2.8 Long Duration Mission

2.8.1 Description - The in-orbit configuration of the long duration (30-45 days) mission orbital spacecraft involves the addition of a "Mission Section" to the Agena and a combination "access tunnel/living quarters" segment which is used for transfer from the Gemini to the mission section. The mission section contains the food, water, personal gear, and emergency oxygen supplies. The section is 60 inches in diameter and 165 inches long. The access tunnel provides direct access to the mission section. It is inflatable and is erected by one man outside the spacecraft. The tunnel is composed of multi layers of dacron, polyurethane, and vinyl foam. The tunnel volume appears to be adequate for crew activities to be performed. The major modification to the Gemini consists of redesign to include another smaller hatch within the present right hand hatch.

2.8.2 Technical or Scientific Benefit - Possible benefits or a technical or scientific nature are:

A. Long duration weightlessness experiments (up to 45 days)

B. Extravehicular assembly and development of techniques for assembly in space outside a spacecraft or space station.

C. Expandable living quarters development and demonstration

2.8.3 Effects on U.S. Space Program

A. Apollo - Development and proof testing of Expandable Structures for temporary shelters on the moon for short periods of time, or for lunar instrumentation housings is obtained.

B. AES and Advanced Missions in General - The utility of expendable structures for space living quarters on long duration missions, and for use as

2.8.3 (Continued)

access tunnels or environmental hangers for advanced space stations, would be evaluated.

C. DOD - A tunnel of this type was considered for MOL crew transfer.

2.8.4 Prestige Value

A. Domestic - The mission would be an advancement in the state of the art in the development of equipment and techniques to be used in future space stations and lunar bases and exploration.

B. International - Extravehicular erection of a structure and the extended mission should enhance prestige.

2.8.5 Performance Feasibility - Over all performance appears to be feasible based on preliminary analysis of test requirements and existing hardware and technology. Preliminary estimates indicate the payload requirements to be within the capabilities of the GLV and the Atlas-Agena.

2.8.6 Cost Feasibility - The first unit cost is estimated to be $36 million, with each additional unit costing $6.0 million, plus the costs of the launch vehicles (see Section 4).

2.8.7 Schedule Feasibility - The development and qualification of the Access Tunnel/Living Quarters and associated equipment will take approximately 18 months.

2.8.8 Operations Feasibility - The present Gemini program will provide proof of the feasibility of extravehicular activity and a level of experience in free space action. Erection of the access tunnel and activation of the mission section appears to be an operationally feasible mode.

2.8.9 Impact on Gemini Program - Based on present schedule estimates, it appears the long duration mission could be fitted into the present Gemini program.

2.9 Land Landing

2.9.1 Description - Land landing is considered to be very desirable, and a parasail-landing rocket system for Gemini is currently under development by NASA. Tests of the parasail-landing rocket system are now in progress. The first full scale Gemini parasail drop test produced some unsatisfactory results: (1) the control motors apparently locked, which caused the spacecraft to turn while landing and probably caused the spacecraft to tumble; (2) the landing rockets fired, dislodged rocks of considerable size, and created a fairly large crater which may also have caused the spacecraft to tumble.

Consideration has been given to other methods for providing a land landing capability, should the present parasail landing system prove unacceptable. It is felt that major design changes to Gemini would not be acceptable and landing schemes involving minimum change to Gemini were considered. Two alternative approaches are:

A. The landing rockets for the parasail could be suspended from the parasail bridle.

B. The parasail-landing rocket configuration could be replaced with a cloverleaf parachute which would decrease the descent velocity to 13 fps prior to impact. Thirteen feet per second is the allowable impact velocity for the present Gemini structure.

Some other landing schemes considered, but which would require extensive change to Gemini, are:

A. Installation of Impact Bags between Heat Shield and large pressure bulkhead, plus the installation of a toroidal shape impact bag around the recovery section.

2.9.1 (Continued)

B. Installation of cable-spike arrangement in conjunction with an impact bag, an extendable heat shield, or vertical landing rockets and utilizing heat shield deformation for energy absorption.

C. Horizontal and vertical landing rockets.

D. Larger landing gear (increased stroke and strength).

2.9.2 Technical or Scientific Benefit - Land landing capability is desirable in the U.S. Space Program, particularly from a military standpoint since security would be more easily maintained. In addition, "dry" landings should reduce refurbishment required and is therefore desirable from a reusability standpoint.

A. Apollo - Future Apollo missions could benefit from addition of a land landing capability.

B. AES and Advanced Missions in General - Land landing techniques would be developed and available for possible application to future missions.

C. DOD - Land landing capability could be incorporated on the Gemini B.

2.9.4 Prestige Value - Domestic prestige should be increased by demonstrating this capability. Television could show the actual landing. Probably little international prestige can be gained since the Soviet Union presumably has a land landing capability.

2.9.5 Performance Feasibility - Weight estimates show that the clover leaf system could be incorporated within GLV payload capability by elimination of the rendezvous capability.

2.9.6 Cost Feasibility - For the parasail system employing landing rockets suspended from the risers, it is estimated that the first unit cost would be $4.6 million, with each additional unit costing $0.35 million. For the cloverleaf system, the first unit cost is estimated to be $15.2 million, with each additional unit costing $0.20 million. Spacecraft and launch vehicle costs are to be added.

2.9.7 Schedule Feasibility - It is estimated that cloverleaf system would require about 30 months to develop for Gemini. Qualification of a landing rocket suspended from the parasail risers would require about 24 months.

2.9.8 Operations Feasibility - Development of the cloverleaf chute appears to be well under way. Suspension of a landing rocket on the risers should be a relatively straightforward development.

2.9.9 Impact on Gemini Program - Land landing capability would be incorporated in conjunction with one or more other experiments.

2.9.10 Other Aspects - The installation of the present parasail system on Gemini (S/C #12) will incur a weight increase. The rendezvous system (radar, OAM's propellant tanks and pressurant, etc.) will be removed to provide for the additional weight of the parasail land landing system over the parachute water landing system. There are approximately 400 lbs. of experiments on S/C #12 which could be eliminated to permit retention of some rendezvous capability. It is doubtful that the rendezvous capability could be retained with a landing rocket stored in the rendezvous and recovery section, at least without extensive redesign.

The parasail land landing system has not been approved for installation on Gemini as of this date. However, McDonnell has submitted a work statement to NASA for approval.

3. SUPPORTING INFORMATION

3.1 Rendezvous with an Unmanned Satellite - The basic mission plan for rendezvous with the Pegasus is presented in Figure 3.1-1. The plan is to: (1) inject into a low orbit coplanar with the Pegasus orbit for gross catch-up, (2) transfer open loop, based on tracking data, to a slow catch-up orbit slightly lower than the Pegasus orbit, and (3) perform a closed-loop rendezvous after radar contact is made.

The amount of OAMS propellant remaining at injection of the Gemini into a 87-100 na. mi. orbit as a function of total OAMS propellant loaded at lift-off is shown in Figure 3.1-2.

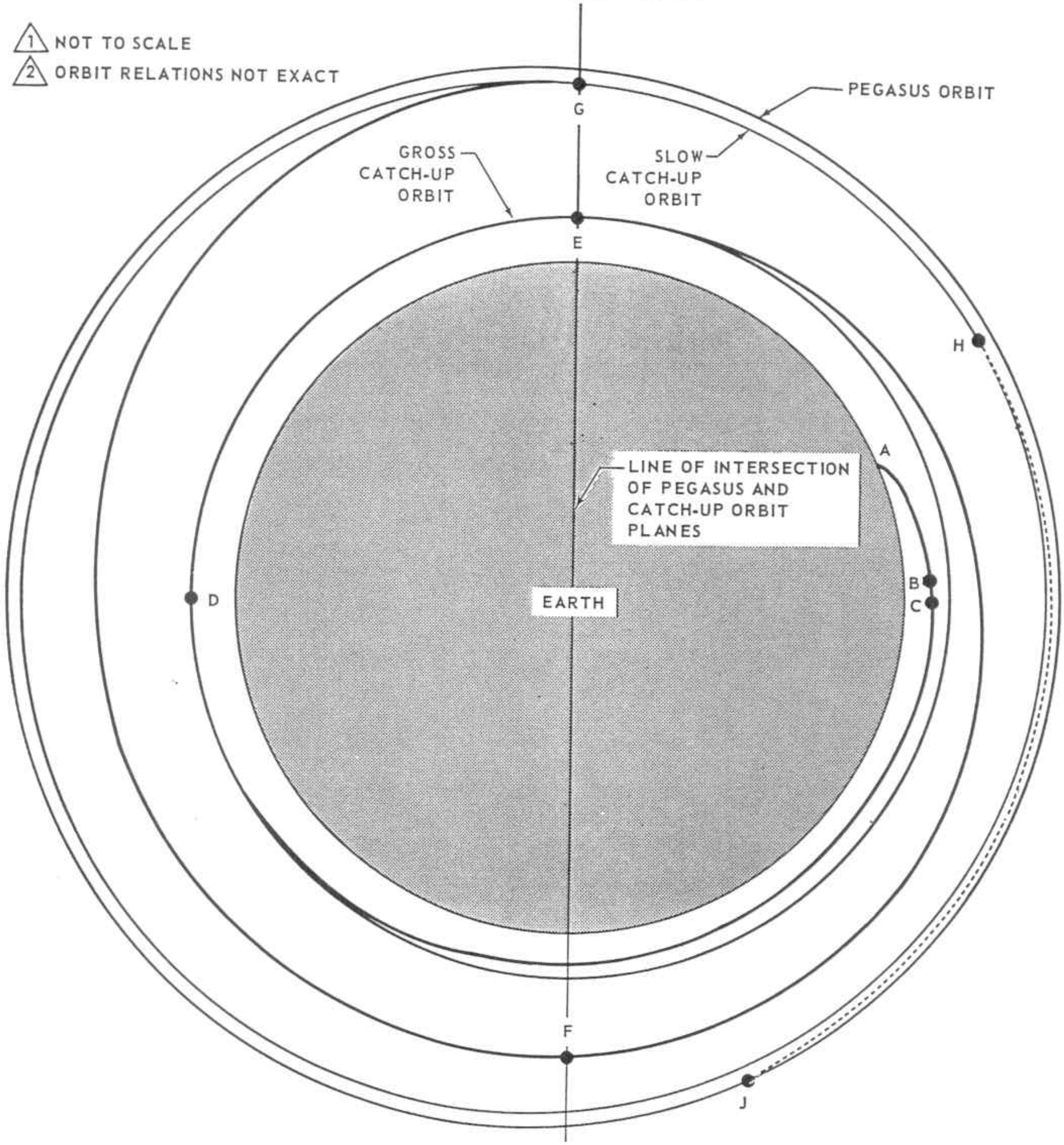
A preliminary estimate of the propellant required for the mission is shown in Table 3.1-1. This is also noted on Figure 3.1-2.

The weight data used in calculating the propellant required for rendezvous with Pegasus is given in Figure 3.1-3 as a function of OAMS propellant loaded. Current Gemini I_{sp} of 256 seconds was used in estimating propellant required for ΔV. An I_{sp} of 225 seconds (pulse mode) was used when estimating attitude control requirements.

Radiation dose rate estimates are summarized in Table 3.1-2. The mission dose depends on the detailed mission profile. The dose rates given allow the mission dose to be estimated when the time spent in each particular profile segment is known. As shown, the radiation dose is almost entirely received in the region south of the equator and between South America and Africa. Within this region, the dose received is dependent upon altitude, increasing rapidly with altitude.

For June 1966, a Lockheed radiation model (Reference 3.1-1) predicts that, for a 30^{o} inclination circular orbit at 325 nautical miles altitude, a man in a 1 lb/ft^2 space suit would receive 25 rads/day, with a range of uncertainty of

SCHEMATIC OF MISSION PROFILE



A. LAUNCH
B. AUGMENTED OAMS USED TO COMPLETE INJECTION
C. INJECTION INTO 87–100 N.M. ORBIT
D. CIRCULARIZE AT 100 N.M. FOR GROSS CATCH-UP
E. FIRST PULSE OF TRANSFER
F. SECOND PULSE OF TRANSFER
G. INJECTION INTO SLOW CATCH-UP ORBIT
H. START OF CLOSED LOOP RENDEZVOUS
J. COMPLETION OF RENDEZVOUS

FIGURE 3.1-1

OAMS PROPELLANT REMAINING AT INJECTION AS A FUNCTION OF OAMS PROPELLANT AT LIFTOFF

87–100 N.M. PARKING ORBIT

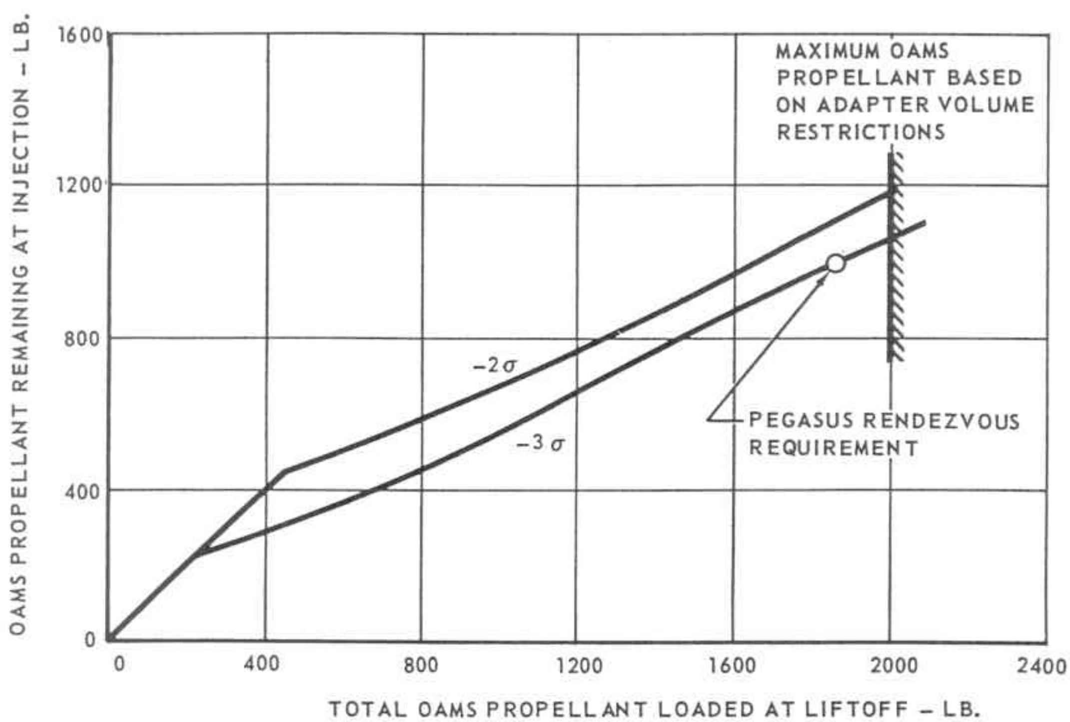


FIGURE 3.1-2

TABLE 3.1-1

ESTIMATED RENDEZVOUS ΔV

CIRCULARIZE @ 100 N.M.	23 FT./SEC.
TRANSFER TO SLOW CATCH-UP	790 FT./SEC.
CLOSED LOOP RENDEZVOUS (PERFECT)	17 FT./SEC.
GUIDANCE AND NAVIGATION ERRORS	50 FT./SEC.
DOCKING	50 FT./SEC.
TOTAL	930 FT./SEC.
OR IN POUNDS (8540 LB. INITIAL SPACECRAFT WEIGHT IN ORBIT)	920 LB.
ESTIMATED ATTITUDE CONTROL PROPELLANT (BASED ON .1 FT. c.g. ECCENTRICITY, 1 DEGREE THRUSTER ALIGNMENT AND 2.5 DEG./SEC. ORIENTATION RATES AND 20% MARGIN)	78 LB.
TOTAL WEIGHT	998 LB.

LAUNCH WEIGHT VS. PROPELLANT ON BOARD

8 RETROROCKETS
ESTIMATED RETROGRADE WT. – 5955 LB.
DESIGN RETROGRADE WT. – 6295 LB.

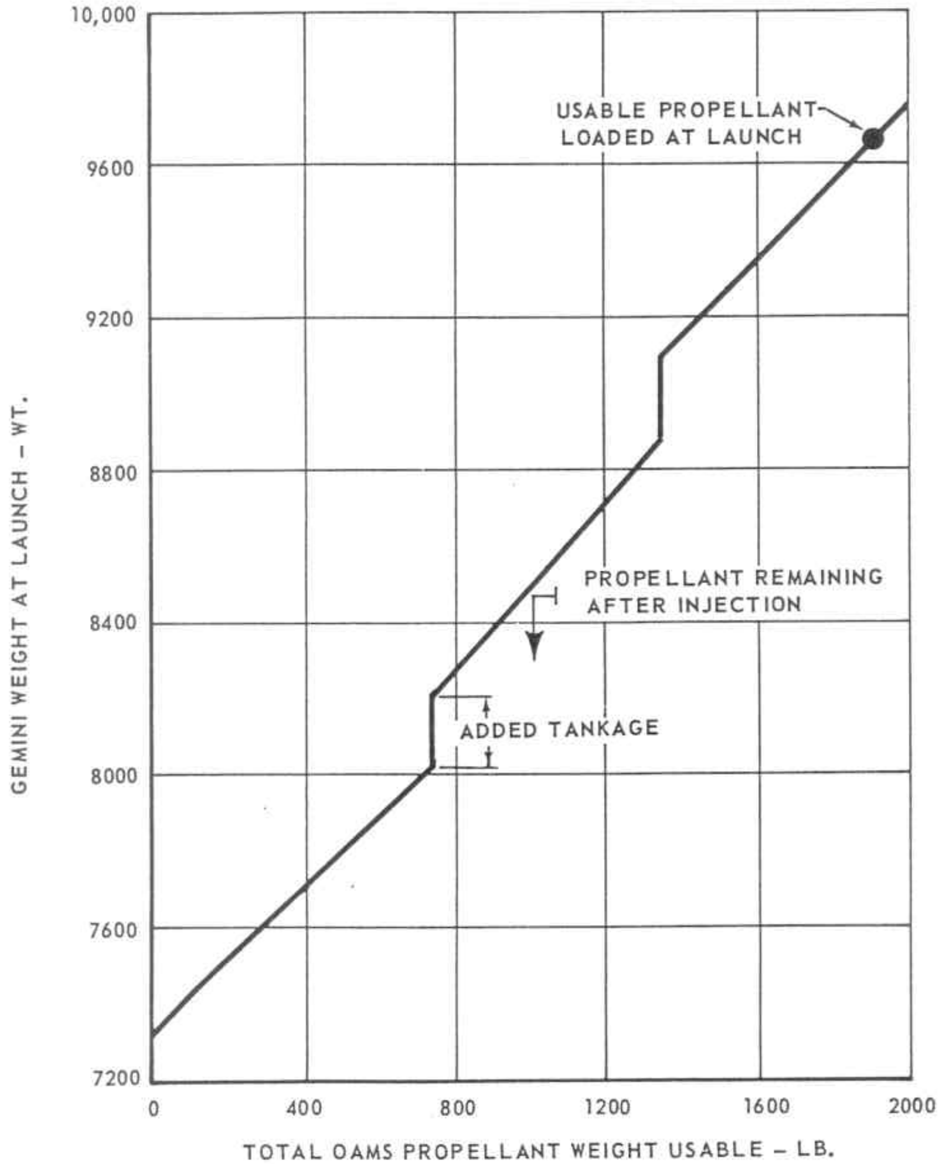


FIGURE 3.1–3

TABLE 3.1-2

RADIATION DOSE RATE ESTIMATES FOR
GEMINI-PEGASUS RENDEZVOUS MISSION

RADIATION ENVIRONMENT MODEL 30° ORBIT INCLINATION		EXTRAVEHICULAR OPERATIONS WITH 1 LB./FT.^{2} SPACE SUIT				GEMINI			
		325 N.M. CIRCULAR	100 N.M. CIRCULAR	PERIGEE 270 N.M. OVER S. ATLANTIC	APOGEE 394 N.M. OVER S. ATLANTIC	325 N.M. CIRCULAR	100 N.M. CIRCULAR	PERIGEE 270 N.M. OVER S. ATLANTIC	APOGEE 394 N.M. OVER S. ATLANTIC
MCDONNELL AIRCRAFT CORPORATION	JANUARY 1966	16 6 TO 45	.3 .12 TO .9	10.2 3.75 TO 28.5	81 30 TO 225	5.2 2.1 TO 15	.12 .06 TO .3	3.3 1.5 TO 9.3	16.5 7 TO 48
	JUNE 1966	5.4 2 TO 15	.1 .04 TO .3	3.4 1.25 TO 9.5	27 10 TO 75	1.8 .7 TO 5	.04 .02 TO .1	1.1 .45 TO 3.1	5.5 2.3 TO 16
	JANUARY 1967	1.8 .7 TO 5	.03 .01 TO .1	1.1 .4 TO 3.1	9 3.3 TO 25	.6 .23 TO 1.7	.013 .01 TO .03	.4 .15 TO 1.1	1.8 .8 TO 5.3
LOCKHEED MISSILES AND SPACE COMPANY	JANUARY 1966	75 24 TO 240	.15 .06 TO 4.8	46.5 15 TO 150	375 120 TO 1200	25 8.1 TO 81	.6 .15 TO 1.5	15.6 5.1 TO 51	78 25.5 TO 255
	JUNE 1966	25 8 TO 80	.05 .02 TO 1.6	15.5 5 TO 50	125 40 TO 400	8.3 2.7 TO 27	.2 .05 TO .5	5.2 1.7 TO 17	26 8.5 TO 85
	JANUARY 1967	8.3 2.7 TO 27	.02 .007 TO .53	5.2 1.7 TO 17	41.7 13.3 TO 133	2.8 .9 TO 9	.07 .02 TO .2	1.7 .6 TO 6	8.7 2.8 TO 28

NOTE:

1. DOSE RATES ARE IN RAD/DAY. UPPER FIGURE IS THE BEST ESTIMATE. LOWER FIGURES ARE THE RANGE OF THE ESTIMATE.
2. LOCKHEED MODEL BASED ON REF. 3.1-1

3.1 (Continued)

from 8 rads/day to 80 rads/day. Inside the Gemini, the astronauts would receive about 30% of the dose.

For June 1966, a McDonnell radiation environment model predicts that, for the same orbit and altitude, a man in a 1 lb/ft^2 space suit would receive only 5.4 rads/day, with a range of uncertainty of from 2 rads/day to 15 rads/day. Inside the Gemini, the astronauts would receive about 30% of the above M.A.C. dose

Since both sets of values are predictions, based for the most part on the same or equivalent data, a choice as to the correctness of either model can not be made at this time. However, the M.A.C. model is less by a factor of nearly 5 than the Lockheed.

When the Gemini is in an elliptical orbit at an inclination of 30° and with a perigee of 290 nautical miles and apogee of 394 nautical miles, the dose to the astronauts will depend sharply on whether the apogee or the perigee is located south of the equator and between South America and Africa. With the perigee in this region, the dose figures above are divided by a factor of 1.6. With the apogee in this region, the dose figures are multiplied by a factor of 5. If the upper limit based on the Lockheed radiation model of 80 rads/day is correct, then an astronaut in a space suit would receive approximately 400 rads/day to his skin. This would cause erythema (skin burn) and would probably limit severely any future missions for the astronaut. Thus, one criterion for minimizing radiation dose in an elliptical orbit has been established; stay in the elliptical orbit only when the perigee is over the South Atlantic region, or when the orbit plane is outside that region.

The time spent in a 100 nautical mile circular orbit is of comparatively small consequence to the total mission dose, since the dose rates given for a

3.1 (Continued)

325 nautical mile circular orbit are less by about a factor of 50.

The mission dose will also depend on the month and year of the mission. For a January 1966 mission, the dose rates are a factor of 3 higher than the dose rates for June 1966. For a January 1967 mission, the dose rates are a factor of 3 lower than the dose rates for June 1966. This is primarily due to the slow decay of the Starfish artifical radiation zone.

The weight added to the Gemini is summarized in Table 3.1-3.

TABLE 3.1-3

WEIGHT SUMMARY
RENDEZVOUS WITH UNMANNED SATELLITE (PEGASUS)

WEIGHT ADDED TO GEMINI ADAPTER		(2052)
PROPELLANT		1380
FUEL	622	
OXIDIZER	758	
TANKAGE		79
FUEL TANKS (4)	36	
OXIDIZER TANKS (4)	43	
PRESSURIZATION SYSTEM		82
HELIUM	8	
HELIUM TANKS (4)	74	
MOUNTING		130
THRUSTERS		51
100 LB. TCA (4)	37	
MOUNTING	9	
CIRCUITRY	5	
RETROGRADE SYSTEM		330
ROCKET MOTORS	270	
MOUNTING AND CIRCUITRY	60	

3.2 One Man Gemini-Earth Surface Mapping - Low altitude earth surface photography applicable to topographic mapping, geological reconnaissance, and to studies of oceanography, hydrology, and meteorology, can be accomplished with a One-Man Gemini. For illustration, from a 120 n.m. low inclination orbit, stereographic photographs can be obtained with nadir resolution of 22.5 ft. for 2:1 contrast and estimated mapping accuracies of better than 1000 feet. Film is provided for an area of 14.3 million square nautical miles. Complete coverage with 50% overlap at the equator, between $\pm$ 45° lateral swathwidths on successive days, can be completed in a seven day mission at the 120 n.m. orbital altitude and 35° inclination orbit.

3.2.1 Experiment Equipment Description and Illustrative Performance - The equipment complement, installed as shown in Figure 3.2-1, with the characteristics indicated in Table 3.2-1 consists of the following:

Panoramic Mapping Cameras (2) - Utilizing a 13 inch focal length, f/3.5, Petzval lens design by Itek (Figure 3.2-2) in a nodding lens pan design, a 100 lines per m.m. resolution is realizable at low contrast (2:1) on SO-206 thin base 70 m.m. film. A 10° x 90° scanning field is employed. The resolution at a 120 n.m. altitude and a contrast of 2:1 is 22.5 feet per cycle. A one millisecond or shorter exposure time is required with a sun altitude above 20°. The short exposure time coupled with the focal length permits loose image motion compensation, attitude rates, and attitude tolerances. Image motion compensation accuracy as poor as 15% of nadir V/h results in negligible resolution loss.

The cameras operate unpressurized, but are thermally controlled electrically to maintain alignment and focus. The included angle between the two cameras is 25° for stereoscopic viewing (map contouring). Fiducial marks and data annotation are included on each frame for correlation with post flight orbit data. With time

EARTH SURFACE MAPPING CAMERA INSTALLATION
ONE-MAN GEMINI

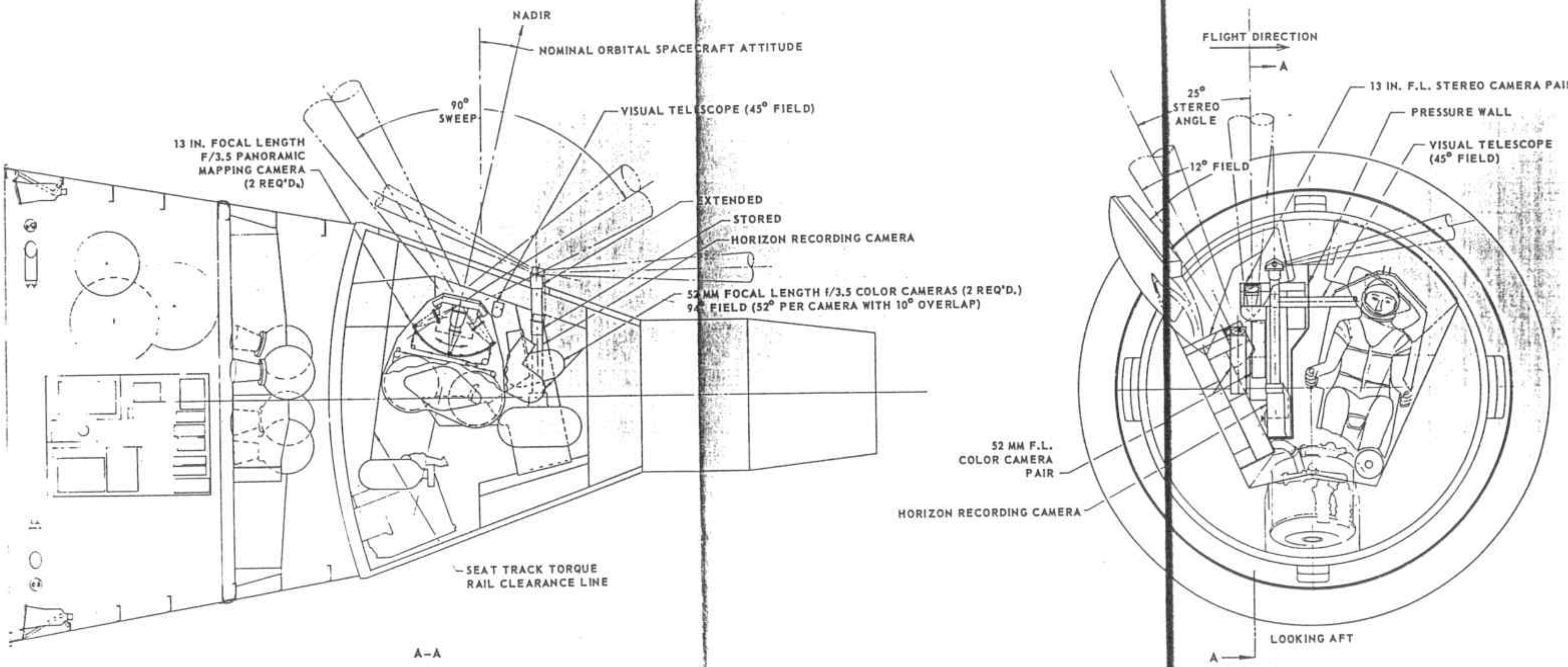


FIGURE 3.2-1

EARTH SURFACE MAPPING CAMERA INSTALLATION
ONE-MAN GEMINI

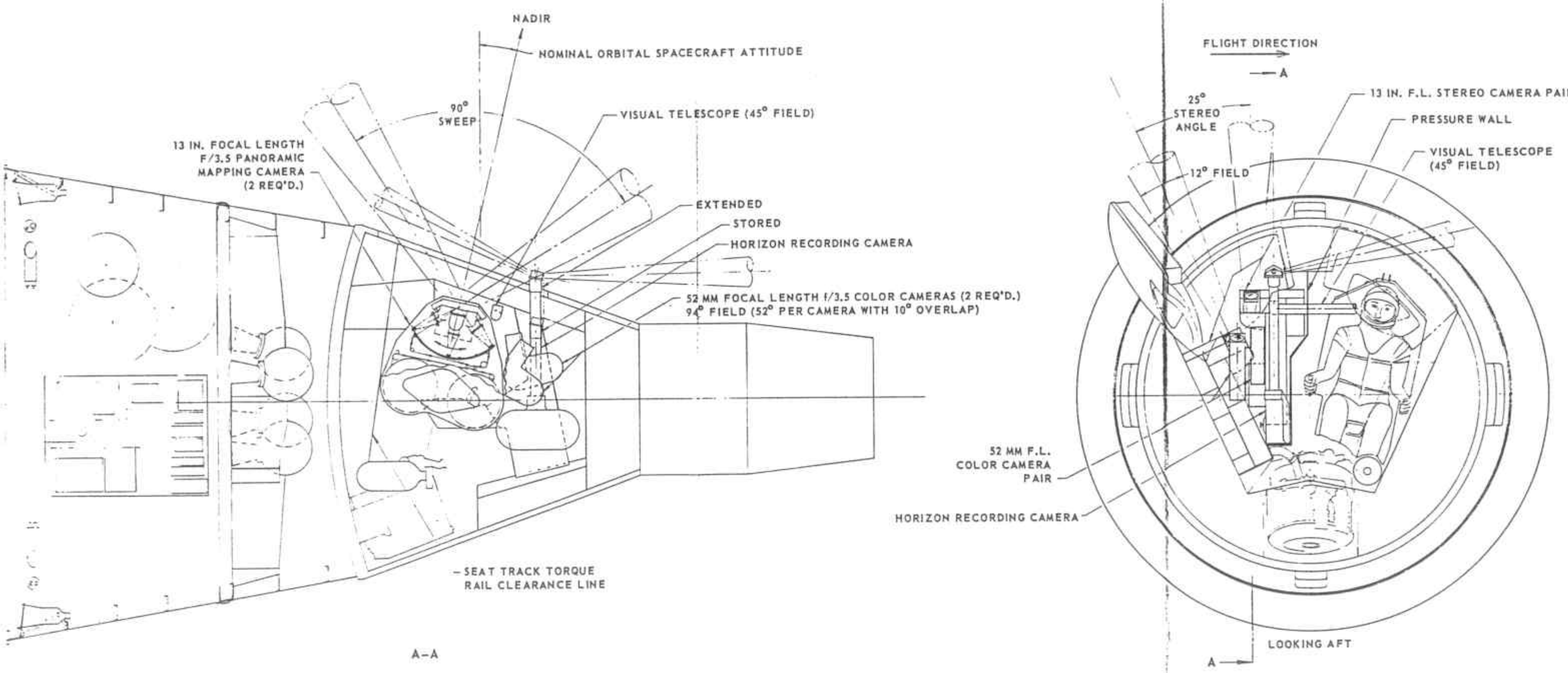


FIGURE 3.2-1

TABLE 3.2-1

MISSION EQUIPMENT CHARACTERISTICS
ONE-MAN GEMINI EARTH SURFACE MAPPING

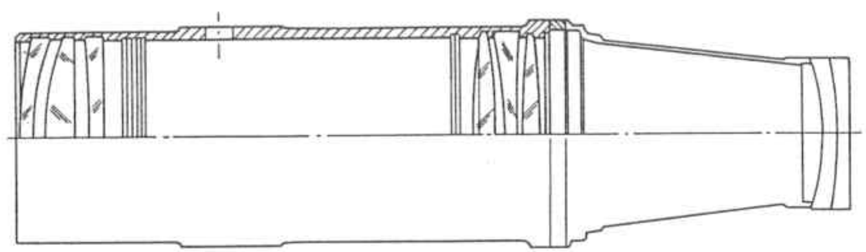
	QUANTITY	WEIGHT LB.-EACH	FIELD OF VIEW	FORMAT INCHES	PERFORMANCE @ 120 N.M.	FILM WT. LB.	MAPPING AREA 10% OVERLAP @ 120 N.M.	STEREO
PANORAMIC CAMERA – 13 IN f.l., f/3.5 PETZVAL FOLDED LENS, SO-206 FILM	2	75 W/O FILM 107.7 W/FILM	$10^\circ \times 90^\circ$	2.25 × 20.4	22.5 FT. RES. @2:1 CONTRAST	32.7 EA.	14.3×10^6 N.M. EACH CAMERA	YES 25° ANGLE
HORIZON CAMERA	1	20 W/O FILM 29 W/FILM	7°	2.25 × 2.25	<5 MIN.	9	COVER PAN PHOTOGRAPHY	–
COLOR CAMERA – 52MM f.l. f/3.5, FRAME	2	6 W/O FILM 9 W/FILM	52° EACH $+90^\circ$ CROSSTRACK BOTH	2.25 × 2.25	375 FT.	6	COVER PAN PHOTOGRAPHY	YES 55% O.L. FORWARD
VIEWFINDER – × 1.5 MAGNIFICATION YAW RETICLE	1	10	45° TRUE FIELD	–	–	–	–	–
HORIZON SENSOR	1	10	–	–	–	–	–	–

NOTE: SYSTEM POWER CONSUMPTION RUNNING IS 250 WATTS, WHEN SYSTEM ON STANDBY, THERMAL CONTROL CONSUMES 50 WATTS

Itek/TECHNICAL BULLETIN

13-inch, f/3.5 PETZVAL LENS

13-inch, f/3.5 PETZVAL LENS

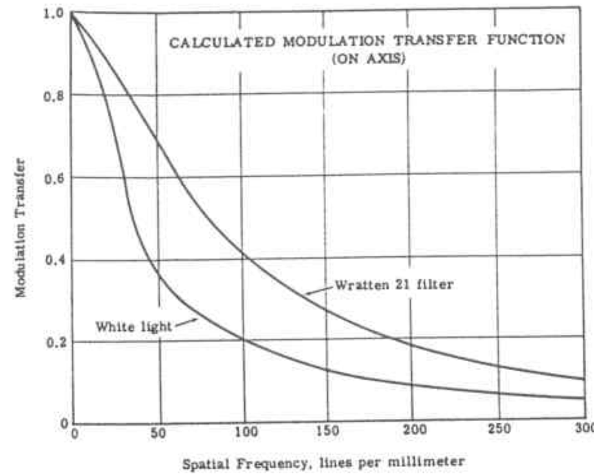


Itek's 13-inch, f/3.5 Petzval lens is an extremely high acuity lens that provides a high contrast AWAR of over 235 lines per millimeter. Low contrast AWAR is over 150 lines per millimeter. Performance is excellent in either white light or over more limited spectral regions.

Designed primarily for use in photographing distant objects, the lens uses a field flattened Petzval design of eight elements, and in a magnesium cell, weighs 6 pounds.

The size of the lens and the negligible distortion of the system make it suitable for frame, strip, or panoramic camera applications. It can also be produced in other focal lengths, and is available either in the straight configuration shown above, or folded for special applications.

Further information can be obtained by calling or writing: Government Systems Marketing, Itek Corporation.



CHARACTERISTICS

Lens type	Field flattened Petzval, 8 elements
Relative aperture	f/3.5, T/3.8
Field of view	±6°, 12° total
Image format	$2\frac{1}{4} \times 2\frac{1}{4}$ inches
Spectral range	0.52 to 0.70 μ
Equivalent focal length	13.0 inches
Back focal length	0.7 inch
Overall length	17.06 inches
Overall diameter	4.6 inches
Overall weight	6 pounds
Transmission	84 percent (axially)
Distortion	Less than 3 μ across the field

PERFORMANCE

Wratten 21 Spectral Range (0.52 to 0.70 μ)

	Resolution*	
Angle, degrees	High Contrast Target (1,000:1)	Low Contrast Target (2:1)
0	250	160
2	250	160
4	250	155
6	180	125

* All resolution figures are given in lines per millimeter using Eastman Kodak 4404 film and MIL-STD $\sqrt[4]{2}$ targets.

FIGURE 3.2-2

3.2.1 (Continued)

between camera frames and velocity accurately known, a baseline is established which in conjunction with a mechanically controlled camera convergence angle of 25° permits uncertainties in altitude (and scale) to be minimized by analysis of stereoscopic parallax between matching stereo photos. Coupled with a few arc-minutes error in local vertical from a horizon camera, mapping accuracies of 1000 feet are anticipated.

Film used is SO-206 thin base on 18 inch diameter reels. The reel diameter in the right hand crew compartment is the limiting factor rather than film weight. Sufficient film is carried to map 14.3 million square nautical miles in stereo from 120 n.m.

Horizon Camera (1) - A 70 m.m. frame camera which views the horizons is slaved to the pan cameras. Local vertical to a few minutes of arc is ascertained at the instant of each pan camera sweep center. A boresight center reticle is exposed on each horizon segment of the frame so that attitude excursions can be compensated in pan camera frame nadir point determinations.

The horizon camera head is retracted during launch, and is extended in orbit to provide a clear view of the horizon quadrants. Upon completion of the mission, the head may be either retracted or jettisoned to permit hatch closure.

Sufficient film is carried to correspond to complete exhaustion of the pan camera film (one frame/pan camera frame).

The design is similar to that of the horizon camera built by Wild of Heerbrugg.

70 m.m. Color Cameras (2) - Continuous color coverage is provided for the areas mapped by the pan cameras by the inclusion of two-52 m.m. focal length f/3.5, 70 m.m. color cameras. The cameras, each with a 52° field of view, look down and to the side with a slight nadir lateral overlap of about 5°, thus providing more than 90° lateral viewing using both cameras. Forward overlap of 55% is

3.2.1 (Continued)

is programmed so that stereoscopic color analysis can be performed. Sufficient film is carried to map the entire area mapped by the pan cameras. A resolution of slightly greater than 400 ft. from a 120 n.m. altitude is obtained.

Viewfinder (1) - The astronaut's viewfinder consists of a simple optical train using periscopic wide field optics. A real field of 40° at a magnification of 1.5 is employed to enable the astronaut to look down at part of the field to be mapped. The entire system is operated manually over areas of extensive cloud cover to conserve film. The viewfinder will include a yaw reference reticle for manual control of yaw attitude. The astronaut controls the spacecraft in yaw to within a few degrees of the nadir ground track.

Control Panel (1) - A simple control panel for astronaut operation consisting of on-off-warmup system positions as well as a display of film used is included. A clock display and up-date feature are essential to proper system operation, since operation is manually intiated and stopped. One control knob is used manually to insert ground supplied V/h for control of image motion compensation and the camera intervalometer.

3.2.2 Orbital Mapping Considerations - Selection of the orbit characteristics for the mapping mission is based on several overlapping factors which need to be evaluated to assure a practical and efficient mission. The following paragraphs discuss some of these considerations for a circular orbit with an altitude between 100 and 160 n.m. A sensor with a total lateral field of view of 90° with its line of sight pointed along the local vertical is assumed.

The orbit-to-orbit and the day-to-day shift of the spacecraft ground track with respect to a reference point on the earth is determined by the orbit period.

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 Spacecraft Characteristics - 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.