ratio of the 0.1 arc second signal (1000 counts/second) to the rms fluctuation in the photomultiplier output (100 counts second) will be 10/1. The use of guide stars of 5.5 magnitude of brighter enables the integration time to be reduced to 0.1 seconds while still maintaining a signal to noise ratio of 10/1.

Two sources of background counts superimposed on the guide star signal are the photomultiplier thermionic dark current emission and the integrated star light in the detector field of view. The dark current background can be reduced to less than

STAR POSITION DETECTOR OUTPUT

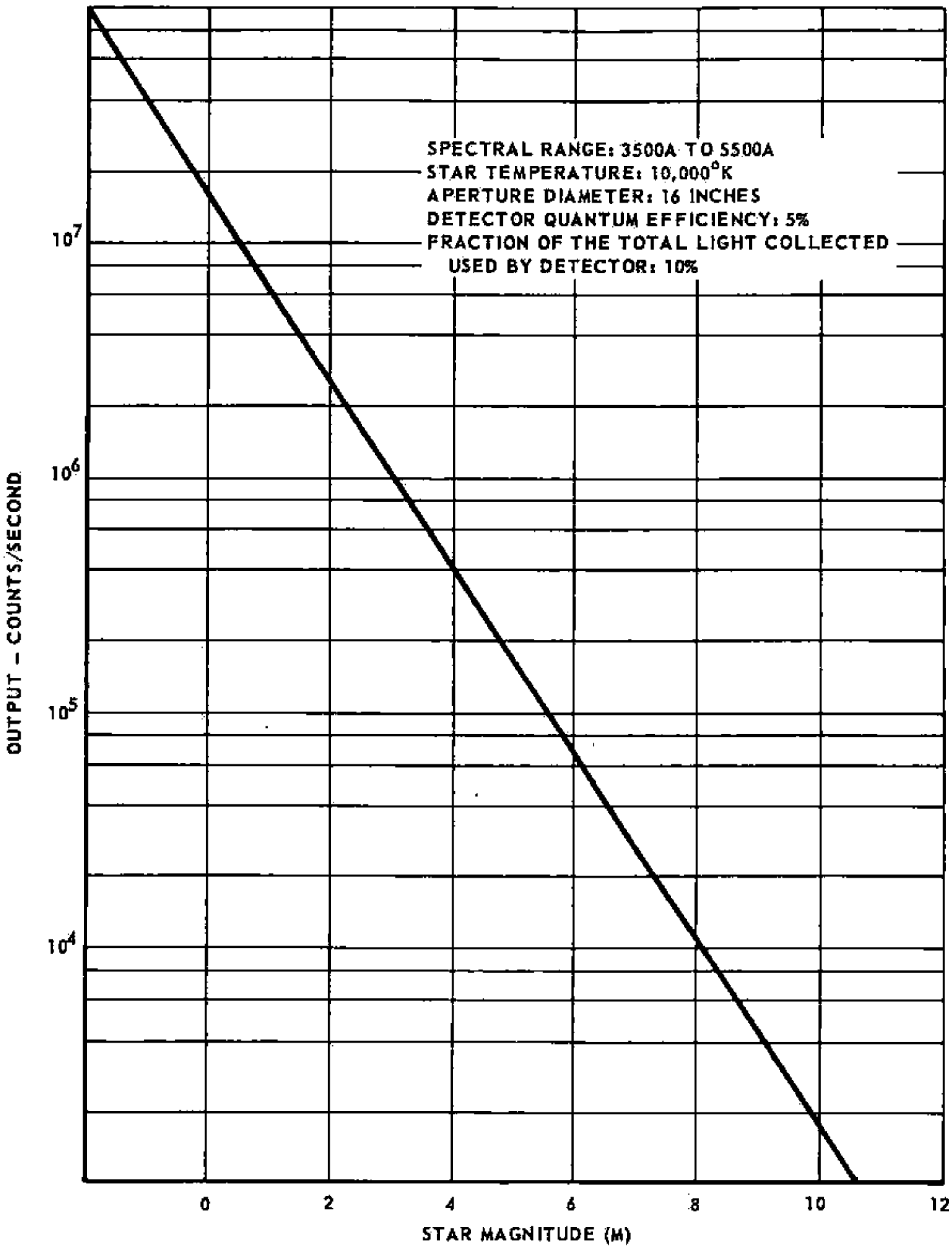


FIGURE 3.3–13

3.3.3.4 (Continued)

200 counts per second by cooling the phototube, as shown in Figure 3.3-14, or by using a tube with a restricted long wavelength sensitivity. The restricted long wavelength sensitivity detector is satisfactory for hot guide stars, but is undesirable for the cooler stars which have a larger fraction of their output in the red region.

The background due to integrated starlight is not consequential for fields of view of 30 arc seconds or less since even near the galactic equator, the photomultiplier output caused by this background is less than 80 counts/second.

3.3.4 Electrical Power System - A third fuel cell section is used for the one-man Gemini to provide redundancy with an increased peak power load due to the added telescope instrumentation and fine attitude control system. The power and energy requirements and the resulting electrical power system reactant weights are listed in Table 3.3-5.

3.3.5 Operational Description of Mission - After insertion into an 87 na. mi. perigee, 200 na. mi. apogee orbit, 185 lbs. of OAMS fuel is used at apogee for circularization. The attitude is manually adjusted with 0.5 degrees of the perpendicular to the orbit plane using the horizon sensors sequentially. The separation of attitude adjustments is 90 degrees in orbit position.

For star acquisition, the roll attitude control system is used with a roll angle indicator to manually position the spacecraft within about one degree of the required roll attitude for a selected star. The telescope optical axis pitch angle is then adjusted using a pitch angle indicator and the telescope or telescope mirror dri·

Following the rough pointing, the 10 to 30 power acquisition telescope, which ha a 1 to 3 degree field of view, is used to make fine adjustments in the spacecraft roll angle and telescope optical axis pitch angle to bring the guide star within the

STAR POSITION DETECTOR DARK CURRENT BACKGROUND

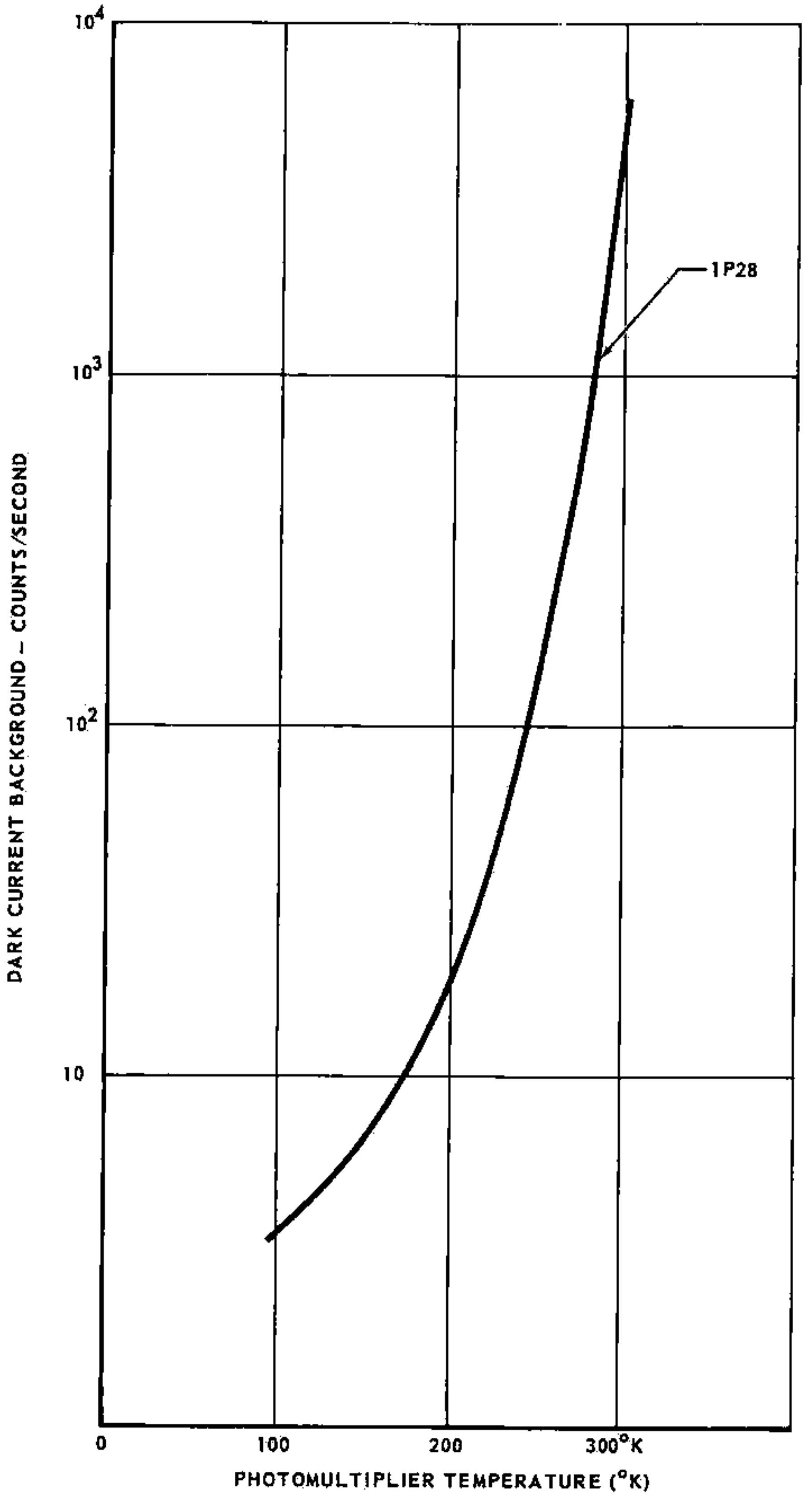


FIGURE 3.3-14

TABLE 3.3-5

ASTRONOMICAL TELESCOPE MISSION ELECTRICAL POWER SYSTEM
MISSION DURATION: 5 DAYS

PEAK POWER	WATTS
BASIC GEMINI EQUIPMENT STEADY LOADS	382
BASIC GEMINI EQUIPMENT INTERMITTENT LOADS	578
TELESCOPE INSTRUMENTATION	115
TELESCOPE ACTIVE THERMAL CONTROL	40
FINE ATTITUDE CONTROL SYSTEM	280
PEAK ELECTRICAL LOAD	1,395
DIODE AND DISTRIBUTION LOSSES	112
PEAK FUEL CELL OUTPUT WATTS	1,507
ENERGY	WATT-HOURS
BASIC GEMINI EQUIPMENT	52,200
TELESCOPE	11,600
FINE ATTITUDE CONTROL SYSTEM	12,000
TOTAL LOAD	75,800
DIODE AND DISTRIBUTION LOSSES	7,020
TOTAL MISSION FUEL CELL OUTPUT	82,820
POWER SYSTEM	
FUEL CELL REACTANTS	104 LB.
REACTANTS TANKAGE	SAME AS 14 DAY GEMINI
RE-ENTRY BATTERIES	4-SAME AS 14 DAY GEMINI

3.3.5 (Continued)

acquisition range of the star position detector. The pitch drive is then locked and the automatic fine tracking is initiated.

3.3.6 Measurement of the Effects of Astronaut Motions on Telescope Pointing Stability - For the re-entry module installation of the telescope, the effectiveness of compensation for the harmful effects of astronaut motions by using a fast spacecraft attitude control system, optical image motion compensation, and electronic image motion compensation can be determined. The telescope image stability is measured by the star position detector and photographic star image. An experiment on flexible support isolation of astronaut motions can be accomplished; however, the available space is very limited.

For the adapter module installation of the telescope, the tunnel provides space for other experiments such as a controlled floating support for the astronaut. If the telescope diameter is reduced from 26 to about 22 inches, 100 lbs. is made available for the support. The support has six servo driven reaction weights and six photodetectors to sense the position of the support with respect to the spacecraft. As the astronaut moves an arm, for example, the body and support move in the opposite direction. The photodetectors sense the motion and provide a signal to the reaction weight servo drives which move the weights to balance the arm motions and keep the support in a fixed position. This type of support should be very useful in a large space station observatory since the astronaut can use an eyepiece for optical alignment and star acquisition while mechanically isolated from the spacecraft.

3.3.7 Astronomical Measurements - The high resolution and extended spectral range for a space telescope offer many possibilities for optical measurements as discussed in References 3.3-1 to 6:

1. Ultraviolet flux from hot stars (Important for stellar evolution and interstellar gas dynamics studies).

3.3.7 (Continued)

2. Chemical composition of the stars and interstellar gas (Search for unknown components).
3. Photometery of dense clusters (Determine mass of luminous giant stars in the center of the globular clusters).
4. Absolute intensity of zodiacal light.
5. Ultraviolet sky survey.
6. Observations of close double stars with rapid orbit period.
7. Spectrometry at wavelengths less than 1000 Å.
8. IR spectrometry.
9. Interferometric spectrometry.
10. Visual observations.
11. Photographic observations.
12. Coronagraphic observations.
13. Observations using an image intensifier.
14. Measurement of manual tracking accuracy.

3.4 Artificial Gravity - Three methods of providing artificial gravity are discussed. The first method involves rotation of the Gemini spacecraft while directly connected to the burned-out stage II of the Gemini Launch Vehicle. The second method rotates the docked Gemini-Agena orbiting vehicle. The general configuration and characteristics of these methods are shown in Figure 3.4-1. The third method is accomplished by rotating the Gemini spacecraft while cable-connected to either the Agena or to Stage II of the Gemini Launch Vehicle. These configurations are shown in Figures 3.4-2 and 3.4-3.

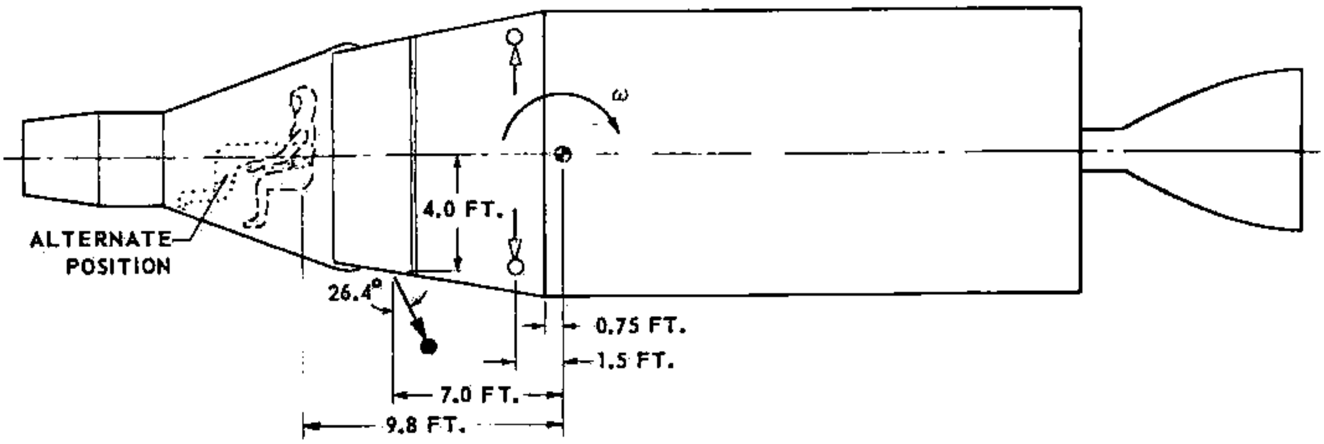
Operational conditions presented in Table 3.4-1 indicate that methods of using Gemini thrusters are feasible if their application to operations other than spin or despin are limited. Ablation thruster specifications require a guaranteed life of 425 seconds of burn time for the attitude control thrusters and 557 seconds of burn time for the maneuver thrusters. It is anticipated that the maneuver thrusters can be used for both rendezvous and for artificial gravity. Since rendezvous requires predominantly longitudinal thrusts, the major portion of the design life of the lateral and vertical thrusters can be used for artificial gravity. The Agena attitude control system could also be used if provided with additional storage of cold gas propellant.

The alternate position of the crew members for the first method, shown in Figure 3.4-1, would provide an eyeballs-in g force, but would require some restraint for artificial gravity operation. In addition, viewing the Gemini instrument panels would be difficult.

At rotational speeds greater than 4 to 5 rpm, normal head motions generally result in nausea due to Coriolis effects. The relatively high rates of rotation for directly connected vehicles, shown in Table 3.4-1, may be justified by restricti duties and movements during spin or by head restraint. The cable-connected

INTEGRAL SPIN CONCEPTS

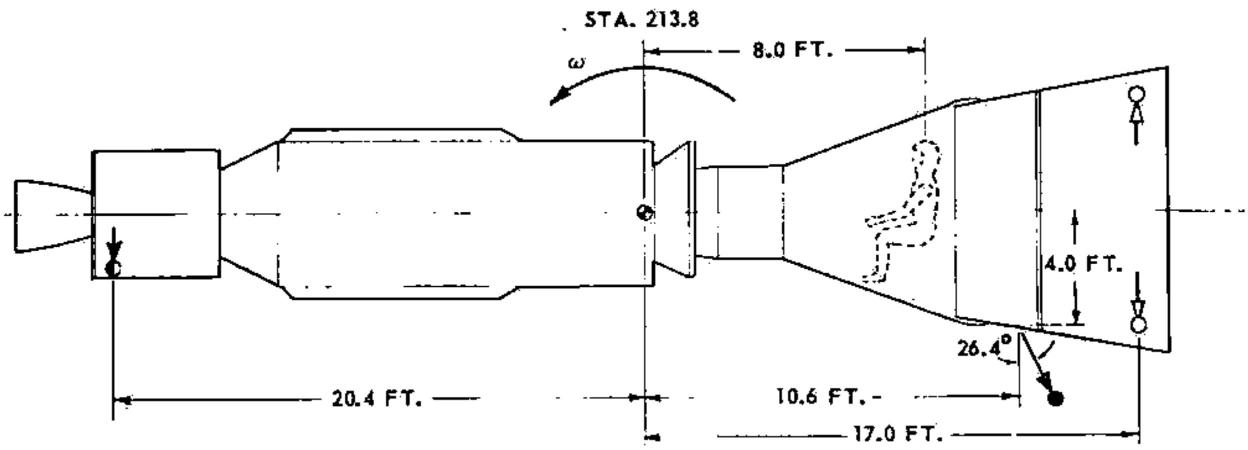
A) GEMINI DIRECTLY CONNECTED TO STAGE II OF GLV



TOTAL WEIGHT = 12,930 LB.
$I_y = I_z = 52{,}100$ SLUG-FT.^{2}

THRUSTER CODE	
○◁—	23 LB. THRUST CHAMBER
●◀—	94.5 LB. THRUST CHAMBER
⊖◀—	10 LB. COLD GAS CHAMBER

B) DOCKED GEMINI-AGENA VEHICLE



TOTAL WEIGHT = 14,130 LB.
$I_y = I_z = 62{,}000$ SLUG-FT.^{2}

FIGURE 3-4-1

CABLE-CONNECTED GEMINI-AGENA CONFIGURATION

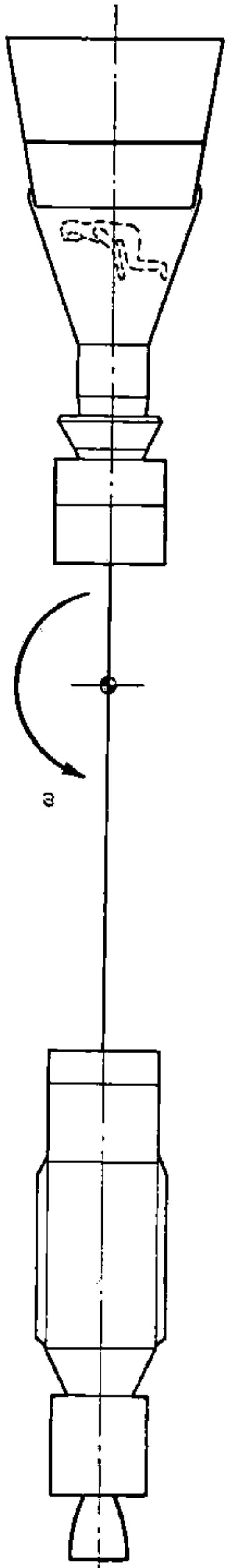
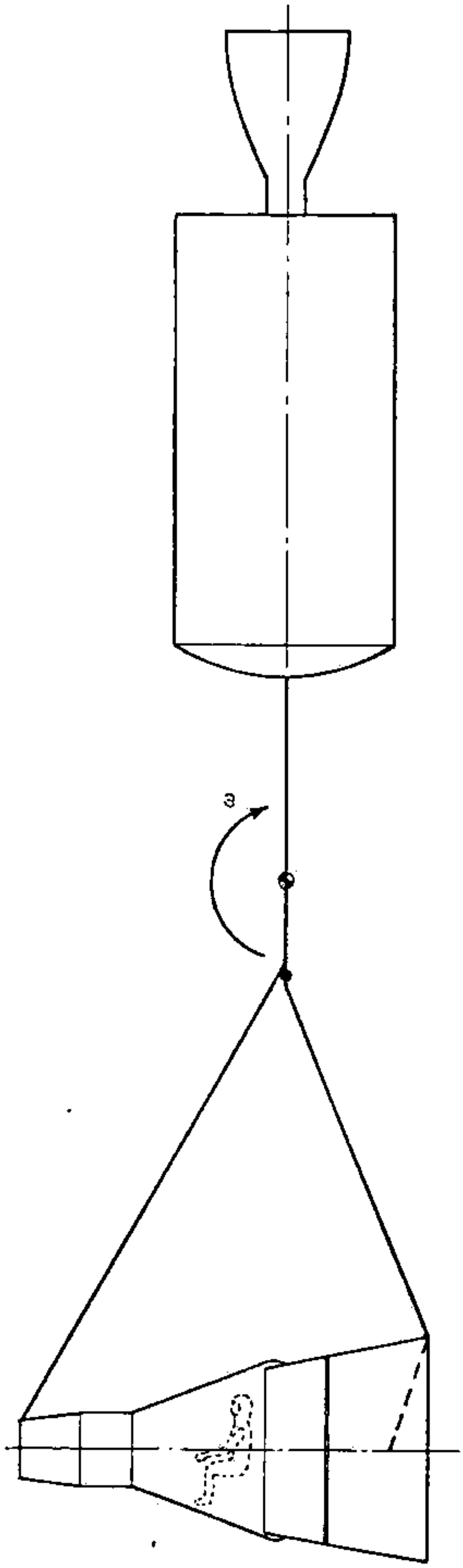


FIGURE 3.4-2



CABLE-CONNECTED GEMINI-STAGE II OF GLV CONFIGURATION

FIGURE 3.4-3