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GEMINI SPACECRAFT

ADVANCED MISSIONS (U)

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MCDONNELL AIRCRAFT CORPORATION

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1. INTRODUCTION

Nine advanced missions, or experiments, for the Gemini Spacecraft are discussed in this report. In Section 2, a qualitative, narrative discussion of the following aspects of the missions is presented:

1. Title
2. Description
3. Technical or Scientific Benefit
4. Effects on U.S. Space Program
    (a) Apollo
    (b) AES and Advanced Missions in General
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5. Prestige Value
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10. Impact on Gemini Program
11. Other Aspects

Additional technical detail on each of the missions is presented in Section 3. The missions are summarized, and cost and schedule information are presented in Section 4.

The information presented is more comprehensive for some missions than for others, reflecting differences in background information available and previous work performed in related areas at McDonnell. The cost and schedule information presented herein are for planning purposes only and apply to efforts associated directly with the experiments and spacecraft. Launch vehicle availability is assumed and the costs that might be incurred on launch vehicles are not included.

2. ADVANCED MISSIONS

2.1 Rendezvous with an Unmanned Satellite

2.1.1 Description - The objective of the flight is to rendezvous with a non-cooperative target, namely the Pegasus satellite, photograph the meteoroid puncture panels to corroborate telemetered data, and remove and return a piece of one of the panels by extravehicular activity, if possible.

The basic mission 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 contact is made. An alternate plan would be to first rendezvous with an Agena and then use the Agena propulsion for the open loop transfer to the slow catch-up orbit. The Agena would then be discarded and a second closed loop rendezvous performed with Pegasus. The alternate is operationally complicated and would not be considered unless more extensive analysis shows the basic plan unworkable or undesirable.

After rendezvous is completed, a slow pass is made to photograph the meteoroid puncture panels. After the photograph run is completed, the two craft are "docked" and a crewman secures the specimen of the panel by EVA.

2.1.2 Technical or Scientific Benefit - The greatest benefit of the mission is the accomplishment of rendezvous with a non-cooperative target, thus opening the possibility of obtaining additional data using spacecraft with this capability. The information returned from the Pegasus should provide, in addition to substantiating data received from it by telemetry, direct information of the effects of meteoroid impacts on structures for use in future designs.

2.1.3 Effect on U.S. Space Program - The experience obtained would be directly applicable in the areas of satellite data retrieval, resupply, maintenance, repair, and recovery. The knowledge of meteoroids and their impact with a spacecraft

2.1.3 (Continued)

should be considerably increased, which will benefit the Apollo and other future space programs.

The mission will provide information for DOD directly applicable in the areas of satellite interception, inspection, and surveillance.

2.1.4 Prestige Value - The return of a piece of a spacecraft from orbit, a feat which has yet to be accomplished, would demonstrate advanced space skills and carry implications of an ability to exercise access to any orbiting object at will.

2.1.5 Performance Feasibility - Preliminary analysis shows that if the OAMS is augmented by the addition of two sets of tanks and four 100 lb. thrusters, and if some of the additional ΔV capability is used to extend the GLV payload capability, rendezvous with Pegasus is possible. Analysis shows that 860 lbs. of the 1860 lbs. of propellant loaded in the OAMS at liftoff will be used to inject the Gemini into a 87-100 na. mi. orbit. The remaining 1000 lbs. of propellant should be sufficient to complete the mission. The weight change associated with the change in spacecraft configuration and propellant loading, and the use of eight rockets for retrograde from the Pegasus orbit, are considered in the analysis.

Extensive analysis of: (1) injection performance, (2) rendezvous with a spacecraft in an elliptic orbit, and (3) of retrograde and re-entry will be required to more definitely establish the ΔV capability of the spacecraft and the ΔV required for rendezvous and retrograde.

2.1.6 Cost Feasibility - The first unit cost is estimated to be $19.75 million with each additional unit costing $1.75 million, plus the cost of the spacecraft and launch vehicle. (See Section 4.)

2.1.7 Schedule Feasibility - It is estimated that a Gemini could be modified in approximately 20 months.

2.1.8 Operations Feasibility - The effect of the changes required, OAMS augmentation and possible computer program changes should make little difference in over-all ground operations. Flight operations are similar to those for Gemini and are expected to be straightforward.

Radiation hazard over the South Atlantic may require flight operations at the Pegasus altitudes to be performed when the orbit does not enter this zone, according to a radiation analysis conducted using two different radiation models.

2.1.9 Impact on Gemini Program - The impact of the Pegasus mission on the Gemini Program would be principally that of sustaining a fairly sizeable engineering effort to accomplish the propulsion system configuration changes and to insure the integrity of the associated structural changes. In addition, changes to checkout equipment and AGE would have to be made. The magnitude of these changes have not been ascertained to date.

Aside from the actual hardware changes, the suitability of: (1) the Gemini scheme of rendezvous when in elliptic orbit, (2) the re-entry control scheme when re-entering from high orbits, and (3) the launch guidance back-up for controlling an OAMS augmented injection will have to be established. If any of the three were to give unsatisfactory performance, a new scheme would have to be devised, programmed, and procedures revised.

2.2 One Man Gemini-Earth Surface Mapping

2.2.1 Description - Low latitude earth surface photography applicable to topographic mapping, geological reconnaissance, and to studies of oceanography, hydrology, and meteorology, can be accomplished with a GLV launched One Man Gemini. A camera system which can be mounted in the right hand side of the crew compartment can provide useful resolution and mapping accuracies. A mission duration of 7 days allows complete coverage at the equator with 50 percent overlap. Low resolution auxiliary color cameras provide for correlation between color tonal gradations and the higher resolution black and white pictures.

The Gemini is flown upside down and sideways, with the right-hand hatch door open to expose the thermally controlled, unpressurized camera system. Mapping camera system design is such that close tolerance attitude, attitude rate, or image motion compensation control are not needed. However, a horizon camera is included to obtain precisely the local vertical when imaging the nadir. An auxiliary horizon scanner for coarse pitch and roll reference is also included. Manual yaw sensing and control to the ground track is employed.

Gemini ground tracking network data are used in conjunction with post flight photogrammetric data reduction. The mission is considered to have high confidence of success since reliable, relatively simple, state-of-the-art hardware is used throughout.

2.2.2 Technical and Scientific Benefit - The mission would be of scientific and technical benefit, in terms of topographical mapping of underdeveloped areas, determination of oceanographic characteristics, better definition of Earth folliation outlines, and geological refinements.

2.2.3 Effects on U.S. Space Program - Advanced missions in general would be aided by the experience gained with orbital mapping techniques and interpretations.

2.2.4 Prestige Value - International prestige would be enhanced by the scientific and commercial contributions to be gained by the increased accuracy and detail of the large scale mapping.

2.2.5 Performance Feasibility - Since the estimated mission performance is based upon use of existing equipment and technology, feasibility is not considered to be in question. The payload requirements are within GLV capability.

2.2.6 Cost Feasibility - The first unit cost is estimated to be $10.3 million with each additional unit costing $1.35 million, plus the cost of the spacecraft and launch vehicle. (See Section 4.)

2.2.7 Schedule Feasibility - It is estimated that about 24 months would be needed to perform the necessary engineering and fabrication.

2.2.8 Operations Feasibility - Ground tracking and monitoring for orbital period control is obtainable with the existing ground network. One Man Gemini operations have been previously studied at McDonnell and are considered quite feasible.

2.2.9 Impact on Gemini Program - The 24 months acquisition time would result in a delay of a few months if the mission were scheduled for spacecraft number 12. If a refurbished spacecraft were utilized, the mission could be accomplished without interference to the Gemini program.

2.3 One-Man Gemini with Astronomical Telescope

2.3.1 Description - Two types of telescope installations for the one-man Gemini were examined: (1) a 16-inch diameter telescope located in the right-hand crewman's seat, and (2) a 26-inch diameter telescope mounted in the adapter with access to the telescope through a hatch in the heat shield. Both installations are discussed in Section 3.3. Although minimum change to Gemini is a ground rule for the missions presented in this report, it is felt that the adapter-mounted design offers sufficiently greater return, scientifically and technologically, to make it the preferred approach. The hatch in the heat shield should be a developed item by 1967 due to the Gemini B program. The adapter mounted telescope is discussed in this section.

A one-man Gemini spacecraft with a 26-inch diameter, 560 pound, astronomical telescope in the adapter can be placed in a 200 na. mi. circular orbit by the Gemini launch vehicle in 1968. The two mission goals are: (1) to demonstrate the ability to make astronomical measurements with a pointing accuracy of 0.1 arc-seconds for periods over ten minutes, and (2) to obtain new astronomical data.

To provide a steady vehicle base for the telescope, the altitude and attitude are chosen to keep the external disturbance torques on the spacecraft low. A fine attitude control system is added to stabilize the spacecraft in the presence of the low disturbance torques. A heat shield hatch and tunnel are added to provide pressurized access to the telescope, and to provide room for experiments on the isolation of astronaut motions from the spacecraft and telescope. The spacecraft roll is used for roll pointing of the telescope, and the single-axis gimbal on the telescope is used for pointing in pitch. Measurements on any star can be made either by selecting launch time or by operating in the presence of a large gravity gradient torque.

2.3.2 Technical and Scientific Benefit - The one-man Gemini experiment could remove one of the major obstacles to the advancement of space astronomy by demonstrating a practical means for eliminating undesirable effects of astronaut motions on telescope pointing angle stability. Pointing angle stability is essential to the development of a capability for orbiting observations with telescope apertures over 100 inches, utilizing resolutions better than 0.04 arc seconds, for purposes such as seeing faint stars never before observed and for searching for planets associated with nearby stars.

The optical alignment and adjustments for such large observatories involves many tasks which are facilitated by manned operation. However, the astronaut disturbances must be kept small. Three techniques for keeping astronaut disturbances small are: (1) isolation of the telescope, (2) isolation of the astronaut in the observatory or in a separate spacecraft, and (3) compensation for the disturbances. The use of a separate spacecraft is not appealing due to time lost in transfer. Compensation for maximum likely disturbance torques caused by body or limb motions is difficult because the torques are large and change greatly in a short time interval. The isolation of the telescope from the spacecraft is difficult for the very large optics since the telescope is essentially part of the spacecraft.

The isolation of the astronaut from the spacecraft and telescope in a controlled floating or elastic support in the spacecraft offers a promising way of simplifying the problem of precise attitude control of the manned spacecraft. The astronaut could perform direct viewing, visual alignment, focusing, star acquisition, and other functions while mechanically, but not visually, isolated from the telescope. In the extreme, a controlled floating crewman support can be mechanically reaction balanced using drive signals from photo detectors which measure the support position with respect to the spacecraft. The use of an adapter mounted telescope and associated access tunnel permits testing crewman isolation techniques in the one-man

2.3.2 (Continued)

Gemini. In addition, techniques for compensation for disturbances can be evaluated.

The scientific benefits of the one-man Gemini include visual planetary observations and photographic plate and film data return.

2.3.3 Effect on U.S. Space Programs - The experiment provides a desirable foundation for manned orbiting observatories having large, greater than 100-inch, apertures.

2.3.4 Prestige Value - A "first" would considerably enhance prestige, particularly in the scientific community. Considerable additional prestige would accrue by publication of photographs hitherto unavailable, particularly if new astronomical phenomena were discovered.

2.3.5 Performance Feasibility - The 0.1 arc-second pointing angle stability for the telescope line of sight is considered to be a reasonable design goal. Payload requirements are within GLV capability.

2.3.6 Cost Feasibility - The first unit cost is estimated to be $53.3 million, with each additional unit costing $4.9 million, plus the cost of the spacecraft and the launch vehicle. (See Section 4.)

2.3.7 Schedule Feasibility - It is estimated that 30 months are required from go-ahead to delivery for the adapter mounted version.

2.3.8 Operations Feasibility - One-man operation of Gemini has been previously studied at McDonnell and appears to be quite feasible. The hatch in the heat shield will have been developed for Gemini B. Otherwise, operations are similar to those for Gemini.

2.4 Artificial Gravity Experiment

2.4.1 Description - Three methods of providing centrifugal force for artificial gravity with a Gemini spacecraft were investigated: (1) rotation of the Gemini directly connected to the burned-out Stage II of the GLV (Gemini Launch Vehicle), (2) rotation of the docked Gemini-Agena vehicle, and (3) rotation of a cable-connected Gemini and either the Agena or Stage II of the GLV.

The first method provides an eyeballs-out g force, unless the crewmen are repositioned, and the second method provides an eyeballs-in g force. Only the third method, by using the paraglider bridle, affords a means of providing a spinal g force. In addition, the cable-connected method provides a larger rotation radius which results in lower Coriolis effects. The cable system results, however, in increased weight, design, and operational complexity. The second method involves a rendezvous with an Agena. For each method, spin-up would be accomplished manually using the appropriate attitude control or maneuver thrusters.

In each concept, it will be of interest to consider several directions of spin. Because of Gemini cockpit confinement, spin about the various human body axes is accomplished by spinning the orbiting vehicle in various attitudes.

2.4.2 Technical Benefit - The test would provide data on the effects of in-space artificial gravity by rotation, increase the yield from data from subsequent investigations conducted on earth, and help substantiate or repudiate the need for, or amount of, artificial gravity provisions in future space station designs.

2.4.3 Effect on U.S. Space Program - Advanced manned missions, including Apollo Extension Systems and Manned Orbital Laboratories for extended periods of time, will derive the technical benefit cited. The Gemini test will provide data for making decisions concerning artificial gravity provisions at an earlier stage of development of advanced manned missions.

2.4.4 Prestige Value - Prestige would undoubtedly result from being first to produce artificial gravity in space and establish meaningful design criteria.

2.4.5 Performance Feasibility - The operational conditions studied indicate that methods using Gemini Orbit Attitude and Maneuver System (OAMS) thrusters are feasible if application to operations other than spin or despin are limited. It is anticipated that the maneuver thrusters will be used for both rendezvous and for artificial gravity. For non-rendezvous, methods any of the thrusters can be used for artificial gravity. A major portion of the design life of the lateral and vertical thrusters can be used for artificial gravity. The Agena attitude control system also could be used with additional storage of the cold gas propellant.

It is estimated that the experiment can be conducted within the GLV and Atlas-Agena payload capabilities.

2.4.6 Cost Feasibility - For the two methods involving direct coupling to either the GLV or Agena stages, the first unit cost is estimated to be \$3.75 million with each additional unit costing \$0.25 million. For the cable connected system, with either the Agena or GLV stage, the first unit cost is estimated to be \$22 million with each additional unit costing \$2.0 million. Costs of spacecraft and launch vehicles are to be added. (See Section 4.)

2.4.7 Schedule Feasibility - Flights with the directly-connected vehicle methods appear to be readily feasible early in the current Gemini program. The cable method is estimated to require a 15-month development period. However, with any one of the approaches, the experiment could be accomplished within the present Gemini schedule, assuming an immediate go-ahead.

2.4.8 Operational Feasibility - The directly connected vehicles result in an operationally simple method of producing centrifugal force for artificial gravity. The thrusters are used directly to obtain the angular velocity. Opera-

2.4.8 (Continued)

tions are similar to those involved in a normal rapid slew maneuver. Thus, the docked Gemini-Agena concept appears feasible and achievable at an early date. Rotation of the Gemini directly connected to Stage II of the GLV, with the crew members positioned to minimize adverse physiological effects, presents complications in viewing the required displays and in controlling the entire vehicle.

A relatively complex operation is required with the cable system. This operation includes cable attachment; cable reel-out to the extended position, re-orientation of the Gemini capsule, spin-up to the artificial gravity level while keeping cable slack from becoming excessive, and disconnecting the cable at the end of artificial gravity operation. However, previous McDonnell studies have shown this method to be completely feasible.

2.4.9 Impact on Gemini Program - The impact on the Gemini program should be minor, as discussed in Sections 2.4.6 and 2.4.7.

2.5 Simulation of LEM Rendezvous

2.5.1 Description - The rendezvous of the Lunar Excursion Module (LEM) of the Apollo spacecraft with the Command Module (CM) in lunar orbit is one of the essential elements of the return phase of the Apollo lunar mission. Therefore an early earth-orbit test of the LEM rendezvous equipment and technique would be highly desirable. However, the LEM does not possess a re-entry or launch vehicle escape capability so a manned launch would be extremely risky. Automating the LEM probably would not provide a very satisfactory simulation since in the actual Lunar Orbit Rendezvous (LOR) heavy reliance is made on manual methods and human capabilities. A Gemini spacecraft, with its inherent re-entry and escape capability, could be used as a test bed for early simulation of the LEM LOR technique. Such a test flight would involve outfitting a Gemini spacecraft with LEM equipment to perform computations and radar observations, or modifying Gemini equipment in order to simulate the LEM rendezvous operations. A rendezvous target with actual Apollo CM equipment or equivalent hardware would either be launched separately or carried into orbit aboard the Gemini.

2.5.2 Technical Benefit - The guidance laws of LEM rendezvous have already been evaluated by digital computer simulations during the development of the rendezvous method. Also, detailed studies using statistical models of LEM hardware have been performed to predict the effects of hardware limitations on the lunar rendezvous. Human factors and other unpredicted variations could be discovered by an early test with man in the loop. The Gemini test flight would improve the technical understanding of LEM rendezvous by providing a test using actual hardware in a manned, space environment. The experiment would be of greatest technical importance if actual Apollo CM-LEM equipment is used. Without this hardware, the test would nevertheless be quite important since it would demonstrate the soundness of the LEM rendezvous laws.

2.5.3 Effects on U.S. Space Program - It is difficult to evaluate these effects without a detailed knowledge of the Apollo flight test program. A Gemini flight test would provide a very early indication of possible deficiencies, and would be an asset to the Apollo program. If deficiencies should be discovered, early diagnosis and design "fix" would be possible.

2.5.4 Prestige Value - A successful flight test would be a demonstration of the soundness of a key element for success in the United States lunar exploration program.

2.5.5 Performance Feasibility - The effect of the additional weight of LEM systems aboard the Gemini could, if necessary, be compensated by either off-loading maneuver propellant or shortening the mission, although it appears this would not be necessary. If LEM hardware is not used, there would not be a significant weight increase. The present Gemini orbit propulsion system would provide adequate propulsion performance for LEM rendezvous evaluation.

2.5.6 Cost Feasibility - It is estimated that the first unit cost would be $18.5 million with each additional unit costing $5.1 million if LEM equipment were used. If modified Gemini equipment were used, the first unit cost would be $7.5 million with each additional unit costing $3.1 million. Costs of spacecraft and launch vehicles are to be added. (See Section 4.)

2.5.7 Schedule Feasibility - Unknown if LEM equipment is used. A time from go-ahead of thirteen months seems reasonable with modified Gemini hardware, assuming the LEM guidance system description is firmly established at the time of go-ahead.

2.5.8 Operations Feasibility - With or without the utilization of LEM hardware, the proposed Gemini extension would be feasible from an operations viewpoint. Development of test requirements and procedures, data handling, and use of ETR and

2.5.8 (Continued)

the mission control center should be straightforward since the mission is a natural follow-on to planned Gemini Rendezvous missions.

2.5.9 Impact on Gemini Program - The test flight could be added as a piggy-back experiment to a Gemini flight in late 1966 or as the primary mission of an additional Gemini flight in early 1967. If the auxiliary computer tape memory unit is installed on later Gemini flights regardless of the LEM flight test requirements, the only cost increase would be in the development of a new flight tape to program the LEM system equations in the computer.

2.6 Structural Assembly in Orbit

2.6.1 Description - Man is capable of performing many useful extravehicular (EV) functions in space, such as assembly, maintenance, inspection and alignment of radiators, sensors, antennas, and propulsion systems, as well as serving as a back-up to automatic systems. Man's usefulness can be demonstrated by an experiment involving the structural assembly in orbit of a 40' diameter parabolic antenna, and the disassembly and recovery of a 100-lb. OAMS Thrust Chamber Assembly (TCA).

The antenna is folded and stowed on the nose of the Gemini for launch. An ascent fairing covers the entire assembly. One crewman leaves the Gemini with 30' extension umbilicals, and manually erects the antenna and extends the feed horn. Some structural modification to the Gemini nose to accommodate the increased launch loads is required. The surface of the antenna disk is an aluminum coated polyethylene mesh, irradiated, and formed in parabolic segments. Voids in the surface allow for passage of the crewman and afford clearer visibility for manual pointing. After erection, the antenna can be used for a variety of communication experiments.

The TCA is removed by the crewman and brought within the re-entry module. A special mounting to allow removal of the TCA is needed. Tube connectors are guillotined prior to egress, allowing time for cooling and dissipation of propellant downstream of the propellant isolation valves.

2.6.2 Technical or Scientific Benefit - Construction of the large communications antenna described would: (a) demonstrate the use of special design features to minimize assembly time, and (b) determine the effectiveness of this type of extravehicular operation.

The retrieval of the thrust chamber assembly simulates a type of repair activity which may be needed in future space operations. It would also establish the

2.6.2 (Continued)

feasibility of performing various tests of rocket motors and other equipment under orbital conditions, and returning the test specimen to earth for evaluation. These tests would be desirable in order to better establish the safety margin of thrusters or other spacecraft equipment.

2.6.3 Effect on U.S. Space Program - The demonstration that man can erect large structures in orbit is of significance for the planning of future missions.

Information derived from the proposed examination of the rocket specimens would be of advantage in designing propulsion systems for various space programs.

2.6.4 Prestige Value - The international and domestic prestige value of extravehicular activities has been demonstrated. U.S. prestige would be enhanced in this case since the experiments simulate operations of a practical nature.

2.6.5 Performance Feasibility - The experiment is considered to be feasible since advance in the state-of-the-art will not be required for the development of the erectable antenna. The payload is within the GLV capability.

2.6.6 Cost Feasibility - The first unit cost is estimated to be $16.75 million, with each additional unit costing $1.75 million, plus the cost of the spacecraft and launch vehicle. (See Section 4.)

2.6.7 Schedule Feasibility - It is estimated that it is feasible to have this mission ready for flight in approximately 16 months after go-ahead. The estimate allows for wind tunnel tests and structural proof tests.

2.6.8 Operations Feasibility - Considered entirely feasible since extravehicular capability should be proven prior to this mission.

2.6.9 Impact on Gemini Program - Re-design (beef-up) of nose section and increased emphasis on development of extravehicular capabilities of both the suit and re-entry module are needed.

2.6.10 Other Considerations - This mission can be logically combined with the Long Duration Gemini mission. The payload capability of the Agena allows for heavier and more complex (or more ambitious) structures than the antenna. Also, the erection of the tunnel/living quarters in the long duration mission constitutes a structural assembly in space experiment.

2.7 Propellant Transfer Tasks

2.7.1 Description - In-orbit transfer of storable propellants between tanks which do not utilize positive expulsion bladders can be accomplished with a minimally modified Gemini and Agena (Gemini Agena Target). The Gemini is of the rendezvous configuration. It is equipped with additional tankage to receive the propellant to be transferred from the Agena. These tanks, located in the adapter equipment section of the spacecraft, contain centrifugal separators which part the liquid from the vapor. The Agena is modified to accommodate the propellant supply tankage. These tanks have propellant collection or retention devices. The transfer lines are brought forward and connected by extravehicular (EVA) activity. Following transfer, the propellants are discarded with the adapter equipment section when it is jettisoned.

2.7.2 Technical or Scientific Benefit - The accomplishment of this task will provide the following technical benefits for future space missions:

A. Develop the EVA capability to accomplish the transfer line assembly and hook-up

B. Develop the capability to adapt to the use of special tools required for this assembly while in the EVA environment.

C. Evaluate the effectiveness of propellant separation devices (gas/fluid separator in receiving tank)

D. Evaluate the effectiveness of propellant retention devices (fluid retention in the supply tank).

E. Establish methods for quantity monitoring of fluid (propellant) remaining in the supply tanks and fluid accepted by the receiving tanks.

2.7.3 Effects on U.S. Space Programs - The technical benefits derived from this task which are applicable to future U.S. space programs are many. These include:

2.7.3 (Continued)

A. Early assessment of propellant transfer. Resupply will involve large quantities of propellants, thus necessitating the use of tanks without positive expulsion devices. The tanks used in this experiment are smaller than those which will be utilized in logistic resupply. However, they are sufficiently large to test and prove the basic techniques.

B. Early checkout of the Apollo LEM propellant quantity gauging system. System monitoring includes quantity monitoring of the supply tanks and the receiving tanks. The tanks used can be designed to the Apollo LEM diameter (12.5 in.) in order to utilize a prototype gauging system.

C. Assessment of man's capability to accomplish the assembly of transfer plumbing while in the EVA environment, including the handling of special tools and equipment.

2.7.4 Prestige Value - Accomplishment of the propellant transfer experiment will be viewed as a step in developing future space station and interplanetary capability.

2.7.5 Performance Feasibility - The feasibility of the mission is established by the use of scaled tanks to minimize weight, and by the use of existing gauging equipment. Modifications to the Gemini and Agena appear to be straightforward. Payload requirements are within the GLV and Agena capabilities. Because of the scale effect of the gauging devices, further analysis will be necessary to determine the exact test configuration.

2.7.6 Cost Feasibility - The first unit cost is estimated to be $20.5 million, with each additional unit costing $1.5 million, plus the cost of the spacecraft and launch vehicles.

2.7.7 Schedule Feasibility - It is estimated that approximately 24 months

2.7.7 (Continued)

will be needed to complete the modifications to the Gemini and Agena.

2.7.8 Operations Feasibility - The mission makes use of a standard Gemini/Agena rendezvous and docking. EVA will have been accomplished in previous Gemini flights. Therefore, new operational procedures, other than the actual manual coupling and subsequent propellant transfer, are not required.

2.7.9 Impact on Gemini Program - Since the acquisition time is estimated to be 24 months, delays would result if the experiment were incorporated on later Gemini flights. If a refurbished spacecraft were utilized, the mission could be accomplished without interference with the Gemini program.

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.

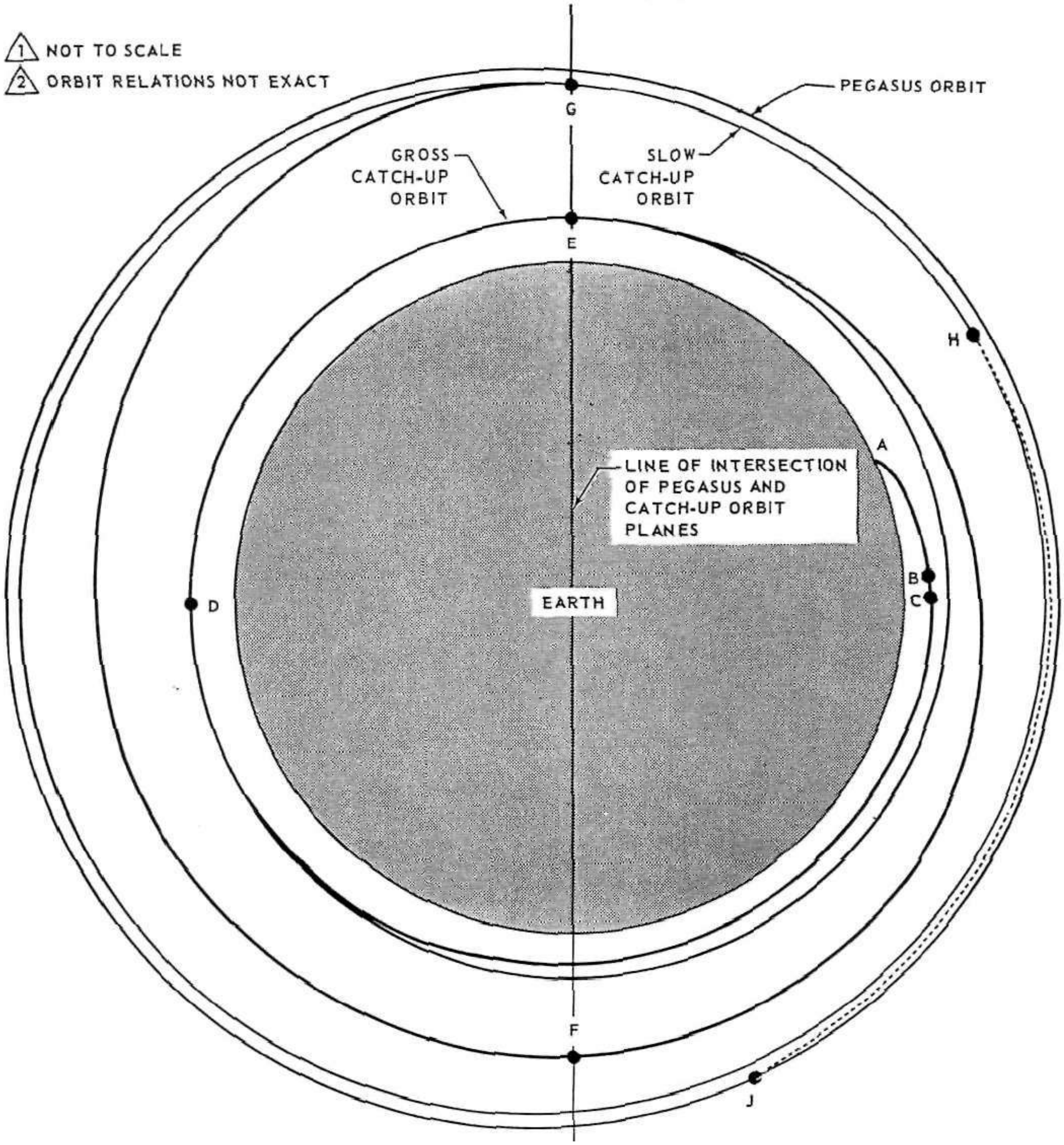
A preliminary estimate of the propellant required for the mission is shown in Table 3.1-1. This is also noted on Figure 3.1-2.

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° 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

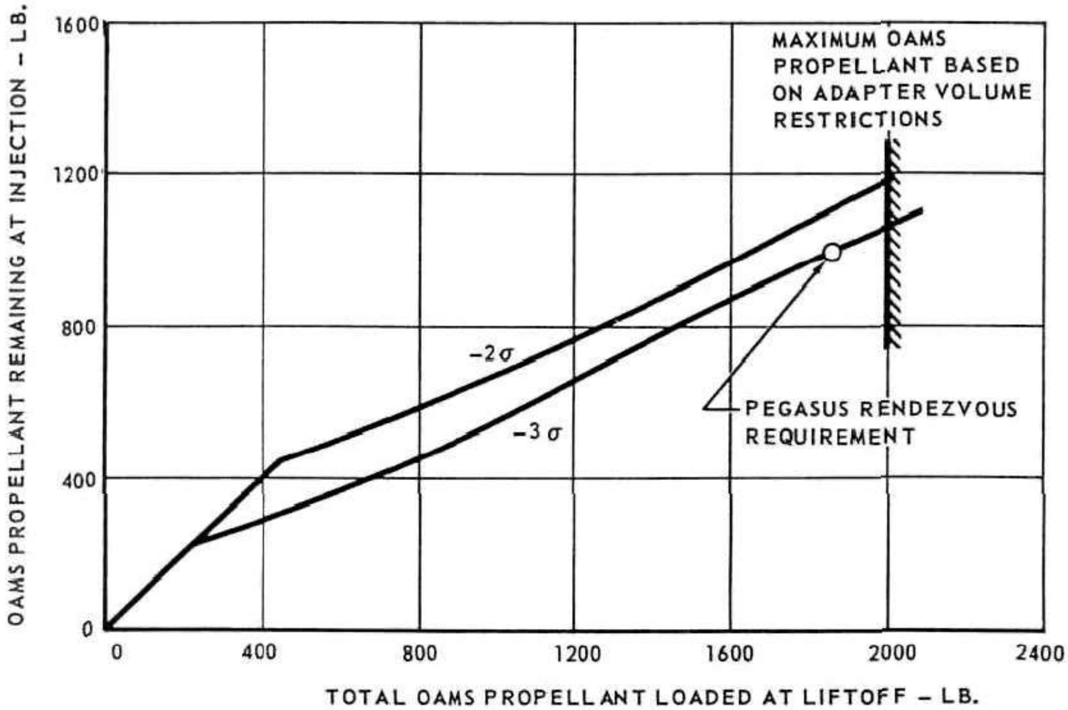


FIGURE 3.1-2

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.

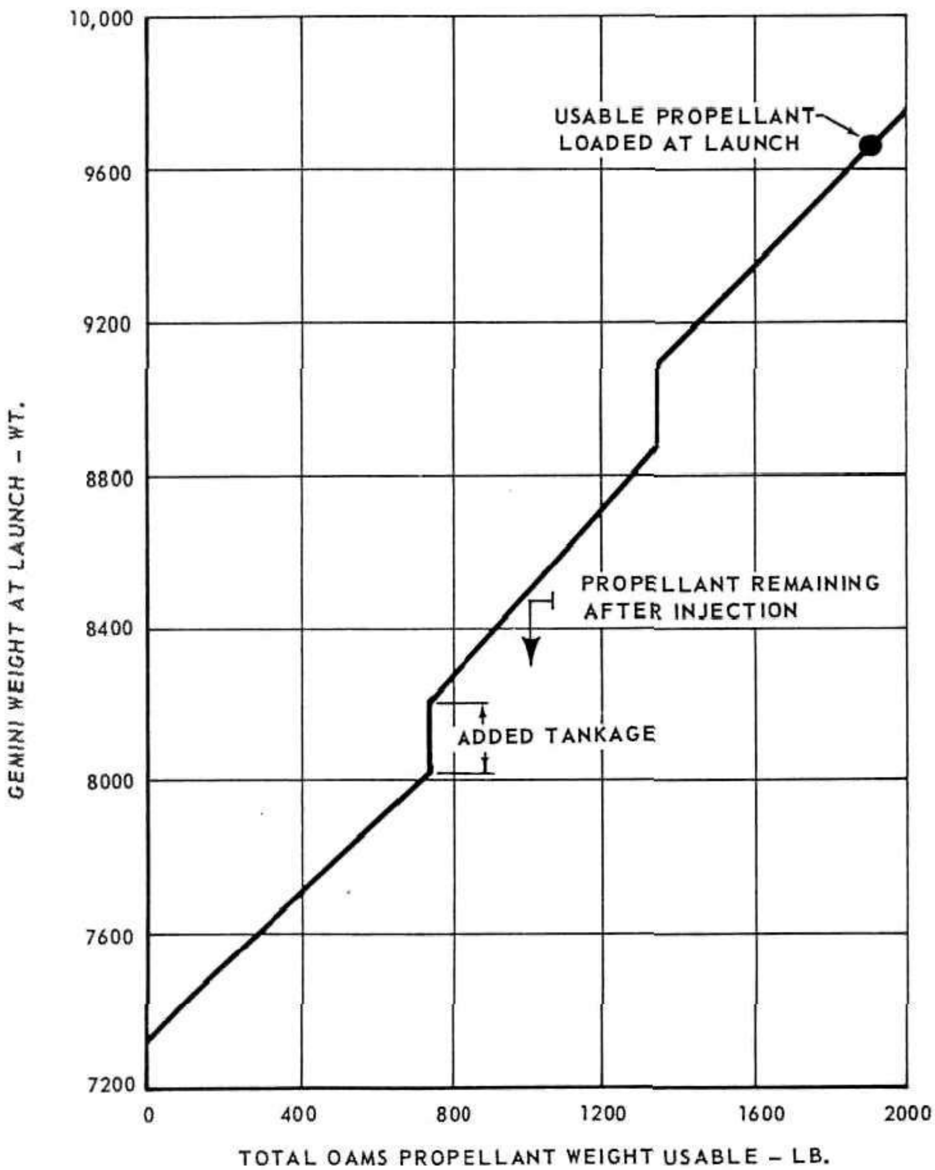


FIGURE 3.1–3

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

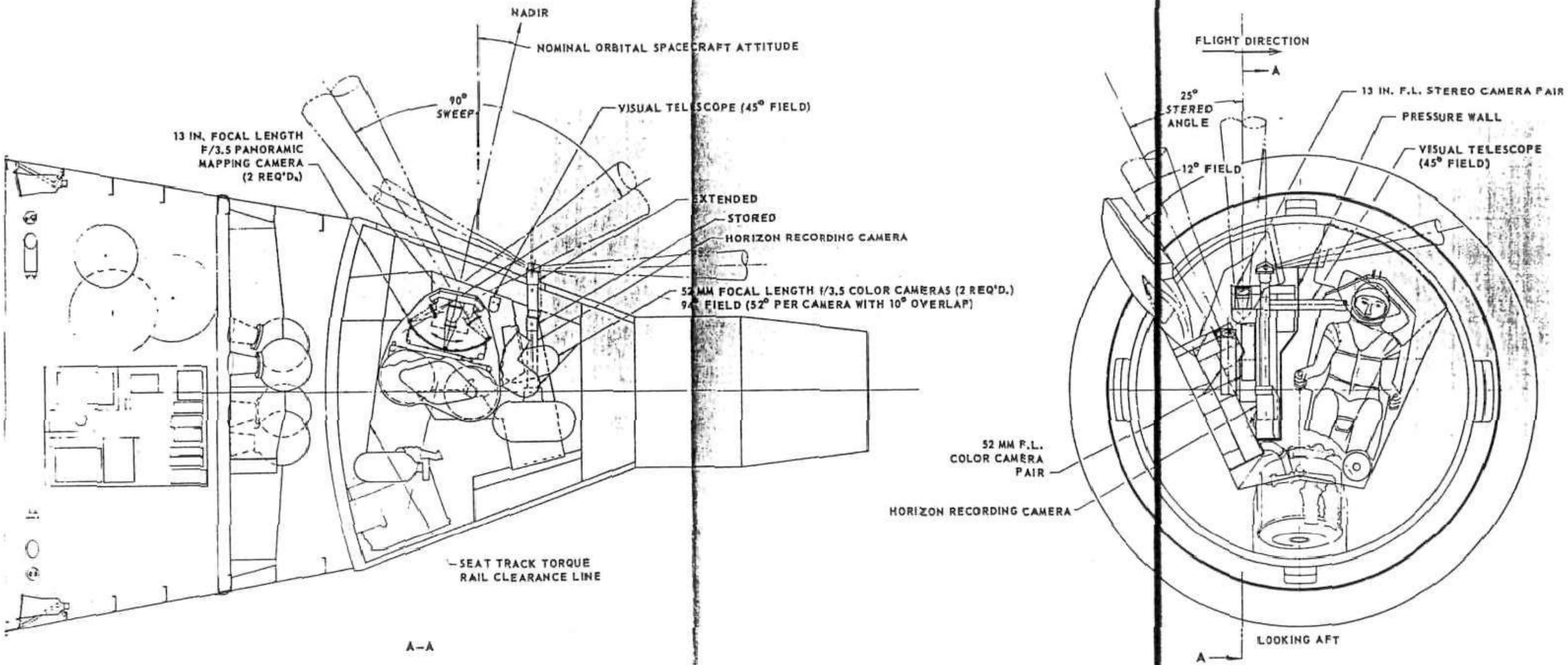


FIGURE 3.2-1

EARTH SURFACE MAPPING CAMERA INSTALLATION
ONE-MAN GEMINI

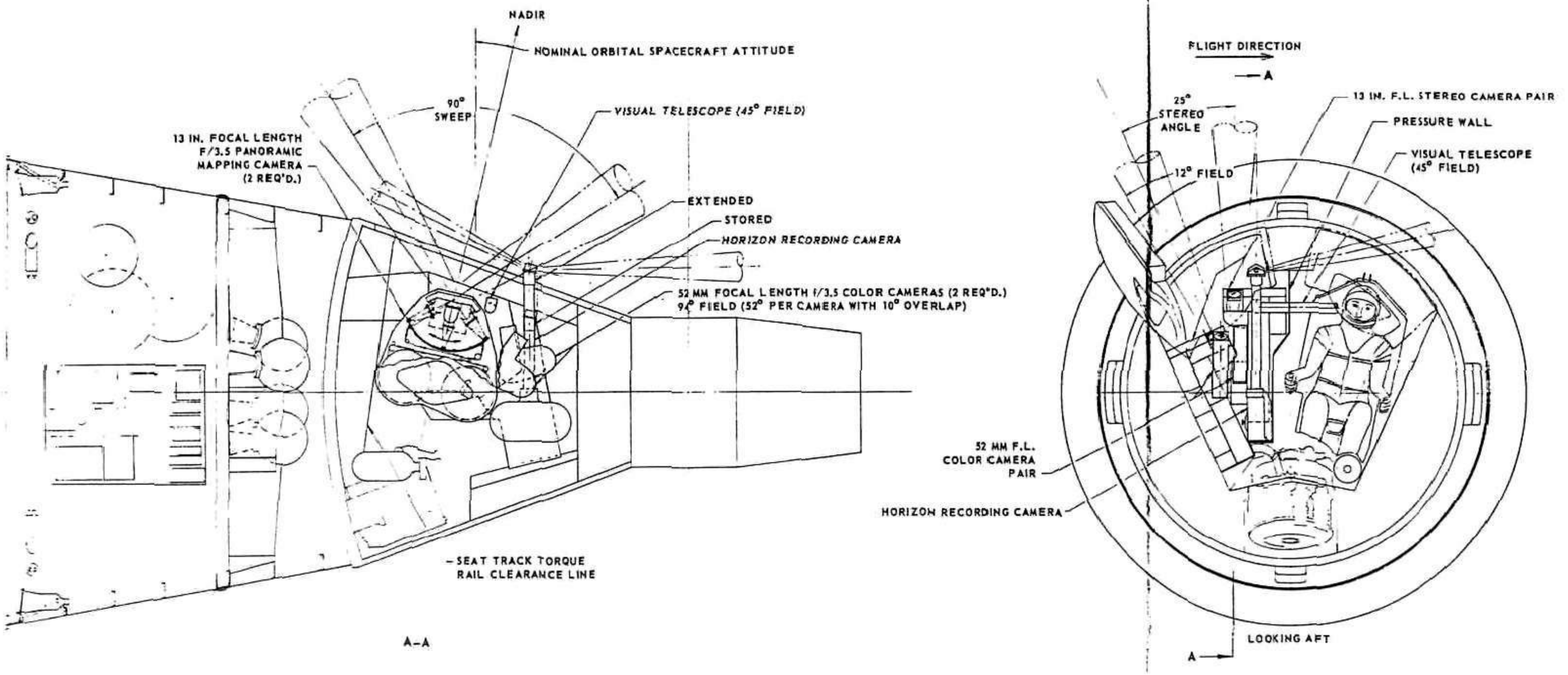


FIGURE 3.2-1

TABLE 3.2-1

MISSION EQUIPMENT CHARACTERISTICS
ONE-MAN GEMINI EARTH SURFACE MAPPING

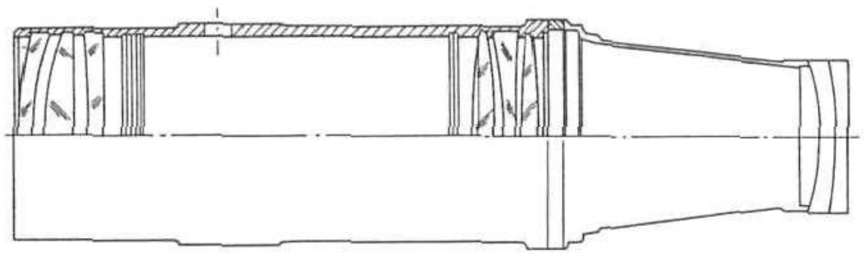
	QUANTITY	WEIGHT LB.-EACH	FIELD OF VIEW	FORMAT INCHES	PERFORMANCE @ 120 N.M.	FILM WT. LB.	MAPPING AREA 10% OVERLAP @ 120 N.M.	STEREO
PANORAMIC CAMERA – 13 IN f.l., f/3.5 PETZVAL FOLDED LENS, SO-206 FILM	2	75 W/O FILM 107.7 W/FILM	$10^\circ \times 90^\circ$	2.25×20.4	22.5 FT. RES. @2:1 CONTRAST	32.7 EA.	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 – $\times$ 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.

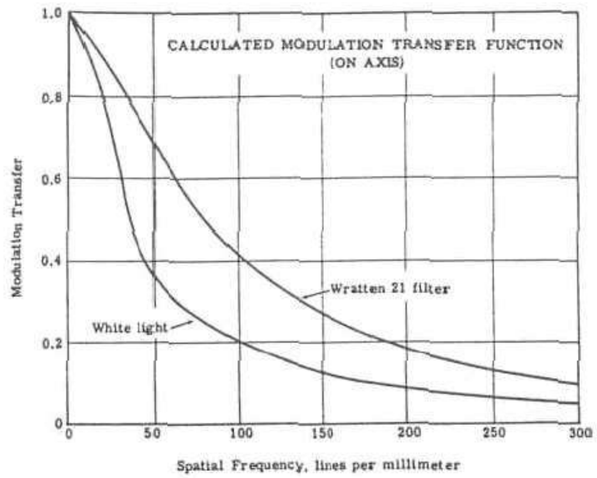
Designed primarily for use in photographing distant objects, the lens uses a field flattened Petzval design of eight elements, and in a magnesium cell, weighs 6 pounds.

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.

Itek

Itek Corporation
10 MAGUIRE ROAD, LEXINGTON 73, MASSACHUSETTS



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)

Angle, degrees	Resolution* 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^o and 45^o) 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^o and 45^o. 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.

ORBITAL PERIOD vs. ALTITUDE
(INCLUDING OBLATENESS EFFECTS)

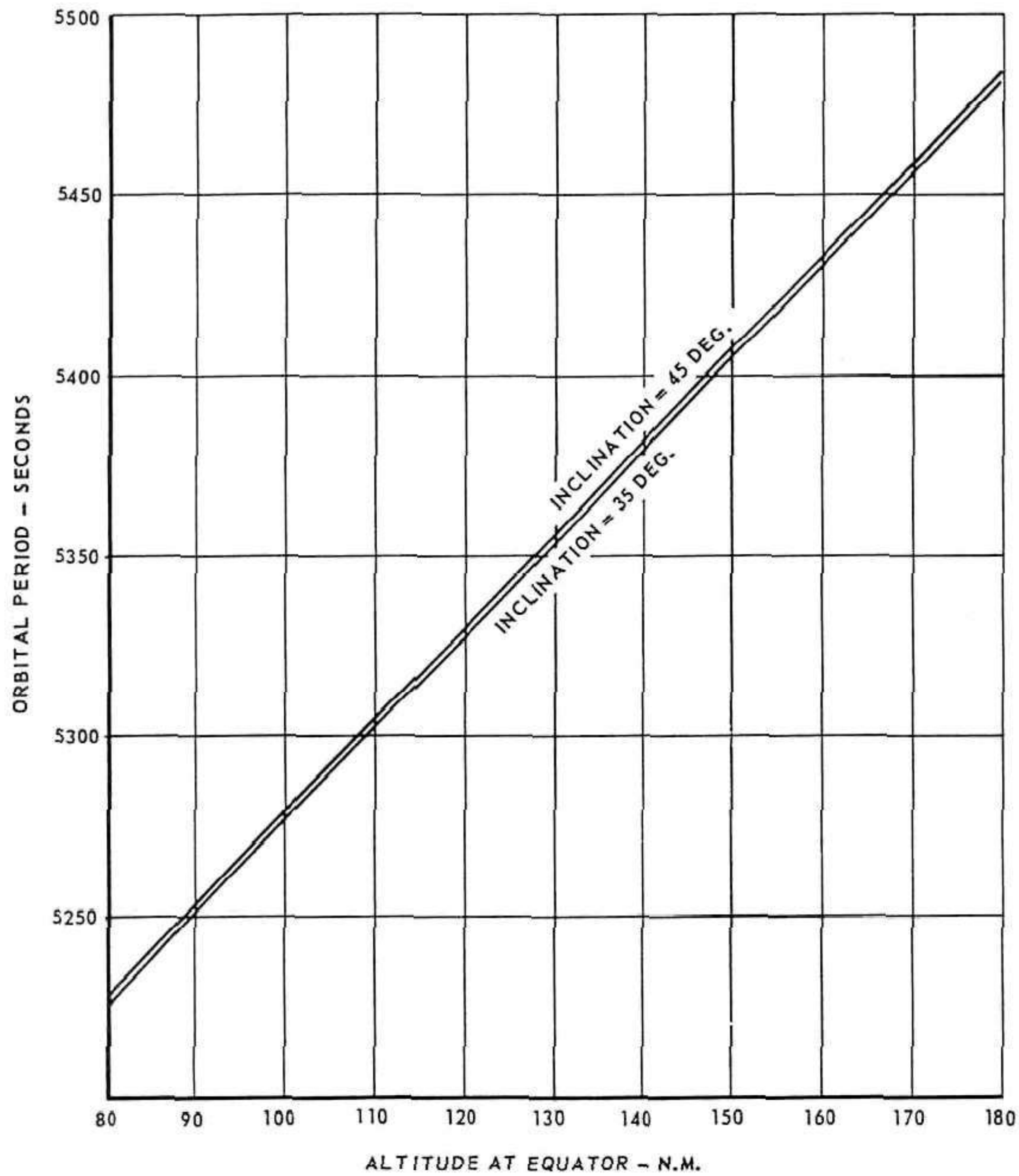


FIGURE 3.2-3

PERCENT OVERLAP vs ALTITUDE

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

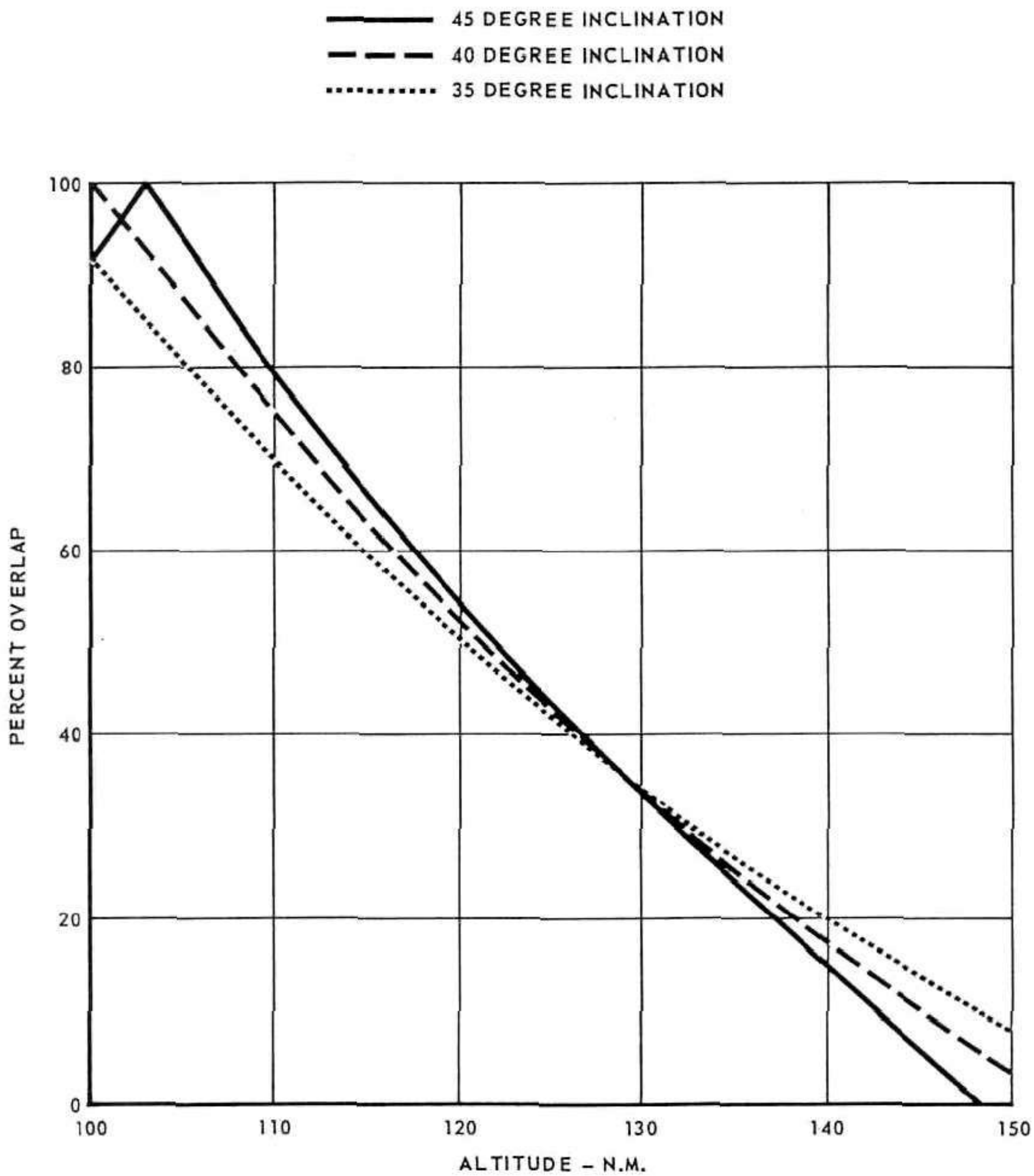


FIGURE 3.2-5

MISSION DURATION vs ALTITUDE
FULL COVERAGE AT ZERO DEGREE LATITUDE
NO ORBIT DECAY
CAMERA FIELD = ±45 DEGREE CROSSTRACK

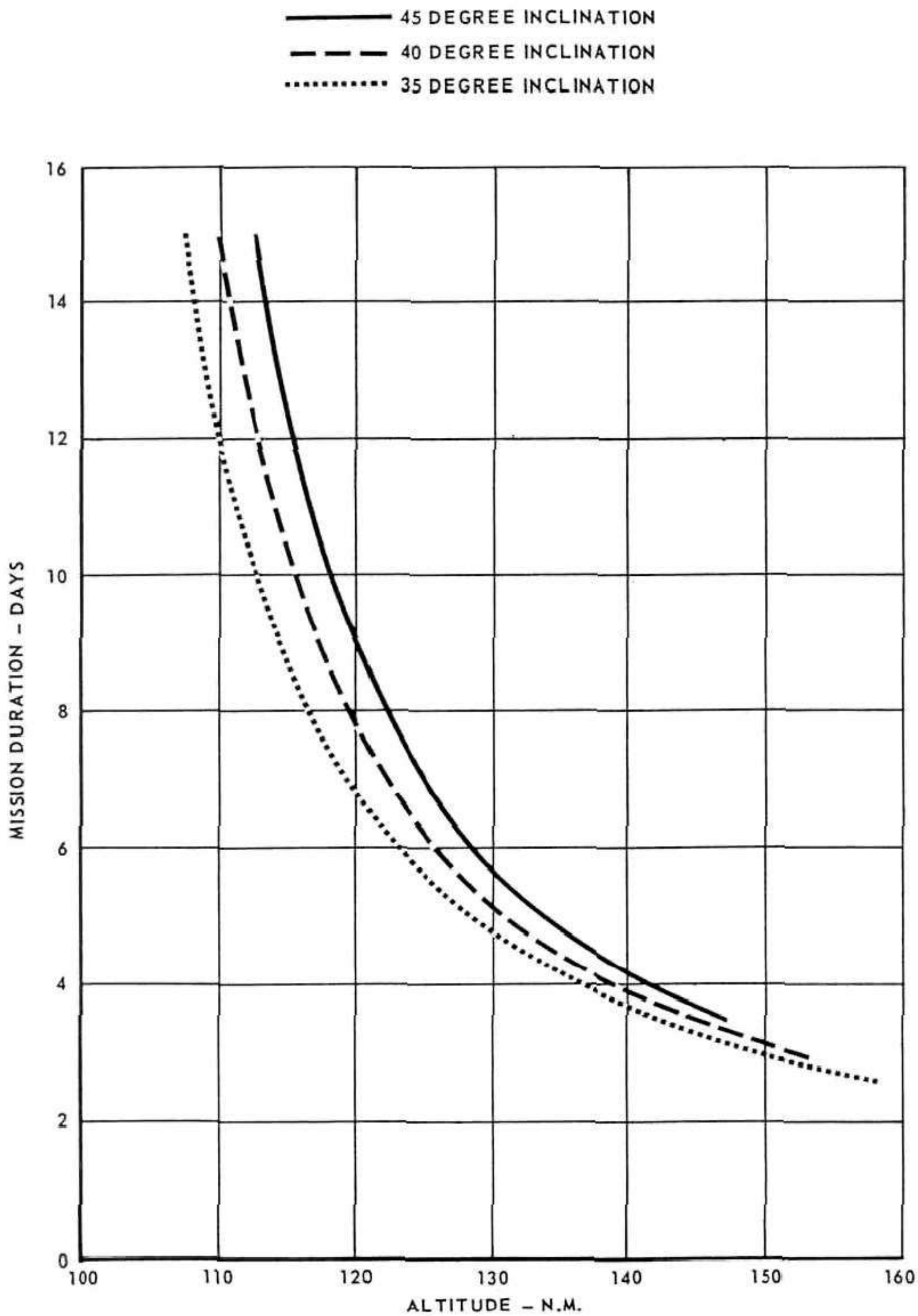


FIGURE 3.2-6

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.

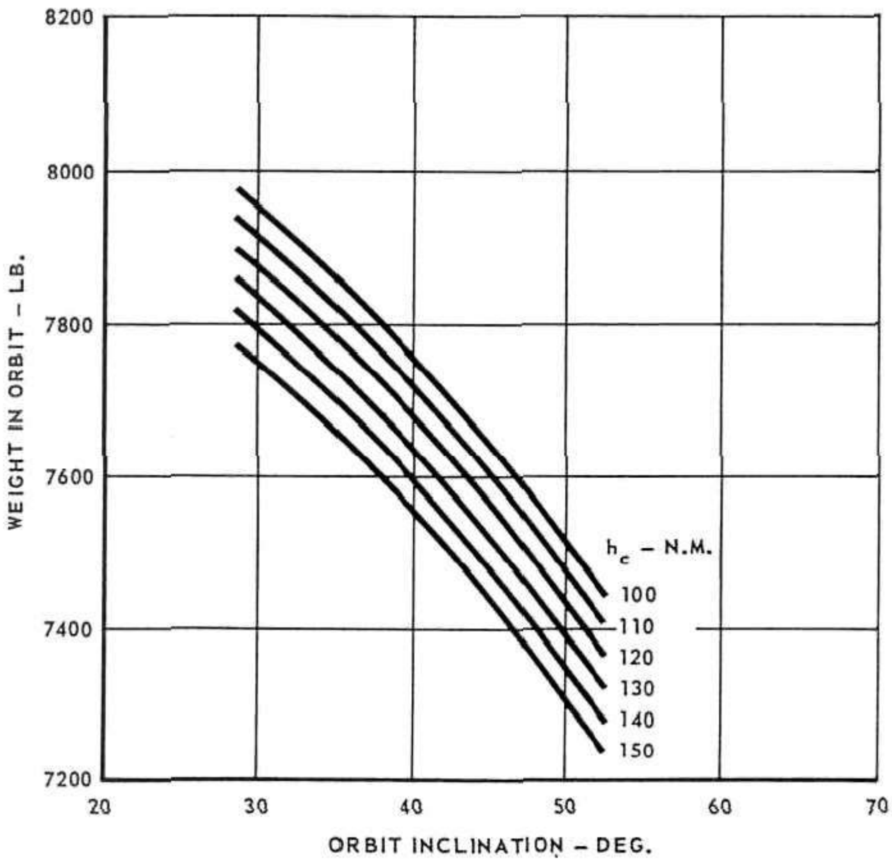


FIGURE 3.2-7

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.

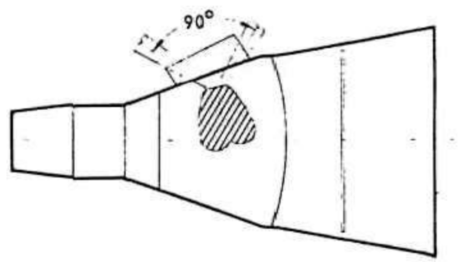
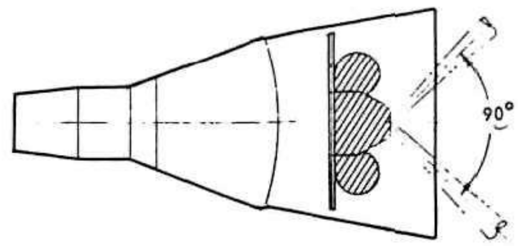
A horizon scanner mounted on the horizon camera head is utilized to maintain pitch and roll attitude. The orientation of the spacecraft during mapping operations precludes the use of the Gemini horizon sensors as presently mounted.

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

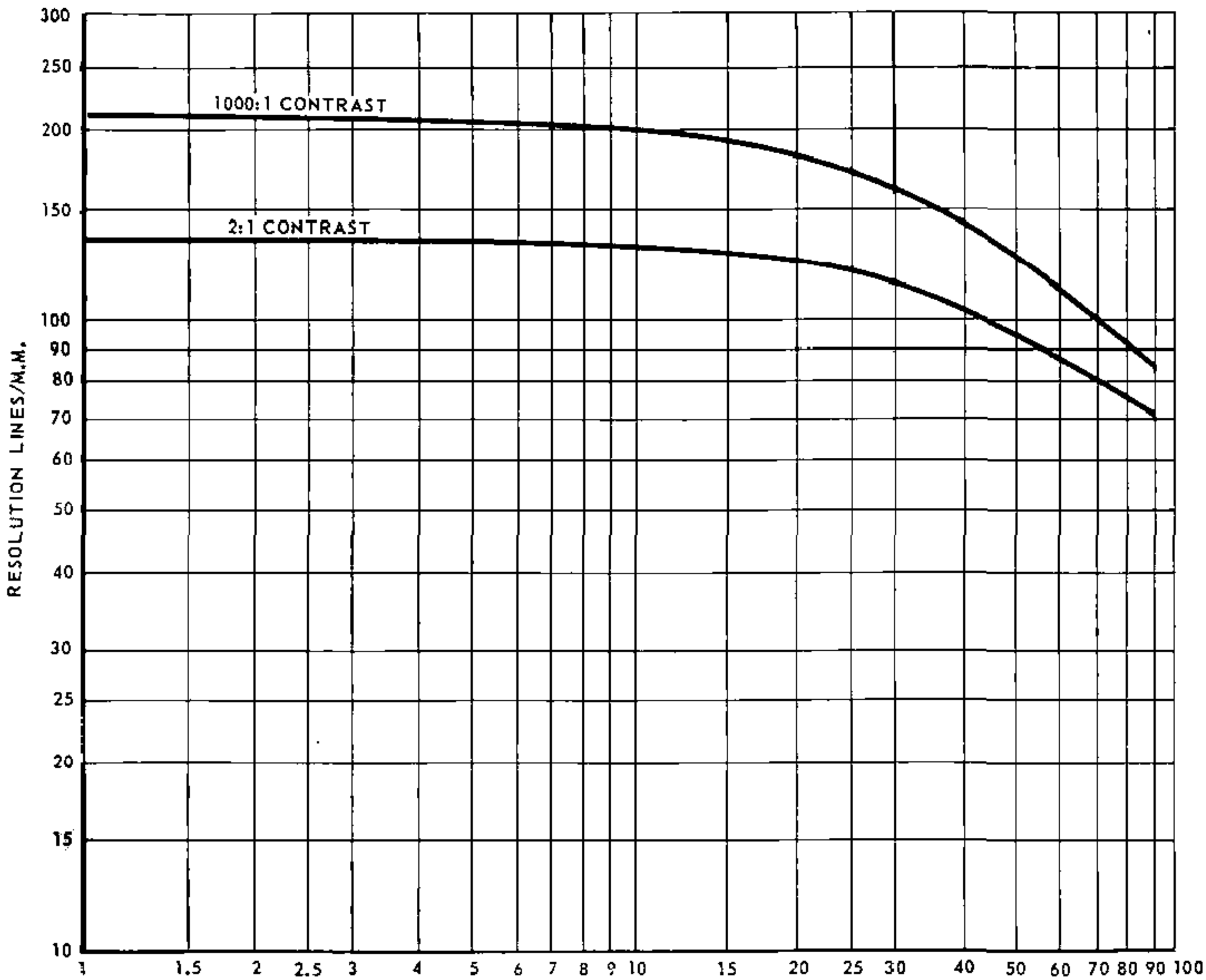


FIGURE 3.2-8

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%

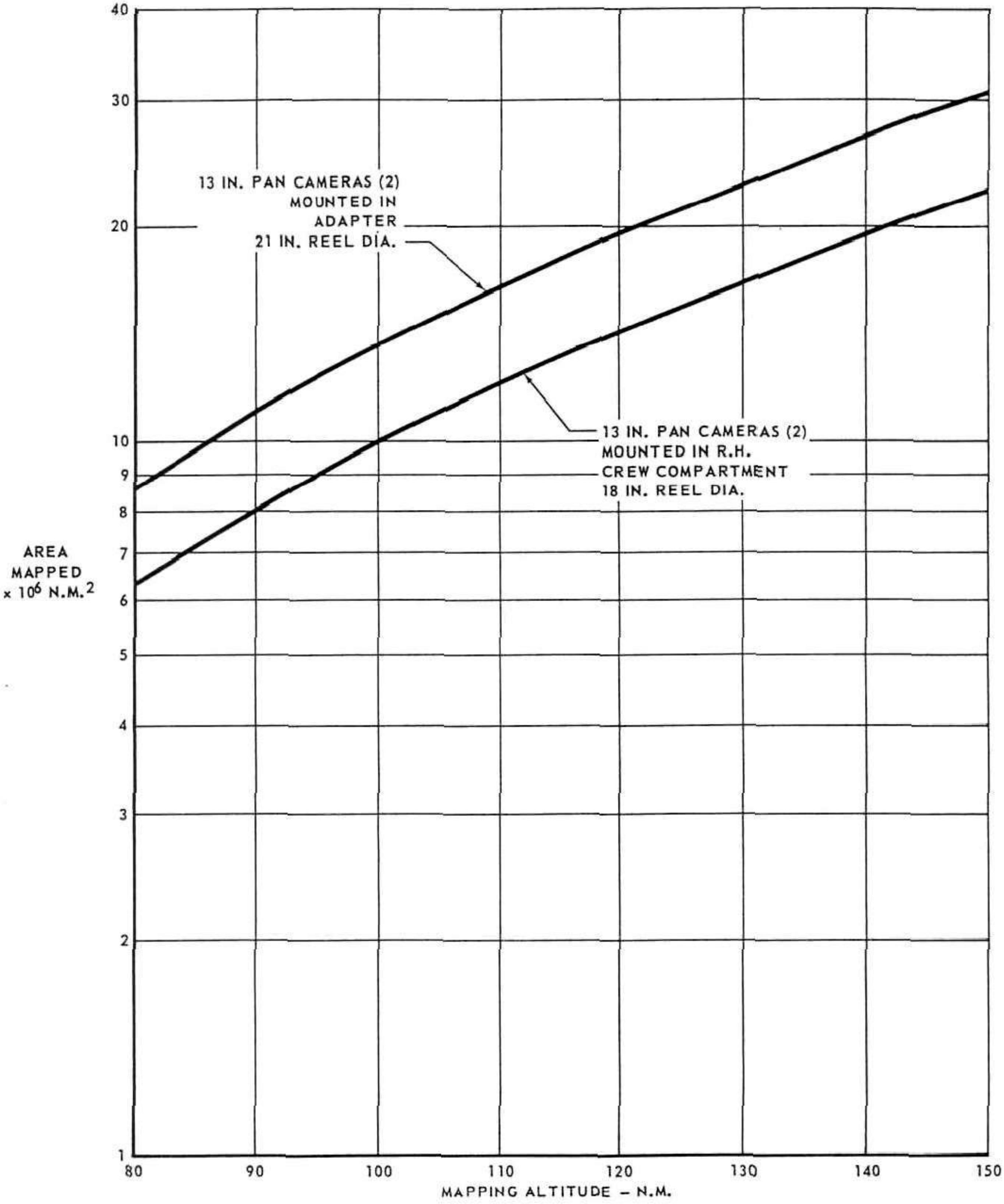


FIGURE 3.2-9

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.

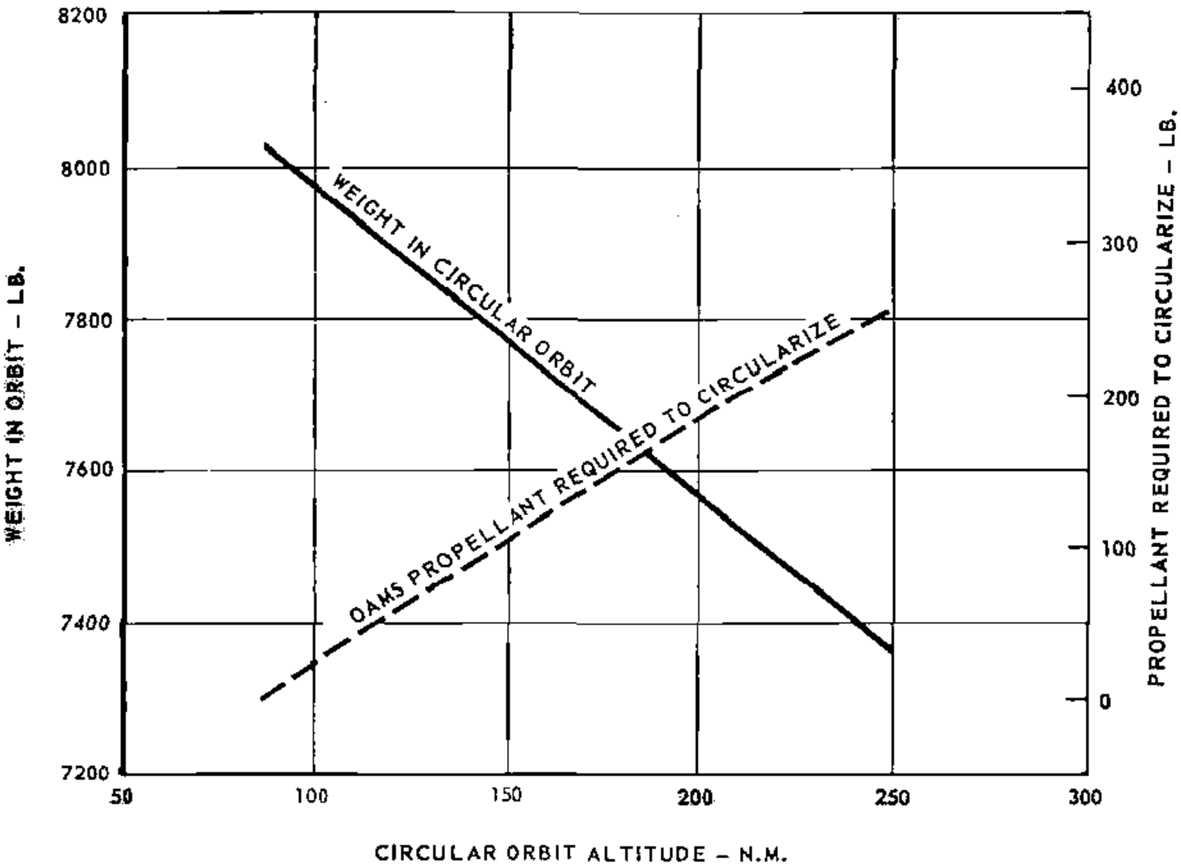


FIGURE 3.3-1

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.

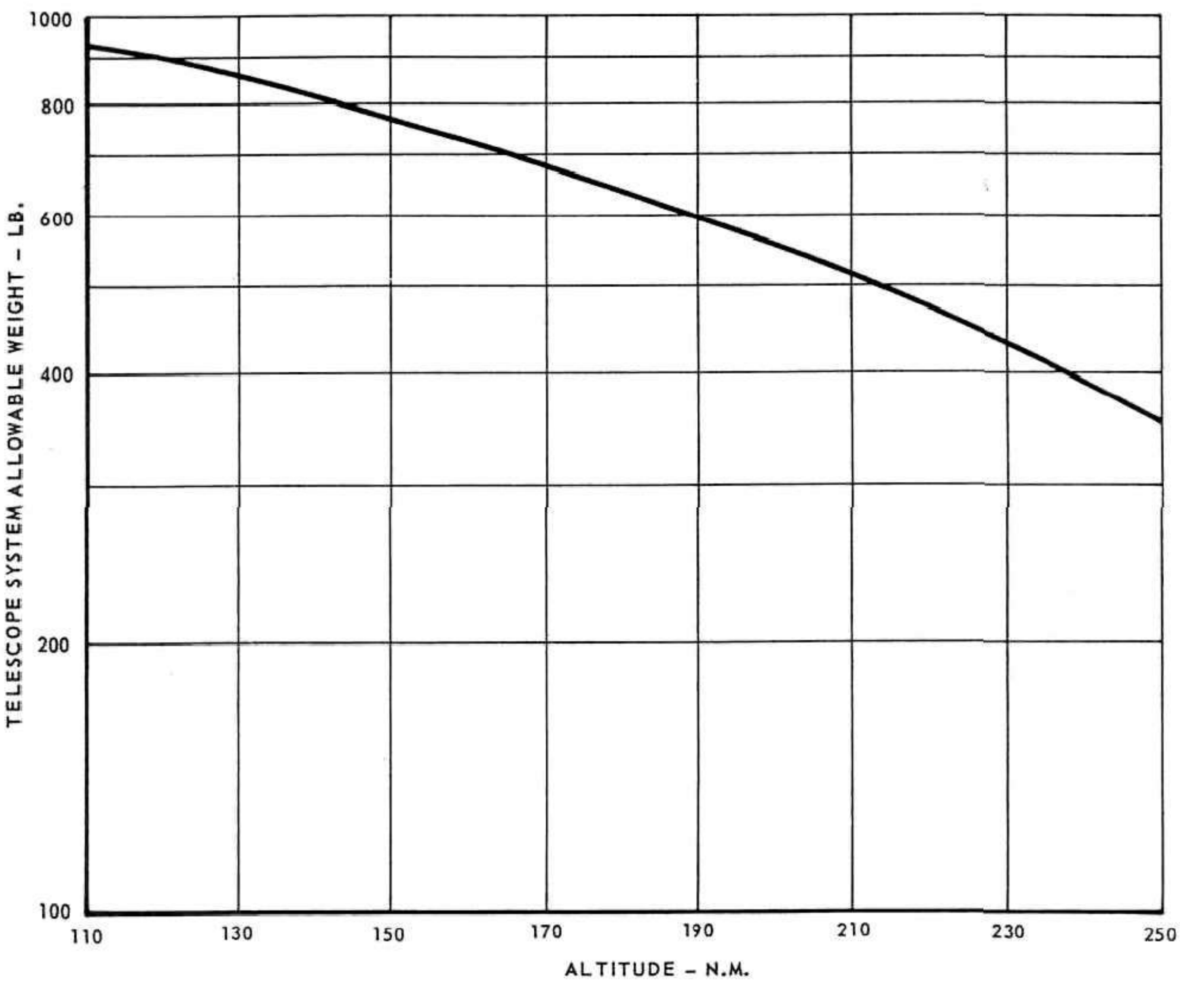


FIGURE 3.3-2

ASTRONOMICAL TELESCOPE WEIGHT

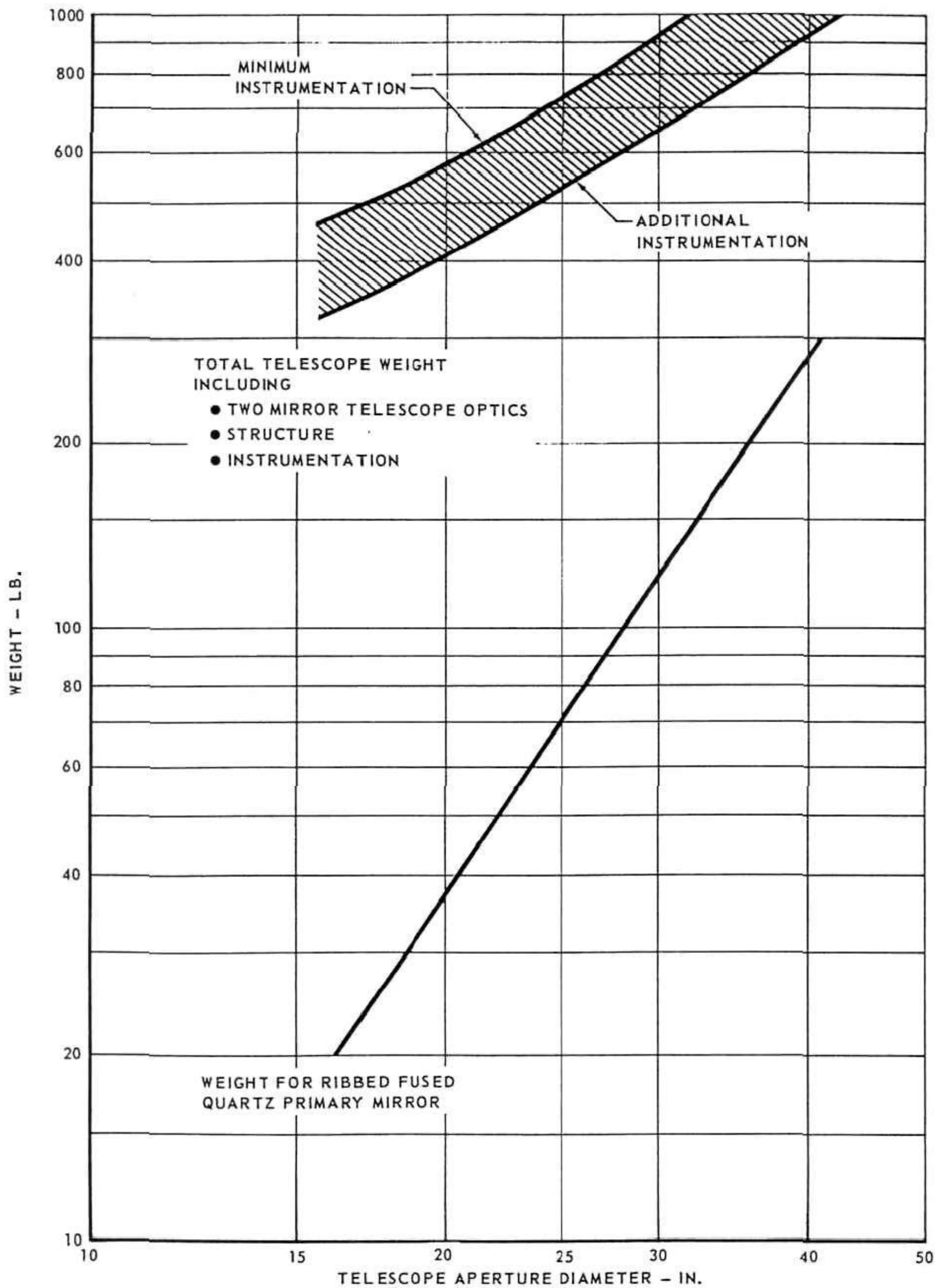


FIGURE 3.3-3

MAXIMUM APERTURE DIAMETER
OF THE TELESCOPE

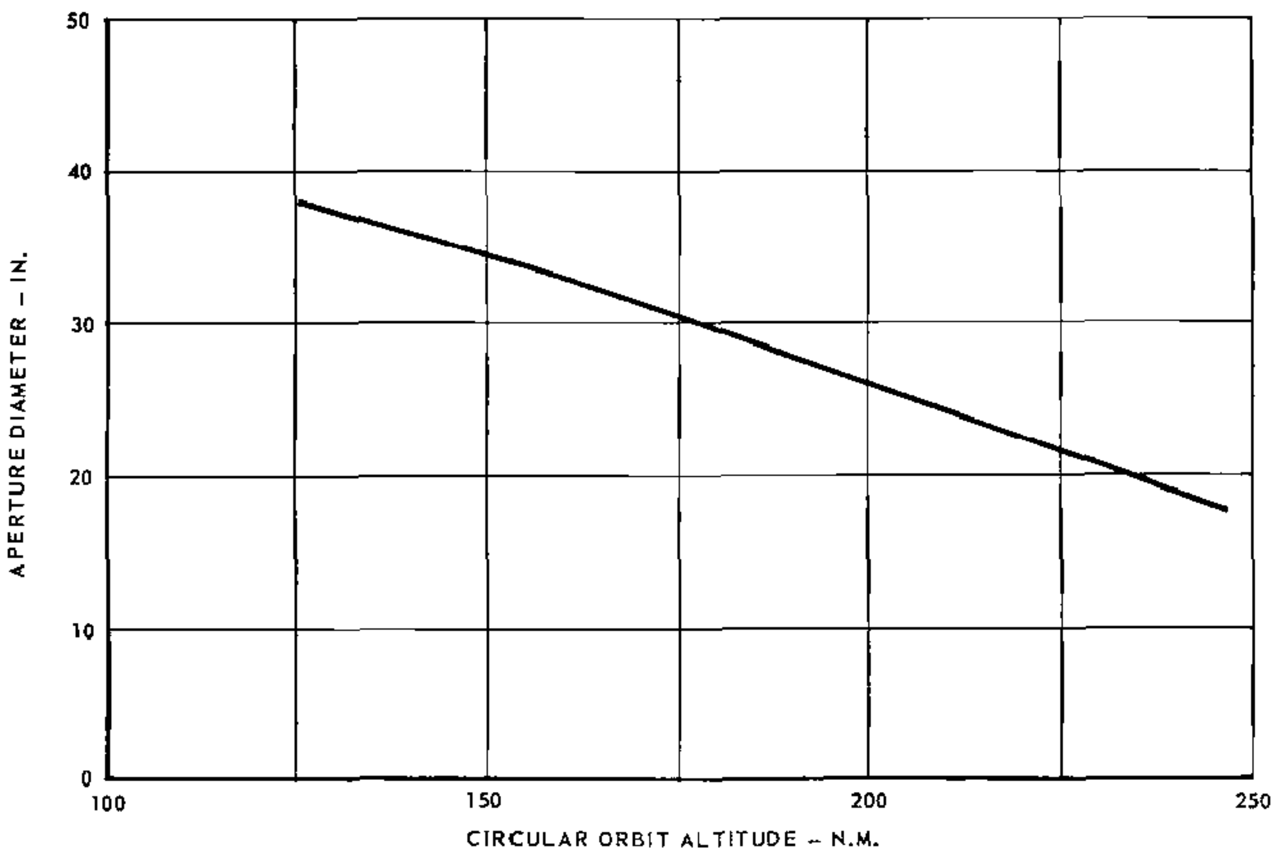


FIGURE 3.3–4

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

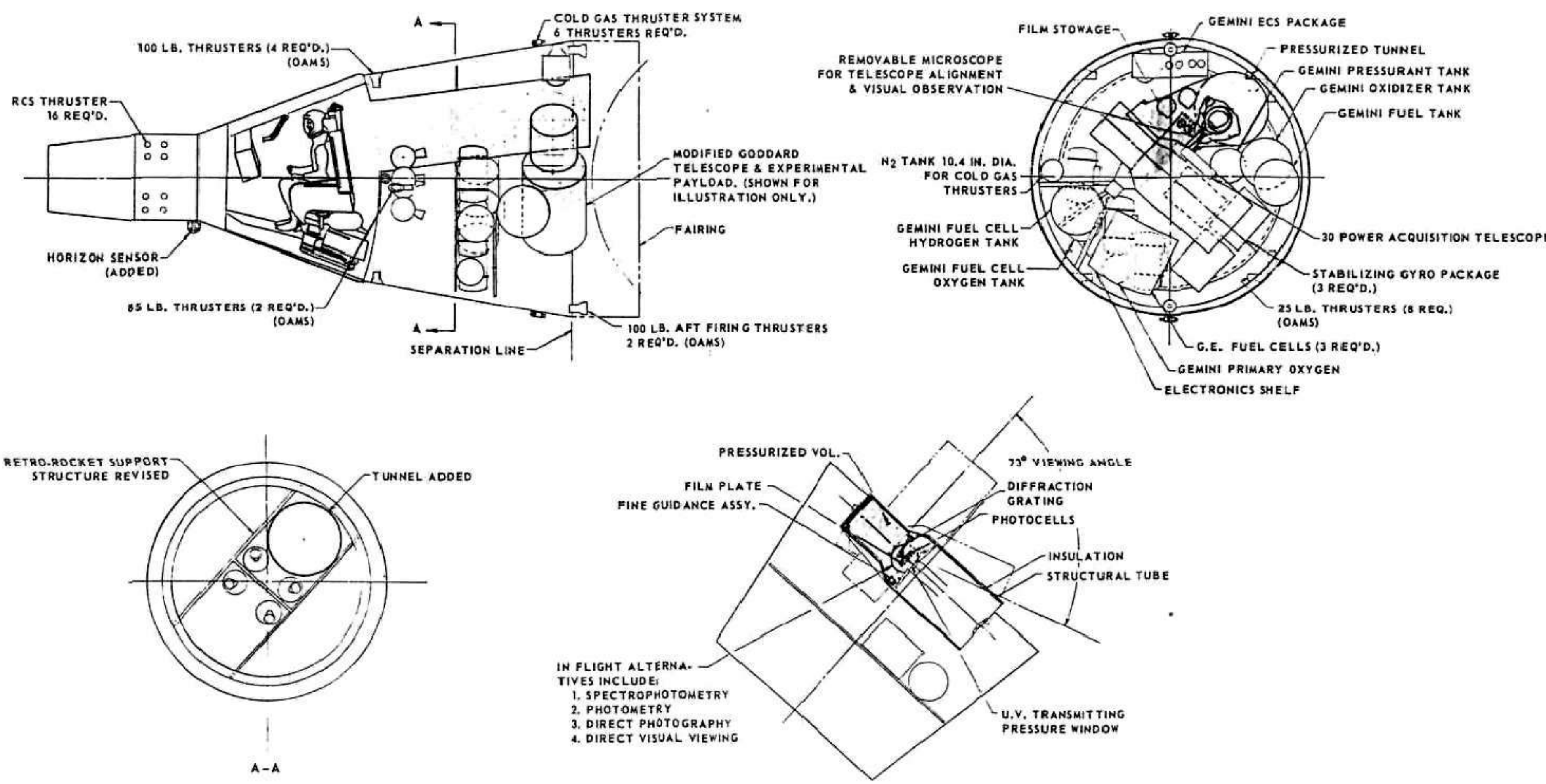


FIGURE 3.3-5

ONE MAN GEMINI WITH ASTRONOMICAL TELESCOPE

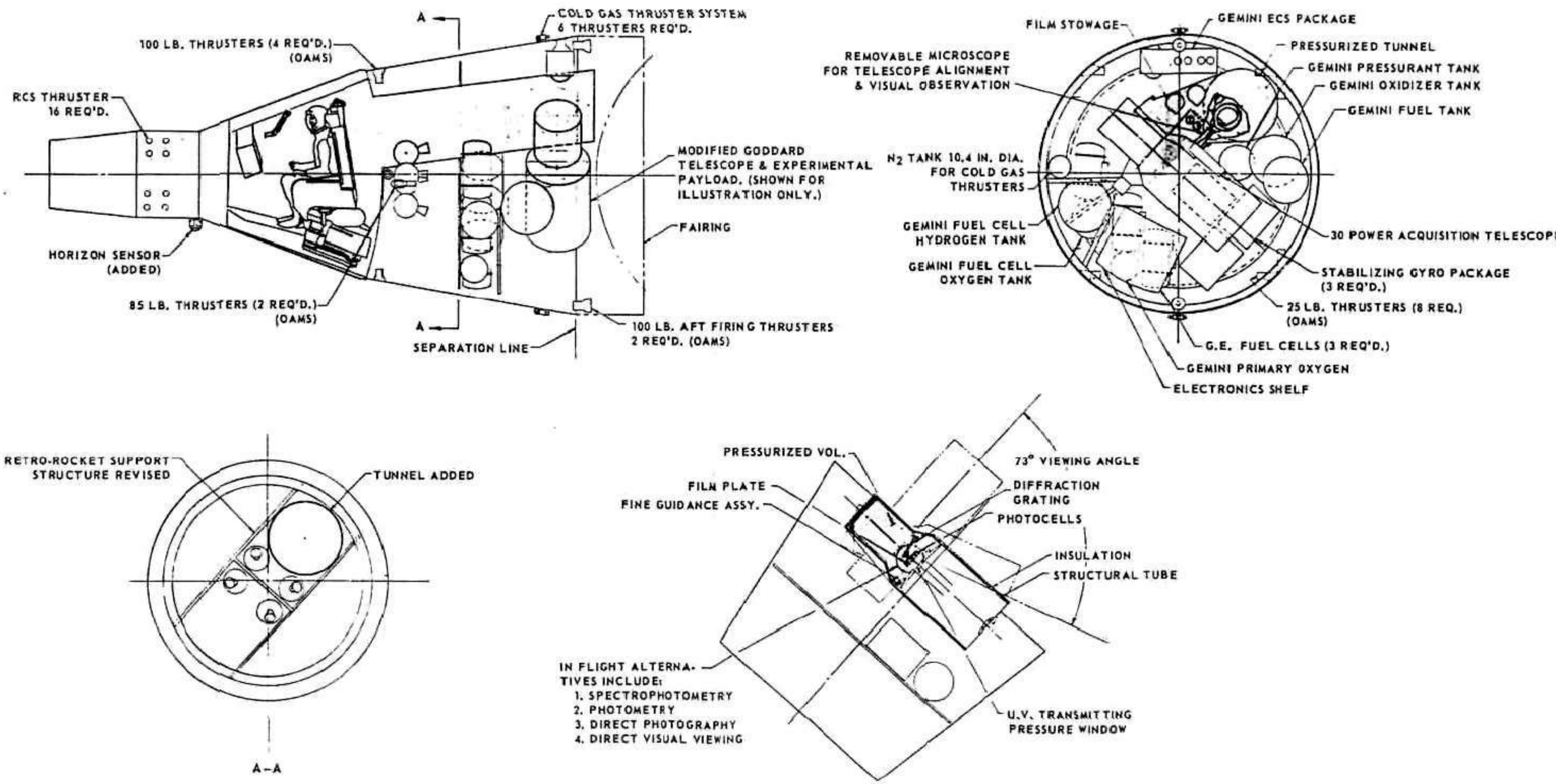
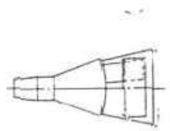
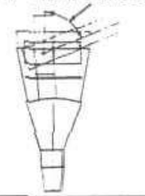
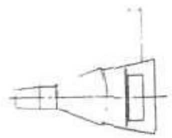
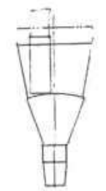
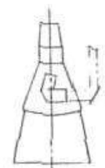


FIGURE 3.3-5

ASTRONOMICAL TELESCOPE INSTALLATION ALTERNATIVES

ONE MAN GEMINI WITH ASTRONOMICAL TELESCOPE		 			
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. - f/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 CONNECTION	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

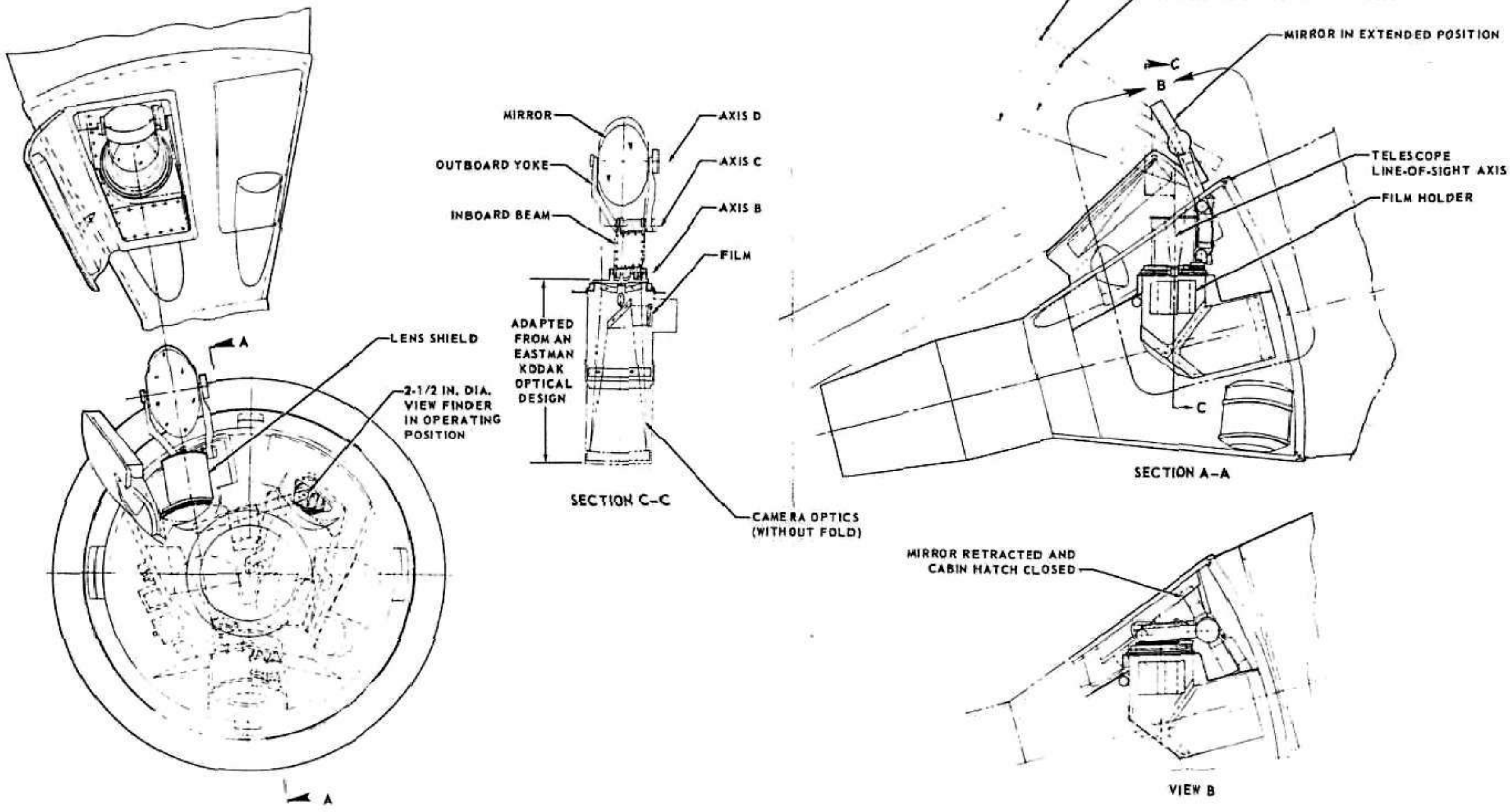


FIGURE 3.3–7

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

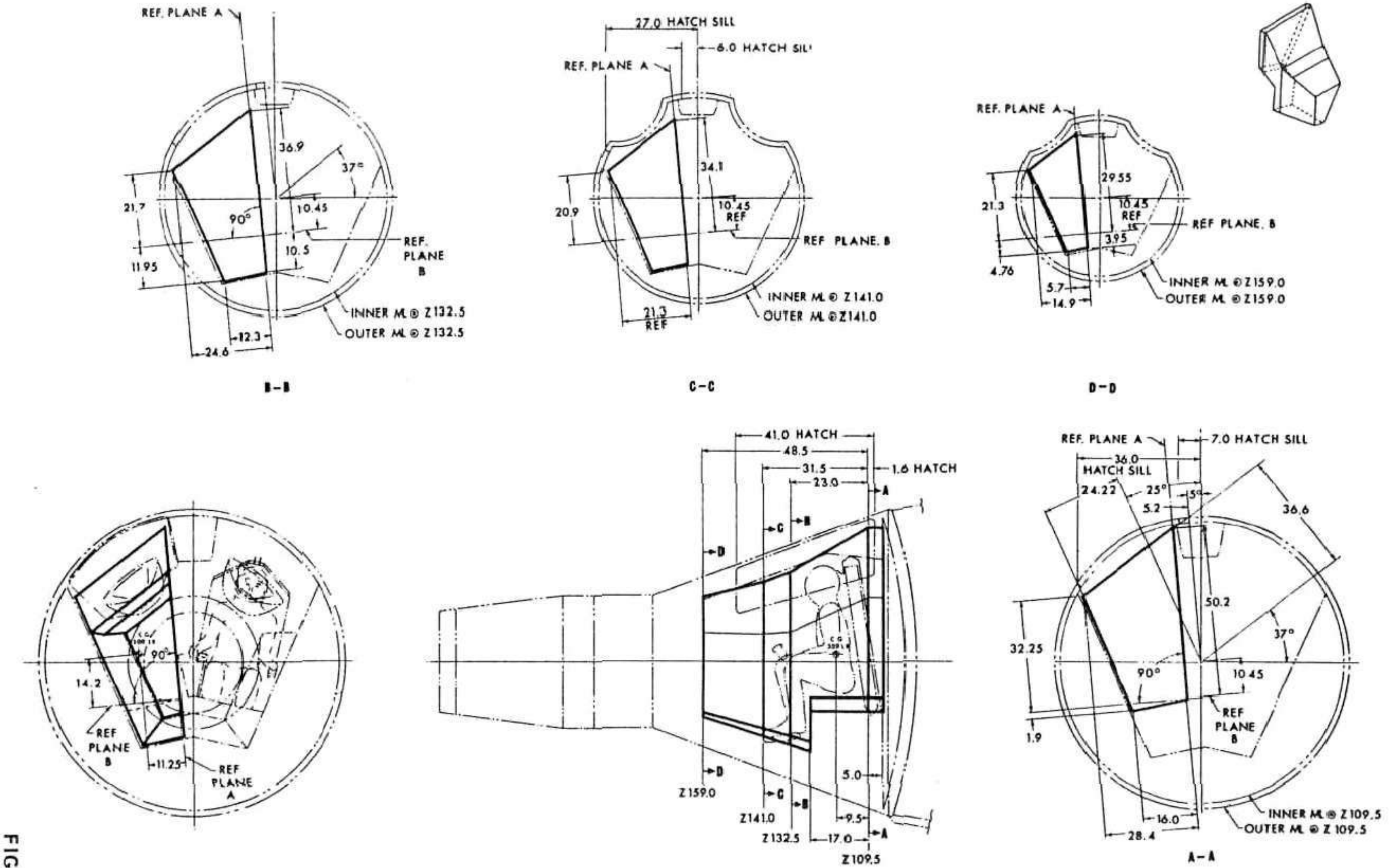


FIGURE 3.3-8

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 misalignment 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

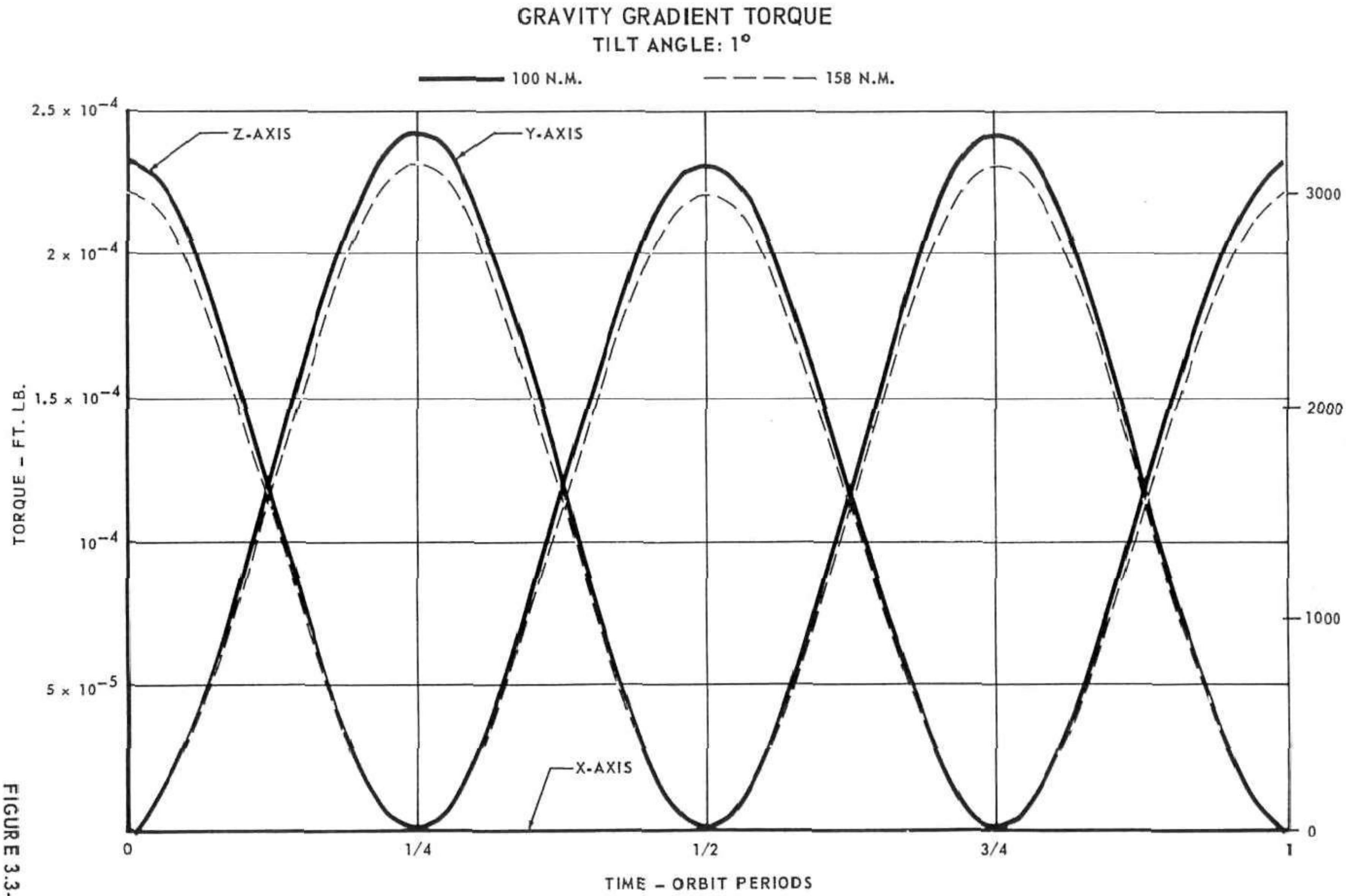

GRAVITY GRADIENT TORQUE
TILT ANGLE: 1°
100 N.M.
158 N.M.
Z-AXIS
Y-AXIS
X-AXIS
TORQUE - FT. LB.
2.5×10^{-4}
2×10^{-4}
1.5×10^{-4}
10^{-4}
5×10^{-5}
TORQUE - DYNE-CM
3000
2000
1000
0
1/4
1/2
3/4
1
TIME - ORBIT PERIODS

FIGURE 3.3-9

AERODYNAMIC TORQUE

ALTITUDE = 158 N.M.

$C_D = 2.2$

CROSS SECTIONAL AREA = 112 FT.^{2}

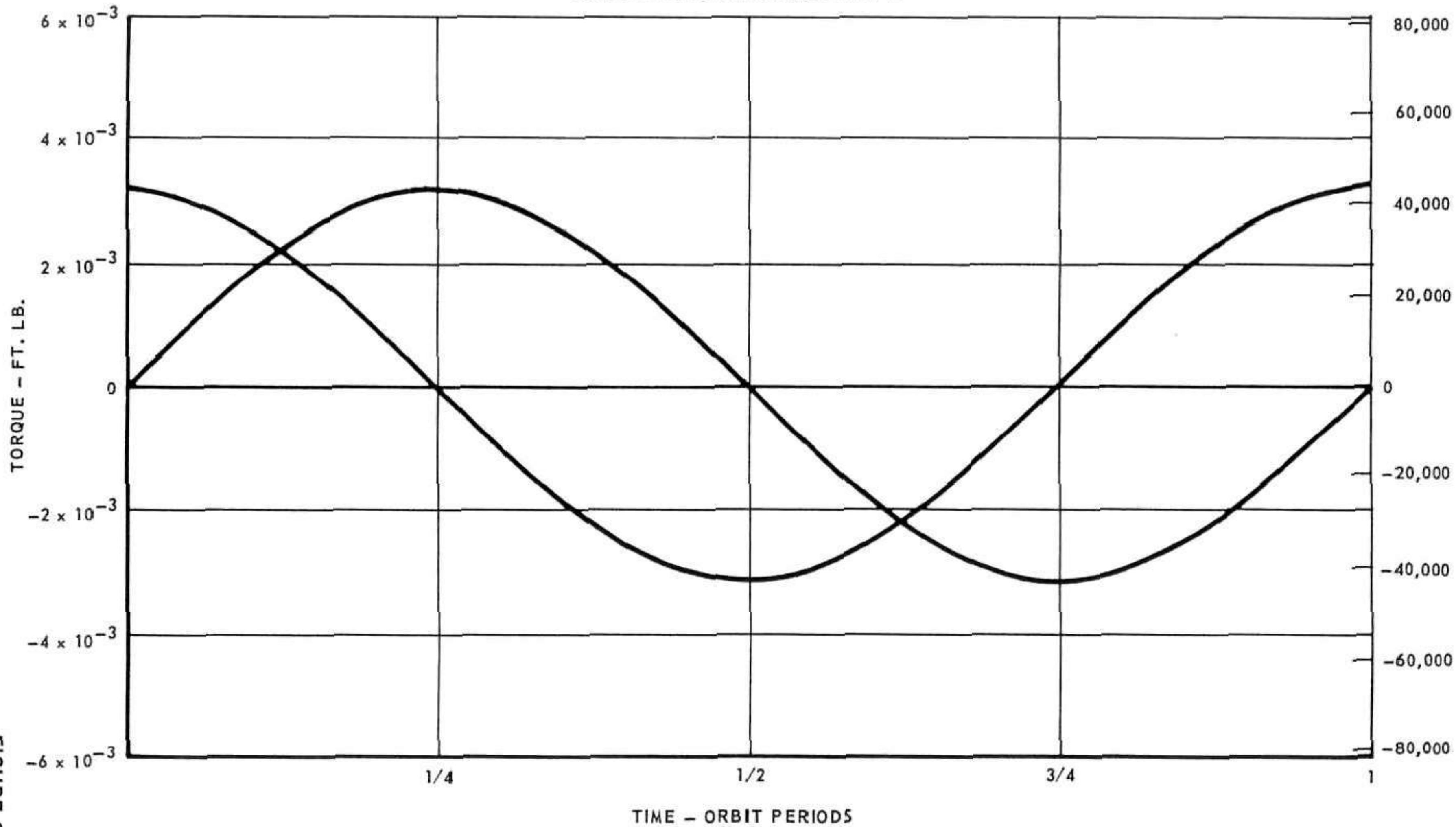


FIGURE 3.3-10

GRAVITY GRADIENT TORQUE
ON THE SPACECRAFT

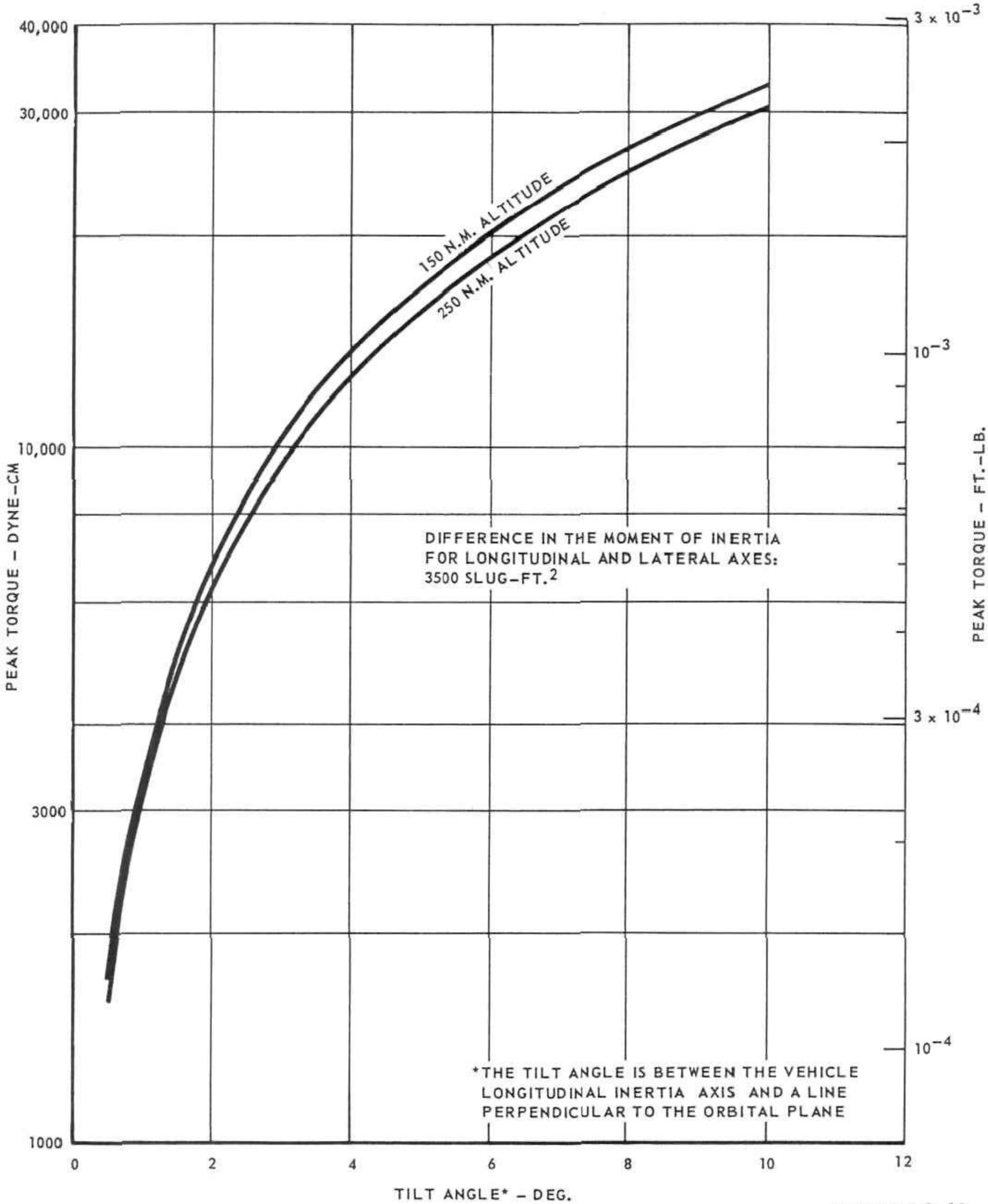


FIGURE 3.3-11

AERODYNAMIC TORQUE
ON THE SPACECRAFT

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

ATMOSPHERE: 1962 STANDARD

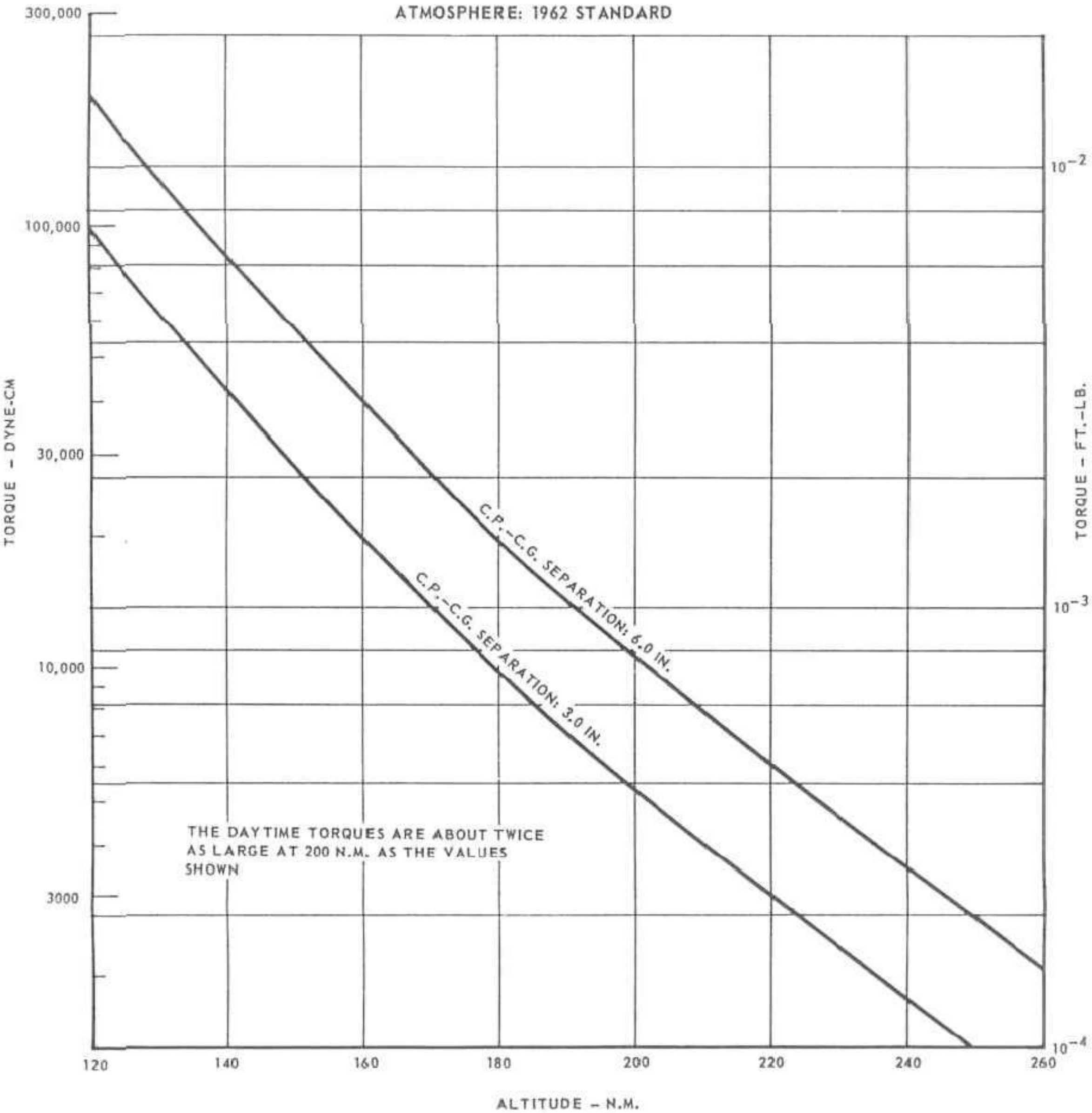


FIGURE 3.3-12

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.	
$\theta = \pm$ 0.03 DEG./SEC.	
ATTITUDE SLEW:	40 LBS.
NORMAL SLEW RATE: θ 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

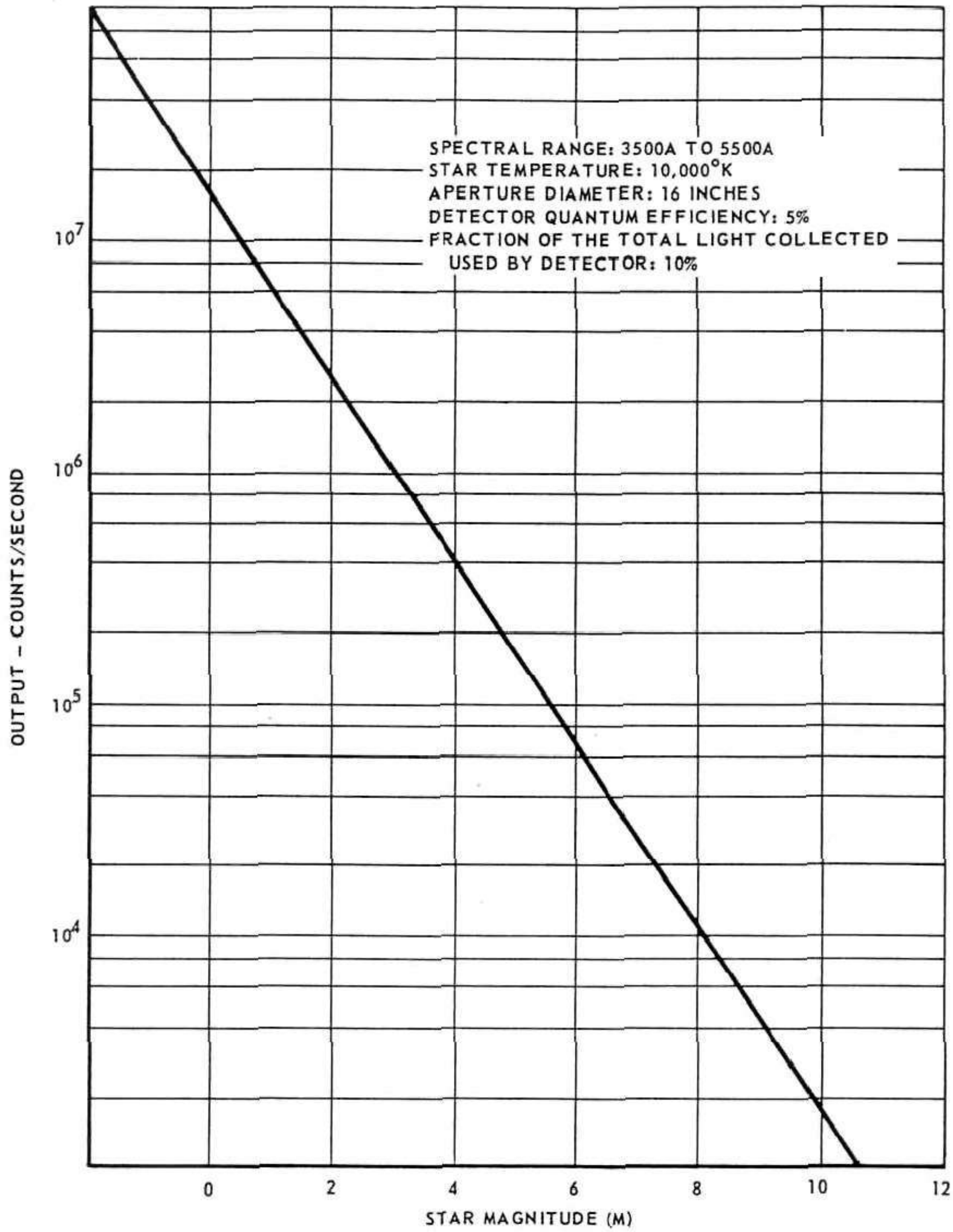


FIGURE 3.3–13

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

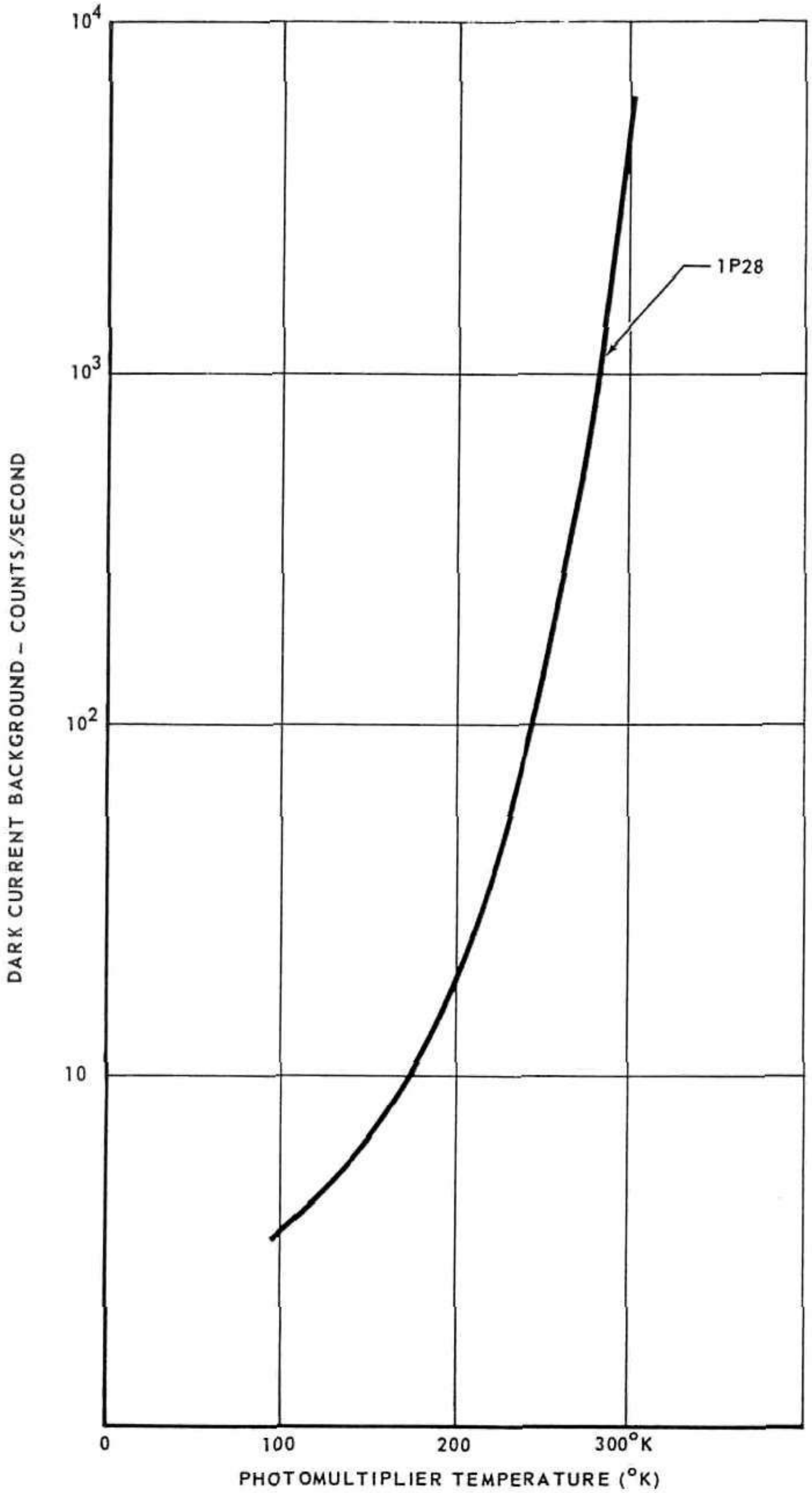


FIGURE 3.3-14

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 cabin-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.

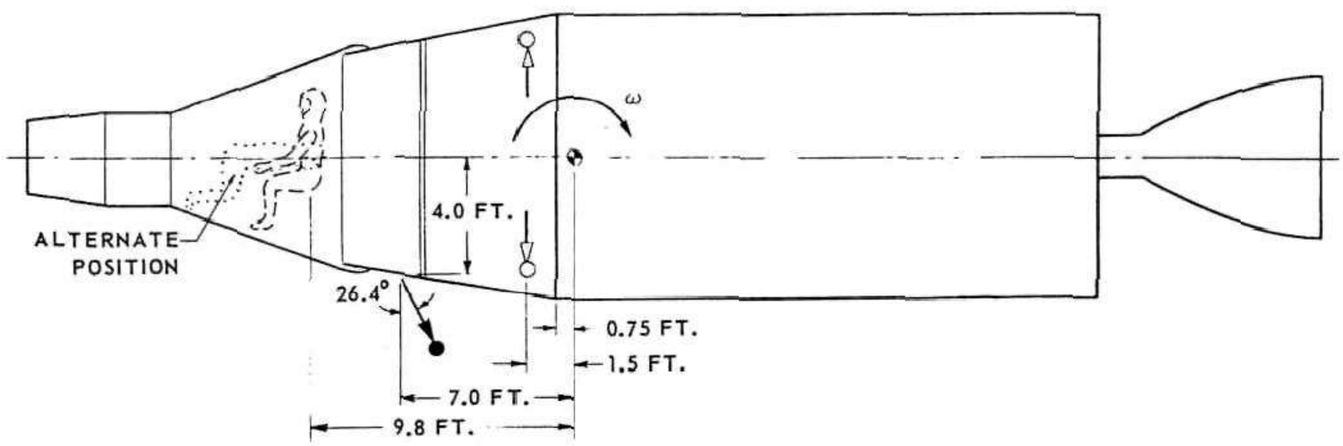
Operational conditions presented in Table 3.4-1 indicate that methods of using Gemini thrusters are feasible if their application to operations other than spin or despin are limited. Ablation thruster specifications require a guaranteed life of 425 seconds of burn time for the attitude control thrusters 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

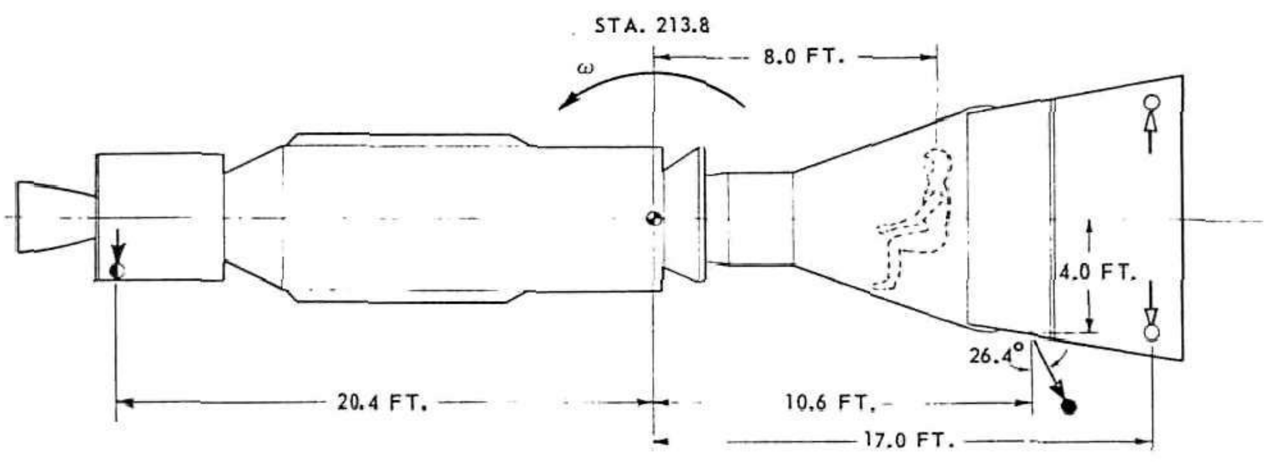


TOTAL WEIGHT = 12,930 LB.
$I_y = I_z = 52,100$ SLUG-FT.^{2}

THRUSTER CODE

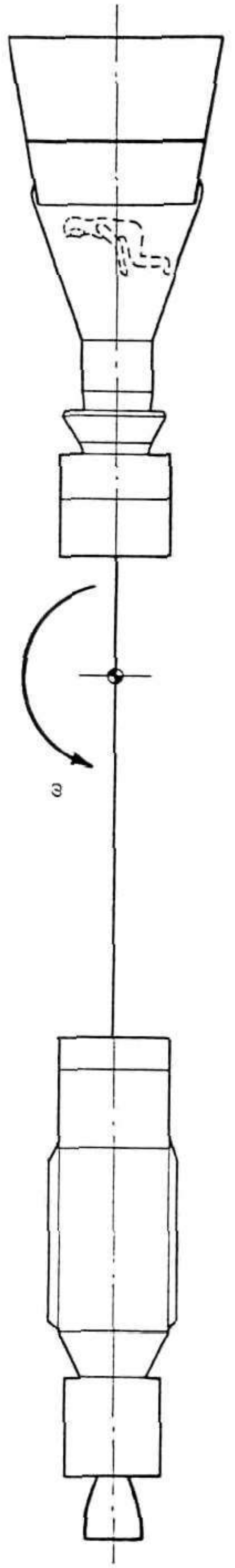
○◁—	23 LB. THRUST CHAMBER
●◀—	94.5 LB. THRUST CHAMBER
◒◀—	10 LB. COLD GAS CHAMBER

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

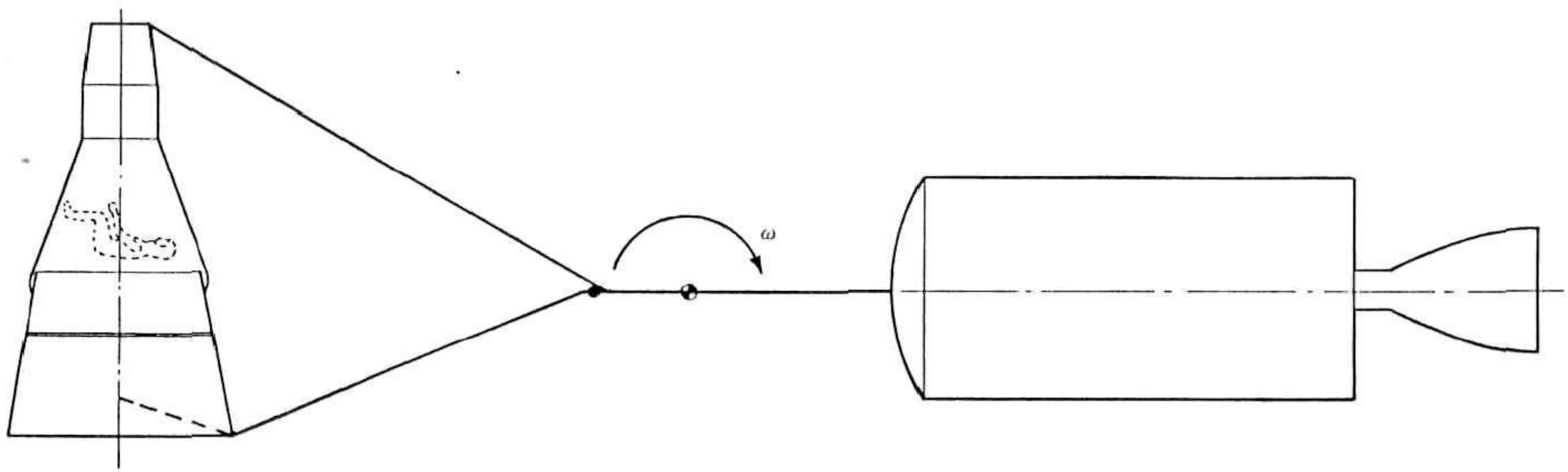


FIGURE 3.4-3

TABLE 3.4-1

ARTIFICIAL GRAVITY OPERATIONAL CONDITIONS

METHOD	PITCH OR YAW SPINNING	THRUSTER	THRUSTER LEVER ARM (FT.)	ROTATED MOMENT OF INERTIA (SLUG-FT.^{2})	CREW MEMBER'S HIP ROTATING RADIUS (FT.)	CENTRIFUGAL ACCELERATION (g's)	SPIN RATE (RPM)	ONE SPIN (OR DESPIN) PROPELLANT (LB.)	ONE SPIN (OR DESPIN) BURN TIME (SEC.)	CABLE WEIGHT (LB.)
DIRECTLY-CONNECTED GEMINI – STAGE II	EITHER	MS 1–94.5 LB.	8.0	52,100	9.8	1.0 0.5 0.1	17.3 12.2 5.5	46.3 32.8 14.6	125 88 39	0 0 0
DOCKED GEMINI – AGENA VEHICLE	EITHER	MS 1–94.5 LB.	11.3*	62,000	8.0	1.0 0.5 0.1	19.1 13.9 6.1	63.9 45.2 20.2	172 122 55	0 0 0
	EITHER	GACS 2–23 LB.	17.0	62,000	8.0	1.0 0.5 0.1	19.1 13.9 6.1	29.2 20.7 9.2	159 112 50	0 0 0
	YAW	AACS 2–10 LB.	20.4	62,000	8.0	1.0 0.5 0.1	19.1 13.9 6.1	122.0 88.5 27.3	306 221 96	0 0 0
CABLE-CONNECTED GEMINI-AGENA	EITHER	MS 1–94.5 LB.	152	9,580,000	150.	1.0 0.5 0.1	4.4 3.1 1.4	114.3 80.9 36.2	309 218 98	70 35 35

I_{sp} = 50 LB.-SEC./LB. FOR AACS (AGENA ATTITUDE CONTROL SYSTEM)
= 250 LB.-SEC./LB. FOR GACS (GEMINI ATTITUDE CONTROL SYSTEM)
= 255 LB.-SEC./LB. FOR MS (GEMINI MANEUVER SYSTEM)

*EFFECTIVE ROTATIONAL THRUST IS .675 OF ACTUAL THRUST

3.4 (Continued)

system provides a better simulation of artificial gravity and less danger of vertigo because of the larger radius and correspondingly lower angular velocity. More effective head-to-foot gravity forces result from using a parawing bridle to obtain the Gemini orientation shown in Figure 3.4-3.

The reliability of the cable method is lower than for the other methods as a result of the cable attachment and handling facilities required. To minimize the amount of additional equipment required, cable reel-in provisions might be omitted.

The propellant requirements listed in Table 3.4-1 do not include allowances for spinning during extension of the cable; it is assumed spinning is accomplished at the final cable extension. Partial spin-up prior to and during cable extension, which will probably be required to eliminate slack cable conditions, might double the amounts of propellant needed.

Since rendezvous is not required with Stage II of the GLV as the counterweight, equipment such as the rendezvous radar will not be needed. Elimination of this equipment and a reduction of the propellant for the orbit attitude and maneuver system results in adequate weight and space allowance for the cable and its associated equipment. With the Agena as the counterweight, a separate Agena section adjacent to the target docking adapter would be provided for the cable and its associated equipment.

3.5 Simulation of LEM Rendezvous - The Gemini flight test of Apollo LEM-CM rendezvous would evaluate the rendezvous phase of the Apollo mission using actual LEM equipment or modified Gemini hardware. During rendezvous, the rendezvous radar and inertial measuring unit aboard the LEM are used to accumulate range data of the CM relative to the LEM. These data are processed by the LEM guidance computer, where the required velocity change for rendezvous is computed. The LEM propulsion system is then used to change the LEM velocity vector so that a collision course with the Apollo CM is attained.

In the rendezvous test flight, the Gemini spacecraft would have the role of the LEM vehicle with the velocity change computations performed aboard the Gemini and the Gemini propulsion system used to execute the velocity change maneuvers. The Apollo CM would be simulated by either a Gemini Agena Target launched into orbit prior to the Gemini, or by a smaller target launched with the Gemini and separated from the Gemini while in orbit.

With Apollo hardware available for the flight, the LEM computer, IMU, and radar would be used aboard the Gemini to compute the velocity change maneuvers. Also, an Apollo CM transponder would be installed on the rendezvous target vehicle. Should Apollo equipment not be ready for the test flight, Gemini equipment, including a modified Gemini computer, would be used to perform the same operations as the LEM hardware. If only some of the Apollo hardware could be used for the test flight, e.g., the radar, then modified Gemini equipment could be substituted and the flight performed using a combination of Gemini and Apollo equipment.

A preliminary estimate of the added weight of the Apollo equipment is 223 lbs. This includes the computer, power supply, IMU, radar, and Gemini structural changes. This weight addition is within the launch weight capabilities for a two day Gemini rendezvous mission. For example, several spacecraft have experiments with weights

3.5 (Continued)

in excess of the Apollo equipment weights. Therefore, this equipment can be incorporated into future spacecraft without affecting mission capability. If modified Gemini equipment without any Apollo hardware is used, the weight increase would be a 25 lb. auxiliary tape memory for the Gemini computer.

3.6 Structural Assembly in Orbit - The antenna to be erected in orbit is shown in Figure 3.6-1. The manual erection of this type of structure by EVA will be most valuable in the performance of assembly, maintenance, inspection, and alignment of such items as:

Radiators

Sensors

Solar arrays

Gimbal mounts

Fuel lines, valves, and connections

Antennas

Propulsion systems

Docking ports

3.6.1 Types of In-Space Operations - Man's role, as applied to the various types of in-space assembly separations, is discussed in the following paragraphs.

A. Fluid or Gas Transfer Connections and Large Electrical Connections - Manual connection and manual activation are preferred for simplification of these operations. However, automatic connections may be desirable for safety reasons. Visual inspection of the connection will be important.

B. Assembly of Heavy Structures

1. For Space Station build-up or assembly, manual operations are desirable in order to omit complex hinges, locks, and automatic or motor driven fasteners. However, built-in positioning and holding devices will be needed because of the inability to position and hold large masses by hand.

2. Large Telescopes - Manual alignment and calibration is desirable.

ANTENNA ERECTION AND ASSEMBLY IN ORBIT

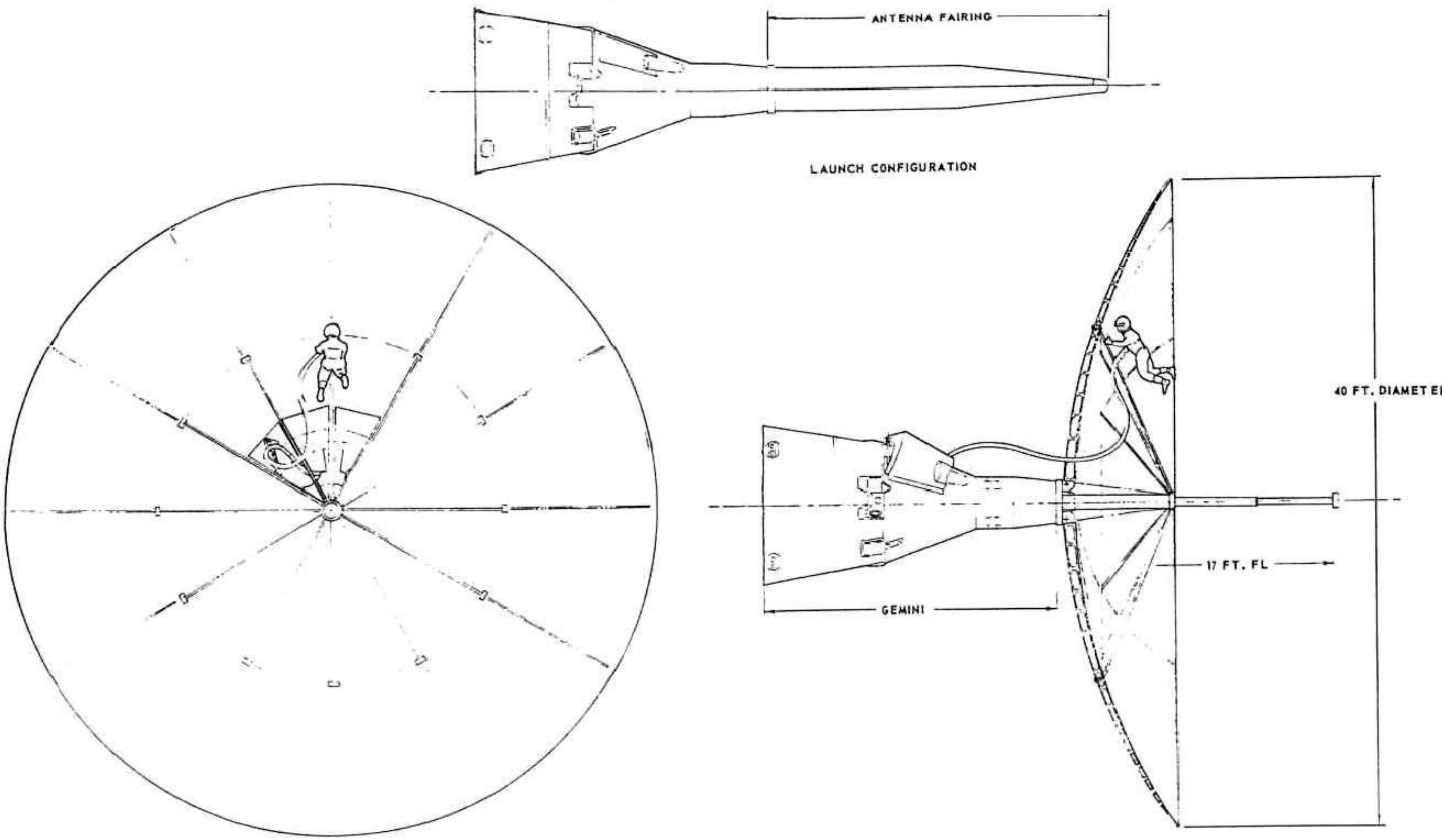


FIGURE 3.6-1

ANTENNA ERECTION AND ASSEMBLY IN ORBIT

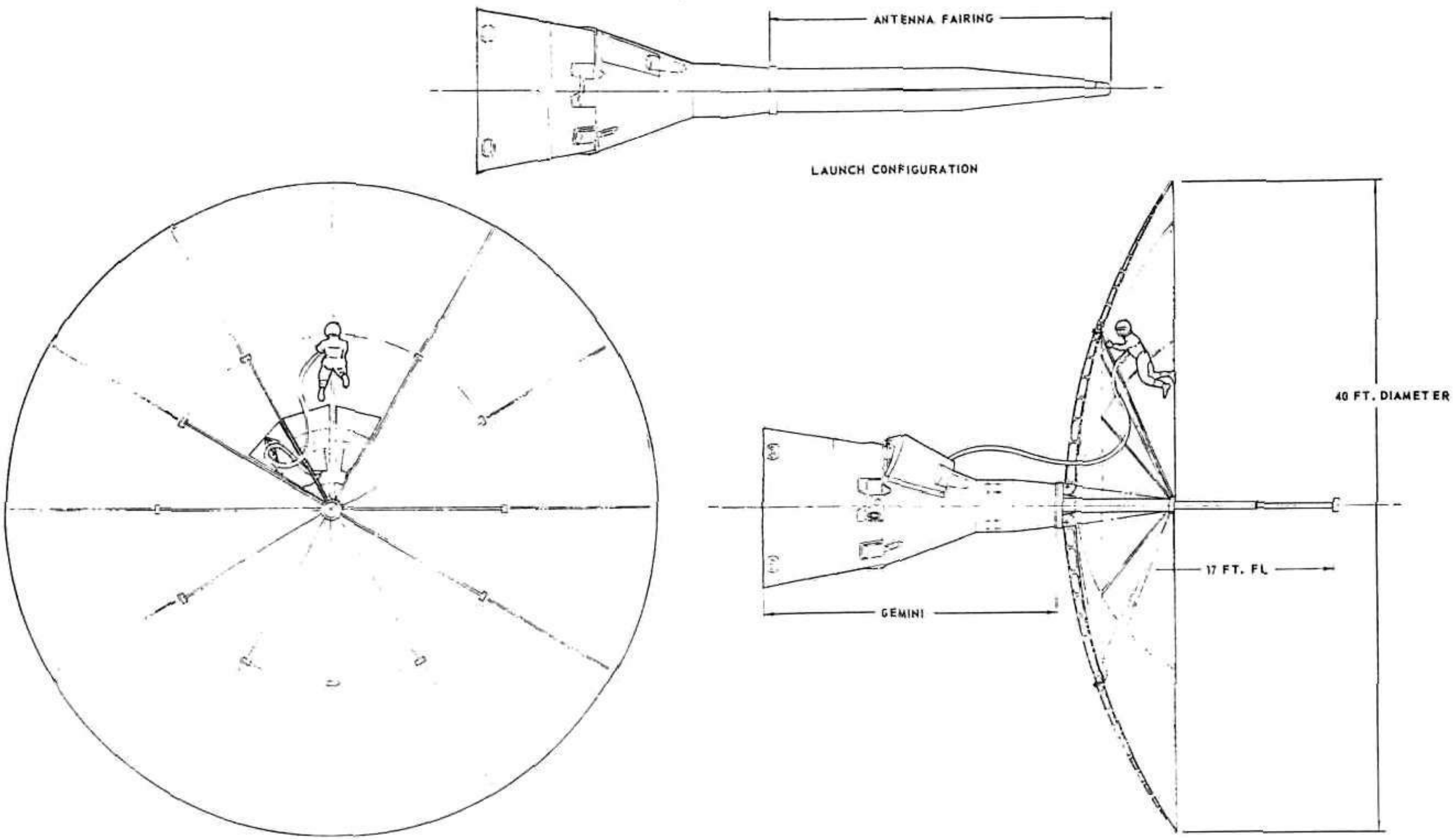


FIGURE 3.6-1

3.6.1 (Continued)

C. Assembly of Large, Low Density Structures

1. Antennas, solar arrays and radiators - Very little actuator power is required to mechanize erection because of weightlessness. However, man is required as backup to this operation as well as to provide final lock-up, alignment, and inspection.

2. Pressurized Transfer Tunnels - Man can more effectively perform sealing operations and also perform seal inspection and seal maintenance.

D. Return of Gemini Thrust Chamber Assembly - Rocketdyne has accomplished a breakthrough on thruster life capability for the Gemini spacecraft. Although the Gemini TCA's have an appreciable margin of safety for the Gemini missions, the precise margin has not been determined due to the face that endurance testing of rocket motors at orbit pressure altitude has not been accomplished to date on ablative-type motors. Information of this type is of particular interest because of the speculation of shorter life capability in space than that estimated from tests conducted at altitudes intended to be representative of orbital altitudes, but which do not duplicate actual operating conditions. It is especially important to obtain endurance results from a pulsing mode of operation.

3.6.2 Erectable Antenna Weight Estimates - The preliminary weight estimate is given in Table 3.6-1.

TABLE 3.6-1

STRUCTURAL ASSEMBLY IN ORBIT

	WEIGHT – LB.
WEIGHT REMOVED	(-817)
SPACECRAFT 12 EXPERIMENTS	-279
RADAR	-87
DOCKING SYSTEM	-18
OAMS TANKS	-44
OAMS PROPELLANT	-288
WEIGHT ADDED	(818)
ANTENNA	685
STRUCTURE	103
ECS LINES	30
NET WEIGHT INCREASE – LB.	1

3.7 Propellant Transfer

3.7.1 Orbiting Vehicle Configuration - The orbiting vehicles utilized to accomplish this task are a rendezvous configuration Gemini and Gemini Agena Target (G.A.T.). The basic configuration of the Gemini is altered to incorporate two propellant tanks, one pressurant tank, and transfer plumbing in the adapter equipment section. The G.A.T. is altered to accommodate two propellant tanks and a pressurant tank externally accessible.

3.7.2 System Configuration - The transfer system schematic is shown in Figure 3.7.1. The Gemini equipment includes liquid/vapor separators in the propellant tanks, propellant quantity gauging devices externally mounted, in-line flow meters, pressure transducers, thermocouples, and latching solenoids. The latter are used to control the transfer, purge the system, regulate the receiving tank pressure, and pressure check the system prior to transfer. The G.A.T. equipment is similar, except the propellant tanks contain collectors, the pressurant tank utilizes a heater blanket to provide maximum transfer, and the pressurant switch is regulated. The propellant quantity gauge is used on these tanks for complete transfer monitoring, but flow meters which would be redundant, are felt not to be necessary.

3.7.3 Typical Transfer Procedure - The following is a typical mission sequence for storable propellant transfer:

A. Gemini rendezvous with Agena (nose dock)

B. Latch up vehicles (rigid)

C. Connect propellant and pressurant lines (EVA)

D. Secure and check connection (EVA)

E. Pressurize transfer lines and turn off pressurizing source

F. Pressure-check lines by monitoring transfer line pressure to verify good connection

PROPELLANT TRANSFER SYSTEM

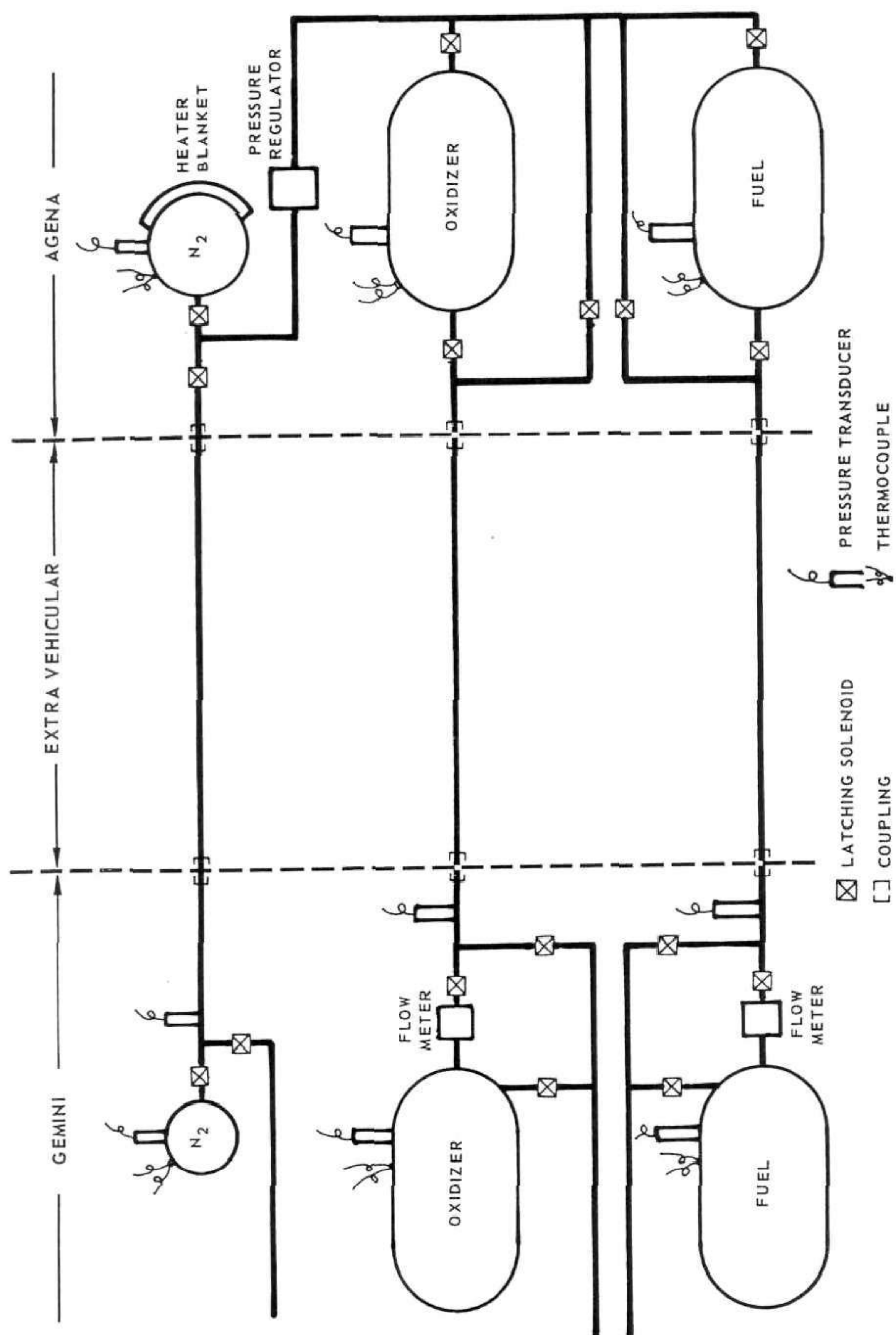


FIGURE 3.7-1

3.7.3 (Continued)

G. Release transfer line pressure and close dump valve

H. Pressurize oxidizer (N_2O_4) receiving tank to 90 PSIA

I. Check oxidizer tank temperature (not to exceed 150°F)

J. "Zero" oxidizer transfer integrating flow meter

K. Commence oxidizer transfer

(1) Introduce regulated pressure to oxidizer tank (300 PSIA)

(2) Open oxidizer tank isolation valves (2)

(3) Monitor integrating flow meter

(4) Bleed receiving tank to maintain 70 PSIA

L. Halt transfer when complete

(1) Shutdown regulated pressure to supply tank

(2) Shutdown receiving tank bleed valve

(3) Shutdown isolation valves

M. Purge transfer line

(1) Open dump valve

(2) Open regulated pressurant to transfer line

N. Halt purge

O. Close dump valve

P. Pressurize fuel (MMH) minimum 10 PSIA

Q. Repeat steps I to N inclusive

R. Open pressurant receiving tank isolation valve

S. Open pressurant storage tank isolation valve

T. Open and control pressurant transfer valve until tank pressures are equal

U. Turn on tank heater

V. Close down pressurant isolation valves

3.7.3 (Continued)

W. Turn off pressurant tank heater

X. Open oxidizer transfer line dump valve

Y. Disconnect transfer lines

3.7.4 Weight Summary - The weight added to Gemini is summarized in Table 3.7-1.

TABLE 3.7-1
WEIGHT SUMMARY
PROPELLANT TRANSFER

	WEIGHT – LB.
WEIGHT ADDED TO GEMINI (LEM EQUIPMENT)	(80)
FUEL TANK	12
OXIDIZER TANK	12
FUEL DETECTOR	2
OXIDIZER DETECTOR	2
CONTROL UNIT	7
PRESSURIZATION TANK AND GAS	25
MOUNTING AND CIRCUITRY	20

3.8 Long Duration Mission - The in-orbit configuration of the long duration orbital spacecraft is shown in Figure 3.8-1. The mission section to be added to the Agena is 165 inches long and is mounted between the Forward Auxiliary Rack and Forward Rack. The inflatable tunnel is stored in a fairing attached to the mission section. The fuel cells, reactants, and breathing oxygen are housed in an unpressurized section. Food, water, emergency oxygen, and personal needs are contained in a pressurized section.

The combination access tunnel/living quarters is shown after EV erection, which can be accomplished manually by one man. The tunnel provides easy access to the mission section. It is a structural assembly with a volume of approximately 230 cubic feet. The tunnel selected is based on a Goodyear design which was developed under Air Force contract.

A weight summary of the tunnel and associated end attachments to the Gemini is given in Figure 3.8-2. The weights were taken from a Gemini B study of the same type inflatable tunnel and are directly applicable to this case.

A meteoroid penetration evaluation of the tunnel and Gemini Re-entry Module, for a period of 30 days, is given in Figures 3.8-3 and 3.8-4. The evaluation was based on Aerospace meteoroid penetration environment and penetration criterion (Ref. 3.8-1).

A summary and extrapolation of both Gemini and Apollo fuel cell weights for supplying electrical power for the duration of the mission are given in Figure 3.8-5. The electrical power design point is based on previous estimates including an allowance of 1.6 KW, peak, for experiments and operation of the Agena mounted items which would be carried in either the pressurized or unpressurized sections, as appropriate. Based on analyses conducted during the Gemini B study and Gemini Ferry study (Ref. 3.8-2), the Gemini systems, including the re-entry batteries, should operate properly after the orbital storage period. Power will be supplied

LONG DURATION MISSION CONFIGURATION
45 DAYS

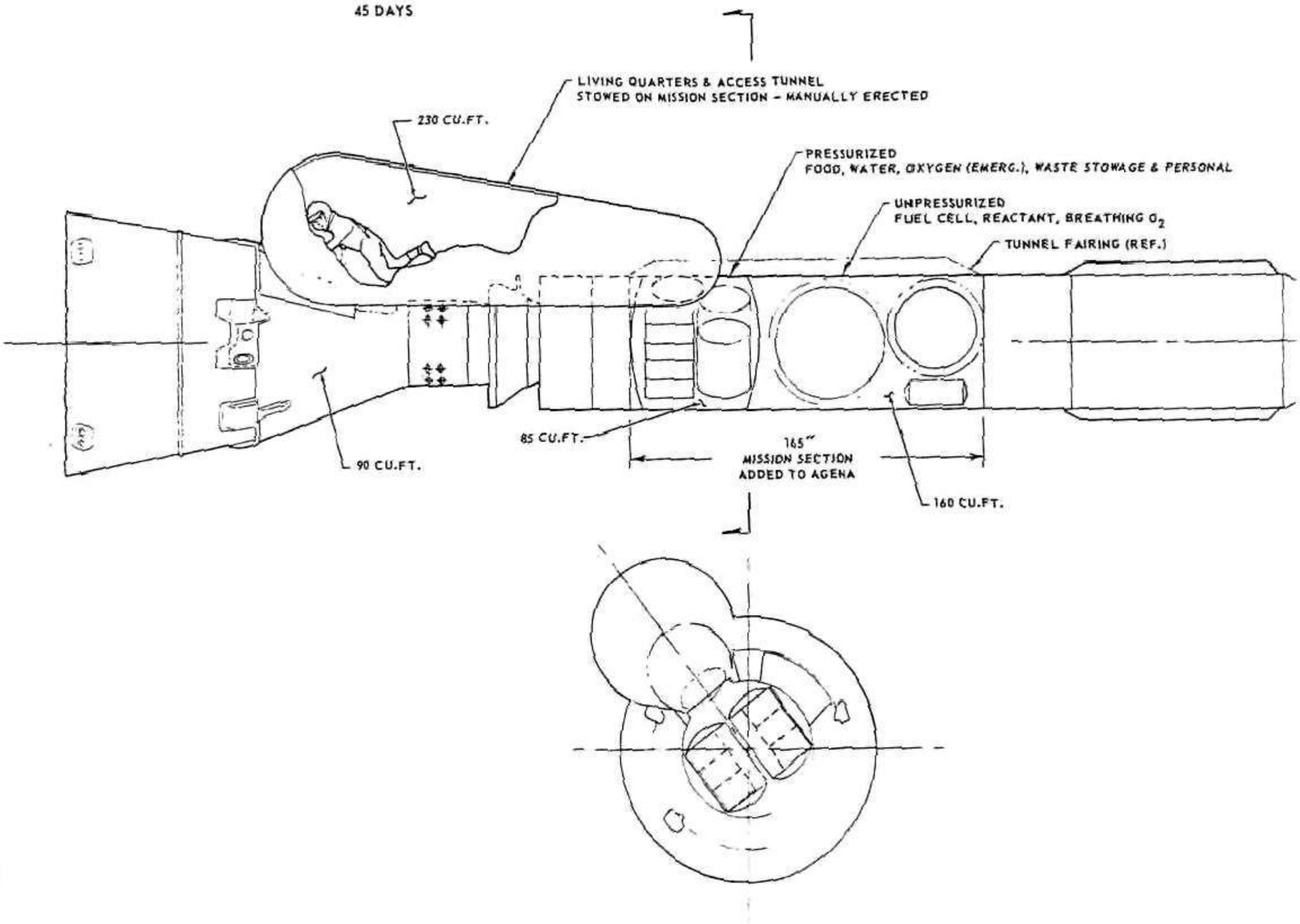


FIGURE 3.8-1

LONG DURATION MISSION CONFIGURATION
45 DAYS

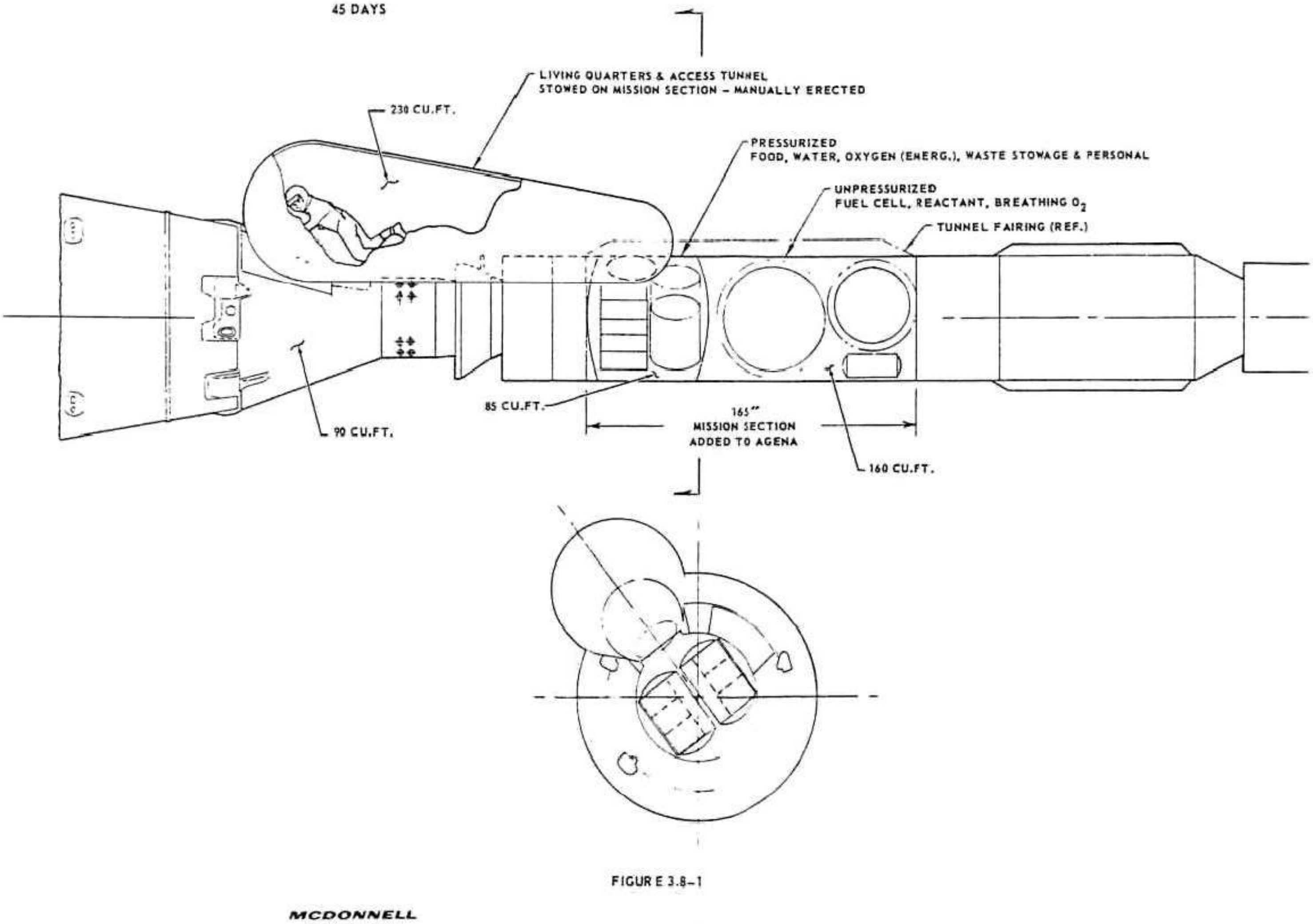
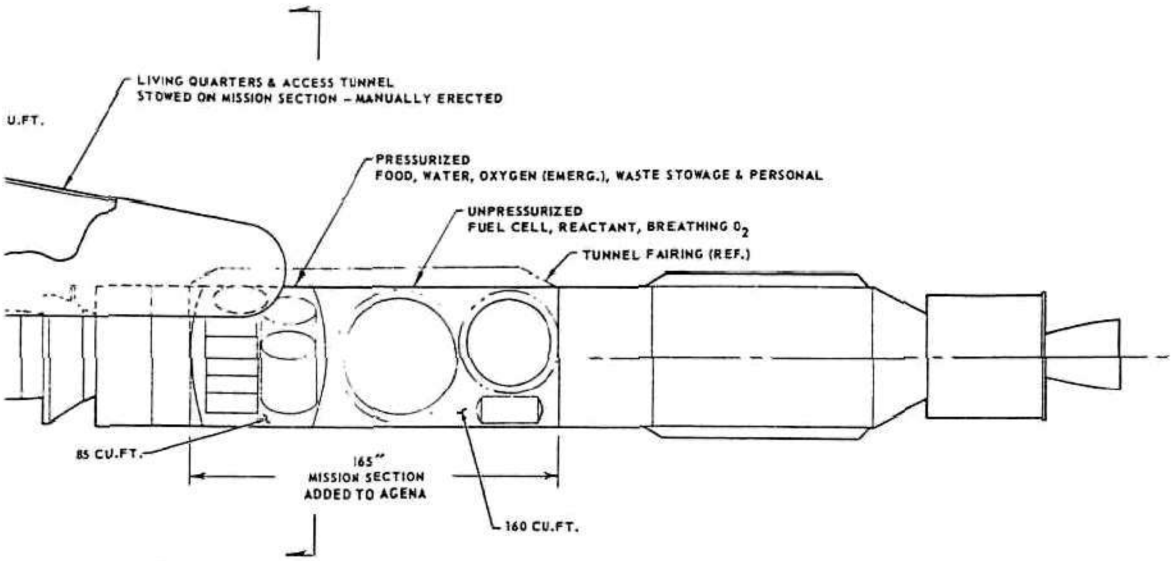
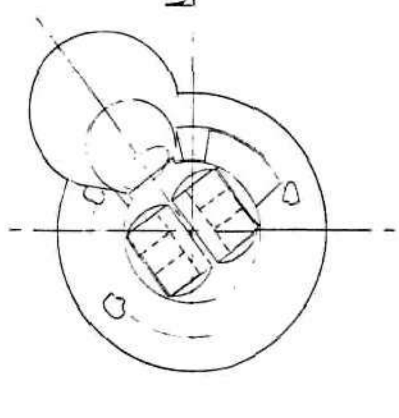
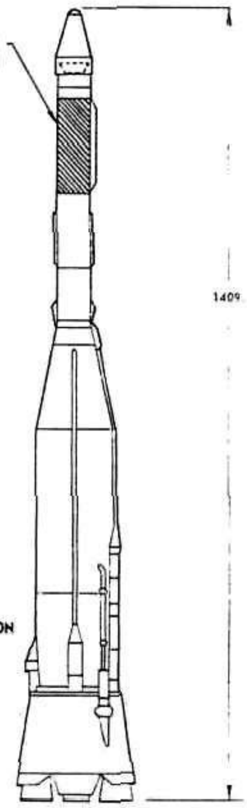


FIGURE 3.8-1

IFIGURATION

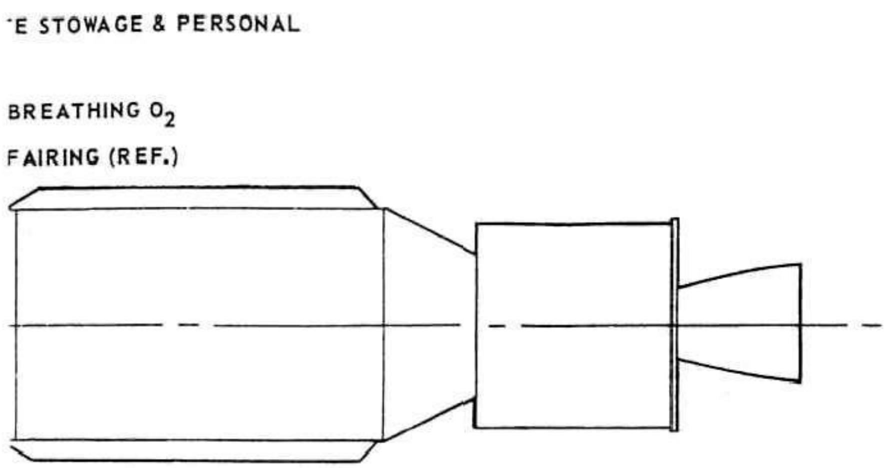


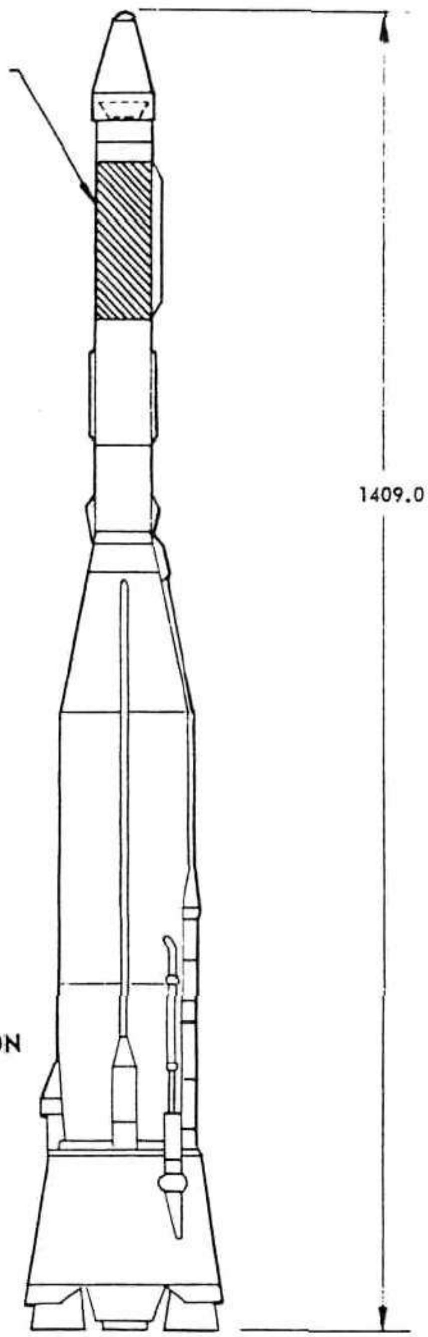




AGENA
LAUNCH CONFIGURATION

FIGURE 3.8-1





AGENA
LAUNCH CONFIGURATION

WEIGHT STATEMENT
INFLATABLE TUNNEL

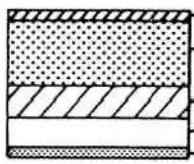
ITEM	DETAILED CALCULATION	WEIGHT (LB.)
TUNNEL/ LIVING QUARTERS	CROSS SECTION  (1) .020 IN. DACRON 52; (2) 1.25 IN. POLYURETHANE; (3) .038 IN. DACRON 52; (4) .060 IN. VINYL FOAM; (5) .020 IN. DACRON 52 — .75 LB./SQ.FT. × 252 SQ. FT. × 1.20%	(226)
		226
FAIRING		(185)
	SKIN – .025 TITANIUM	45
	.025 RENE LEADING EDGE	2
	FRAMES – TITANIUM	29
	STRINGERS – TITANIUM	11
	PADDING – RF300	3
	INSULATION – RF300	25
	FLEXIBLE LINEAR SHAPED CHARGE	30
	ATTACHMENT – TITANIUM	40
HATCH-IN HATCH		(34.0)
	HATCH-IN-HATCH	
	EDGE RING – TITANIUM AT .51 SQ.IN. + RUBBER SEAL	7.8
	STRUCTURE – SHINGLES, SKIN, STIFFENERS,	6.5
	WINDOW AND FRAME	11.5
	MODIFICATION TO PRESENT HATCH	
	REMOVE WINDOW	–13.3
	STRUCTURAL CUT-OUT	–4.2
	ADD HATCH SILL TITANIUM AT .45 SQ.IN.	5.5
	HATCH BEEF-UP	12.3
	LATCHING MECHANISM	2.8
	TUNNEL SILL	5.1
COMMUNICATIONS AND LIGHTS		(10.0)
ENVIRONMENTAL CONTROL SYSTEM	LONG UMBILICAL, 2 – 25 FT. AT 0.3 LB. FT.	(15.0)

FIGURE 3.8–2

INFLATABLE TUNNEL PENETRATION EVALUATION

30 DAY MISSION
EARTH SHIELDING FACTOR = 0.7

NOTES:
1. A_p = 75 SQ. FT. (MAXIMUM)
2. A_S = 150 SQ. FT.
3. FOAM DENSITY = 1.2 LB./CU.FT.

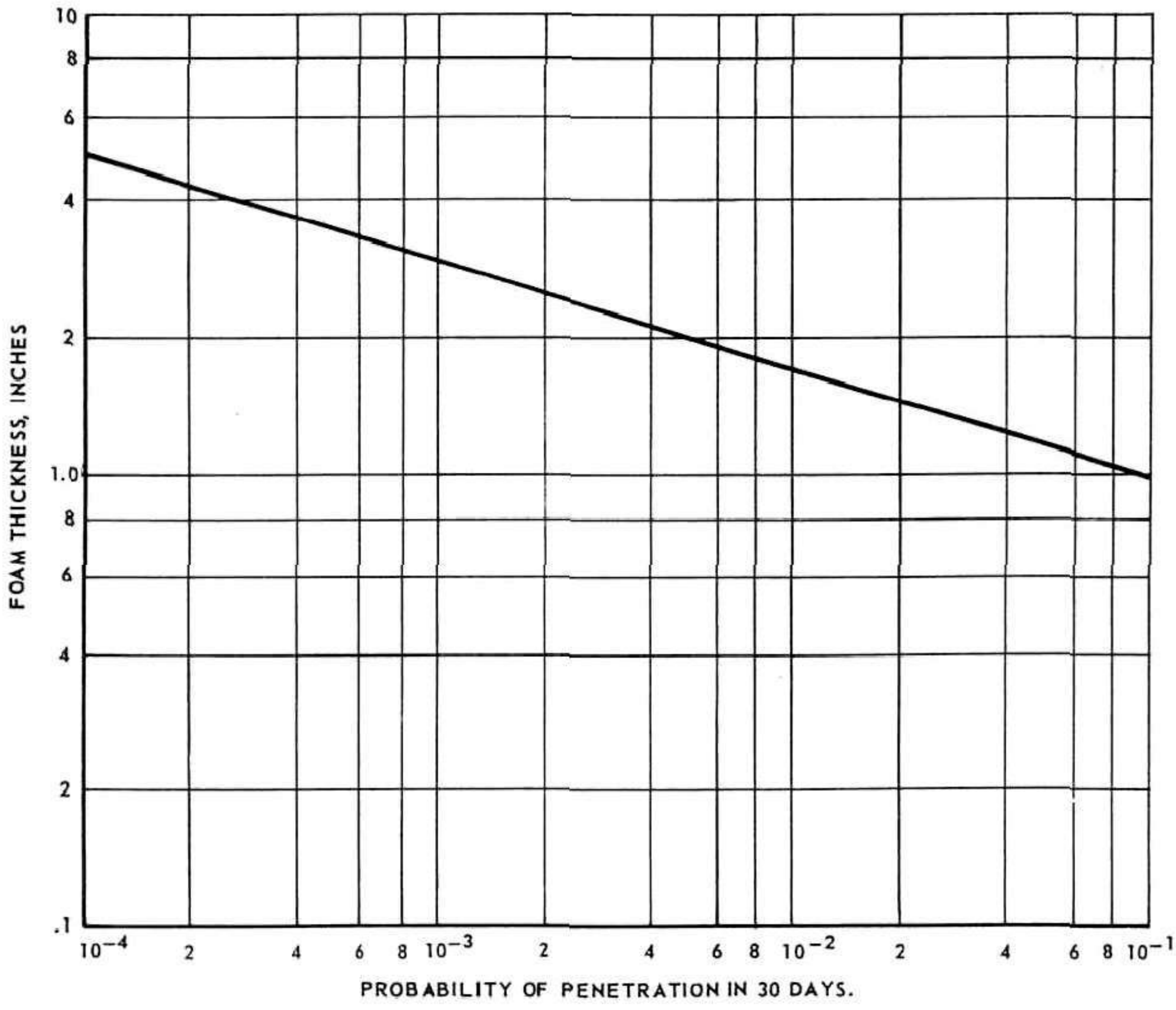


FIGURE 3.8-3

METEOROID HAZARD EVALUATION

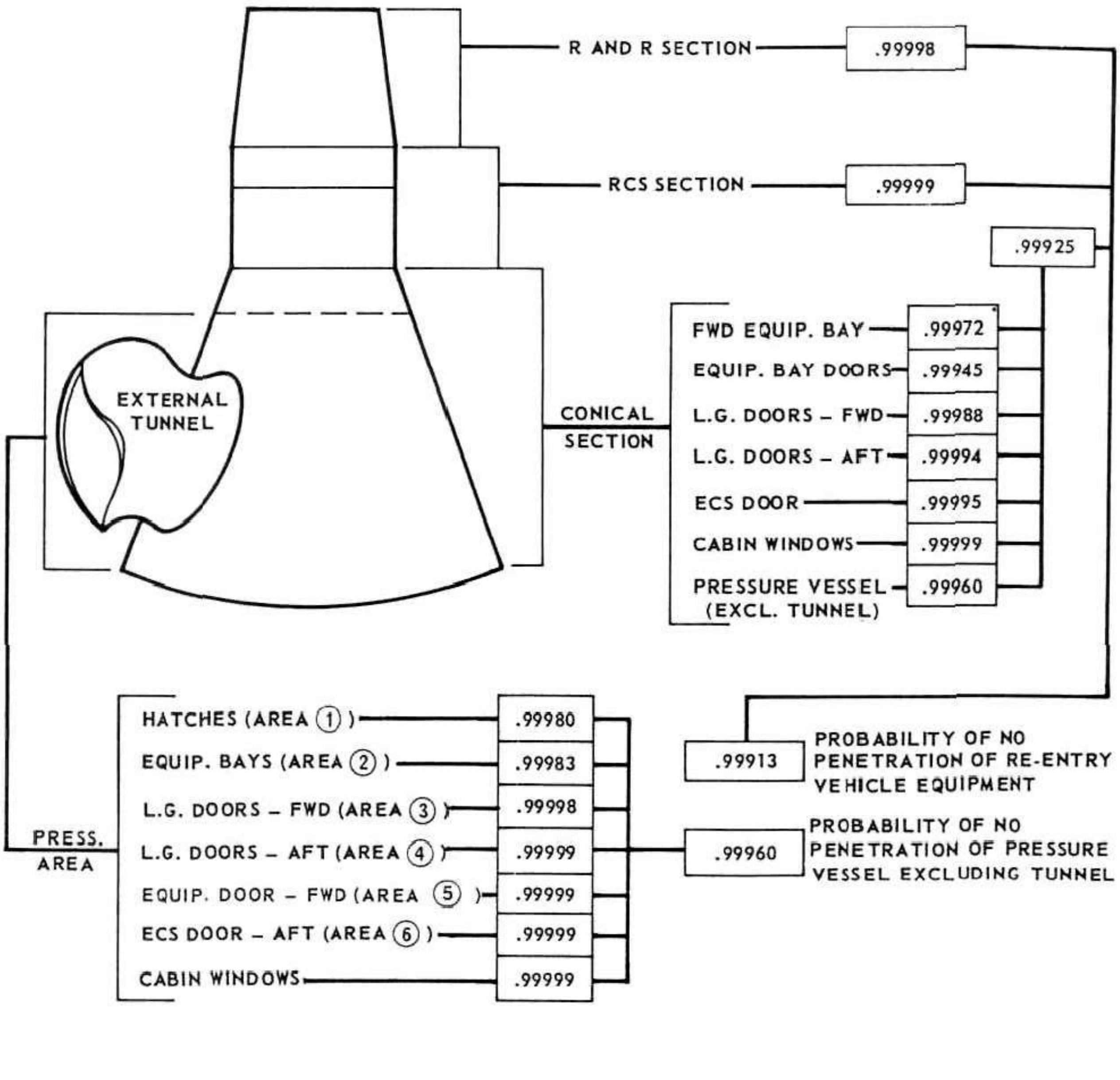


FIGURE 3.8–4

ELECTRICAL POWER SUMMARY

1. FUEL CELLS APPEAR TO BE THE MOST APPLICABLE POWER SOURCE TO BE USED IN THE CAN, ATTACHED TO THE AGENA, FOR ORBITAL OPERATIONS.
2. POWER TO OPERATE GEMINI ORBITAL STORAGE LOADS (RCS HEATERS, WARMANT LOOP, TELEMETRY) SUPPLIED FROM CAN TO THE GEMINI THROUGH THE DOCKING ADAPTER UMBILICAL CONNECTOR.

ITEMS	GEMINI	APOLLO
FIXED HARDWARE	414 LB.	508 LB.
REACTANTS (1,000 KWH)	1,342 LB.	1,212 LB.
TANKAGE	610 LB.	542 LB.
TOTAL (30 DAYS)	2,366 LB.	2,262 LB.
TOTAL (45 DAYS)	3,342 LB.	

ASSUMPTIONS: AVERAGE LOAD – 1,400 WATTS
PEAK LOAD – 2,000 WATTS
INITIAL CAPACITY INSTALLED – 4,000 WATTS

FIGURE 3.8–5

3.8 (Continued)

to the appropriate Gemini systems through the Gemini target docking adapter umbilical.

A preliminary weight breakdown of the mission section and associated equipment, to be launched on the Atlas-Agena, is given in Table 3.8-1.

Habitability aspects of a spacecraft have a direct relationship to mission duration and crew performance. One of these aspects is the free space, which is defined as the pressurized cabin space less that occupied by the man himself, instruments, and other equipment (a pressure suited crewman will normally require 5 cubic feet). In contrast to other provisions, such as life support, the requirement for free space cannot be precisely defined. The design goal is to provide a spacecraft volume which is just adequate, particularly when increased volume might add substantially to cost or delay the achievement of operational flight capability.

To put the requirement of free space in better perspective, 37 studies reporting the behavioral aspects of confinement were reviewed (Ref. 3.8-3). These were studies judged to be relevant to spacecraft design and involved the use of experimental chambers, simulated vehicles, and operational vehicles. The results of the analysis, which integrated mission duration, volume per man and performance, and physicological factors are depicted in Figure 3.8-6, taken from Reference 3.8-3. Relating this analysis to the 30-45 day Gemini mission, the following extrapolations are pertinent:

A. Volumes per man of less than approximately 40 cubic feet can result in severe degradation in performance and physiological functioning.

B. For missions up to thirty days, volumes per man between 100-200 cubic feet appear to be satisfactory.

C. For missions longer than thirty days, additional volume allocations beyond 200 cubic feet per man becomes relatively less important as a determinant of habitability.

TABLE 3.8-1

WEIGHT SUMMARY
LAND DURATION MISSION

WEIGHT ADDED TO AGENA	POUNDS	
SECTION ADDED TO AGENA		327
STRUCTURE	304	
HATCH	23	
DOCKING ADAPTER		360
INFLATABLE TUNNEL		276
TUNNEL	226	
RINGS AND PORT HOLE	50	
FAIRING		185
COMMUNICATIONS AND LIGHTS		10
ENVIRONMENTAL CONTROL SYSTEM		1,210
FOOD AND CONTAINERS	190	
DRINKING WATER	585	
TANK AND MOUNTING	104	
BREATHING OXYGEN	180	
TANK, MOUNTS AND VALVES	76	
HOSES	15	
EMERGENCY OXYGEN AND TANK	60	
ELECTRICAL POWER SYSTEM		3,482
FUEL CELLS AND HARDWARE	414	
REACTANTS	2,013	
TANKAGE, MOUNTS AND VALVES	1,055	
TOTAL WEIGHT ADDED TO AGENA		5,850 [1]
WEIGHT ADDED TO GEMINI		
HATCH IN HATCH		34

(1) ATLAS – AGENA B CAPABILITY AT 150 NA.MI. $(-3\sigma) = 6200$ LB. (REF. 3.8-6)

3.8 (Continued)

The most directly related study was done in the School of Aerospace Medicine (SAM) 2-man Space Cabin Simulator, Reference 3.8-4. Two Air Force pilots were confined for 30 days in a chamber providing 190 cubic feet per man. The mission plan called for operational tasks approximately 50 percent of the time. Some decrement in work capacity occurred during the period. Inter-crew compatibility was satisfactory where incidents that elicited hostility were minimized. The most recent study (Ref. 3.8-5) involved six pilots (attired in pressure suits) who spent 34 days in a space cabin simulator providing 167 cubic feet per man. The crew members were required to perform operational tasks 50 percent of the time. Preliminary results indicate that there were not major problems. It should be noted that an allotment of 167 cubic feet per man in a large crew is probably equivalent to 200 cubic feet per man in a two man station where there is less opportunity to share space.

Based on this analysis of the space requirements, the mission section on the Agena for the long duration mission should provide a free space minimum volume allotment of 150 cubic feet per man. As shown on the concept presented in Figure 3.8-1, the access tunnel/living quarters provides a volume of approximately 230 cubic feet. This should be adequate since one crew man should remain on duty in the Gemini cabin at all times.

The long duration mission discussed offers a number of appealing aspects including:

A. Utilization of existing Gemini (GLV) Atlas/Agena/TDA equipment to the maximum extent possible.

B. Development of E.V. operation.

C. Utilization of an inflatable tunnel developed under Air Force Contract.

D. Accomplishment of structural assembly in space.

3.8 (Continued)

Mission stay times are shown in Figure 3.8-7 for three assumptions: (1) water included for pressure suit environment, (2) water included for shirtsleeve environment, and (3) water assumed produced by fuel cells.

MISSION STAY TIME CAPABILITY

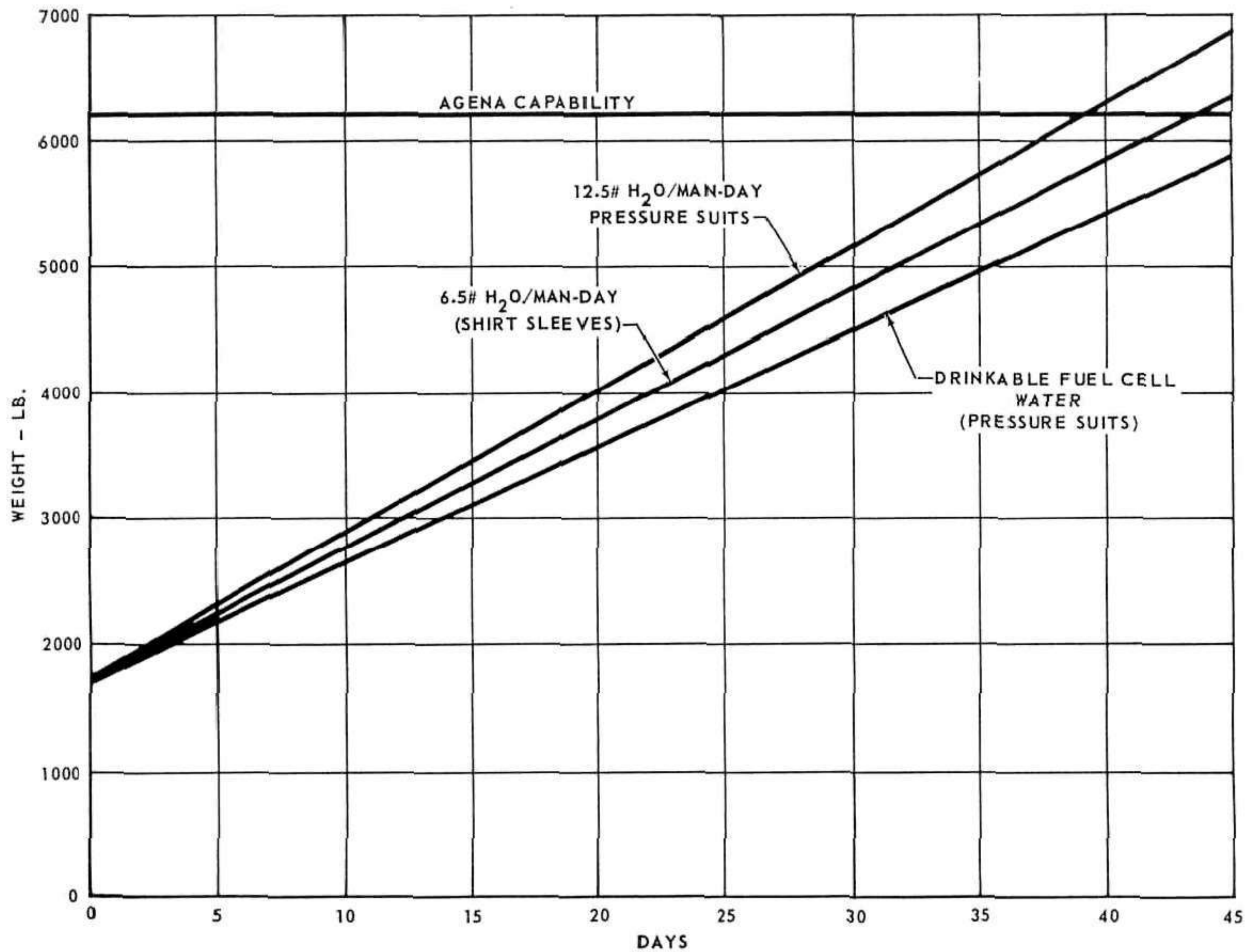


FIGURE 3.8-7

3.9 Land Landings - Two approaches to land landing are presented in Section 2.9. Four other approaches considered but ruled out because they involve extensive change to Gemini, are also discussed. Further descriptions of all six approaches are presented in the following paragraphs.

3.9.1 Landing Rocket Suspended from Parasail Risers - Design changes required to the parasail version of Gemini to incorporate a landing rocket suspended from the parasail, as shown in Figures 3.9-1 and 3.9-2, are as follows:

A. Rendezvous radar is omitted, or relocated, and the parasail cannister changed to allow for installation of the landing rocket in the rendezvous and recovery (R&R) section. Some structural changes are also needed in this section.

B. A device for controlling the deployment of the landing rocket is installed in the R&R section.

Areas requiring thorough analysis before a landing scheme of this type is pursued for Gemini are as follows:

A. Landing rocket deployment effects on spacecraft stability. Particular attention should be given to study of the directional stability of the spacecraft during and following rocket deployment.

B. Effects of plume impingement on the spacecraft and supports.

C. Possibility of burned out rocket motor collision with spacecraft.

3.9.2 Cloverleaf Landing System - Design changes required to incorporate a cloverleaf landing system, as shown in Figure 3.9-3, are as follows:

A. The recovery section is modified to accommodate the cloverleaf installation. At this time, it is believed the changes required are not extensive.

B. A control system, different from the Gemini parasail control system, is located in the top centerline torque box.

PRESENT PARASAIL LANDING CONFIGURATION

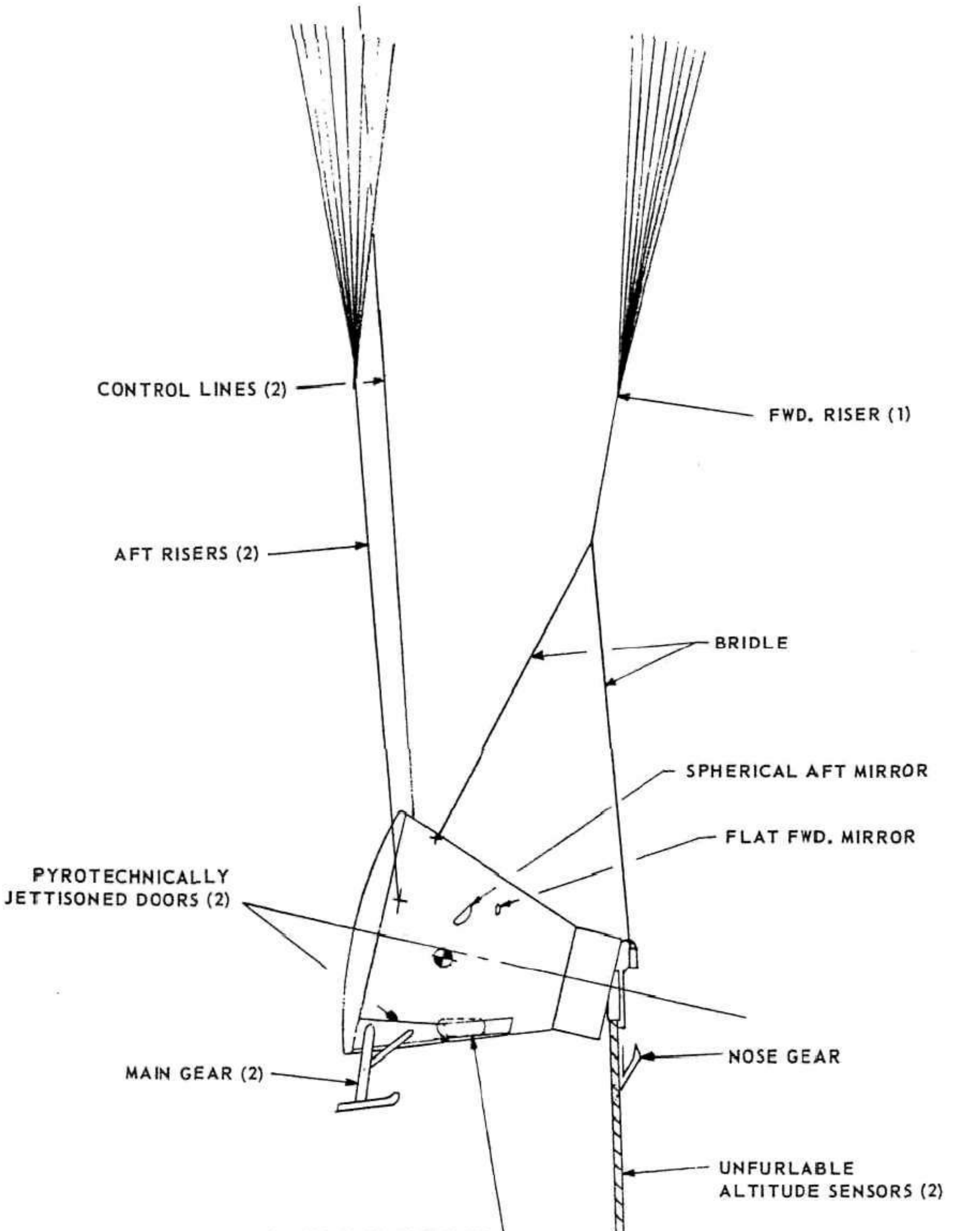


FIGURE 3.9-1

PARASAIL LANDING CONFIGURATION
WITH SUSPENDED LANDING ROCKET

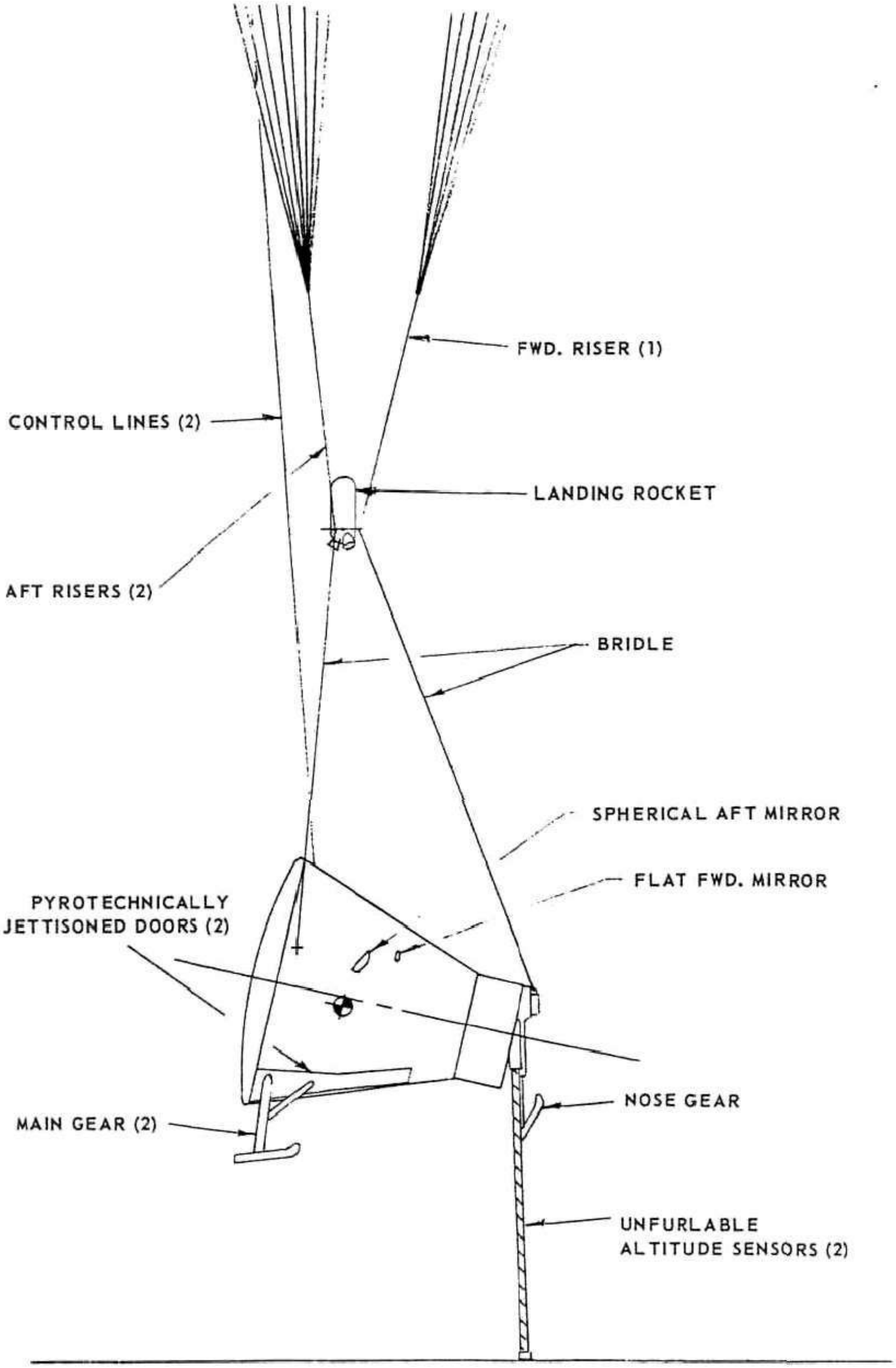


FIGURE 3.9-2

CLOVERLEAF LANDING CONFIGURATION

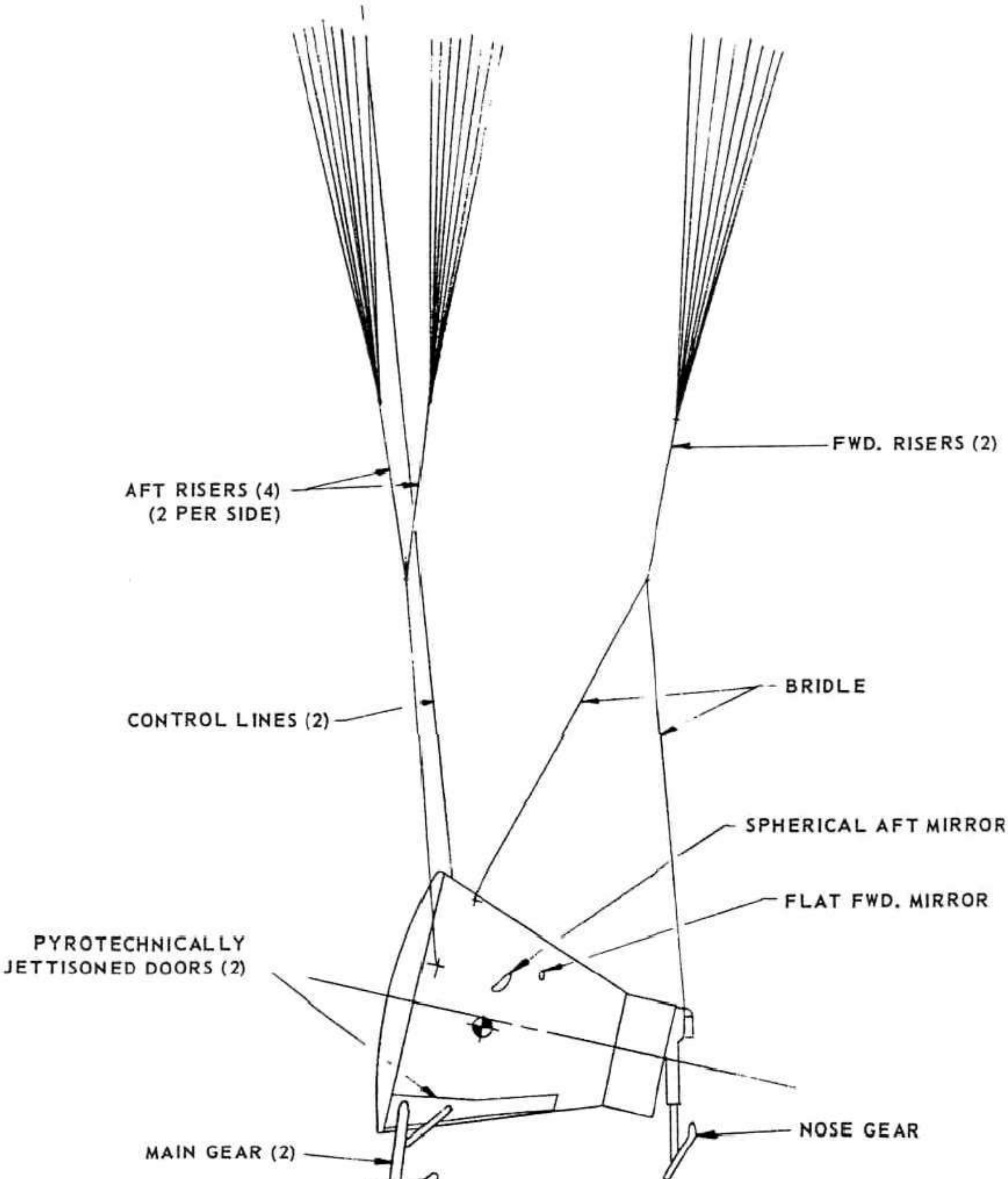


FIGURE 3.9-3

3.9.2 (Continued)

Development of the cloverleaf is in the initial stages and looks promising, but more data is required in the areas applicable to Gemini.

Areas which should be thoroughly investigated in the development of this type of landing system are as follows:

A. Reefing requirements necessary to keep cloverleaf deployment loads below the 16,000 pound limit load of Gemini are to be determined.

B. Control forces required are considerably higher than that of the Gemini parasail and therefore may require a different design approach from that used for the parasail.

The Gemini landing gear design parameters, also used for the cloverleaf analyses, are shown in Table 3.9-1.

TABLE 3.9–1
GEMINI LANDING GEAR DESIGN PARAMETERS

MAXIMUM GROUND WIND VELOCITY – 30 FPS.
LANDING AREA BEARING STRENGTH – 200 PSI MINIMUM.
TOUCHDOWN AREA TO BE CLEAR OF OBSTACLES LARGER THAN TWO INCHES HIGH.
ALLOWABLE IMPACT VELOCITY – 15 FPS FOR NOSE DOWN OF –16 DEGREES, 13 FPS FOR NOSE UP OF 12 DEGREES.
GROUND SLOPE – 5 DEGREES MAXIMUM IN ANY DIRECTION.
LIMIT DEPLOYMENT LOAD – 16,000 LB.

The cloverleaf chute being considered for application to Gemini has a wetted equivalent diameter of approximately 100 feet and horizontal velocity control from 10 to 24 fps. Maximum descent velocity would be approximately 26 fps, with a range at landing of 13-14 fps.

The estimated weights for the various configurations are shown in Table 3.9-2.

TABLE 3.9-2

WEIGHT SUMMARY
LAND LANDING CONFIGURATIONS

ITEM	GEMINI PARASAIL CONFIG. (LB.)	ALTERNATE PARASAIL CONFIG. (LB.)	CLOVERLEAF (LB.)
REMOVE:	(-618)	(-618)	(-618)
RADAR	-88	-88	-88
OAMS TANKS & PRESSURANT	-44	-44	-44
OAMS PROPELLANT	-288	-288	-288
DOCKING SYSTEM	-19	-19	-19
PARACHUTE SYSTEM	-159	-159	-159
ROLL BAR, FLOTATION AIDS IN RCS SECT.	-20	-20	-20
ADD:	(619)	(644)	(577)
PARASAIL	269	269	-
LANDING ROCKETS	93	93	-
LANDING GEAR	236	236	236
EQUIPMENT RELOCATION	21	21	21
CLOVERLEAF	-	-	300
ROCKET DEPLOYMENT MECHANISM	-	15	-
R & R STRUCTURAL MODIFICATION	-	10	20
BALLAST ADJUSTMENT	(-16)	(-12)	(-60)
TOTAL SPACECRAFT WEIGHT CHANGE	-15	+14	-101

3.9.3 Impact Bags - Installation of an impact bag between the large pressure bulkhead and heat shield, plus the installation of a toroidal impact bag around the recovery section, was investigated for application to Gemini. The arrangement is shown in Figure 3.9-4.

The descent system used for land landings requires sufficient controllability to permit maneuvering in the landing area. For this reason a parasail was selected as the descent system.

The cylindrical aft impact bag is designed to attenuate the descent velocity and the toroidal impact bag to stop any tumbling which may occur. This system is not adversely affected by variations in impact area sliding coefficients of friction since tumbling is assumed to occur.

Behavior of the spacecraft upon impact and its physical and psychological effects upon the crew may prove this system to be an undesirable landing scheme for manned vehicles.

3.9.4 Cable and Spike Landing Schemes - Alternate approaches for using a cable attached to a spike for horizontal velocity attenuation are shown in Figures 3.9-5 and 3.9-6. The re-entry module is maneuvered into a suitable landing area where the spike attached to the cable is driven pyrotechnically into the ground.

The method shown in Figure 3.9-5 employs an impact bag for vertical velocity attenuation. After the cable has been anchored to the ground, the glide chute is trimmed to behave as a parachute and the spacecraft continues its descent, drifting with the wind. A winch mounted in the spacecraft reels in the cable slack, until the cable is perpendicular to the flight path. The spacecraft then follows an arc defined by the cable. Upon impact with the ground horizontal velocity has been attenuated.

IMPACT BAG LANDING SYSTEM

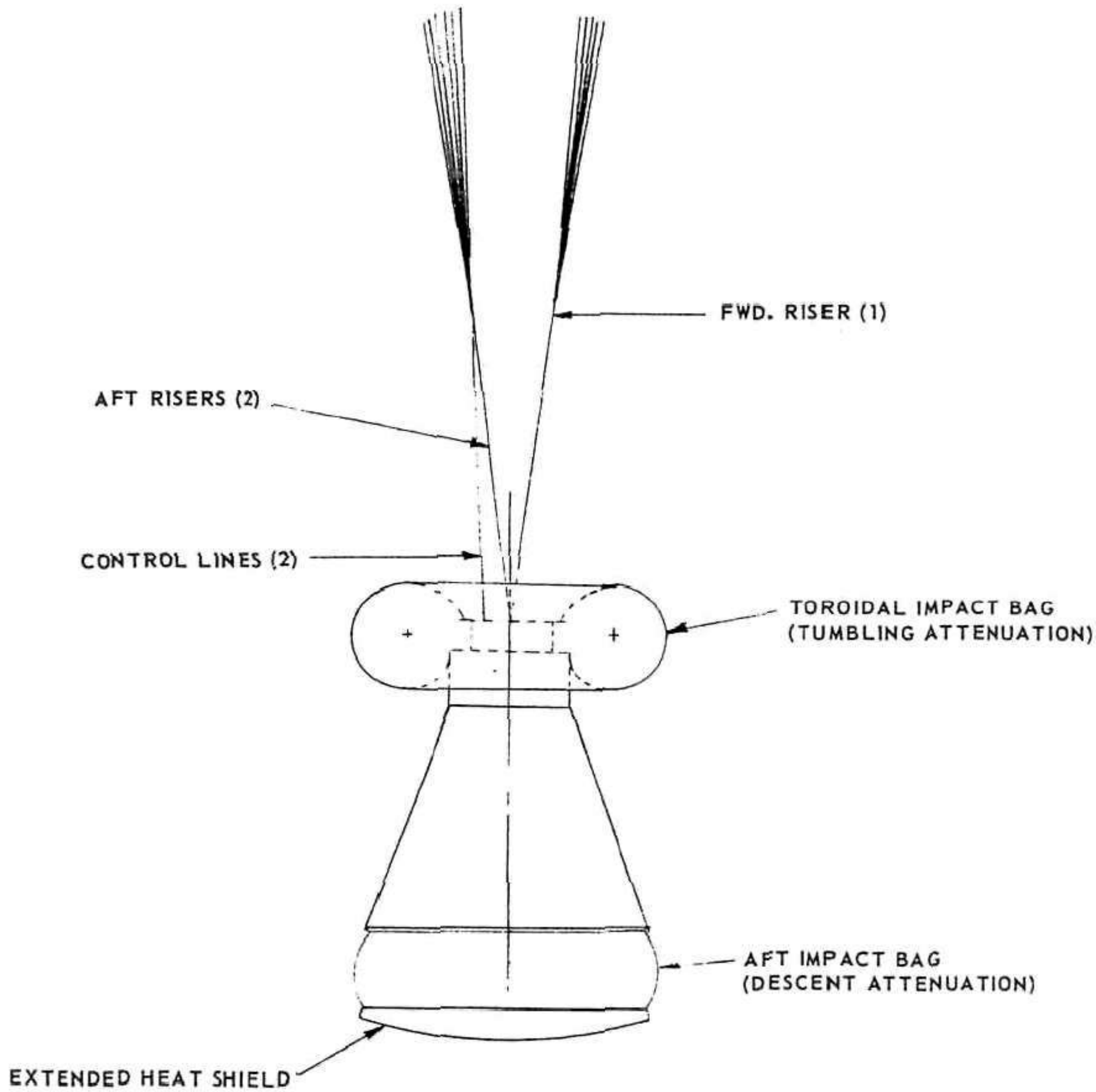


FIGURE 3.9-4

CABLE – SPIKE & IMPACT BAG

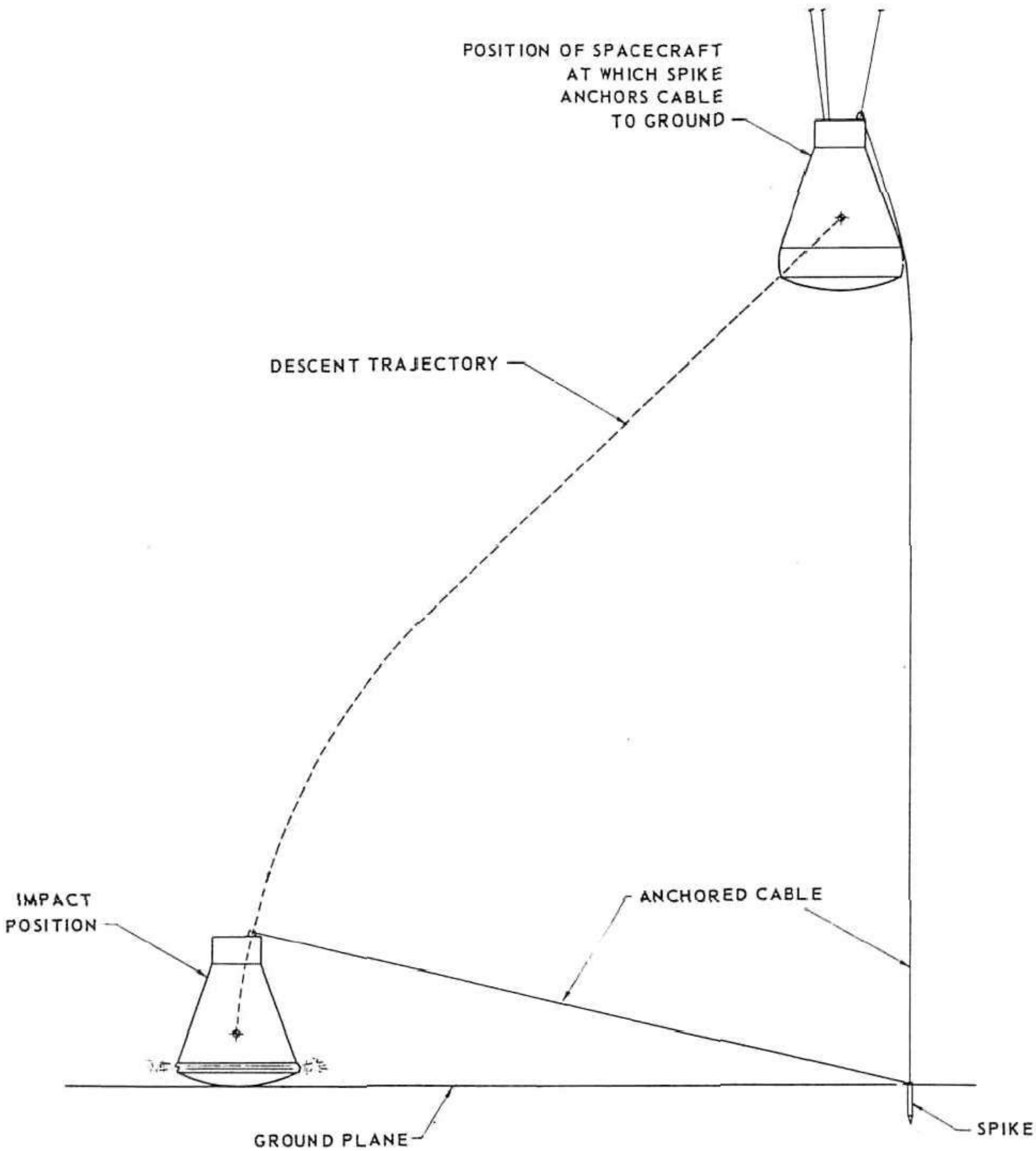


FIGURE 3.9-5

3.9.4 (Continued)

The method shown in Figure 3.9-6 employs a landing rocket and passive attenuation between the large pressure bulkhead and heat shield for vertical velocity attenuation. After the cable has been anchored to the ground, the cable is maintained taut with a minimum force until the landing rocket attitude sensor initiates landing rocket ignition and engages the cable load brake located on the spacecraft. The amount of cable tension applied by the load brake depends on the length of cable extended, i.e., cable length is dependent upon horizontal velocity. The horizontal velocity is then dissipated in the load brake and the spacecraft impacts without horizontal velocity.

3.9.5 Horizontal and Vertical Landing Rockets - A landing rocket arrangement which uses horizontal firing rockets for horizontal velocity attenuation and a vertical firing rocket for vertical velocity attenuation is illustrated in Figure 3.9-7. The same altitude sensing system used to ignite the vertical rockets is used to ignite the horizontal rockets. The number of horizontal rockets fired depends on the relative ground speed of the re-entry module. The ground speed would be determined either by crew judgement or automatically by radar. The attenuation system could be used in conjunction with a parasail descent system.

3.9.6 Larger Landing Gear - Providing a larger gear with increased strength and stroke is not applicable to Gemini without beefing up the landing gear support structure. The landing gear support structure fittings are integral parts of the Gemini structure. Therefore, modification of the present landing gear to increase its capability to accommodate the higher descent velocities associated with the parasail landing system would require extensive redesign of the re-entry module.

CABLE – SPIKE & LANDING ROCKET

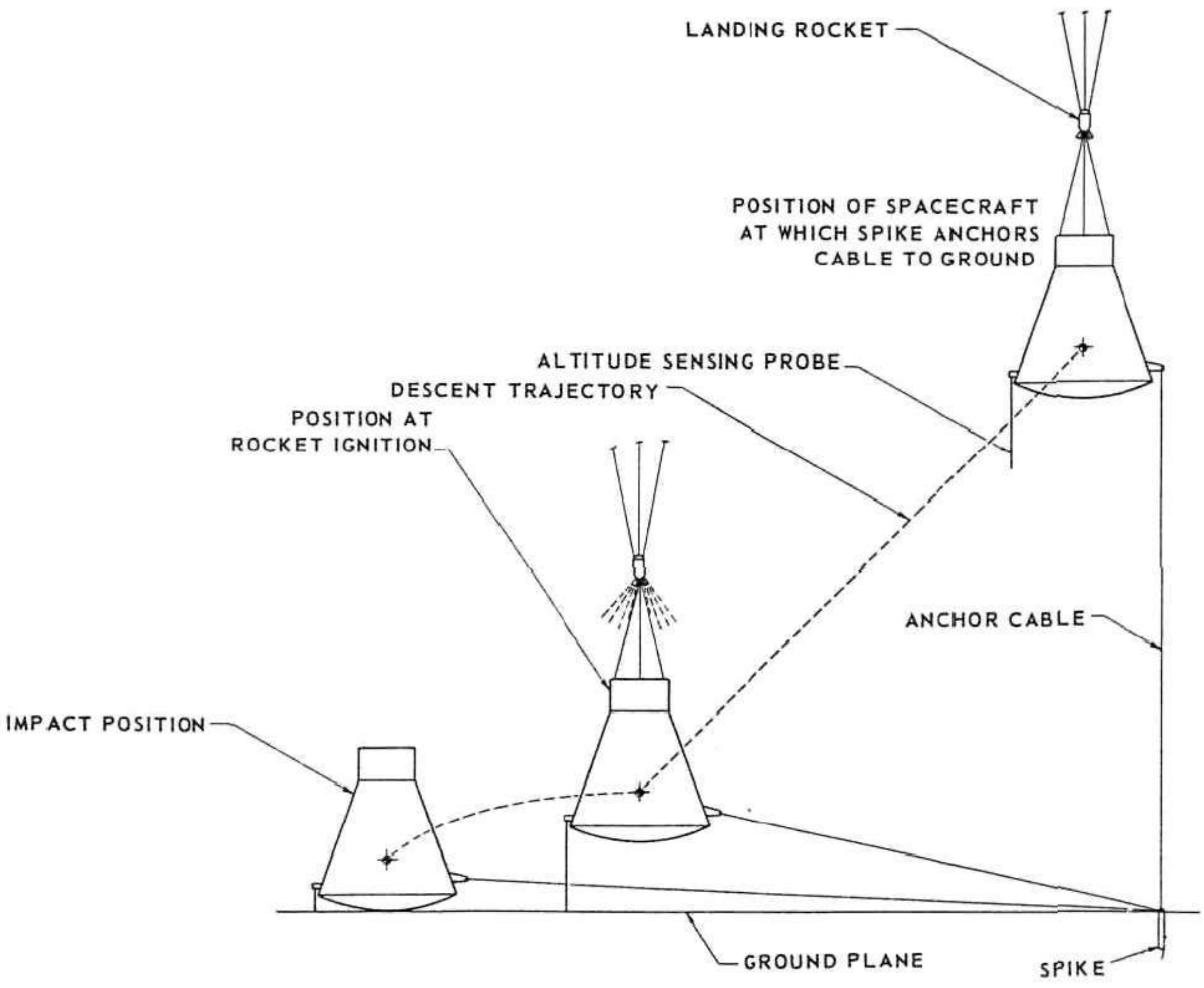


FIGURE 3.9–6

HORIZONTAL & VERTICAL LANDING ROCKETS

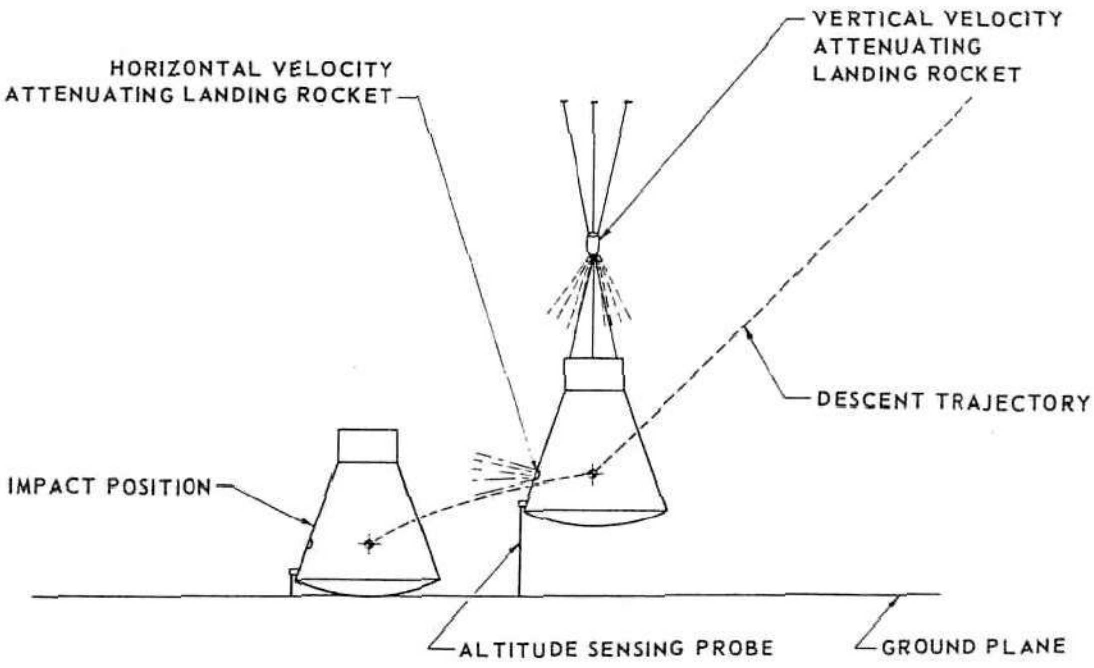


FIGURE 3.9-7

4. MISSION SUMMARY, COSTS, AND SCHEDULES

4.1 Mission Summary - The major experiment hardware and equipment changes, significant development tasks, and estimated acquisition time are summarized in Table 4.1-1. Compatibility with currently contracted spacecraft with respect to schedule, other experiments and to hardware and mission requirements are also indicated.

4.2 Costs - The estimated experiment incremental costs are presented in Table 4.2-1. These costs, in current dollars, are derived using standard MAC cost analysis procedures and are based on documented cost records of comparable efforts.

The McDonnell portion of the launched cost for a particular experiment may be obtained by adding to the experiment cost the contract price of the Gemini spacecraft utilized and the appropriate launch vehicles. If a new spacecraft is required to be built outside the current contract period, the estimated cost to be added to the experiment cost is twenty-five million. Spacecraft refurbished during and outside the current contract period are estimated to cost five million and fifteen million, respectively.

Experiment costs are based on the experiments being incorporated in the basic Gemini with a parachute recovery system and rendezvous capability, except for except for experiment 9A. The cost of this experiment is additive to the contract change proposal, submitted by MAC for a parasail configuration without rendezvous capability.

4.3 Schedules - The schedules for the various experiments, based on an assumed go-ahead of 1 July 1965, are presented in Figures 4.3-1 through 4.3-13. These schedules are predicated on past Mercury and Gemini experience.

The manufacturing flow times and subsequent deliveries are fixed in relation to estimated availability of experimental or developed system hardware, costs, facilities, AGE, and selection of spacecraft structure, (existing, new or refurbished).

4.3 (Continued)

Continued investigation and further analysis may affect changes in estimated availability of determining factors, and consequently vary target delivery dates of spacecraft for respective experiments.

TABLE 4.1-1
MISSION SUMMARY

NO	TITLE	EQUIPMENT	WEIGHT - LB	HARDWARE CHANGES	DEVELOPMENT TASKS	ACQUISITION TIME (MONTHS)	COMPATIBLE WITH
1	RENDEZVOUS WITH UNMANNED SATELLITE	Augment OAMS for self-inject – 2 additional sets of OAMS tanks and propellant – 4 additional 100 lb. thrusters, augment retro – 4 additional retro rockets	1922	Modify adapter to accommodate additional OAMS and retro (approximately 150 lb. additional structural weight).	Extensive computer studies of injection, rendezvous, and retro maneuvers. Structural tests of adapter and installations. Develop OAMS fuel system	20	None
2	ONE MAN GEMINI – EARTH SURFACE MAPPING	Panoramic cameras (2) Horizon camera Color cameras (2) Viewfinder	300	Remove RH crew station, LH crew member 7 day supplies, OAMS propellant, and experiments. Add bulkhead between seats, automatic hatch actuator, camera mounts, horizon sensor, and control panel.	Qualify panoramic camera, horizon camera, and viewfinder. Develop automatic hatch actuator, and horizon camera extension system.	24	None
3A	ONE MAN GEMINI WITH ASTRONOMICAL TELESCOPE (MOUNTED IN ADAPTER)	26 inch diameter optical system	560	Remove RH crew station, LH crew member 9 day supplies, OAMS propellant, and experiments. Modify retrograde rocket beams. Add fine attitude control system, horizon sensor, data recording device, hatch in heat shield, adapter tunnel, telescope rotating device, telescope mounts, fuel cell for gyro packages, and adapter extension.	Qualify hatch in heat shield, retrograde rocket impingement, retrograde rocket beams, and telescope fairing. Develop 26 inch diameter optical system, fine attitude control system, adapter tunnel seal, tunnel to telescope interface, and telescope rotating device.	28	None
3B	ONE MAN GEMINI WITH ASTRONOMICAL TELESCOPE (MOUNTED IN RE-ENTRY VEHICLE)	16 inch diameter optical system	400	Remove RH crew station, LH crew member 9 day supplies, OAMS propellant, and experiments. Add fine attitude control system, horizon sensor, data recording device, bulkhead between seats, automatic hatch actuator, telescope mounts, fuel cell for gyro packages.	Develop 16 inch diameter optical system, fine attitude control system, and automatic hatch actuator.	24	None
4A	ARTIFICIAL GRAVITY EXPERIMENT (SOLID COUPLE TO EXPENDED STAGE II OF GLV)	Angular velocity display gyro with rate range ± 40 degrees/second (92 lb. of available propellant required for 0g – 1g – 0g maneuver)	7	Mount angular velocity display and gyro	Qualify gyro and angular velocity display. Develop restraint system for repositioning crewmen for eye balls down in acceleration. Conduct attitude control and stability analysis. Determine physiological limitations of mission. Determine mission communication restraints.	11	S/C 10, 11, 12 Experiment 5, 6 and 7
4B	ARTIFICIAL GRAVITY EXPERIMENT (SOLID COUPLE TO AN AGENA)	Angular velocity display gyro with rate range ± 40 degrees/second (128 lb. of available propellant required for 0g – 1g – 0g maneuver)	7	Mount angular velocity display and gyro	Qualify gyro and angular velocity display. Conduct attitude control and stability analysis. Determine physiological limitations of mission. Determine mission communication restraints	11	S/C 10, 11, 12 Experiments 5 and 7
4C	ARTIFICIAL GRAVITY EXPERIMENT (CABLE COUPLE TO AGENA OR EXPENDED STAGE II OF GLV)	Angular velocity display and gyro, cable and cable reeling system (drum and motor) (approximately 200 lb. of available propellant required for 0g – 1g maneuver)	127	Mount angular velocity display and gyro. Mount cable and associated equipment (for Stage II of GLV – approximately 30 lb. structure) (For Agena – separate Agena section adjacent to target docking adapter is required. Approximately 75 lb. structure)	Qualify gyro and angular velocity display. Conduct attitude control and stability analysis. Determine physiological limitations of mission. Determine mission communication restraints. Develop cable system and installation. Conduct structural tests, dynamic model tests and extensive analog simulation study	15	S/C 12 Experiment 5 and 7
5	SIMULATION OF LEM RENDEZVOUS	5A. Add LEM equipment to Gemini to perform computations and radar observations (computer and power supply IMU and radar) 5B. Modify Gemini equipment to simulate LEM rendezvous	223	Install LEM computer and power supply, IMU and radar	Extensive digital computer simulations. To develop operational computer program. Define crew operations and provide crew training	17	S/C 11, 12 Experiments 4, 7 and 9B
6	STRUCTURAL ASSEMBLY IN ORBIT	40 ft. diameter erectable antenna. ECS umbilical and miscellaneous handles	713	Install antenna and fairing to Gemini (include structural beef up to Gemini – structural weight approximately 100 lb.)	Wind tunnel tests of launch configuration. Develop erectable antenna. Conduct structural tests of antenna installation for launch loads	16	S/C-12 Experiments 4A and 9
7	PROPELLANT TRANSFER	Add to Gemini: Propellant tanks, Pressurant tank flow meters, Gauging systems, Transfer lines. Add to Agena: Propellant tanks, Propellant, Pressurant tank, Pressurant	100 / 200	Modify Gemini for installation of propellant tanks and pressurant tank. Modify Agena for installation of propellant tanks and pressurant tank	Qualify propellant transfer system. Develop extravehicular activity capability, propellant centrifugal separator, and propellant retention device	24	Experiments 4, 5, and 9B
8	LONG DURATION MISSION	Agena mission section: Expandable living quarters, ECS extension and living quarters circulation, Additional electrical power, Drinking water, food and containers, Breathing O_2, bottles and mounts, Agena structure, Docking adapter	5850	Add hatch in hatch and launch fairing (approximately 34 lb.)	Wind tunnel tests of launch configuration. Develop tunnel and living quarters. Develop mission section. Hatch in hatch and fairing structural tests	18	None
9A	LAND LANDING PARASAIL	Parasail, Landing rocket, Landing gear, Rocket deployment mechanism	644	Remove radar, OAMS tanks, and pressurant, OAMS propellant, docking system, parachute system, roll bar, flotation aids in RCS, and adjust ballast. Modify R & R section for installa-	Qualify landing system: Parasail, Rocket ejection mechanism, Landing rocket, Landing gear. Develop parasail, landing rocket	24	Experiment 6

		propellant required for θ_g – 1g – 0g maneuver)			Determine physiological limitations of mission Determine mission communication restraints		[illegible]
4	ARTIFICIAL GRAVITY EXPERIMENT (CABLE COUPLE TO AGENA OR EXPENDED STAGE II OF GLV)	Angular velocity display and gyro, cable and cable reeling system (drum and motor) (approximately 200 lb. of available propellant required for 0g – 1g maneuver)	127	Mount angular velocity display and gyro Mount cable and associated equipment (for Stage II of GLV – approximately 30 lb. structure) (If on Agena – separate Agena section adjacent to target docking adapter is required. Approximately 75 lb. structure)	Qualify gyro and angular velocity display Conduct attitude control and stability analysis Determine physiological limitations of mission Determine mission communication restraints Develop cable system and installation Conduct structural tests, dynamic model tests and extensive analog simulation study	15	S/C 12 Experiment 3 and 7
5	SIMULATION OF LEM RENDEZVOUS	5A. Add LEM equipment to Gemini to perform computations and radar observations (computer and power supply IMU and radar) 5B. Modify Gemini equipment to simulate LEM rendezvous	723	Install LEM computer and power supply, IMU and radar	Expensive digital computer simulations To develop operational computer program Define crew operations and provide crew training	13	S/C 11, 12 Experiments 4, 7 and 9B
6	STRUCTURAL ASSEMBLY IN ORBIT	40 ft. diameter erectable antenna ECS umbilical and miscellaneous handles	115	Install antenna and fairing to Gemini (include structural beef-up to Gemini – structural weight approximately 100 lb.)	Wind tunnel tests of launch configuration Develop erectable antenna Conduct structural tests of antenna installation for launch loads	18	S/C 12 Experiments 4A and 9
7	PROPELLANT TRANSFER	Add to Gemini Propellant tanks Pressurant tank flow meters Gauging systems Transfer lines Add to Agena Propellant tanks Propellant Pressurant tank Pressurant	100 300	Modify Gemini for installation of propellant tanks and pressurant tank Modify Agena for installation of propellant tanks and pressurant tank	Qualify propellant transfer system Develop extravehicular activity capability, propellant centrifugal separator, and propellant retention device	24	Experiments 4, 5, and 9B
8	LONG DURATION MISSION	Agena mission section: Expendable living quarters ECS extension and living quarters circulation Additional electrical power Drinking water, food and containers Breathing O_2, bottles and mounts Agena structure Docking adapter	5850	Add hatch in hatch and launch fairing (approximately 24 lb.)	Wind tunnel tests of launch configuration Develop tunnel and living quarters Develop mission section Hatch in hatch and fairing structural tests	18	None
9A	LAND LANDING PARASAIL	Parasail Landing rocket Landing gear Rocket deployment mechanism	644	Remove radar, OAMS tanks, and pressurant, OAMS propellant, docking system, parachute system, roll bar, flotation aids in RCS, and adjust ballast. Modify R & R section for installation of parasail, landing rocket, and rocket deployment mechanism. Modify re-entry vehicle for installation of landing gear (approximately 630 lb. removed)	Qualify landing system Parasail Rocket ejection mechanism Landing rocket Landing gear Develop parasail, landing rocket, and landing gear	24	Experiment 6
9B	LAND LANDING – CLOVERLEAF CHUTE	Cloverleaf chute landing gear	577	Remove radar, OAMS tanks and pressurant, OAMS propellant docking system, parachute system, roll bar, flotation aids in RCS, and adjust ballast. Modify R & R section for installation of cloverleaf chute. Modify re-entry vehicle for installation of landing gear (approximately 678 lb. removed)	Develop 100 ft. diameter cloverleaf chute and landing gear.	30	Experiments 5, 6 and 7

TABLE 4.2-1
ADVANCED MISSIONS

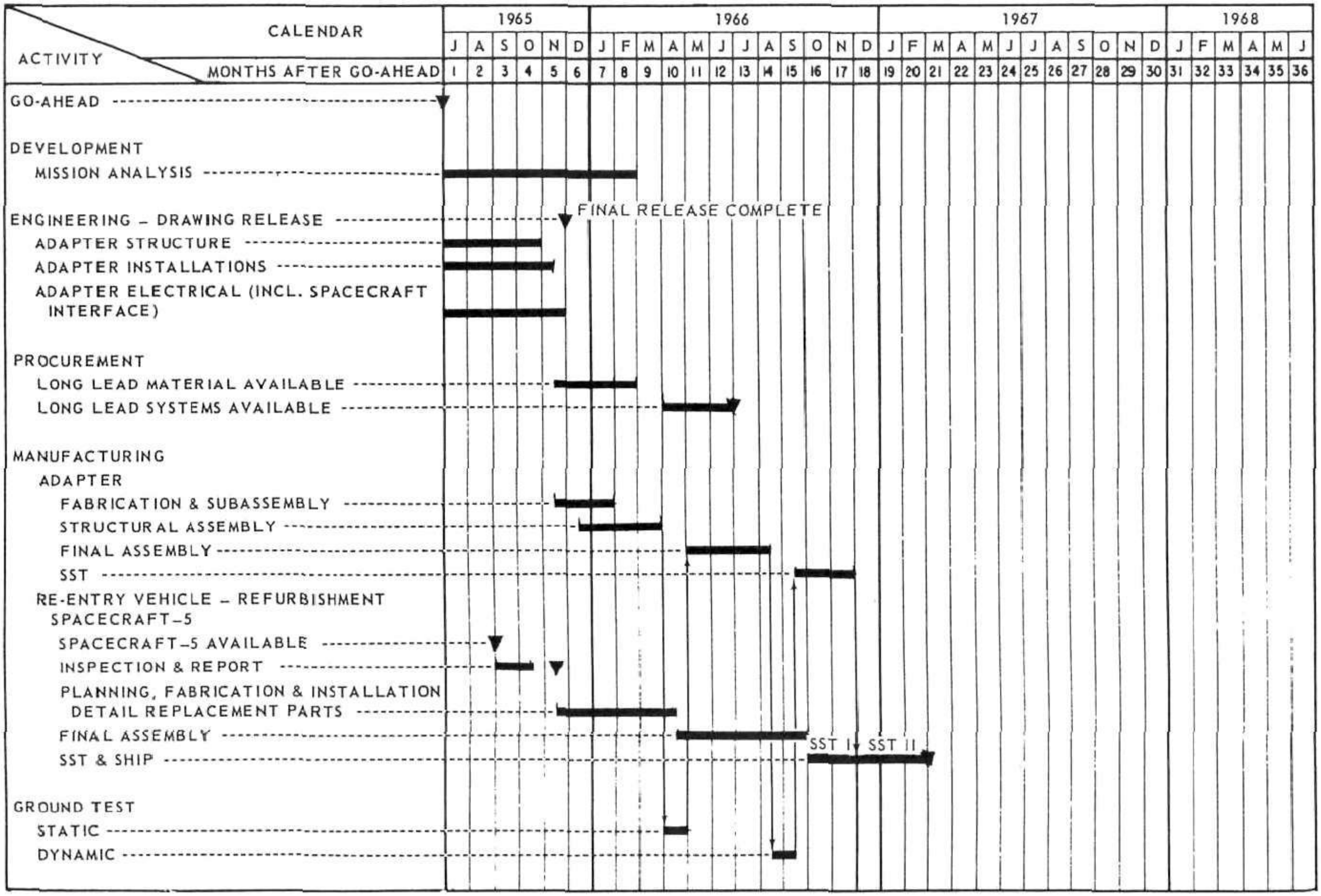
SUMMARY OF ESTIMATED COSTS
(MILLIONS OF DOLLARS)

EXPERIMENT AND COST ITEM	FIRST UNIT COST	EACH ADDITIONAL UNIT COST
1. RENDEZVOUS WITH AN UNMANNED SATELLITE		
a. OAMS MODIFICATIONS	3.70	.27
b. NEW ADAPTER STRUCTURE	3.60	.55
c. GUIDANCE SYSTEM-COMPUTER STUDIES, PROGRAMMING AND MODIFICATION	2.75	-
d. NEW RETRO ROCKET SYSTEM	.80	.09
e. MISCELLANEOUS SYSTEM CHANGES	2.70	.23
f. SUPPORT (AGE, SPARES, MISSION PLANNING, SPACECRAFT SYSTEMS TESTS, GROUND TEST, PUBLICATIONS, SPECIFICATIONS, REPORTS)	6.20	.61
	19.75	1.75
2. ONE MAN GEMINI-EARTH SURFACE MAPPING		
a. CAMERAS	2.35	.45
b. HORIZON SENSOR	1.20	.05
c. ANCILLARY STRUCTURAL MODIFICATIONS	2.15	.20
d. MISCELLANEOUS SYSTEM CHANGES	1.45	.17
e. SUPPORT	3.15	.48
	10.30	1.35
3. ONE MAN GEMINI WITH ASTRONOMICAL TELESCOPE		
3A (MOUNTED IN ADAPTER)		
a. 26" DIAMETER OPTICAL SYSTEM	4.30	.55
b. FINE ATTITUDE CONTROL SYSTEM	16.30	.70
c. HORIZON SENSOR	1.20	.05
d. FUEL CELL	1.55	.13
e. ADAPTER EXTENSION AND TUNNEL	4.50	.42
f. ANCILLARY STRUCTURAL MODIFICATIONS	.95	.67
g. MISCELLANEOUS SYSTEM CHANGES	7.50	.63
h. SUPPORT	17.00	1.75
	53.30	4.90
3B (MOUNTED IN RE-ENTRY VEHICLE)		
a. 16" DIAMETER OPTICAL SYSTEM	3.50	.44
b. FINE ATTITUDE CONTROL SYSTEM	16.30	.70
c. HORIZON SENSOR	1.20	.05
d. FUEL CELL	1.55	.13
e. ANCILLARY STRUCTURAL MODIFICATIONS	2.15	.20
f. MISCELLANEOUS SYSTEM CHANGES	6.30	.40
g. SUPPORT	14.00	1.08
	45.00	3.00
4. ARTIFICIAL GRAVITY EXPERIMENT		
4A (SOLID COUPLE TO STAGE II OF GLV)		
a. ADD NEW RATE GYRO	1.60	.07
b. MODIFY CREW DISPLAYS	.45	.05
c. MISCELLANEOUS SYSTEM CHANGES	.50	.05
d. SUPPORT	1.20	.08
	3.75	.25

4B (SOLID COUPLE TO AGENA)		
SAME AS 4A ABOVE	3.75	.25
4C (CABLE COUPLE TO AGENA OR STAGE II OF GLV)		
a. CABLE SYSTEM	8.60	.80
b. MODIFY ATTITUDE CONTROL SYSTEM	3.30	.15
c. MODIFY CREW DISPLAYS	.45	.05
d. MISCELLANEOUS SYSTEM CHANGES	2.85	.30
e. SUPPORT	6.80	.70
	22.00	2.00
5. SIMULATION OF LEM RENDEZVOUS		
5A (UTILIZE LEM EQUIPMENT)		
a. LEM COMPUTER AND IMU	7.50	1.75
b. LEM RADAR	1.50	.42
c. GEMINI TDA (MODIFIED)	1.60	.50
d. MISCELLANEOUS SYSTEM CHANGES	2.40	.23
e. SUPPORT	4.00	1.60
	17.00	4.50
5B (UTILIZE GEMINI EQUIPMENT)		
a. GEMINI EQUIPMENT MODIFICATION	3.90	1.60
b. SUPPORT	2.10	.90
	6.00	2.50
6. STRUCTURAL ASSEMBLY IN ORBIT		
a. ANTENNA	5.20	.55
b. SUPPORT STRUCTURE AND STRUCTURAL BEEF-UP	3.00	.39
c. ECS PROVISIONS	2.50	.07
d. OAMS TCA DISASSEMBLY PROVISIONS	.25	.02
e. MISCELLANEOUS SYSTEM CHANGES	.60	.12
f. SUPPORT	5.20	.60
	16.75	1.75
7. PROPELLANT TRANSFER		
a. PROPELLANT TRANSFER SYSTEM	9.30	.60
b. AGENA STRUCTURAL MODIFICATIONS	2.00	.20
c. MISCELLANEOUS SYSTEM CHANGES	2.80	.20
d. SUPPORT	6.40	.50
	20.50	1.50
8. LONG DURATION MISSION		
a. AGENA MISSION MODULE	9.80	1.60
b. LIVING QUARTERS AND ACCESS TUNNEL	6.80	1.30
c. RE-ENTRY VEHICLE ADDITIONAL EQUIPMENT	2.80	.15
d. MISCELLANEOUS SYSTEM CHANGES	5.40	.95
e. SUPPORT	11.20	2.00
	36.00	6.00
9. LAND LANDING		
9A (PARASAIL)		
a. PARASAIL SYSTEM CHANGES	2.10	.02
b. LANDING ROCKET	1.00	.16
c. MISCELLANEOUS SYSTEM CHANGES	.10	.05
d. SUPPORT	1.40	.12
	4.60	.35
9B (CLOVERLEAF CHUTE)		
a. CLOVERLEAF CHUTE	9.00	.10
b. CHUTE CONTROLS	.90	.03
c. MISCELLANEOUS SYSTEM CHANGES	.50	.02
d. SUPPORT	4.80	.05
	15.20	.20

ADVANCED MISSIONS – NO. 1

RENDEZVOUS WITH AN UNMANNED SATELLITE



NOTES: (1) ASSUMES ADAPTER AVAILABLE FOR OTHER TESTING AS REQUIRED.
(2) ASSUMES MINOR ADAPTER TOOL MODIFICATION ONLY.

FIGURE 4.3–1

ADVANCED MISSIONS – NO. 2

ONE-MAN GEMINI – EARTH MAPPING

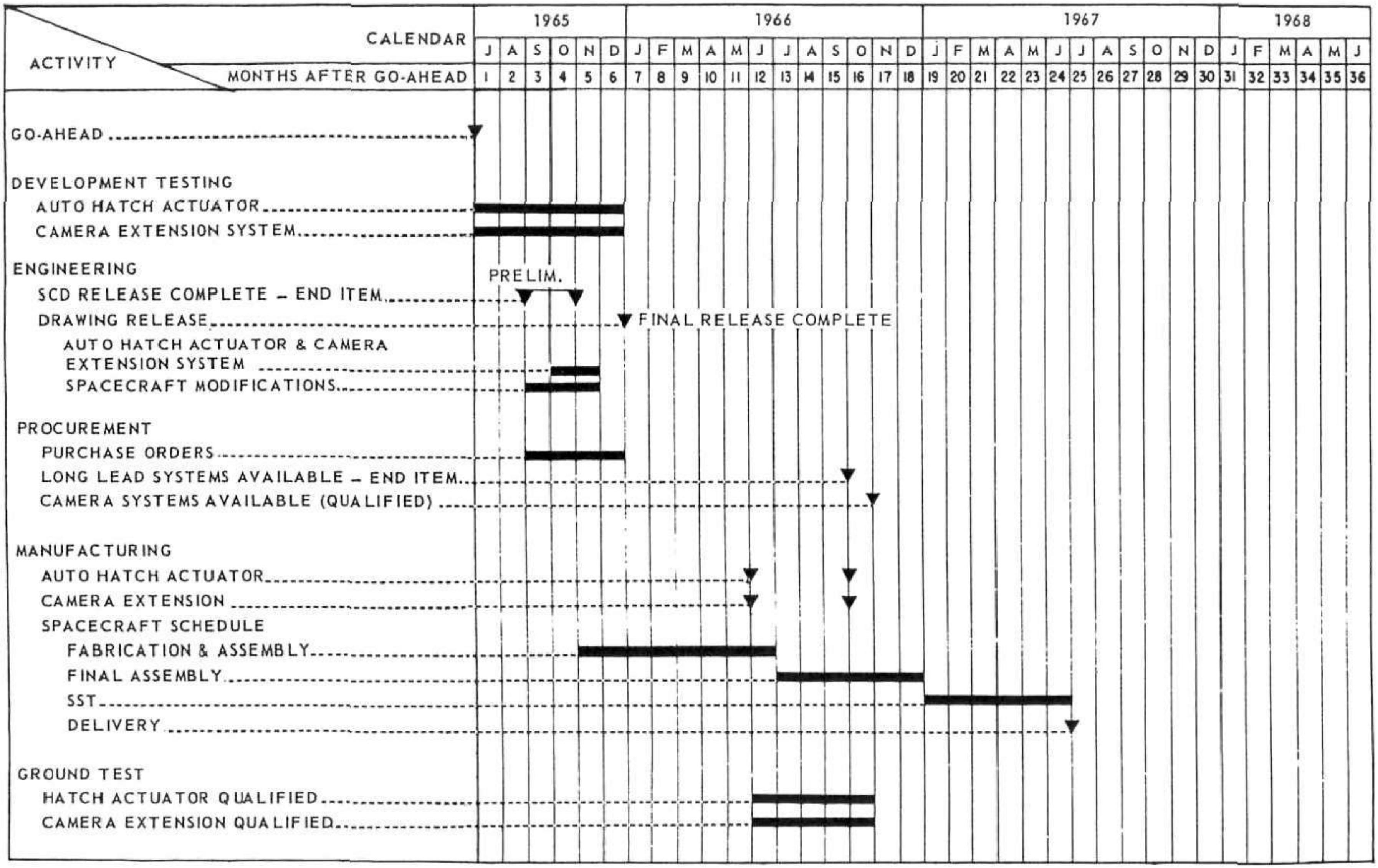
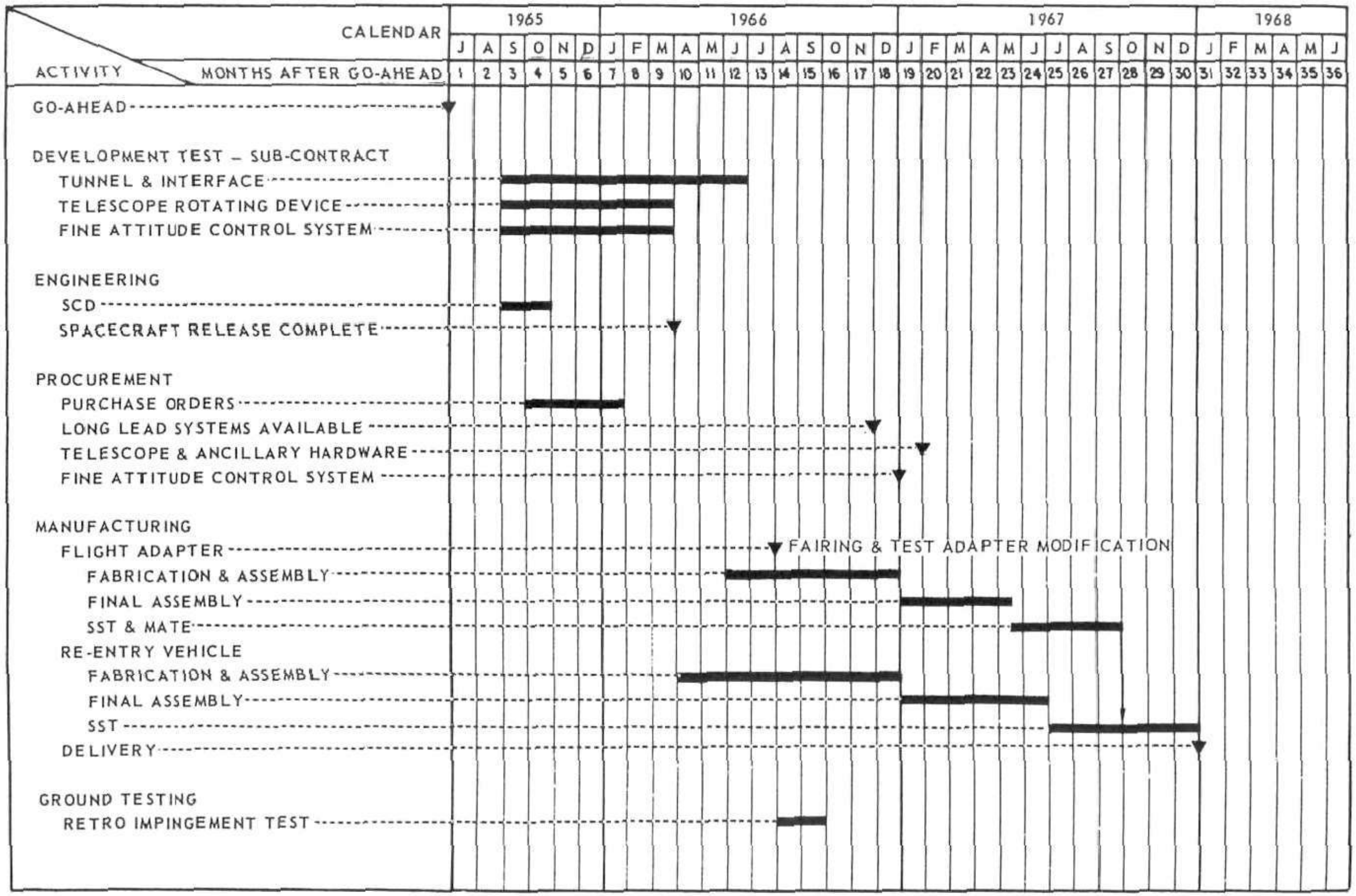


FIGURE 4.3-2

ADVANCED MISSIONS – NO. 3A

ONE-MAN GEMINI WITH ASTRONOMICAL TELESCOPE MOUNTED IN THE ADAPTER



NOTES: (1) HEAT SHIELD FROM HSQ

FIGURE 4.3–3

ADVANCED MISSIONS – NO. 3B

ONE-MAN GEMINI WITH ASTRONOMICAL TELESCOPE MOUNTED IN THE RE-ENTRY VEHICLE

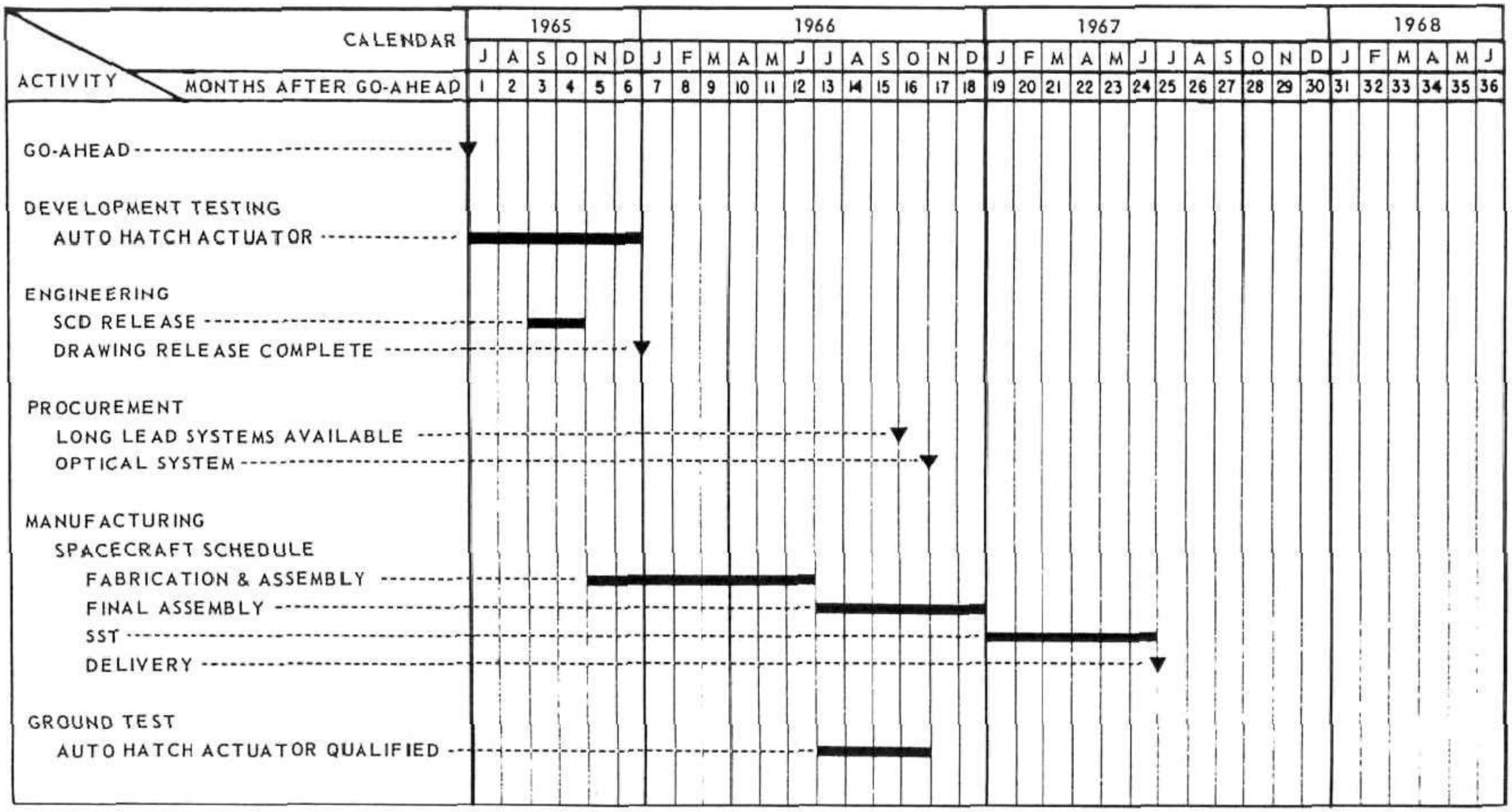


FIGURE 4.3-4

ADVANCED MISSIONS-NO. 4A
ARTIFICIAL GRAVITY-SOLID COUPLE TO STAGE II OF GLV

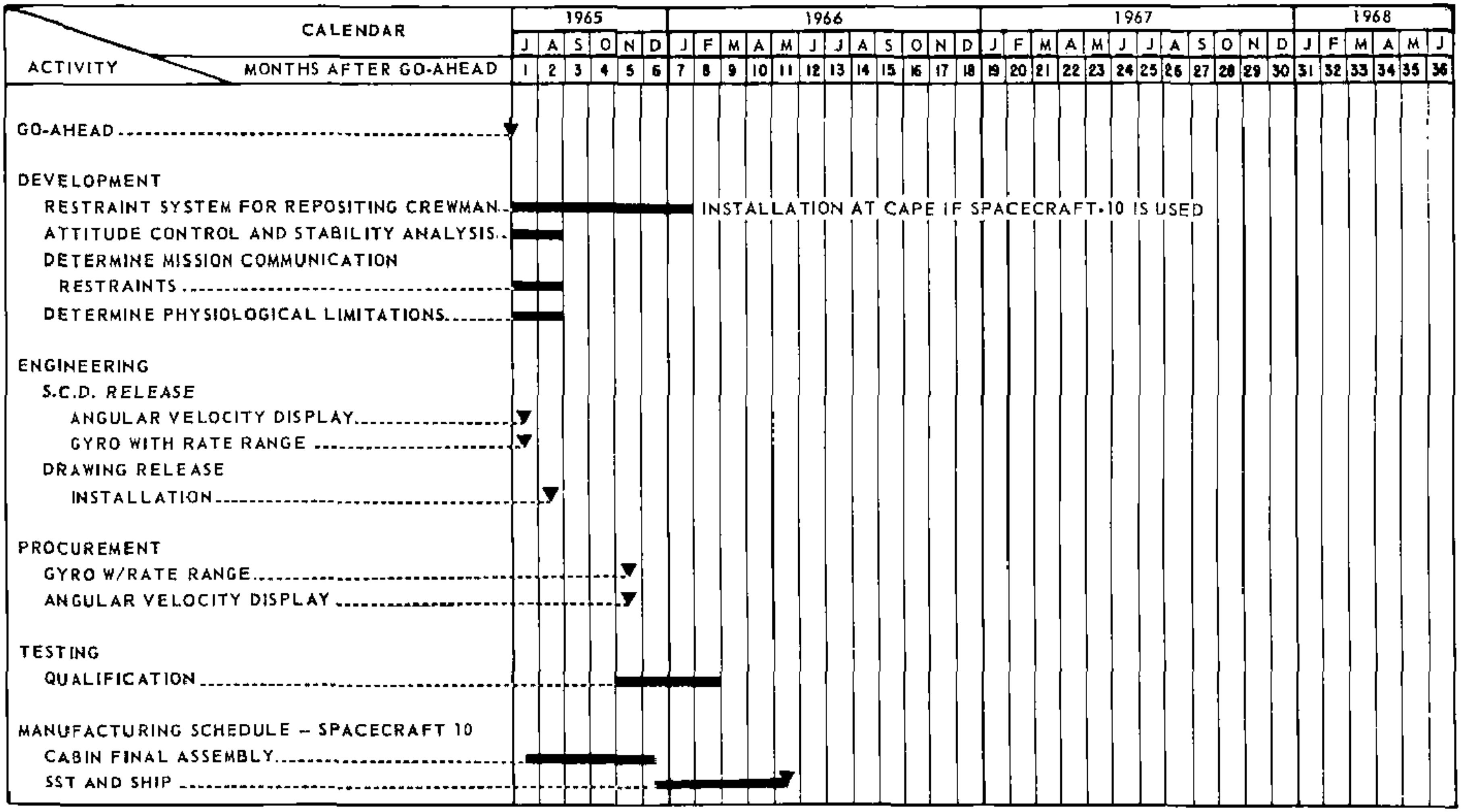
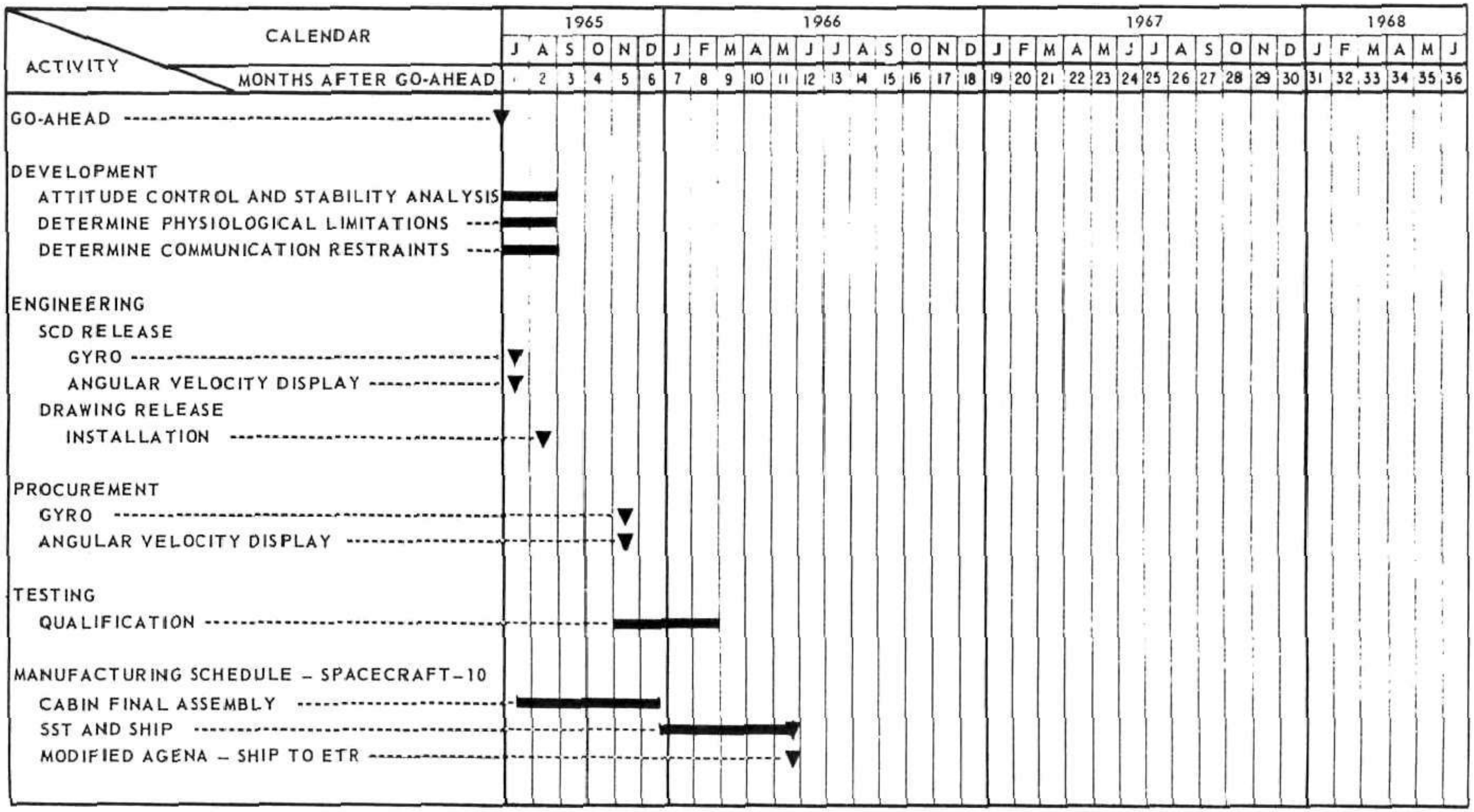


FIGURE 4.3–5

ADVANCED MISSIONS – NO. 4B

ARTIFICIAL GRAVITY – SOLID COUPLE TO AGENA

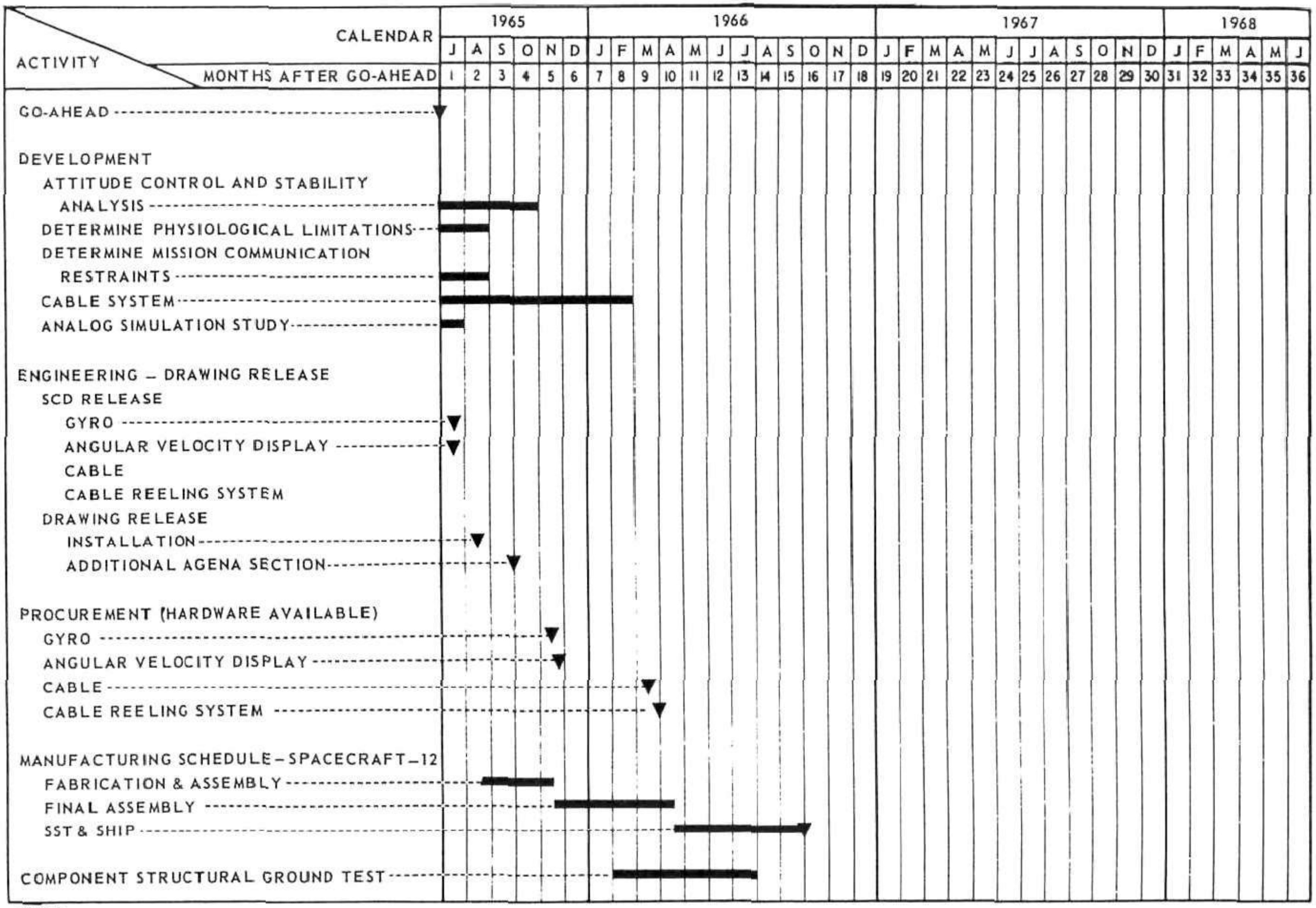


NOTES: (1) ASSUMES NO GROUND TESTING REQUIRED.
(2) ASSUMES AVAILABILITY OF MODIFIED AGENA SECTION.

FIGURE 4.3-6

ADVANCED MISSIONS – NO. 4C

ARTIFICIAL GRAVITY – CABLE COUPLE TO AGENA OR STAGE II OF GLV



NOTE:

(1) COMPONENT STRUCTURAL TEST TO BE PERFORMED BY RESPONSIBLE CONTRACTOR.

FIGURE 4.3-7

ADVANCED MISSIONS – NO. 5

SIMULATION OF LEM RENDEZVOUS

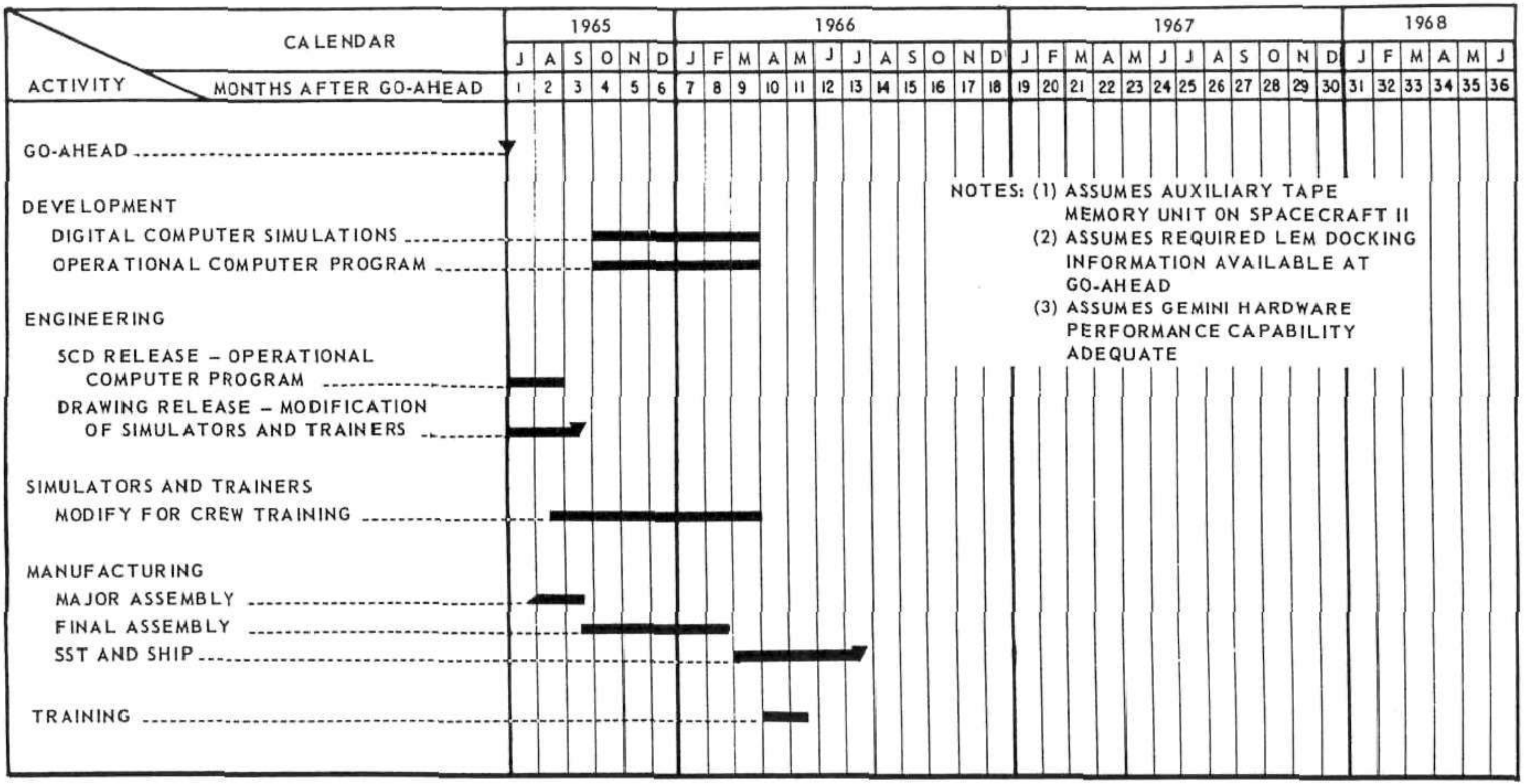


FIGURE 4.3-8

ADVANCED MISSIONS – NO. 6

STRUCTURAL ASSEMBLY IN ORBIT

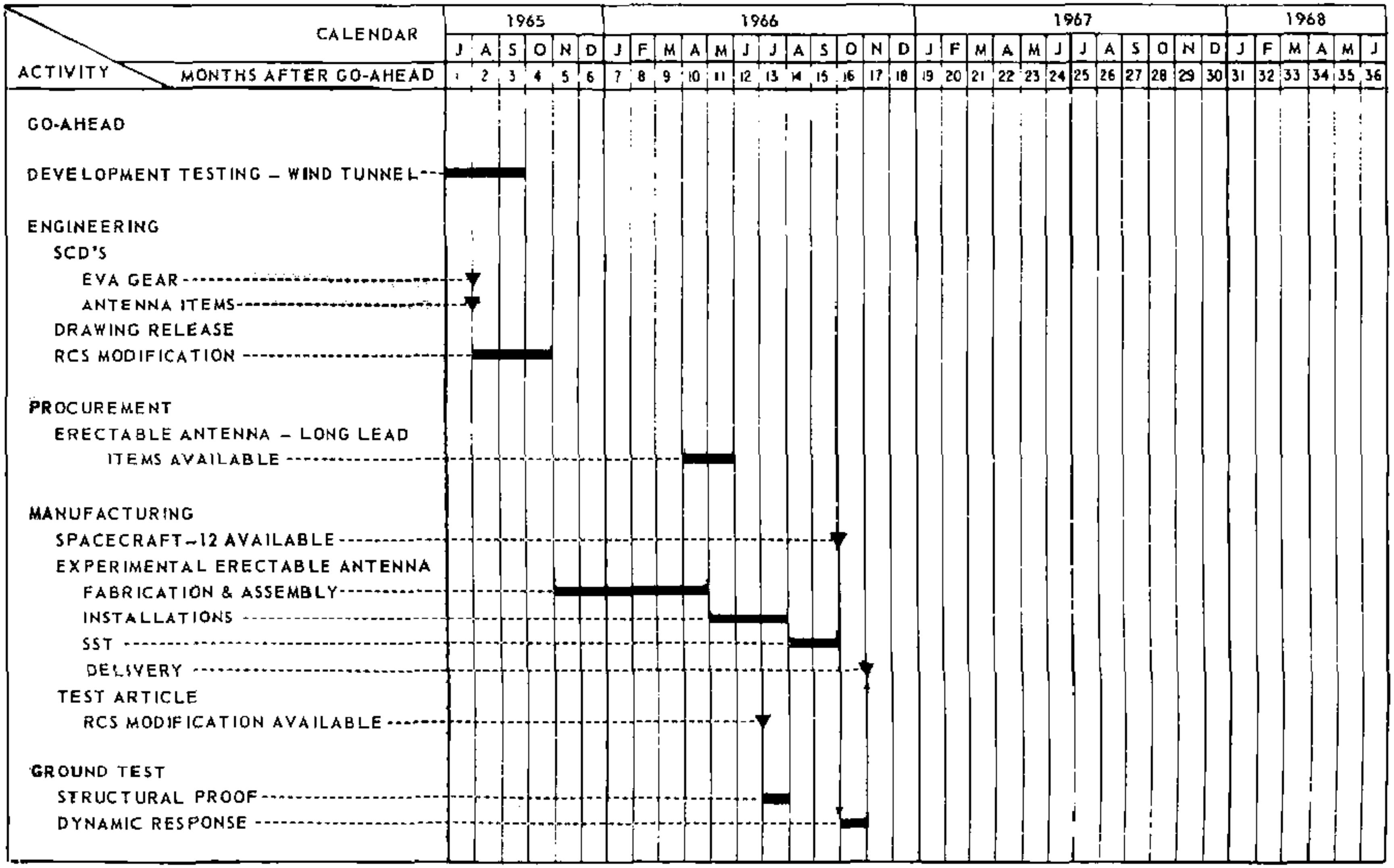


FIGURE 4.3-9

ADVANCED MISSIONS – NO. 7

PROPELLANT TRANSFER

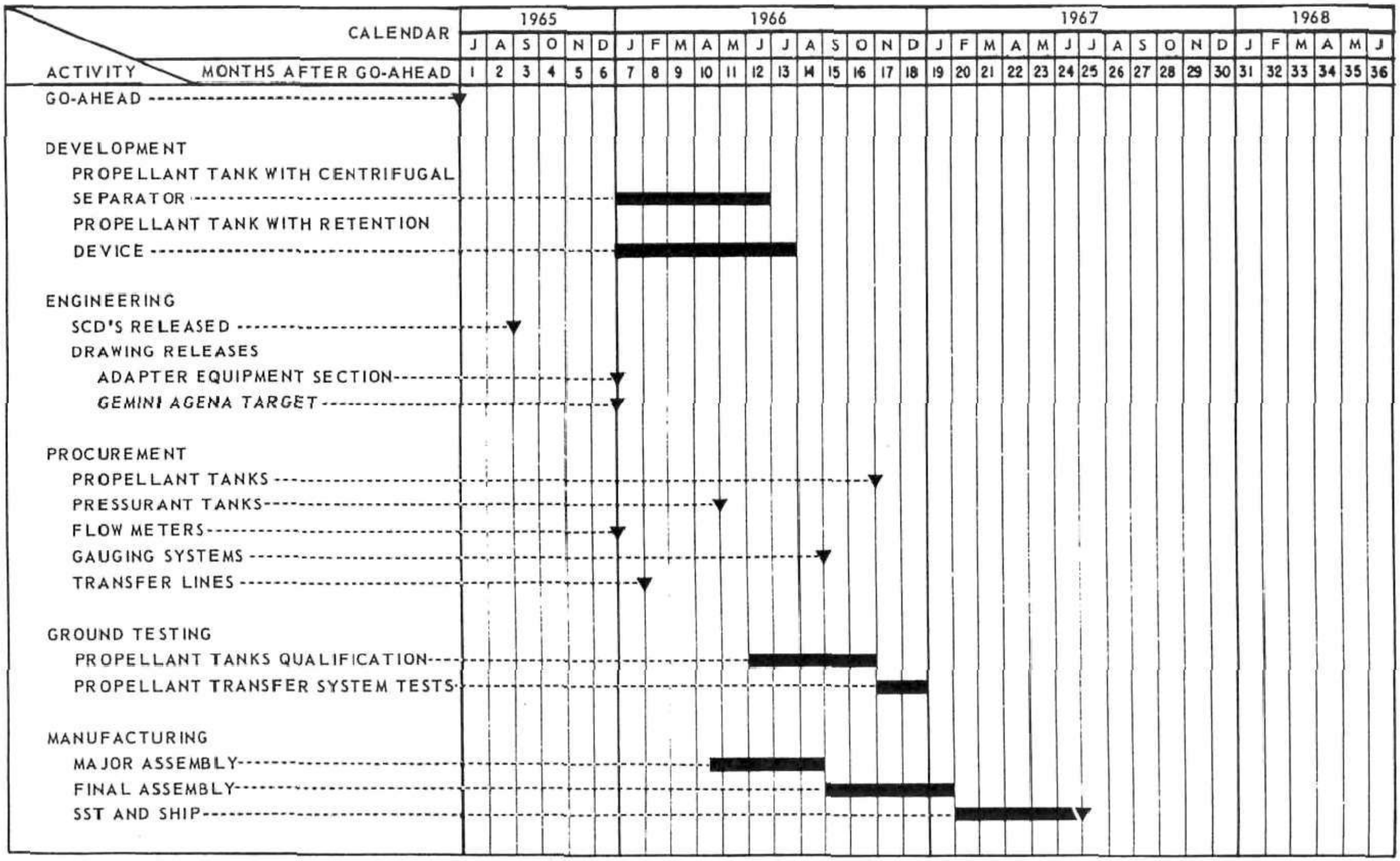
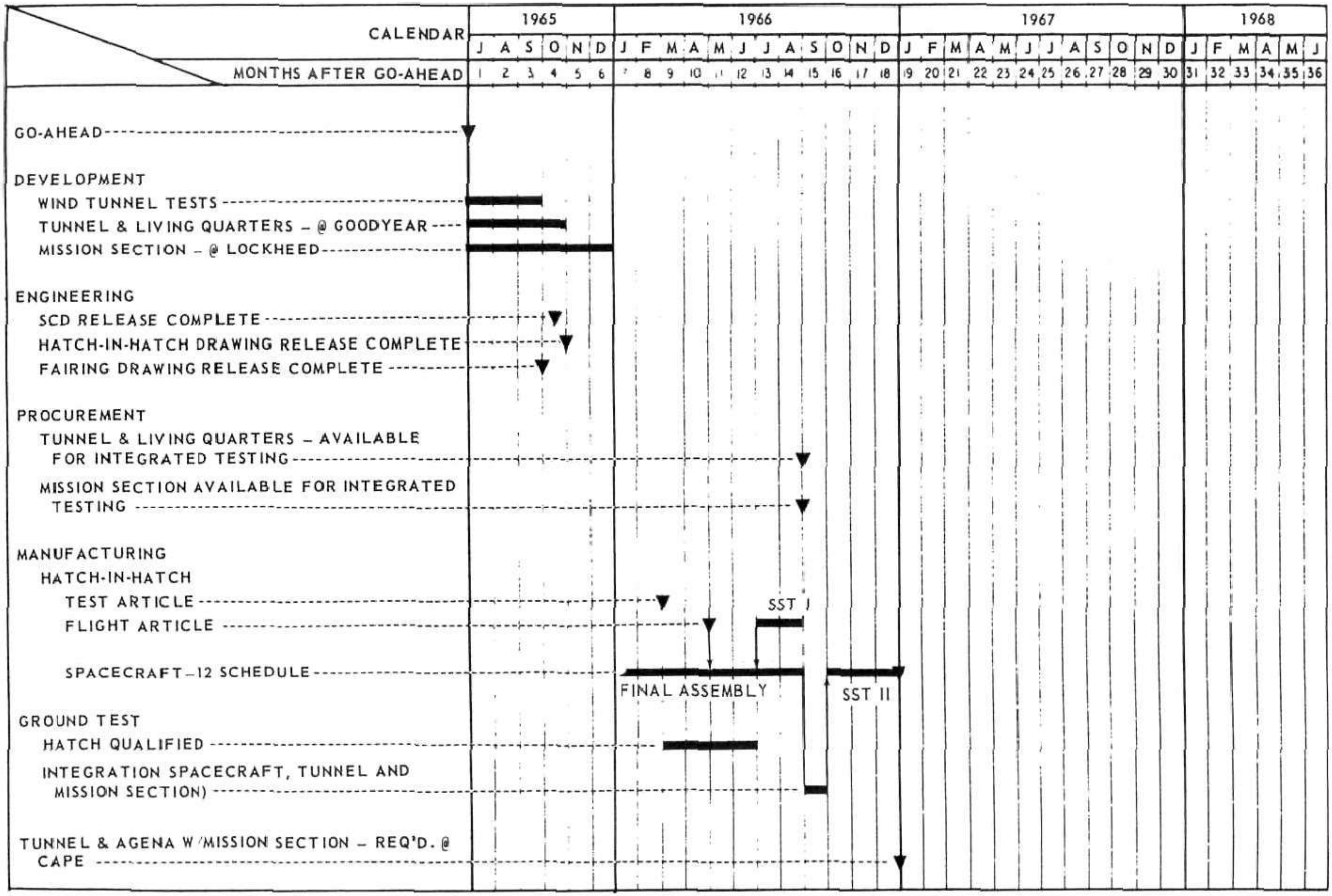


FIGURE 4.3-10

ADVANCED MISSIONS – NO. 8
LONG DURATION MISSION



NOTE: (1) SCHEDULE ASSUMES TUNNEL & AGENA W/MISSION SECTION AVAILABLE
(2) SPACECRAFT–12 DELIVERY SLIPS 3 MONTHS

FIGURE 4.3-11

ADVANCED MISSIONS – NO. 9A

LAND LANDING – PARASAIL

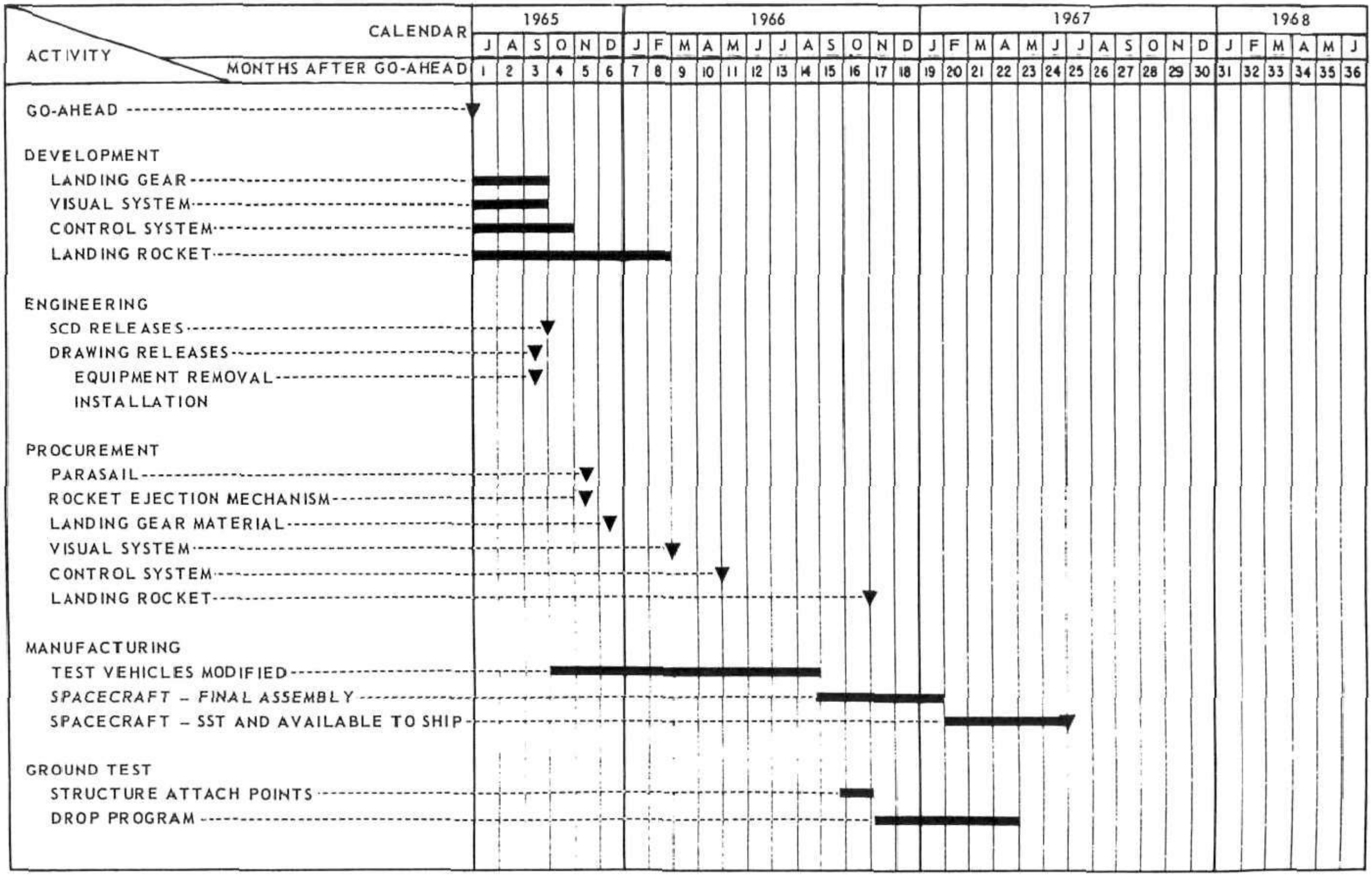


FIGURE 4.3-12
4-16

ADVANCED MISSIONS – NO. 9B

LAND LANDING – CLOVERLEAF CHUTE

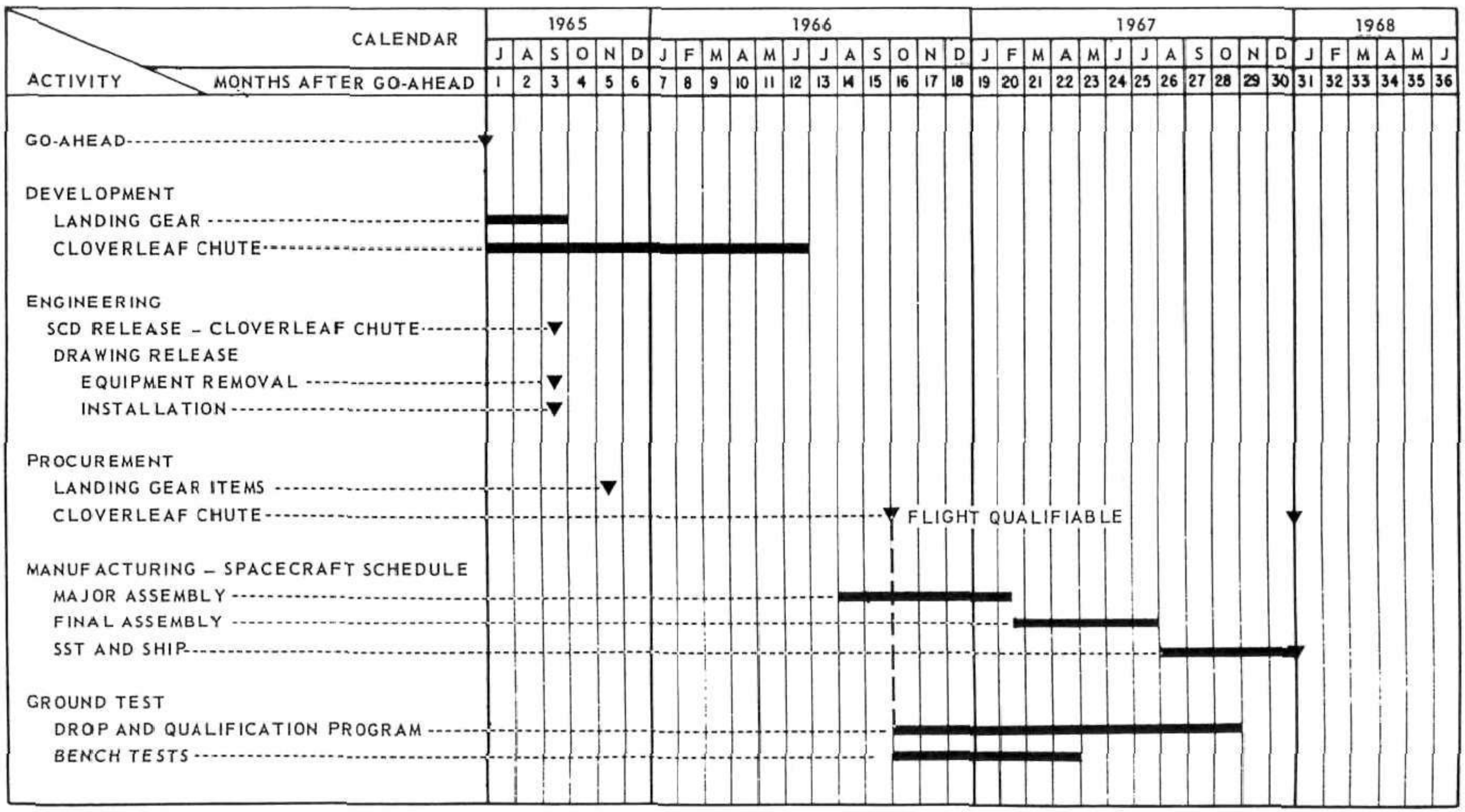


FIGURE 4.3-13

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