MAR 29 1966

DEVELOPMENT TESTING OF SPACE VEHICLES

Col Donald D. Carlson, USAF

and

Sqn Ldr George MacFarlane, RCAF

DCS/Plans and Technology

Hq., AEDC

February 1966

ARNOLD ENGINEERING DEVELOPMENT CENTER

AIR FORCE SYSTEMS COMMAND

ARNOLD AIR FORCE STATION, TENNESSEE

NOTICES

DEVELOPMENT TESTING OF SPACE VEHICLES

Col Donald D. Carlson

Sqn Ldr George MacFarlane

DCS/Plans and Technology

Hq., AEDC

FOREWORD

This technical report has been reviewed and is approved.

George MacFarlane
Sqn Ldr, RCAF
Technology Division
DCS/Plans and Technology

Donald D. Carlson
Colonel, USAF
DCS/Plans and Technology

ABSTRACT

This report reviews the salient characteristics of various classes of spacecraft, unmanned and manned, present and future. The test objectives and testing techniques used for development of aerospace vehicles are discussed. This portion includes the range of required ground test facilities and the role that flight test programs play. The report also discusses the hypothetical ground test facilities that will be needed to test adequately the conceptual systems of the future, and concludes with the role of earth-orbiting research laboratories in the overall spectrum of flight testing.

CONTENTS

ILLUSTRATIONS

TABLES

NOMENCLATURE

C_L	Lift coefficient
G	Acceleration of gravity
L/D	Lift-drag ratio
M	Mass
P	Surface atmospheric pressure
S	Wing area
W	Weight
ΔV	Velocity increment
ϵ	Orbital eccentricity
ρ	Density

SUBSCRIPTS

⊕	Earth
♂	Mars

SECTION I
INTRODUCTION

The United States, in conjunction with the nations of the North Atlantic Treaty Organization (NATO), has taken a logical step-by-step or building block approach in an attempt to solve the numerous technical problems involved in the aerospace program. This approach can essentially be categorized into three complementary parts (Ref. 1).

1. The first phase of the program deals with the exploration of the environments encountered in space.

2. The second phase determines the best ways to exploit the potential advantages of space for the benefit of all mankind. The development of both an unmanned and a manned capability in space will be essential to support a comprehensive program.

3. The third phase identifies and develops the complex technologies necessary to achieve mankind's goal of conquering space travel.

The space program was subsequently divided into many types of missions. The spectrum of missions includes: sounding rockets, planetary and interplanetary probes, exploratory unmanned satellites, and manned space flight. Vehicle payloads were designed to gather environmental data, to determine ranges and limitations of mechanisms which were to operate in space, and to test man's ability to function and survive in the hostile environments of space.

In Ref. 2, the authors defined the major space regions, discussed the more important space environments, and evaluated the effects of the hostile environments on spacecraft. This report reviews the important characteristics of the many types of space vehicles, discusses test objectives and testing techniques used for development of a satisfactory flight vehicle, and hypothesizes somewhat on the requirements for specialized facilities and flight testing methods. This report concludes with a discussion of the future trends evident in space environmental simulation and flight testing.

SECTION II
CLASSES OF SPACE VEHICLES

There are about as many ways to classify or categorize space vehicles as there are definitions of "space". For example, one might use the classifications of military and nonmilitary, another might use

mission profiles, and still another might use planetary or interplanetary as a classification. However, the general classifications of spacecraft to be used by the authors in this report are unmanned exploratory vehicles and manned space vehicles.

2.1 UNMANNED EXPLORATORY VEHICLES

The prime activity of the unmanned exploratory vehicles has been the gathering of basic information pertaining to the environments encountered during space flight. When one recognizes the many regions of space to be explored and analyzed, it soon becomes apparent that a wide variety of vehicles and mission profiles will be involved. Mission requirements normally include unmanned exploratory vehicles such as sounding rockets, planetary and interplanetary probes, earth-orbiter, lunar-orbiter and lander, and planetary and deep-space vehicles. Some representative mission profiles are shown in Fig. 1 (Ref. 1).

Early in the rocket and space program it became quite evident that, before suitable vehicles and propulsion systems could be developed, more information relating to the upper atmosphere must be obtained. In the mid-1940's, following World War II, the German-developed V-2 rocket became available as a vehicle that could carry an instrumented payload into the upper atmosphere. The United States, along with several other nations, soon developed a sounding rocket program.

2.1.1 Sounding Rockets

A generally accepted definition of a sounding rocket is a vehicle used to obtain information concerning the atmospheric conditions or environmental characteristics surrounding the earth. One might also consider the sounding rocket as a class of instrumented vehicles which do not achieve orbital or escape velocities. In 1946 the U.S. military initiated a high altitude research program using instrumented V-2 rockets. The instrument package carried equipments to measure ambient temperature and pressure, as well as equipment to measure the electron density of the E-layer (Ref. 3). Once the sounding rocket program was under way, many disadvantages in using the V-2 rocket for the purpose of atmospheric research soon became apparent. Heavier instrumented payloads and collection of information at higher altitudes became a requirement. Accordingly, modifications to existing rockets were made and new launching vehicles were designed. The AEROBEE and subsequent boosters were soon put into operation. Since these early exploratory flights, the launch and data gathering techniques have steadily improved. A typical family of sounding rockets, developed by the United States, is shown in Fig. 2 (Ref. 4).

It might be of interest to review the purpose of the sounding rockets which are used for data gathering and vehicle development. Sounding rockets, normally quite small, are relatively inexpensive and extremely versatile vehicles. Examples of some of the primary uses for sounding rockets are shown below:

1. Sounding rockets can be flown from selected places to cover special events.

2. Preparation time is usually short and the program can be very flexible.

3. Instrumentation packages can be designed for a specific mission without a compromise of compatibility between the many experiments involved on a satellite.

4. Verification or scatter data can be achieved by flying one experiment many times on a sounding rocket, whereas costs normally limit attempts to only once or twice on a satellite.

The sounding rocket will remain a basic tool for the scientist to acquire data on the properties of the space environments and for the engineer as a flight test bed for development of instrumentation for the more complex manned and unmanned satellites. Representative sounding rockets with their altitude and payload capabilities are shown in Fig. 3 (Ref. 4).

The sounding rocket program has provided data related to the region of space surrounding the earth, as well as information about the earth itself. The unmanned exploratory program during its evolution has now been expanded to include scientific research about other planets, the sun, and to some degree even distant stars.

2.1.2 Planetary and Interplanetary Probes

The requirement for vehicle design engineers to have environmental data beyond the earth's atmosphere soon dictated the need for a new generation of space vehicles. When propulsion techniques, guidance systems, instrumentation packages, and many other technical disciplines were sufficiently far advanced, an exploratory space vehicle having the capabilities to obtain scientific data beyond the earth's atmosphere evolved. This class of vehicles is often referred to as the planetary and interplanetary probe class. Probes relating to specific planets within our solar system are usually called planetary probes, whereas vehicles measuring properties at great distances from any given planet are often referred to as interplanetary probes. In special cases of investigating the space environments in the vicinity of the moon, one might term these vehicles lunar probes.

Some of the most important lunar and interplanetary space measurements have been accomplished by the NATO and Soviet probe programs. Probes, such as the American PIONEER and Soviet LUNIK, have been instrumented to obtain detailed measurements of the fields and particle environments in interplanetary space and also to determine the effect of solar activity on these environments. Probes, of course, are instrumented for a specific mission or task to be performed. A typical space probe with the instrumentation package identified is shown in Fig. 4 (Ref. 5, p. 185).

It might be of interest to examine the scientific information obtained by the flights of the PIONEER probe. A typical vehicle in this class is shown in Fig. 4. PIONEER I made the first radial measurements of the Van Allen radiation belt and the first recorded measurement of the strength of the magnetic field in interplanetary space. PIONEER III discovered the second radiation region and assisted in defining its boundaries. PIONEER IV made additional measurements of the Van Allen radiation belts, passed the moon at a distance of approximately 50,000 km, and is now in a solar orbit. PIONEER V was used to measure interplanetary radiations, determine radio communication capabilities, and verify the value of the astronomical unit (Ref. 6). With this impressive record of data gathering, it is quite apparent that probes will play an important part in our exploration of the space environments and space vehicle development programs.

2.1.3 Earth-Orbiting Satellites

The primary advantages of sounding rockets and probes were discussed in Sections 2.1.1 and 2.1.2. However, it is quite obvious that both classes of vehicles are limited in useful flight times, which are considered to be in the short-duration category. The earth satellite is the outgrowth of the requirement for long-duration flight times. When evaluating the earth-moon orbital period and the earth-sun orbital period, one can readily establish the need for flight times in the periods of months to even years. The earth satellite is the basic vehicle that the scientist and engineer rely upon for measurements of the environmental conditions surrounding the earth.

The satellite path or region of space under investigation can be controlled by varying both the orbital inclination and ellipticity. Earth satellites now orbit from altitudes just over 100 km to greater than 35,000 km at various inclinations. The satellite mission profile is determined by the objective of the specific vehicle. Types of earth satellites are many; however, most of the vehicles may be classified in accordance with the instrumentation payload. Table I, Earth Satellites, is representative of the many vehicles in this class.

4

Table I is included, not as a consolidated listing of earth satellites, only to indicate to the reader the variety of space vehicles, all having the prime objective of gathering space environmental data and/or determining environmental effects on spacecraft. The open literature is flooded with articles on each specific earth satellite, its instrumentation, and its mission profile. Most of the exploratory earth satellites have been developed with a specific purpose in mind and, consequently, the vehicles designed accordingly. Rather than discuss any of these limited, single-purpose vehicles in detail, it might be more beneficial to evaluate one of the most demanding near-earth scientific satellites, the Orbiting Astronomical Observatory (OAO).

The largest of the NASA observatories will be the OAO shown in Fig. 5 (Ref. 5, p. 123). The OAO has a 1635-kg payload that will be launched into an 800-km circular earth orbit. The basic instrumentation will be a series of telescopes; these will include telescopes to observe the ultraviolet (UV) spectrum, to study selected bright stars and nebulae, and to conduct detailed studies of interstellar matter. Sounding rockets, probes, and primitive satellites have provided sufficient data points to enable the design of some rather precise and sophisticated observatories. For example, the OAO will require a stable platform for pointing the telescopes to specific stars with accuracies of 0.1 arc-sec, i. e., star trackers must have the capability of locking on a star with tolerances equivalent to 0.05 cm at a distance of 1 km while recording data (Ref. 8). The observatories will have the necessary equipment to transmit data on a real-time basis or by stored data readout.

The observatory scientific satellites are extremely complex vehicles; however, they are representative of the technologies which have been developed during the space era. The orbiting astronomical observatory program will provide astronomers with the basic celestial data unattainable by ground-based telescopes.

2.1.4 Lunar Exploratory Vehicles

The lunar vehicle program might be divided into two broad categories: unmanned lunar exploration and manned space flight. Both categories are focused and geared to achieving manned flight to the moon in this decade. Two significant questions must be answered prior to attempting a manned lunar landing. Where can men land upon arriving in the vicinity of the moon, and what environments and surface characteristics will be encountered? Exploratory programs are presently under way to obtain information of this nature. There are three general types of unmanned vehicles contemplated for data gathering: lunar hard lander, lunar soft lander, and lunar orbiter.

2.1.4.1 Lunar Hard Lander

2. 1. 4. 1 Lunar Hard Lander

The United States' RANGER program and the Soviet Union's LUNIK program are representative of vehicles that have obtained high resolution photographs of the lunar surface. The best pictures of the lunar surface available prior to the LUNIK and RANGER programs were taken through earth-based telescopes with a resolution of approximately 1.1 km. In contrast, the last (P-3 camera) picture transmitted by RANGER IX prior to impact covered an area of only 540 m^2 at a resolution of approximately 50 cm. When one evaluates the information obtained from the RANGER and LUNIK programs, there are still many controversial items relating to the surface characteristics. It has been necessary to proceed with a more complex vehicle and instrument package in hopes of obtaining the required data. The second step in the lunar exploration will be accomplished by soft-landing vehicles on the moon.

2.1.4.2 Lunar Soft Lander

An example of this class of vehicle can best be described by evaluating the SURVEYOR program. The main objectives of this vehicle are to demonstrate the soft landing technique required for future manned programs, to measure the physical and chemical properties of the lunar surface, and to photograph various landing sites. Instrumentation packages on SURVEYOR will include: a television, surface sampler of soil mechanics, alpha scattering, micrometeorite seismometer, and touchdown dynamics. These instruments will provide stereo photographs, chemically analyze the soil, measure particle radiations, determine the environment and strength of moonquakes, and provide data on the compressive and shear strength of the lunar surface. Figure 6 is representative of this type of vehicle (Ref. 9).

2.1.4.3 Lunar Orbiter

The LUNAR ORBITER is a follow-on vehicle which complements the lunar soft lander. The primary objective of this class of vehicles is to photograph the specific area of interest to the APOLLO and the lunar soft lander at high resolution for the exploration and selection of suitable landing sites. It is planned that an eccentric lunar orbit will be used as the trajectory for obtaining the high resolution pictures. The perigee and apogee will be approximately 46 and 250 km, respectively. The LUNAR ORBITER can also be instrumented to provide surface properties data, selenological data, and cislunar environmental measurements. It is interesting to note in Fig. 7 the proposed mission profile for accomplishment of the LUNAR ORBITER program (Ref. 10).

2.1.5 Planetary and Deep-Space Vehicles

The last class of unmanned vehicles to be discussed might properly be called planetary and deep-space exploration vehicles. The first step in the space program is quite obviously to explore our own planet and its moon. However, to advance the knowledge of space, one must explore the other planets and deep space. Programs are being developed to investigate other planet environments, surface characteristics, magnetic and gravitational qualities, and biological properties.

Vehicles, such as the United States MARINER and Russian ZOND, have been developed to explore deep space. There are so many unknowns in the regions of interplanetary space that the instrumentation scheduled for the vehicles is of prime concern. Instrumentation packages will normally include television, magnetometers, radiation detectors, telescopes, dust and micrometeorite detectors, photometers, and communication equipment. The amount of instrumentation or payload carried on deep-space vehicles is a direct function of the booster and mid-course propulsion requirements. There is wide disagreement among the experts in planetary astronomy as to the properties of the other planets within our solar system. Answers to some of these disagreements will be obtained by planetary and deep-space vehicles.

Progress to date on planetary environments has been obtained from earth-based telescopes, balloon observations, radar techniques, and the MARINER II flyby of Venus. Results are summarized below (Ref. 5, p. 131):

1. Venus surface temperature - 700°K

2. Venus surface pressure - 10 atm

3. Low magnetic field surrounding Venus

4. Little or no rotation of Venus

5. Water vapor detected on Mars

6. Low surface pressure on Mars

The composition of the Venus and Mars atmospheres must be known to a fair degree of accuracy if our space exploration vehicles are to progress. It will be necessary to have information such as the near-surface pressure of Mars when a Mars landing vehicle is designed. How far and how fast the unmanned exploratory vehicle program will progress is dependent upon the launch vehicles available. Booster sizes, propulsion techniques, logistics, economics, and political aspects will be pacing factors in determining the milestones in planetary exploratory programs.

The specific configuration of a deep-space vehicle will be determined by the instrumentation package involved, mission profile, guidance and control system, and perhaps useful lifetime in orbit around the planet prior to returning to earth. The VOYAGER vehicle, shown in Fig. 8, might be considered representative of a deep-space exploratory vehicle (Ref. 11, p. 16).

2.2 MANNED SPACE VEHICLES

Man's desire to conquer space has been imaged for centuries; however, the first positive steps were taken by President Dwight D. Eisenhower in 1955 when it was announced that the United States planned to orbit a scientific satellite during the International Geophysical Year - July 1, 1957 to December 31, 1958. This was followed by the USSR's successful flight of the first earth-orbiting satellite, SPUTNIK I, launched October 4, 1957.

The real impetus to the space program occurred on that historic date of October 4, 1957. Since the inception of the NATO and Soviet space programs, the primary objective has been oriented toward man in space. Unmanned exploratory vehicles have been in operation for the past 8 yr gathering environmental data and information on the effects of these environments on spacecraft. This effort has been the primary building block in achieving manned space flight. Research programs to study the biomedical aspects of space flight have also been in progress for several years. The Russians launched SPUTNIK V in August of 1960 carrying two dogs and some additional biological specimens (Ref. 12), and the United States accomplished its early exploratory biological research using primates. The important fact is that information relating to space environment and its effects on biological specimens was obtained by these exploratory research vehicles.

The Russian VOSTOK and the American MERCURY programs were subsequent building blocks to the overall manned space effort. It might be of interest to review some of the manned vehicles used in the space program. For the sake of familiarity, the authors will review the classes of vehicles by referencing the American space program. However, when thoroughly analyzed, the Russian and American manned space programs are seen to be quite comparable in scope.

2.2.1 The Mercury Program

The MERCURY program was established to determine man's capability to perform technological projects during orbital flight when

exposed to the unnatural condition of weightlessness. The MERCURY program completed six missions; two suborbital flights were accomplished in 1961, three orbital missions took place in 1962, and the program was successfully completed in 1963 by the long-duration flight of astronaut Gordon Cooper.

Project MERCURY has provided information on how to design, build, and test a spacecraft to take man more than 160 km from the surface of the earth. The earth-orbital flights have also provided sufficient physiological and psychological data on human behavior during weightless conditions to demonstrate man's ability to cope with the exploration of space. The MERCURY vehicle, shown in Fig. 9, is representative of an early manned earth-orbital vehicle. The satisfactory completion of Project MERCURY subsequently led into the follow-on program of GEMINI (Ref. 13).

2.2.2 The Gemini Program

The primary objectives of the GEMINI program are to increase operational proficiency in space and develop the technology required for manned space flight. The flight program will include long-duration manned flights, extra-vehicular activity, development and testing of rendezvous and docking techniques, determination of the reliability of electrical and mechanical spacecraft equipment, and establishment of limits on spacecraft maneuverability. These techniques must be developed for man to accomplish successfully the lunar mission as presently proposed by the U.S. and the USSR.

The GEMINI spacecraft is about 30 percent larger than the MERCURY capsule and weighs approximately 3200 kg. Instrumentation includes: an inertial guidance system; rendezvous radar; on-board digital computer; bioinstrumentation; food, water and waste systems; and personal hygiene equipment. The GEMINI will be limited to earth-orbital flights; however, the systems have the capability to sustain two men in orbit up to 14 days and thereby will permit the gathering of behavioral data over periods of time anticipated for flights to the moon and return.

The initial flight of the GEMINI spacecraft was made in March 1965 when astronauts Virgil Grissom and John Young orbited the earth and landed after three orbits. The next flight was launched on June 3, 1965, and astronauts Edward White and James McDivitt circled the earth for 66 orbits, logging a flight time of slightly less than 98 hr. Later GEMINI launches included: August 21, 1965, when astronauts Cooper and Conrad circled the earth for 190 hr and 56 min; December 4, 1965, when astronauts Borman and Lovell circled the earth for 330 hr and 35 min; December 15, 1965, when astronauts Schirra and Stafford circled the earth for 25 hr and 51 min.

The GEMINI vehicle, as shown in Fig. 10, is representative of a two-man earth-orbital vehicle (Ref. 14). The U.S. GEMINI and the USSR VOSTOK and VOSKHOD programs are making significant contributions to the technical disciplines involved in man's survival in space. It is interesting to note the manned space flight experience gained in these programs. The accumulated flight hours logged in manned spacecraft are shown in Fig. 11 (Ref. 12). The successful completion of the GEMINI objectives will evolve into a lunar landing vehicle.

2.2.3 The Apollo Program

The APOLLO program is the integration of the entire spacecraft configuration for the final lunar landing mission. This spacecraft will be composed of three separate modules, each designed to fulfill specific mission requirements (see Fig. 12). The command module (CM) (Fig. 13) will contain the three-man crew until the lunar orbit is achieved. The command module serves as the control center for the spacecraft operation, and it will also provide the reentry vehicle to the earth upon return from the moon. The service module (SM) (Fig. 14) will house the spacecraft's life-support systems and a major propulsion system that will be used for mission abort, mid-course corrections, and injection into and out of lunar orbit. The lunar excursion module (LEM) (Fig. 15) will ferry the two men to and from the lunar surface (Ref. 14).

There are several mission profiles which can be used to accomplish the lunar landing. The four primary trajectories which have been investigated are shown below:

1. Direct flight from earth to the moon.

2. Earth-orbital rendezvous (EOR) technique.

3. Lunar-orbital rendezvous (LOR) technique.

4. Double rendezvous technique with an earth and lunar orbit.

After an exhaustive analysis of these various mission profiles, the lunar orbiting rendezvous technique was selected, based on the following facts (Ref. 15).

1. In the direct mode, the spacecraft weight dictated developing a primary booster stage more powerful than the SATURN V booster. This would be very expensive and would also extend the time period for a manned lunar landing. The landing vehicle of the direct mode would necessarily be quite tall (from 20 to 30 m) to accommodate the propellants required

for the return to earth. In addition, the pilot's depth perception, since large and small lunar craters are strikingly similar in appearance, would be poor. Landing the vehicle would therefore be quite difficult.

2. The spacecraft accomplishing the EOR mode would weigh as much as the direct-mode vehicle. However, it could be launched as two SATURN V-sized payloads. Obviously, this mode requires two launch pads and two successful flights with very small launch windows. The EOR method also requires that orbital assembly and refueling techniques be developed in addition to the orbital rendezvous procedure. Furthermore, to minimize the weight of the EOR and direct-mode vehicles, high energy (cryogenic) propellants were necessary with all of their attendant storage problems.

3. In summary, the LOR method, which is the technique selected by the United States, has the following advantages:

   a. The required weight of the booster to achieve earth escape velocity is more than 50 percent less than that needed for either the direct or EOR mode. This weight saving is possible because the LOR method uses the philosophy of rocket staging to the ultimate (i. e., discarding useless or unnecessary mass as soon as possible).

   b. Mission probability of success is greater.

   c. The insertion of the LEM into a very low perigee elliptical lunar orbit greatly increases the inherent safety of the flight crew for the following reasons:

      1. The astronauts may reconnoiter their proposed landing area on their first pass before committing themselves to a descent on the second pass.

      2. The descent engine is tested as the LEM transitions from the circular to the elliptical lunar orbit.

      3. Should the decision be made to abort the flight at any time prior to the initiation of the descent maneuver, the astronauts retain the rendezvous capability.

      4. The chances of a successful return from the lunar surface are greatly increased. If the LEM engine malfunctions or its fuel is exhausted after lunar orbit is achieved but before rendezvous can be accomplished, the crewman remaining in the command module may assume the active role and complete the rendezvous.

The significant milestones of the lunar journey are shown in Fig. 16. After launch and injection into an earth orbit, the systems on the three-module spacecraft and the SATURN IV-B third-stage booster are checked out for launch damage by ground control. The orbital parameters and translunar injection conditions are also calculated. After several orbits, the SATURN IV-B stage is fired for a second time. After the translunar trajectory is confirmed, the command-service module (CSM) turns 180 deg and is mated to the LEM. The SATURN IV-B stage is then jettisoned. Several trajectory corrections may be made in route. Upon arrival at the moon, the three-module spacecraft is injected into a 147-km circular lunar orbit. Next, two crew members enter the LEM and it is then detached. The LEM is next inserted into an elliptical orbit with a perigee of about 16 km. After one observational pass, the landing is made. Sufficient fuel is available to permit limited-duration hovering over the landing point, and a small translation to a more suitable landing site should the first one not be satisfactory. After their stay on the moon, the astronauts in the LEM are boosted into orbit in the LEM ascent vehicle, rendezvous with the orbiting command module, transfer themselves and approximately 180 kg of lunar samples into the command module, and then jettison the LEM. The service module engine is again fired under earth control for the journey back to earth. Approximately 5 min prior to reentry, the service module is jettisoned, and the command module reenters for the final landing.

The first APOLLO flights will be earth-orbital missions to permit observations of the astronauts under weightless conditions for extended periods of time. Subsequent flights will check out the numerous vehicle systems and will involve practice of rendezvous and docking techniques in an earth orbit. This phase will be followed by circumlunar and lunar-orbital flights. Finally, a lunar landing will be attempted.

Upon completion of a successful lunar landing, the space program will have developed scientific and engineering capabilities which will enable mankind to pursue greater goals in manned space endeavors. Follow-on vehicles will probably include lunar roving vehicles, manned orbital laboratories, space stations, and spacecraft to explore and land on near planets, such as Mars. Some of these potential missions are described in the next section.

2.3 FUTURE SPACE VEHICLES

The current U.S. manned aerospace effort, exemplified by the GEMINI program, is aimed at the development and acquisition of operational techniques required for spaceflight. In the immediate future,

steps toward landing men on our nearest neighbor will be taken when the first tests in an earth orbit of an operating, manned APOLLO system commence in 1966.

The initial APOLLO moon landings will represent only a preliminary phase toward exploring the moon. The first astronauts, for example, will remain for only a short period of time on the lunar surface. It is evident that, to carry out extensive mapping and selenological surveys, the astronauts must be equipped to extend stay times on the lunar surface from days to several months. To support this program, it will be necessary to establish permanent, self-sustaining lunar bases at strategic, accessible locations on the moon. Programs in the even more distant future will probably involve the exploitation of the moon's natural resources to reduce the resupply commitment from the earth and to utilize the lunar base as a refueling station for expeditions into deep space.

Without doubt, the lunar exploratory program will entice man to the next big adventure: exploration of the planets. Such an expedition represents a gigantic step forward in complexity. Since man's knowledge of the nearer planets is even more meager than his knowledge of the moon, manned interplanetary flight must be preceded by an extensive supporting program of unmanned flights to gather the detailed knowledge of the planetary atmospheric and surface environments that engineers must have to undertake the design of a manned vehicle.

Extended stays on the lunar surface and manned trips to other planets both involve a crucial and perhaps the dominating factor: time. For example, the round-trip time to Mars is more than one hundred times the length of an earth-moon trip. The crew size for these future missions will be small, perhaps six and certainly fewer than a dozen members. The effects of confinement, lack of exercise and entertainment, and the ever-present environmental hazards will combine to produce intracrew conflicts and degradation of the performance of each individual. Therefore, the first step that must be taken to prepare man for the physiological and psychological stresses of operations of the type described above is the establishment of earth-orbiting laboratories in which to investigate ways of overcoming or alleviating such stresses.

The long operating times also impose very severe reliability criteria on all systems. In addition, a closed-cycle ecological system, computer requirements, onboard communications systems, and other electrical systems will all add to the drain on the spacecraft's power system. This power drain may be made feasible with a nuclear energy source, but the limitations of this solution, namely the weight penalty

imposed by the biological shielding requirements and the restrictions that the reactor radiation field places on rendezvous procedures, docking, and extra-vehicular activity, must be recognized. Furthermore, because of propulsive energy limitations, manned interplanetary flight may not be possible unless orbital assembly and refueling techniques are developed. Thus a space laboratory in orbit would serve as a useful tool for developing such procedures.

2.3.1 Earth-Orbiting Space Stations

Considerable activity in the aerospace industry is now directed toward the formulation of design concepts for earth-orbiting space stations of various sizes and degrees of sophistication. The initial designs will obviously capitalize on existing hardware and state-of-the-art equipment and would be sized to lie within the volumetric and payload mass constraints of existing launch vehicles. As larger, more powerful boosters become available, the laboratories can be expected to become larger and to be more elaborately equipped. Only a brief paragraph is included in this report on advanced concepts because the open literature is available to give detailed information to the interested reader. The following concepts are believed to typify future vehicle requirements.

2.3.1.1 Manned Orbiting Laboratory (MOL)

The primary role of MOL is to investigate and evaluate man's capabilities to function usefully in space and to discover his peculiar limitations. One concept of this vehicle has a cylindrical laboratory, 3 m in diameter by from 6 to 9 m in length (Refs. 16, 17, and 18). A crew of two would ride into, and return from, orbit in a GEMINI capsule. In the early version at least, MOL would be self-sufficient for a minimum duration of from 14 to 30 days.

2.3.1.2 Extended Mission Apollo (APOLLO X)

This proposed program would utilize available APOLLO hardware to create a small laboratory capable of operating from 14 to 45 days in earth orbit (Refs. 16, 17, and 19, p. 265). APOLLO X would consist of the basic APOLLO command and service modules, with the lunar excursion module housing converted into an unpressurized laboratory of about 30 m^3. One crew position would be deleted from the command module to provide additional living space.

2.3.1.3 APOLLO Orbital Research Laboratory (AORL)

Basically similar in concept to APOLLO X, but with a larger (and pressurized) experimental compartment (160 m^3) and a 1-yr operating life, AORL would house six men (Refs. 16, 17, and 19, p. 265). Development of this system obviously hinges upon the establishment of the rendezvous and docking technique. Consideration is being given to providing the station with artificial, reduced gravity; the AORL and the spent S-IVB booster stage would be linked by cables and spun about the common center of mass of the combination.

2.3.1.4 Large Orbital Research Laboratory (LORL)

Stations of the LORL category would be representative of operational, as opposed to strictly research and development, systems. Several different designs have been proposed (Refs. 16, 17, and 19, p. 265). A nonrotating station, cylindrical in shape, would be 10 m in diameter, 43 m long, weigh 5 x 10^5 kg, and support a crew of from 24 to 36 men. A centrifuge would supply gravity conditioning as required. Several types of designs providing an artificial gravity environment are also under consideration; these designs run the gamut from Y-shaped configurations (Fig. 17), with arms that unfold and lock into place once orbit has been achieved (Fig. 18), to expandable or hexagonal structures (Fig. 19). Both of these general types would be about 50 m in diameter when assembled. Since the LORL must possess a long lifetime to be economical (5 yr in orbit is the goal), positioning propulsion must be incorporated to overcome the atmospheric drag forces.

2.3.1.5 Orbital Launch Facility (OLF)

The OLF, envisioned as the ultimate in space stations, would function expressly to support earth-orbit assembly and launches of manned interplanetary vehicles (Ref. 20). In addition, OLF would serve as an orbiting maintenance and repair shop, as a checkout facility that would conduct thorough testing of all lunar ferry vehicles, as an orbiting refueling base, and as a transshipment point for outgoing and returning space crews.

2.3.1.6 Summary of Technological Problems

The more advanced of these concepts (i.e., Sections 2.3.1.3 through 2.3.1.5) require that several complex technological problems be solved before such concepts can be developed into hardware. Such problems might include:

a. An economic and reliable method of personnel rotation and logistical resupply of the orbiting space station with fuel, makeup oxygen, spare parts, and provisions must be developed. The solution to this problem perhaps lies in the development of either an aerospace plane or a reusable booster. The rendezvous, docking, and maneuverable reentry capability must be established.

b. Extended operations in orbit impose strict reliability criteria on all systems.

c. Power requirements for the environmental control system, communications, and auxiliary equipments are very large. Fuel cells, nuclear auxiliary power units, and solar cells are all under consideration for the numerous vehicle demands.

d. Since the attitude of the space station will be adversely affected by the docking impact and by mass unbalance during the unloading of the supplies or resulting from crew movements, large quantities of fuel must be programmed for the attitude control system. In addition, certain experiments will require that the vehicle hold a very precise attitude.

e. As the effective area of these stations will be quite large, aerodynamic drag will be appreciable. In order to maintain the desired orbital altitude of from 200 to 300 km, an intermittent thrust will be required to counteract the drag forces.

2.3.2 Lunar Exploration

Studies are in progress to develop suitable lunar base concepts and to define the operational requirements and environmental considerations which apply to the various subsystems that must be furnished to support these long-duration, multi-manned missions. One such study envisions a series of modules compatible with the constraints of a SATURN V-launched payload, namely 4.6 m maximum diameter and 55,000 kg earth weight (Ref. 21, p. 10). The module designs include 3-, 6-, and 12-man shelters, backed by an environmental control system module, logistics and resupply modules, a communications module, and a power supply module. Bases of various crew sizes and operating periods would be assembled using the required number of these basic units.

One other important payload would be a lunar surface vehicle (LSV). A typical design is illustrated in Fig. 20. Motive power would be provided by electricity generated by hydrogen-oxygen fuel cells; each wheel would be driven independently by a DC motor.

In addition to its basic function as a lunar surface exploration vehicle, the LSV could perform other critical functions at the moon base. For example, it might serve as a hauling tractor to bring in the automatically landed logistic and resupply vehicles from their actual landing point to the base location. If the terrain proves exceptionally rough, the LSV may have to function as a bulldozer and road grader. Finally, the LSV in its soilmoving and backfilling function could be used to bury the nuclear power plant for radiological safety reasons and to cover the walls and roof of the shelter with an insulating, as well as protective, layer of lunar soil.

These base modules must be designed not only to function reliably during the period of their use but they must also be capable of surviving long-duration unattended storage through perhaps several lunar day/night cycles in the harsh lunar environment. Undoubtedly, for safety and cost considerations, all of the requisite modules would be pre-positioned as close as possible to the base site. Their condition would then be monitored for damage by automatic instrumentation prior to the departure of the manned exploration party from earth.

The surface vehicle also presents problems. Special lubricating techniques must be employed if mechanical parts are to function reliably in the high vacuum environment. There are also some indications that the effectiveness of the thermal control surfaces and fuel cell radiators may be rapidly and drastically degraded by the adhesion of lunar dust. Needless to say, the LSV must possess a very high inherent reliability.

. 2.3.3 Planetary Exploration

A modest program for the unmanned exploration of the earth's two nearest planetary neighbors, Mars and Venus, is under way. MARINER II, for example, flew close to Venus in December 1962 and verified the high surface temperature readings deduced from earlier radiotelescope observations made from earth. MARINER IV gave man his first high resolution (the resolution of earth-based telescopes is only from 200 to 400 km) glimpses of the Martian surface when its single TV camera took a sequence of twenty-one photographs as the spacecraft flew by the planet on July 14, 1965. See Fig. 21 for its flight trajectory.

Most of the interest in the exploration of other planets centers on the investigation of Mars. Since Venus is perpetually shrouded in thick clouds, very little is known about its surface except that the measured temperature is quite high. On the other hand, except for localized clouds and sandstorms, the Martian surface is generally always visible. The available spectroscopic and polarimetric evidence indicates that the

surface is quite arid and flat, although the persistence of certain cloud patterns indicates that some mountainous regions may exist. Moreover, the surface coloring changes seasonally, indicating that vegetative life may exist. Accordingly, Mars is the more promising planet for exploration. Table II summarizes the basic astronomical and known environmental data about the two planets.

The recent MARINER IV photographs of Mars indicate that the Martian surface characteristics may resemble some of the lunar features. Information relating to the atmosphere of Mars, surface structure, and somewhat the surface composition has been obtained. However, no conclusive results have been received that could be utilized in determining the existence of life on Mars.

Needless to say, a considerable amount of more definitive information about both the atmospheric structure and composition, and about the surface geography and meteorology, must be acquired before manned exploration can be considered.

The VOYAGER program (a proposed VOYAGER vehicle is illustrated in Fig. 8) has now been initiated to collect this needed information in the following manner:

a. Extended surveillance and atmospheric probing from Orbiters.

b. Soft landings on the planet's surface.

The proposed VOYAGER vehicle is composed of three parts: the orbital spacecraft weighing from 700 to 900 kg, the retro-rocket weighing from 4500 to 7000 kg, and the sterilized landing capsule which may weigh from about 1300 to 4500 kg (of which about 10 to 15 percent would be useful payload) in the more advanced version (Ref. 11, p. 6).

In the soft landing mission, the spacecraft would remain in orbit to make planetary observations and measurements and may also serve as a communications relay station between the landed capsule and the earth. The information transfer rate will be from 5000 to 10,000 bits-sec^{-1} back to earth. The direct capsule-to-earth radio link is expected to handle from 0.5 to 5.0 bits-sec^{-1} depending upon the power of the lander's transmitter. It is worthwhile to compare these bit rates with the 10^6 bit-sec^{-1} rate used to transmit the television pictures from RANGER VII, VIII, and IX. A similar type of mission philosophy has been proposed for exploratory trips to Venus.

18

Manned flights to the planets will require solutions to problems in the following areas:

a. Propulsion considerations

b. Trajectory analysis

c. Earth atmospheric reentry at hyperbolic velocities

d. Aerodynamic braking

e. Systems reliability

Some of these problems are amenable to investigation and solution in research and development programs that will exploit the unique capabilities of orbiting space stations; other problems are unique and will require advances in the technological state of the art. A few comments relating to the above list of problems are presented in the following subsections.

2.3.3.1 Propulsive Energy (Velocity Increment) Requirements

The energy that is available to attain an optimum trajectory and to correct a trajectory error is a basic constraint on interplanetary flight. The energy required for any particular mission is proportional to the absolute value of flight velocity that must be added or subtracted, i. e., ΔV. The ΔV required for interplanetary travel is illustrated in Fig. 22. Note that travel to either Mars or Venus from the earth requires a total velocity increment of slightly less than 13.0 km-sec^{-1}. Note also that exploratory flights to the outer limits of the solar system are possible with ΔV of 17 km-sec^{-1}, provided motion is confined to the plane of the ecliptic. Energy requirements for trips out of the ecliptic plane are prohibitively high.

In the case of Mars, since that planet has a rather eccentric orbit ($\epsilon_{\sigma} = 0.093$ compared to $\epsilon_{\oplus} = 0.017$), the energy requirements for a Mars flight vary considerably with time. Table III summarizes ΔV requirements for a "slow" and "fast" all-propulsive passage during a favorable and an unfavorable period.

If the propulsive or mission constraints restrict the choice of trajectory to the minimum energy one (which is the Hohman or 180-deg transfer), the total elapsed flight time is approximately 1000 days. About 450 days of this total represents an unavoidable stopover at Mars, since a true minimum energy flight is only possible if the flight legs are so timed that the spacecraft arrives at the target planet as the target planet crosses the common line of nodes of Earth and Mars. Otherwise, an additional velocity increment is required to perform the 2-deg orbital

plane change made necessary as a result of the mismatch between the orbital planes of Earth and Mars.

If a velocity margin is available, an optimized trajectory can be flown that will reduce flight time by more than 50 percent, to about 400 or 450 days depending upon the year and the actual magnitude of the ΔV margin. The minimum energy trajectory and two variations of a "fast" trajectory are depicted schematically in Fig. 23.

The higher energy trajectories are characterized by the requirement that, on one leg of the flight, the spacecraft must pass within the orbit of Venus. If this occurs on the Mars-to-Earth segment, the trajectory is termed "direct"; if it occurs during the Earth-to-Mars segment, the trajectory is termed "direct-inverse." These two flight modes and the minimum energy mode are compared in Table IV.

Considering only the propulsive energy requirements, the direct mode is obviously superior to the direct-inverse mode. This results directly from the spacecraft's velocity vector at earth departure being aligned parallel to the tangent to the earth's orbital path; consequently, the heliocentric motion of the earth is utilized to maximum advantage. As a result, the weight that must be launched from earth orbit is minimized. However, the direct mode produces a much higher re-entry velocity upon return to earth. The problems arising from this fact are discussed in a later section.

The thermal design of vehicles that follow the direct and direct-inverse flight trajectories will be difficult since large variations in the solar radiant flux density will be experienced (from 0.5 solar constant at Mars to greater than 2.0 solar constants within the Venus orbit).

2.3.3.2 Trajectory Analysis

The three basic trajectories discussed in Section 2.3.3.1 have been exhaustively analyzed in the open literature. Similar analyses of other trajectories, different time periods, and various propulsion engines, such as solid and liquid chemical engines, nuclear rockets, and nuclear-powered electrical propulsion engines (arc-jet, plasma thruster, or ion engine), are also being conducted.

An interesting result from studies such as these is the discovery that a considerable reduction both in the velocity increment required for an earth-to-Mars-to-earth trip and in the earth reentry velocity can be achieved if a Mars-Venus swingby maneuver is utilized (Ref. 27). This maneuver uses the gravitational attraction of Venus to accelerate

the returning spacecraft and thus change its heliocentric trajectory. The amount of acceleration and ensuing course alteration is a function of the distance of the spacecraft from Venus. The Venus swingby trajectory and the direct flight mode are illustrated in Fig. 24. Table IV, column 5, lists the pertinent data (such as ΔV, reentry velocity, and flight time) for the swingby flight. The outstanding feature of the swingby trajectory is the reduction in earth reentry velocity from 20 km-sec^{-1} to the more manageable value of 13.4 km-sec^{-1}. In all of the trajectories considered above, the spacecraft velocity vector at earth encounter makes a small angle with, and is in the same direction as, the earth's orbital velocity vector. As a result, the spacecraft's velocity relative to the earth is appreciably reduced. In the particular case of the swingby mode, this angle is smaller than in the other cases and, consequently, a larger reduction in planetocentric velocity ensues. The total trip time for the swingby mode is longer by two months. Although a weight penalty to accommodate the additional supplies will occur, the reduction in the heat shield requirements made possible by the decreased reentry velocity may offset this penalty.

2.3.3.3 Earth Reentry Velocity

If a manned vehicle is to reenter successfully, it must be flown down a flight corridor. The boundaries of the corridor are determined by the amount of lift that the vehicle can develop, by the maximum sustained deceleration that the crew can tolerate, and by the maximum temperature and dynamic loading the vehicle structure can withstand. The flight corridor for a vehicle operating with a wing loading of 25 kg-m^{-2} and structural limitations of 2500°K maximum temperature, 1000 kg-m^{-2} maximum dynamic pressure, and a maximum deceleration of 10G is illustrated in Fig. 25. This figure also shows that the flight corridor expands as the wing loading decreases and shrinks as the allowable structural maximum temperatures, dynamic pressure loads, and G-loadings decrease.

However, in the case of vehicles returning to earth from the moon or a planet, gaining entrance to this flight corridor is difficult because the entrance height of the corridor is a function both of the parameters of the atmosphere and of the planetocentric velocity of the space vehicle. In the case of a nonlifting manned vehicle (lift-drag ratio of zero) which is limited to a maximum deceleration of 10G, the entry corridor is approximately 300 km high at earth orbital velocity, about 11 km high at earth escape velocity, and vanishes for velocities in excess of 14 km-sec^{-1} (Ref. 28).

This entry corridor is bounded by an overshoot and an undershoot trajectory (see Fig. 26a). If the spacecraft arrives at a point above the

overshoot boundary, the atmosphere will be too thin to extract the required amount of kinetic energy from the vehicle to reduce its velocity below orbital velocity; consequently, no reentry will be achieved. Of course, if the vehicle's velocity could be reduced below escape velocity, entry could still be effected after several passes through the outer fringes of the atmosphere. Unfortunately, the vehicle will then traverse the intense radiation zones on each pass (thereby increasing the shielding requirements), and will also need very accurate guidance if a landing is to be made at a predesignated spot (Ref. 28). In the other case, if the vehicle enters the atmosphere at a point below the undershoot boundary, it will encounter the dense portion of the atmosphere. The resulting drag, deceleration, and heating rates would probably exceed the vehicle's design limits.

Fortunately, the entrance corridor height can be significantly increased if the vehicle can generate lift. Fig. 26b shows that, by increasing the L/D ratio from 0 to 0.5 for a reentry vehicle of 11.2 $km-sec^{-1}$, the available corridor height increases from 11 to 71 km. Note also that a further increase in the L/D ratio does not provide much additional capability (i.e., L/D = 1.0 gives a corridor depth of 80 km). However, an increased L/D ratio also causes a marked rise in the total heat load and in the heating rate.

Negative lift effectively increases the overshoot boundary limit since the spacecraft can be controlled to fly for a longer period of time through the rarefied atmosphere. An even more effective method of raising this upper boundary is to deploy a large, high drag device. Positive lift, on the other hand, effectively lowers the undershoot boundary since the lift force can either be held constant until the flight path angle reaches zero, at which point it is altered for landing, or be modulated to maintain a constant deceleration rate.

Earth return velocities for the direct mode from Mars are very high; they range from 15 to 17 $km-sec^{-1}$ during favorable years and to greater than 22 $km-sec^{-1}$ during unfavorable years. Unfortunately, the practical upper return velocity, based on the "G" limitations of man, is about 14 $km-sec^{-1}$ for a ballistic reentry; above this velocity, the reentry vehicle must generate lift. As velocity increases, the corridor height decreases (see Fig. 26b). Thus, a vehicle with an L/D = 0.5 (e.g., the APOLLO command module) has a corridor 25 and 2 km high at 15 and 20 $km-sec^{-1}$, respectively. Even if the vehicle L/D were increased to 1.5, the corridor would only expand to 38 and 15 km at 15 and 20 $km-sec^{-1}$, respectively. With the direct trajectory, therefore, the weight penalty of a braking retro-rocket system must be accepted. This emphasizes again the attractiveness of the Venus swingby flight.

However, even at the lowest earth return velocities (from 12 to 14 km-sec^{-1}), the heating rates and total integrated heat load are still formidable and will greatly affect the vehicle design. For example, since convective heating is the dominant heating source during reentry at ballistic missile and orbital velocities, the optimum shape for a manned vehicle reentering at such velocities is a blunt body, e.g, the MERCURY and GEMINI capsules. Such a shape is suitable for reentry at parabolic velocities if lift is employed, e.g., the APOLLO command module. However, as velocity increases above the orbital value, more of the vehicle's kinetic energy goes into ionizing the gas in the vicinity of the stagnation point; radiative heat transfer from the gas cap increases rapidly with velocity, becoming the dominant heat source at hyperbolic velocities. This fact dictates a different-shaped reentry body. In particular, the vehicle should be slender with a pointed nose (Ref. 29).

2.3.3.4 Aerodynamic Braking

Aerodynamic braking of an unmanned Mars lander or a manned Mars Excursion Module, and for the earth return capsule of a Mars Mission Module, is highly desirable in order to reduce the ΔV requirements for the mission. The effectiveness of atmospheric braking is shown in Table V for the all-propulsive mode, earth braking only, and Mars and Earth braking. Note that in the latter case the total velocity increment is reduced by approximately 50 percent.

2.3.3.5 Systems Reliability

Since all equipments (such as the life-support system, electronics, and auxiliary power system) must function continuously with little or no maintenance on a flight that will last from 1-1/2 to 3 yr, the highest reliability will be essential. To achieve this capability as well as to obtain the necessary guidance and navigational precision, a considerable advance in the state of the art in all fields of engineering and science is required. For example, Table VI gives some indication of existing capability in areas of electronics pertaining to guidance and data transmission, together with the capabilities deemed necessary for interplanetary operations (Ref. 30).

Furthermore, to minimize the mass of expendables that must be carried, a closed (ecological) life-support system must be employed. Power requirements, moreover, are likely to exceed the capability of a solar cell system. Since fuel cells appear uneconomical for longduration operation, a nuclear auxiliary power system will have to be developed.

Manned interplanetary missions will probably be made feasible only by employing earth-orbital assembly or refueling operations (Ref. 31). The level of reliability necessary to ensure that the complex systems will operate satisfactorily over long flight periods is of major concern. The psychological problems arising from the ever-present hazard of the space environments will be even more acute than in the case of long-duration flight in orbital laboratories. In the latter type of mission, the astronaut is aware, subconsciously at least, that a reentry to the earth is always possible. However, on an interplanetary mission, the astronauts will be only too well aware of their isolation; should a mechanical, medical, or mental emergency arise, there is no possibility of aborting the mission within any reasonable time period.

SECTION III
TEST OBJECTIVES RELATING TO VEHICLE DEVELOPMENT

The initiation of any exploratory or technological exploitation program in outer space requires the expenditure of a considerable sum of money and the diversion of large numbers of scientists, engineers, and technicians from other sectors of the national economy into the aerospace field. Additional money must also be spent on the various ancillary equipment required to support a space program, such as launch site equipment, communications and tracking networks, control and data processing centers, and ground test facilities. The cost of the launch vehicle and spacecraft may often represent only the visible portion of the cost iceberg. Every effort must therefore be made to extract the maximum amount of information from all flights.

In conjunction with the economic aspects of a vehicle failure, many experiments are time-critical. For example, studies designed to investigate those properties of the space environment that are influenced by solar conditions may be limited to periods of either minimum or maximum solar activity. Such periods exist for about 18 months every 11 yr. Missions to the moon have a launch window of only a few days once a month because of the requirement that the landing must occur on a sunlit portion of the moon visible from the earth and at a time when the attitude sensors of the spacecraft cannot be confused by the sun, earth, or moon being approximately collinear. Planetary expeditions, since energy limitations play very decisive roles, suffer from much stricter launch-time restrictions; trips to Mars, for instance, can only be undertaken with existing propulsion systems during a period of a few months in 1966 and 1971.

Lastly, the demands of national prestige and public opinion associated with manned spaceflight require, at least during this early stage of man's adventure into space, that the astronauts complete their mission in a successful manner. Such reliability can only be achieved by following a rigorous and comprehensive program of ground-based developmental tests and an associated series of flight tests.

Development tests of vehicles are conducted primarily for two reasons. First, it is essential to identify component, subsystem, and system problem areas in sufficient time to permit solution of any difficulties prior to the flight testing or operational phase. In essence, this type of testing involves proving the design adequacy of all equipments. The second reason relates to evaluation and proving of the performance envelope; generally, this portion of the test schedule is conducted by submitting the vehicle to actual flight tests. Operational suitability trials are usually conducted simultaneously. A typical progression of vehicle development tests is illustrated schematically in Fig. 27.

The question normally arises early in vehicle development: can this particular item be proven by ground testing techniques or are flight test programs required? Ground tests and flight testing are complementary in most respects; the criteria for specifying ground testing are usually those of convenience and cost. For example, tests in ground facilities offer the following advantages over those conducted in actual flight:

a. Tests on materials and components may be repeated as often as required at minimal expense.

b. Tests can be made more comprehensive because of the provision of greater quantities of instrumentation and more detailed or sophisticated instrumentation.

c. Should a component or system fail, the test article is readily available for examination to locate the cause or mode of the failure.

d. Tests, particularly those of prototype, manned spacecraft, can be conducted in a safer manner, since rescue of the crew in event of explosive decompression resulting from structural failure, breakdown of the vehicle's life-support system, or an emergency such as an on-board fire, can be accomplished within seconds to minutes.

Flight testing, on the other hand, provides the dynamical background and behavioral data which is lacking in ground facilities. In particular, orbital flight is required to duplicate the long-duration effects of

weightlessness on both physical and biological systems. The actual performance of the vehicle and such associated systems as its guidance system, attitude control system, and the flight control system required to provide maneuvering capability during the re-entry and landing phases can best be determined by actual flight.

Today, it is very difficult to test a space vehicle under the combined environments that it will encounter in space. Indeed, certain environments such as penetrating radiation and meteoroid fluxes cannot be simulated in ground facilities with any great accuracy. Other environments, such as pressures in the range of 10^{-12} torr, cannot be economically achieved in chambers sufficiently large to handle full-scale space vehicles. Thus actual flight tests offer a direct medium for investigation of both the synergistic and the long-term effects of exposure to the space environment. It might be of interest then to attempt to evaluate what the spectrum of ground testing techniques might be in relation to flight testing requirements.

3.1 SPECTRUM OF GROUND TESTING

The spectrum of ground testing covers such items as:

a. Material testing

b. Component testing

c. Subsystem testing

d. Systems testing relating to interfacing problems created by components and subsystems.

e. Full-scale integrated systems testing

f. Familiarization training of launch and mission support ground crews and flight crews.

3.1.1 Material Testing

Material testing has assumed considerable importance because the materials used in the fabrication of a spacecraft and its equipment must be capable of withstanding exposure to the harsh environments of space. In addition, materials must be strong and light in order to maximize the useful payload that is launched into space.

3.1.2 Component Testing

The prime objective of component testing is to prove the adequacy of a specific design when exposed to the anticipated environmental

stresses that will be encountered. Consequently, each component will be vibrated, shocked, heated, cooled, and subjected to all other available desired tests. If the component is electrically, mechanically, or hydraulically operated, it should be tested both in adequate numbers and for a sufficient length of time to permit the derivation of statistically significant reliability and mean time between failure (MTBF) data.

3.1.3 Subsystem Testing

Subsystem testing is the follow-on to component tests. In addition to proving that the subsystem will function properly in the combined environments, tests are also required to demonstrate the redundant features incorporated in the design. One of the most significant facts acquired in the MERCURY project (Ref. 32) was that subsystems and systems must possess redundancies to preclude the failure of any single item leading to the possible critical breakdown of a primary vehicle system. Thus circuits incorporating redundancies must be tested in such a manner that not only is each possible loop tested but also that the system or subsystem is completely isolated from all others.

3.1.4 Systems Testing

As implied in the preceding paragraph, testing of complete systems and assemblies is integrally related to subsystem testing. The emphasis in systems testing, however, is directed toward both integrating the component subsystems into an overall system and making the various vehicle systems compatible with their interfacing. Each system must be functionally exercised to demonstrate that it behaves in a stable manner and in accordance with design intent.

· Systems contained completely within the spacecraft are generally quite readily tested. For example, all electronic and electrical equipment should be operated to ensure that the power drain is within the capability of the vehicle's power supply, that the circuits are adequately grounded and impedance-matched, and that radio-frequency interference is below the design threshold.

However, the trend toward integrating certain of the spacecraft's systems (such as the flight control, navigation and guidance, and automatic abort-sensing systems) with those of the primary and second booster stages complicates matters. Besides the obvious problems that arise from assembling workable systems from "black boxes" manufactured by different vendors (for example, incorrectly wired or mismatched plugs and incompatible signal and noise levels), considerable

time must be devoted to proving the compatibility of computer-controlled ground test equipment with the various packages.

3.1.5 Full-Scale Integrated Testing of the Flight Vehicle

The purpose of this phase of the test program is to ensure insofar as it is possible that no serious design deficiencies exist to impede the successful launching, flight, and recovery, if required, of the spacecraft.

Tests in this category fall normally into three phases:

a. Weight and Balance Data - After all of the vehicle systems have been installed, the vehicle is weighed to determine its center of gravity and rotated about its axis to obtain its moments of inertia.

b. Design Qualification - The vehicle and its systems are then subjected to an environmental program more severe than it is likely to encounter in flight. All mechanical, electrical, and electronic equipments are functionally operated to determine compatibility and design adequacy.

c. Flight Qualification - All subsequent vehicles are functionally tested under the environmental conditions expected in flight. If the facility is sufficiently sophisticated, it may be possible to duplicate all, or a major portion, of the mission profile. In addition, it may be possible to calibrate some of the experimental apparatus and other on-board instrumentation.

3.1.6 Familiarization Training

Since vehicle systems are very complex and few-of-a-kind, both the ground and flight crews require all the training time possible. Testing the vehicle in a space environmental chamber or on the launch pad offers valuable training experience.

3.2 SPECTRUM OF FLIGHT TESTING

Flight testing of manned spacecraft is the only available method to evaluate the overall effectiveness of the vehicle and its equipment, and the performance of the crew, in the operational mode.

Although ground testing can simulate certain of the environments, techniques have not been established to simulate the problems arising

from the dynamical behavior of the vehicle. An example of the latter problem is the difficulty of exercising the attitude control system of an orbiting space station. Such a system must maintain the station in a fixed attitude or at a constant spin rate and, at the same time, must null out the effects not only of the disturbing torques caused by the natural and astronomical environment (see Section 3.1.1 of Ref. 2) but also of the torques produced by the impact forces produced during the docking of space vehicles, the effects of impinging rocket exhausts produced by deorbiting vehicles, and movements of the crew. Ground test facilities are also limited in simulation of weightlessness. Weightless conditions can be reproduced for only a few seconds (i.e., from 2 to 5 sec in a drop tower and from 30 to 40 sec in aircraft flying special flight paths); obviously, crews and equipment must be subjected to long periods of weightlessness before long-duration flights to the nearest planets can be undertaken. In addition, all operations involving in-space repair, orbital assembly, or orbital refueling can only be practiced realistically in the space environment. Finally, ground test facilities are as yet too small to simulate the heating effects and critical aerodynamic loadings on full-scale space vehicles.

It might be of interest to evaluate the specialized ground test facilities required to conduct development tests to solve some of the designers' problems. In many cases, the ground test facility required to test some of the future aerospace vehicles is about as technically complex as the vehicle to be tested. The intent of the next section is to elaborate on some of these facilities.

SECTION IV
GROUND FACILITY TEST PROGRAMS

In this section an attempt will be made to describe in detail the wide spectrum of tests that are normally conducted in ground test facilities. Tests of this nature are required to maximize the probability of achieving an effective flight vehicle. The capability to test components, subsystems, and the complete vehicle under environmental conditions in ground test facilities is one of the prime factors that will assure success to the space program. It should be recognized that ground test facilities will not provide all the answers to the designers' problems; however, when used in conjunction with in-flight data, most of the complex technical problems can be solved. The following subsections will give the reader an insight into the many types of ground tests necessary with some of the prime objectives highlighted.

4.1 MATERIALS TESTING

The first tests to be conducted early in the development cycle of a new system relate to selection of materials and surface finishes required in or on the vehicle. The wide range of materials that must be tested includes thermal control surfaces, protective reentry ablative coatings, plastic products to be used on radomes and antenna covers, dielectrics and insulators, and lubricants for mechanical devices that function in the space environment. Materials must be tested to ensure that they possess suitable properties in the hostile space environment and are compatible with each other. Furthermore, materials must not release toxic or irritating gases which could contaminate the closed-cabin atmosphere of the space vehicle.

The three prime tests usually conducted on materials for space application are the following:

    a. Outgassing

    b. Ultraviolet degradation

    c. Thermal radiative properties

The outgassing test consists of exposing a sample of the material to an ultralow pressure. The sample is weighed periodically to determine the weight loss of its volatile and adsorbed constituents, while the gas composition in the test cell is determined by mass spectroscopy. Depending upon the application of the specimen, additional tests, such as a fatigue strength test, are performed. The second test, ultraviolet degradation, is conducted on any material that will be exposed to direct solar radiation. Organic substances in particular are very susceptible to ultraviolet damage since the energy of a light photon in the wavelength range from 2000 to 4000 Å is of the same order of magnitude as that of the electron bonds. Photochemical changes, such as the general darkening and formation of localized color centers in quartz windows and lenses, and the hardening of flexible substances as a result of bond breakage and subsequent cross-linking of the polymer chains may also occur. The third test, measurement of such thermal radiative properties as absorptance (or reflectance) and emittance as a function of wavelength and temperature, yields significant data to the designer concerned with component, system, and vehicle thermal design.

4.2 COMPONENT TESTING

As the individual components are designed, fabricated, and assembled, they are tested in their anticipated environment to evaluate the adequacy of the design. Such test procedures include vibrating and

shocking the item, exposing it to extremes of heat and cold, and measuring the rate of heat transfer to and from the component by conduction, convection, and radiation. In the case of electrical and electronic equipment, the testing also includes performance operation over a wide range of voltages and frequencies. Components that will be exposed to the space environments have functional tests to evaluate the effectiveness of sealing and lubricating techniques. Certain specialized equipment, such as the latching equipment used to join vehicles upon completion of the docking operation, must be operated after long-duration exposure at pressures below 10^{-10} torr to determine cold-welding characteristics. Other component tests may include leak checking of pressurized equipments and determining the insulation adequacy in high voltage equipment in order to prevent the formation of corona and arcovers.

Thermal design of electronic components has become one of the major problem areas since miniaturization has resulted in high packaging densities. This has resulted in the number of heat-producing components being increased, whereas the surface area available to reradiate this heat away harmlessly has been decreased. To aggravate the situation even more, the weightless condition inhibits the use of free convection cooling, and the effectiveness of forced convection cooling is degraded by the subatmospheric pressure maintained in the cabin.

4.3 SUBSYSTEM TESTING

As the vehicle design progresses, it becomes necessary to test subsystems and systems. For simplicity and cost effectiveness, testing of these items may be performed using the pertinent structural sections of the vehicle. In the case of MARINER III and IV, structural vehicles identical to the flight article were built for certain specialized tests (Ref. 33). Evaluation of the thermal control design is an example of such a test. The thermal design test article was identical in construction and surface finish to the flight vehicle. However, all internal heat-producing equipment, such as electronic boxes, was simulated by strip heaters that produced an equivalent amount of heat. The thermal design was then evaluated in a space environmental chamber that simulated the low pressure, the cold blackness, and the incident solar radiation of space. Changes could thus be made to the thermal design test article very rapidly while the effects of changes in the heat loads of individual components could be rapidly programmed into the test situation merely by altering the appropriate strip heater output.

4.4 SCALE MODEL TESTING

Ideally, it would be desirable to conduct preliminary design and development of spacecraft by conducting scale model tests. Besides simplifying the interplay among the functions of design, modification, and environmental testing, scale model testing in most cases is cheaper. Model tests would permit one to utilize smaller space chambers well-equipped with simulation capabilities.

Extensive theoretical and experimental work is now in progress to understand the similitude or scaling principles that form the foundation for modeling techniques. Several approaches are possible depending on the parameters to be studied. For example, it might be desirable to:

a. Preserve local temperatures

b. Use the same materials and surface finishes as the flight vehicle

c. Maintain geometric similitude, i.e., to keep the radiation configuration factors for the model and the full-scale vehicle identical.

In any event, present experience indicates in the case of thermal modeling that it is impractical to reduce the scale below one-half size.

In addition to thermal tests, the determination of the approximate flexible-body bending and vibrational modes, as well as frequencies, is amenable to simulation by scale model tests (Ref. 34). Such information is of the utmost importance in the design of guidance systems for present and future boost vehicles. Another use of scale models is in the measurement of antenna radiation patterns and the investigation of various interaction phenomena among different types of antennas.

4.5 FULL-SCALE INTEGRATED SYSTEMS TESTING

As mentioned previously, systems testing can be divided into two distinct phases. In the first phase, emphasis is placed on ensuring that the basic design of the prototype vehicle and its equipment is adequate to withstand the environmental effects to be encountered. For example, it is necessary to demonstrate that the spacecraft is structurally sound and that all openings and hatches can be sealed properly. Tests are run to determine the capability of the landing vehicle to withstand the shock loads imposed by a ground or water landing under the various touchdown conditions. For example, a typical drop tower test

on an APOLLO boilerplate command module simulated a flat landing in a heavy sea with two of the three recovery parachutes functioning, resulting in the measured impact acceleration exceeding 50 G. Under this condition, the heat shield and the bottom structure of the spacecraft broke open, and the vehicle sank in less than 4 min. The heat shield and associated structure were strengthened as a consequence of these tests (Ref. 35).

Tests of this nature are normally followed by an experimental program to determine the thermal balance of the vehicle and the temperature distributions of the various components. Although extensive heat transfer computations are usually made early in the design, experimental verification is needed because of the complexity of the problem.

During this phase of ground testing, all of the systems on the vehicle will be functionally operated on their minimum and maximum duty cycle. Particular attention is also devoted to the operation of the environmental control system and to the elimination of any toxic, odorous, or irritating contaminant gases in the manned cabin.

Tests conducted in the second phase of the program might broadly be categorized as quality control tests. The first objective is primarily that of checking actual flight vehicles for defects in workmanship. For example, compartments that must be thermally insulated are carefully checked for undesirable heat leaks. The second objective is to train the ground and flight crews in the necessary procedures for safe and efficient operation. Thus the vehicle systems and crew would be exercised on a simulated mission. The crew would also have the opportunity to practice various emergency procedures involving potentially dangerous situations, such as an on-board fire, meteoroid puncture of the cabin pressure vessel, or the failure of a vital component.

Manned exploration missions to the moon, Venus, and Mars open up another aspect of space simulation: namely, the simulation of environmental conditions on the lunar or planetary surfaces. Provision of ground facilities for this type of testing would permit evaluation of such procedures as the unloading of supply vehicles, the construction of the buildings and workshops associated with a base, and the operation and maintenance of all the mechanical equipment and transport vehicles necessary to sustain life.

To illustrate the interaction problems uncovered by a combined systems test, one might consider a few of the areas investigated and rectified during the space environmental test program that was conducted on the MERCURY capsule (Ref. 36). Tests were run to evaluate the

interfaces and electronic components, electromechanical components, waveguides, and transmitting and receiving antennas on the vehicles. Typical problems encountered and solved are shown below:

    a. Magnetic disturbances from the 400-cps dc to ac inverters created noise and induced spurious signals in several circuits. The inverter was redesigned.

    b. Power spikes and transients originating in inductive electrical components caused many transistor failures. Additional filtering was introduced.

    c. During transmission of signals through an antenna on the upper part of the vehicle, the infrared earth-space horizon tracking circuits picked up the rf energy and produced erroneous signals. The antenna was redesigned.

4.6 GROUND TEST FACILITIES

The many and varied environmental tests described in the preceding section and required for spacecraft development dictate that considerable knowledge must be made available to the design engineer in order to arrive at a satisfactory vehicle design. It must be remembered that the test conditions can vary from simulating a single environment to duplicating combined environments. In addition, the test article may range from a small material sample to a complete spacecraft. However, it is extremely important that the "tools of the trade" be thoroughly understood by the designer and the flight crews. It is in this respect that a review is made of some of the ground test facilities available and contemplated. This section is certainly not meant to be inclusive or a summary of all ground test facilities, but only to give the reader a general idea of the many aerospace facility capabilities available.

4.6.1 Aerospace Environmental Chambers

Tests discussed up to now can be conducted in first generation aerospace environmental chambers. Space chambers range in size from the bell-jar category to extremely large ones such as the Air Force's Mark I Aerospace Environmental Chamber at Arnold Engineering Development Center. The Mark I, for example is a chamber 12.8 m in diameter and 25 m high; it is equipped with diffusion pumps to attain a high vacuum of 10^{-6} torr with a test article in place, liquid-nitrogen-cooled wall panels to simulate the coldness and vacuum of space, a solar simulator which will irradiate the test article with a radiant flux density of 1400 w-m^{-2}, and infrared-emitting lamps to simulate the solar albedo and earth radiations. In addition, the decrease in ambient pressure and

the aerodynamically induced vibrations encountered during the ascent
phase of flight (from sea level to approximately 24,000 m) can be simu-
lated by evacuating the chamber with the compressors associated with
a wind tunnel complex at Arnold Center and by mounting the test article
on vibrating (or shaker) pads located at the bottom of the chamber,
respectively. It is quite common today to have rather large chambers
capable of simulating the environments as mentioned in the discussion
of the Mark I chamber.

4.6.2 Manned Testing in a Space Environmental Facility

Many of the tasks that man must perform inside a spacecraft, in
space, or on the lunar surface can first be investigated under the
controlled test conditions of a space environmental chamber. For
example, it is important to test space suits under the proper space
environmental conditions for comfort, adaptability, and mobility, both
within and outside the vehicle. A closely associated test involves the
capability and reliability of the spacecraft's environmental control sys-
tem (ECS). The performance of the ECS can be monitored by medical
personnel who, besides observing the test subject directly or by tele-
vision, would determine the amount of oxygen consumed, the carbon
dioxide content of the exhaled gases, the pressure and temperature in
the space suit, and such biomedical data as the pulse rate, blood pres-
sure, and heart beat.

In addition to solving various engineering and medical problems,
such tests will enable the crew to acquire valuable experience. If long-
duration tests are conducted with crews in the vehicle, the psychological
compatibility of the individual crew members can also be realistically
evaluated. The crew can thereby experience the dulling monotony of
routine sleep-rest-work cycles and the confinement and lack of privacy
that must be endured in the cramped quarters of the spacecraft.

However, before manned tests can be undertaken in the space
environmental chamber, the environmental chamber must be manrated.
Apart from such obvious changes as the provision of the necessary view-
ports, lighting, and additional instrumentation (primarily biomedical)
required, manrating involves two special requirements:

a. The first requirement involves a double section entry lock.
The proposed installation of such a lock on the Mark I Aero-
space Environmental Chamber is shown in Fig. 28. This
arrangement of locks permits the rotation of vehicle and main-
tenance crews and the resupply of the vehicle while simultane-
ously making a rescue crew available for emergency operations.

b. Another important requirement is to provide a mode of emergency repressurization, since the procedure for returning a conventional chamber to atmospheric pressure takes many hours. In many cases, the rapid repressurization can be conducted in two distinct steps. Since the prime concern is to keep the test subject alive, stored oxygen and nitrogen at high pressure can be used to bring the chamber total pressure up to a level of approximately 300 torr, with an oxygen partial pressure of 180 torr, in 30 sec or less. At this point, rescue personnel on standby in the lock may enter the chamber, apply first aid to the victim, and remove him to nearby medical facilities. While the rescue operation is proceeding, a further supply of stored air can be bled into the chamber, returning it to atmospheric pressure.

4.6.3 Specialized Test Facilities

There are many essential tests of aerospace equipment that cannot be performed in a conventional space chamber. For example, rocket engines cannot be fired in a space chamber without sacrificing altitude simulation; micrometeoroid impact cannot be attempted without chamber damage; high energy particle simulation and operation of nuclear-powered equipment cannot be performed unless expensive and extensive shielding is provided; finally, reentry heating tests cannot be conducted since the required heat fluxes would destroy all of the unprotected systems inside the space chamber. Thus, a variety of specialized facilities must be provided to support the space program. Several of these specialized facilities will be discussed in the subsequent paragraphs.

4.6.3.1 High Altitude Rocket Test Cells

The performance of a rocket engine in space differs markedly from the performance of the same engine at ambient sea-level conditions (Ref. 37). For example, the gross thrust generated by a rocket engine is maximized when the rocket exhaust is expanded to the ambient pressure. Since the ambient pressure is extremely small at high altitude, maximum thrust is obtained with an exhaust nozzle of the highest possible expansion ratio compatible with weight restrictions. However, if such high ratio nozzles are tested at sea-level pressure, the flow would break down because of overexpansion. Some additional difficulties associated with high altitude operations include:

a. Achievement of the proper pressure, temperature, and mixing in the combustion chamber to sustain the combustion process.

b. Capability for multiple restarts in space.

c. Proper functioning of pyrotechnic igniters at high altitude.

Typical high altitude rocket test cells are illustrated. The J-4 cell at Arnold Center (Fig. 29) has a test capsule 15.6 m in diameter and 77 m deep. The cell can handle rocket engines up to 2.25×10^5-kg thrust at simulated altitudes up to 42 km. The smaller cell, J-5 (Fig. 30), is designed to test solid-propellant rocket engines up to 4.5×10^4-kg thrust and up to a simulated altitude of 45 km. Both of these test cells incorporate the following features:

a. After the engine to be tested is installed, the test compartment is sealed off and evacuated to the desired altitude.

b. Since it is desirable to maintain the simulated altitude during the actual firing, steam ejectors and mechanical pumps are operated to remove the large mass flows generated by the engine.

c. Exhauster efficiency can be markedly improved if a diffuser is employed to convert the kinetic energy of the exhaust gases into static pressure by slowing the gas down from supersonic to subsonic velocity.

4.6.3.2 Meteoroid Simulation

The hypervelocity impact effects on space vehicle structures and components caused by micrometeoroid and meteoroid collisions can be simulated by using a hypervelocity impact range, such as the one shown in Fig. 31. The range illustrated can fire projectiles at velocities up to 8 km-sec^{-1}; there are ranges in existence that can achieve velocities up to 18.5 km-sec^{-1}.

A typical hypervelocity launcher (or light gas gun), as illustrated in Fig. 31, consists of five major sections: a firing chamber, high pressure section, launch section, blast chamber, and the target and recovery bed.

a. Firing Chamber - The energy required to accelerate the projectile may be obtained by either burning a conventional gas-producing propellant (cartridge) or by initiating a chemical reaction, such as the chemical combination of hydrogen with oxygen.

b. High Pressure Section - In the cartridge gun, the gases produced act on the base of a plastic piston; this piston compresses the gas ahead of it. In the chemical reaction gun, the shock wave produced by the detonation compresses an inert gas in the second stage.

c. Launch Section - When the pressure in the high pressure section exceeds a preset value, a diaphragm ruptures allowing the gas

pressure to be applied directly to the base of the projectile or to the projectile-sabot combination (a sabot is a device that increases the effective area over which the gas pressure can act; after the combination leaves the barrel, the sabot separates from the projectile).

d. Blast Chamber - The blast chamber absorbs the blast of the gases following in the wake of the projectile. It also serves to trap and contain the remains of the sabot.

e. Target and Recovery Section - This is the instrumented section. Cameras photograph the projectile in flight and at the moment of impact on the target. High speed pulsed X-Ray machines may also be used to record the impact. A soft absorbing bed serves to capture the projectile. A typical crater resulting from one such hypervelocity impact is shown in Fig. 32.

4.6.3.3 High Energy Particle Accelerators

Today, considerable research effort is being exerted on investigations of high energy radiation effects on the physical and thermal qualities of surface finishes. In addition, some investigators have uncovered synergistic effects when some materials are irradiated while exposed to combined environments such as ultralow pressure and high intensity ultraviolet radiation. Considerable experimental effort, as well as engineering and theoretical ingenuity, is being expended on ways to reduce the rate of, and to minimize, the degradation suffered by solar cells as a result of the effects of the space radiations. To produce the high energy particles required for such experiments, a variety of accelerators are now being employed. The major types of accelerators in use include the Van de Graaff generator, the linear accelerator, and the resonance machines such as cyclotrons and synchrotrons.

If the penetrating radiation environment of space is to be adequately simulated, undoubtedly several machines of each type will be required. This is because of the necessity of having to produce not only the correct particle flux but also the proper energy spectrum of the radiation.

4.6.3.4 Magnetic Field Simulator

In Section 3.1.1 of Ref. 2, it was noted that the interaction of the geomagnetic field with the magnetic moment of the space vehicle would produce a torque on an orbiting space vehicle. Since the torque will cause the vehicle to either spin or tumble, this undesirable situation must be prevented by countering the magnetic torque with an equal but opposite torque generated by the attitude control system. It may often be desirable therefore to be able to measure the magnetic moment of a space vehicle.

This is best performed in a facility in which a uniform magnetic field can be maintained despite fluctuations in the geomagnetic field. Such a test area can be constructed within the windings of a large solenoidal coil. By detecting incipient variations in the geomagnetic field and by appropriately controlling the current flowing through the coil winding, it is possible to maintain a very uniform field inside the coil.

Such a facility can also be used to calibrate the magnetometers of spacecraft and to perform demagnetizing (or degaussing) operations on materials that possess remnant magnetization. This latter procedure minimizes the magnetic moment of the vehicle, thereby reducing the fuel expenditure of the attitude control system.

4.6.3.5 Hypersonic Reentry Facilities

The environment associated with a spacecraft's entry into a planetary atmosphere at hypersonic velocities is possibly the most severe condition the vehicle must endure. Vehicles encounter both ever-increasing aerodynamic loads and a substantial heating pulse. The aerodynamic loads vary with the square of flight velocity and are proportional to the atmospheric density and reentry angle. The magnitude of the heating effect, on the other hand, depends not only upon the altitude at which the maximum aerodynamic load occurs, but also upon the mass and shape of the entry body and whether the airflow past the body is laminar or turbulent. Figure 33 illustrates the temperature distribution experienced by a model of the APOLLO command module during a simulated emergency reentry.

Ground facilities for investigating and solving problems in the hypersonic flight regime might best be illustrated by the following examples: (1) The quasi-continuous wind tunnel, such as the arc-heated (or hotshot) wind tunnel illustrated in Fig. 34, exemplifies one class. In this type of facility, high pressure air is forced through a high powered electric arc to raise its enthalpy before it is expanded through a nozzle into the test cell. The test cell is connected to a large tank that is kept at a very low pressure by a pumping system during the test run. (2) Another type of hypersonic facility, the impulse type illustrated in Fig. 35, has two variants: the ballistic range and the shock tunnel. The basic difference between them lies in the fact that the range data are obtained from models in free flight whereas the tunnel data are obtained from static models.

4.7 SUMMARY

It is believed important to reiterate the need for ground testing of spacecraft and their subsystems. Some of the more significant objectives are summarized below:

a. Prove feasibility of a design or concept early in the development program. Total costs are minimized by detailed ground testing.

b. Establish performance characteristics of components and subsystems.

c. Determine reliability and durability of components and systems. Maximum reliability must be gained through suitable ground tests prior to actual flight tests.

d. Validate structural adequacy and integrity.

The effort to put man and machine into space, to ensure proper functioning of the systems while there, and to return man and his re-entry vehicle safely to earth requires a broad foundation of basic and applied research programs. A well planned and highly engineered space program challenges the technical capabilities of the universities, aerospace industries, and governments of the various NATO countries.

SECTION V
FLIGHT TEST PROGRAMS (MANNED)

The Space Age has produced an entirely new gamut of flight test requirements. The space vehicles of today, and even more so in the future, are extremely complex, sophisticated, and expensive. When dealing with manned space vehicles, reliability and performance must be unquestionable. Consequently, suitable ground and flight test programs must be established.

The NASA test philosophy can be stated as follows (Ref. 19, p. 76): "The exhaustive levels of many ground tests are the solid foundation for our flight test program.... Our test plan philosophy is to fly as early as ground testing permits but we will not fly if we are not ready. It is also a basic principle of the program that flight tests are conducted only on those parameters of stages and systems that cannot be proven by ground testing ...."

However, since ground tests cannot provide answers to all of the designers' problems, the final step in the test sequence, after all

possible vehicle development has been performed in the safety and convenience of ground test facilities, must be the commitment of the spacecraft to flight. In particular, operations that depend upon the dynamical behavior of the vehicle can be satisfactorily tested only by actual flight. It is impossible to simulate adequately both the effects of the combined space environments and dynamical effects such as:

a. Aerodynamic loading

b. Shock wave impingement

c. Body bending

d. Fuel sloshing

e. Weightless effects on liquids, gases, mechanical equipment, and man himself

f. Man-machine performance

g. Reentry heating

h. Maneuverability and stability of the reentry vehicle at all points on the continuous flight corridor.

5.1 MAN VERSUS MACHINE

Before discussing Flight Test Programs, it may be worthwhile to examine the reasons for stressing "manned" exploratory space flights. The controversy over the reason for supporting a manned space program when instruments can gather information more cheaply and adequately comes up periodically. However, the collection of information, while important, is only a small portion of the job. Indeed, there are several other functions in which machines surpass men. Everyone knows that man cannot compete in speed or accuracy with a modern, high speed digital computer. Certain tasks also fall outside man's capabilities; for example, a vehicle returning to earth from the moon or another planet must enter a very narrow corridor in space at a very precise entry angle. Should the guidance be in error, it must be corrected within seconds if a successful landing is to be realized. However, computers can only be programmed for the expected. Man's primary advantage is his thinking capability and versatility. With it, he can handle the unexpected and can make intelligent decisions on the basis of sketchy or incomplete information and observations, and on the basis of his previous training and experience. Man also possesses remarkable information-processing ability, again because of his ability to detect trends, anomalies, and peculiarities among vast quantities of data, and to uncover correlations among different types of data. This

latter human trait, for example, would obviously drastically reduce the amount of data that would have to be transmitted to a terrestrial ground station from an orbiting scientific, technological, or military space vehicle. The inclusion of man in the system also greatly increases the probability of achieving a successful mission as was clearly demonstrated in Project MERCURY (Ref. 39). Not only can he employ his decision-making capability to determine whether or not to continue a mission despite a given failure, but he can also isolate malfunctions and repair or replace faulty black boxes or circuit boards. Moreover, in certain situations, man and his senses can outperform automatic mechanisms. The rendezvous and docking operation is an example. The onboard radar and associated guidance equipment can control the propulsive devices to effect the acquisition and gross approach maneuvers in the minimum possible time and with the least fuel expenditure; however, during the final docking maneuver, man is the better controller because he can sense range, range closure rates, and vehicle misalignment more accurately. It is quite apparent that man and machine are complementary and that both must be used in the most advantageous manner to maximize the results from a given mission.

5.2 FLIGHT TESTING OF MANNED VEHICLES

Production of space vehicles, manned and unmanned, unlike that of aircraft in which relatively large numbers of identical vehicles are produced on an assembly line, is severely limited. Each spacecraft in a given class tends to be quite individualistic in design and construction. It is necessary to optimize each vehicle and its contents to take full advantage of the limited payload weight and volume available in order to satisfy the requirements of a specific mission. Whereas aircraft make many flights that soon become routine, a spacecraft normally makes only one flight; therefore, every flight is essentially one of a kind. Let us now examine in detail some of the functions that can be achieved most effectively by manned flight test programs. This will be discussed by the various phases of flight, i.e., launch or boost, orbit, planet exploration, and reentry.

5.2.1 Booster-Spacecraft Flight Qualification Tests

Flight testing of both unmanned and manned space vehicles is generally accomplished in two phases - vehicle qualification tests and the final operation systems tests. We shall elaborate only upon the first phase in this section.

The qualification test program has three basic objectives:

1. To qualify the booster(s) for use in the designated space flight program.

2. To investigate the adequacy of the emergency equipment installed to extricate the crew from the various abort situations that might occur during launch.

3. To qualify the spacecraft (unmanned and manned) for its flight role.

5.2.1.1 Booster Qualification Program

The booster qualification program is a continuation of the extensive engine test firings previously conducted under both sea-level and altitude conditions in ground test facilities. Among the many problems, for example, that require flight investigation is the ability of the flight control system to stabilize an inherently unstable vehicle along a reference trajectory under both normal operating conditions and such abnormal conditions as partial loss of thrust or inability to gimbal a particular engine. The ability of the control system to handle high winds and wind shear constitute another area of investigation. Lastly, the flight test of a booster with a full-scale aerodynamic model (boilerplate model) of the spacecraft, properly ballasted to obtain the correct center of gravity, would provide assurance that the spacecraft - interstage compartment-booster combination was structurally adequate and free of such aerodynamic problems as panel flutter and aeroelasticity during all phases of the flight velocity regime.

5.2.1.2 Abort Testing Program

Abort system tests also use boilerplate spacecraft to ensure that the safety provisions incorporated into the vehicle system do in fact provide the required degree of assurance that these provisions can save the crew. The major abort situations that must be duplicated include the following:

a. Should the booster fail on the launch pad prior to or at the moment of launch, the escape system must carry the crew beyond the radius of the resulting fireball. For example, in the case of APOLLO, the launch escape system will rocket the command module clear of the danger zone.

b. The second trial in the abort sequence occurs when the booster fails at low velocity and low altitude. In the case of GEMINI, the astronauts would extricate themselves by using a conventional rocket-assisted ejection seat.

c. The third flight phase, when the booster is passing through the altitude of peak dynamic pressure and is flying at transonic or supersonic velocity, requires another safety provision because of the excessive wind loadings that an astronaut would experience. In the case of GEMINI and APOLLO, the crew stays in the capsule, which is rocketed clear of the failing vehicle by a solid-propellant escape rocket system.

d. The final abort phase occurs if the vehicle attains hypersonic speed but fails to orbit. The crew must then reorient the spacecraft, fire the retrorocket system, and guide the capsule through the reentry phase to a normal landing.

5.2.1.3 Unmanned Flight Tests

Once the booster rocket has been flight qualified, and flight tests have demonstrated that the abort procedures are satisfactory, a flight model of the spacecraft, unmanned but with all systems operating in their automatic mode, is then launched on a suborbital flight or flights. This test series has the following major objectives:

a. To verify that the spacecraft systems perform as intended under the launch and reentry environmental conditions.

b. To evaluate the effectiveness of the reentry thermal protection system.

c. To evaluate the effectiveness of the landing and recovery systems.

d. To exercise the land and water recovery teams and check the retrieval techniques.

Upon conclusion of the suborbital flights, one or more orbital flights (also unmanned but with all vehicle systems operating in the automatic mode) are carried out to ensure that the systems operate as intended in the orbital environment.

5.2.1.4 Manned Flight Testing and Evaluation

Once the tests of the unmanned booster - vehicle-recovery system have been successfully concluded, manned flight testing will begin. The test pilot crews will play primary and secondary roles in all phases of flight.

The first role that an astronaut can fill is the manned control of the flight from takeoff to landing. Analog computer studies have demonstrated that man is capable of flying the spacecraft not only in orbit but

also in the launch phase (Refs. 40 and 41). The booster-spacecraft combination is aerodynamically unstable; consequently, any failure in the flight control system automatically leads to destruction of the launch vehicle. However, the astronaut, as backup pilot, may be able to salvage the mission or perhaps gain time to initiate the appropriate abort procedures. The limitation of these computer studies must be emphasized: namely, the accuracy of the simulation is dependent upon the accuracy with which the dynamical characteristics of the launch vehicle are known and can be reproduced by the simulator's equipment. However, the dynamical characteristics of the booster can be accurately obtained only by flying it.

The usefulness of man in the loop during launch is very limited at the present time. Other than the astronaut checking the instrument readings during the launch phase, this portion of the flight has always been completed in an automatic mode. However, as the space program progresses, the astronaut may prove to be a suitable manual control for emergencies during the launch portion of the flight.

5.2.2 Orbital Phase

Once the spacecraft is in orbit or on its proper trajectory to the moon or other planet, the astronauts may elect to switch off the automatic control system and take over the attitude control of the space vehicle as a fuel conservation measure. The astronauts are also responsible for performing the necessary maneuvers and velocity changes required to conduct successful rendezvous and docking with an orbiting space station, earth satellite, or companion ship.

Flight testing is also necessary in order to evaluate realistically the psychological and physiological effects of the space environment and of the artificially maintained cabin environment on the astronauts. Space flight is certainly not a very aesthetic experience. The crew members must work, eat, relax, and sleep in a relatively small volume. The consequences are lack of privacy, personal hygiene, and adequate sleep. Experience with pilots making high altitude flights in aircraft indicates that some pilots feel detached from the earth, others develop feelings of oppression, and still others suffer from hallucinations. Up to now, astronauts have been kept so busy preparing and conducting experiments, communicating with earth, and preparing for their re-entry that they have had no time to allow their isolation from their natural environment to affect their subconsciousness. On longer duration flights, such as extended missions in an earth-orbiting laboratory or in deep space, boredom, fatigue, and the development of hallucinations may well prove serious hazards. Again, operational experience

must be accumulated either to disprove these effects or to develop ways to counteract such psychological side effects of space travel. On the medical side, weightlessness is known to cause weakening of the cardio-vascular and muscular systems and decalcification of the skeleton structure. Although astronauts McDivitt and White apparently have had no ill effects from their four-day stay in space, astronauts Schirra and Cooper, and several Russian astronauts, experienced such other adverse symptoms as dizziness, motion sickness, disorientation, and pooling of the blood in the legs. It therefore appears that, although man may be capable of flying to the moon and back without serious effects, men assigned to planetary expeditions involving years of weightlessness will probably require artificial gravity to survive the trip.

In the lunar landing program, man has been assigned the responsibility for making the actual descent, hover, and landing on the lunar surface. Again, although extensive simulator studies have indicated that man can perform this difficult task quite adequately, the conclusive proof can only be acquired through operational experience. In addition, for the sake of reliability, the astronauts will conduct the necessary launch preparations and control the actual firing and launch when the time arrives to return to the orbiting mother ship, the command-service module combination, for the journey back to earth.

5.2.3 Planet or Lunar Exploration

In future lunar programs personnel and quantities of equipment and supplies must be delivered to the lunar surface for construction of the permanent bases required for detailed exploration. Although many of the tasks associated with the construction, operation, and exploration phases of lunar base development can be partially simulated in earth-based space chambers, much of the program will be proof-tested in the lunar environment.

In the event a lunar base is established, methods must be developed to reduce the transportation costs of transferring personnel and supplies. Recoverable earth-launched boosters and nuclear-powered, liquid-hydrogen-fueled rockets appear feasible as ferry vehicles operating between earth and the moon. Both of these systems will require extensive testing. Recoverable boosters will have to be flight tested in the flight corridor shown in Fig. 25. The nuclear rocket engine for this task will have to be operated through many power cycles and over long periods of time in order to demonstrate its reliability for manned flights. Needless to say, this concept of resupply will employ all of the technologies that apply to:

a. Orbital rendezvous and docking.

b. Transfer of personnel and supplies to and from the earth and lunar shuttle vehicles and the earth-lunar cargo vessel.

c. Refueling of the fuel tanks on the earth-lunar ferry and lunar shuttle vehicles under conditions of weightlessness and perhaps in the presence of appreciable radioactivity from a reactor core.

d. Maintenance and repair of spacecraft systems in orbit.

Of course, such techniques as assembly and resupply of space vehicles in orbit will only be possible if man can work effectively outside his spacecraft cabin under the condition of weightlessness. The walks in space by cosmonaut Leonov and astronaut White represent the initial venture of man into space. Subsequent tests of space suits, life-support equipment, tools, and maneuvering units are planned later in the GEMINI and APOLLO programs. The major roles envisioned for man outside his spacecraft include:

a. Inspection

b. Routine maintenance

c. Repair of his own vehicle or some other orbiting craft

d. In-orbit assembly of large space stations

e. In-orbit construction of objects that are either too large to be launched or too fragile to tolerate launch vibrations

f. Deployment of sunshades around tankage containing cryogenic propellants

g. Conduct experiments, including the placement and set-up of instrumentation remote from the disturbing effects of the space vehicle.

5.2.4 Reentry Phase

The final phase of flight, namely atmospheric reentry, again provides a role for man that can only be evaluated by actual flight testing. For instance, both Russian and American astronauts have experienced automatic control system malfunctions; if the crews had not been able to orient their spacecraft properly and fire the retrorockets at the proper time, mission failure could have occurred. In the newest space-craft, some lift is generated during reentry and, by adjusting the lift force relative to the center of gravity, the vehicle may be controlled to land anywhere in a fairly large "footprint" area. Figure 36 shows the footprint of a typical reentry vehicle. Again, the pilot's basic skill will have a marked effect upon the success of this landing procedure.

5.3 SUMMARY

Ground test facilities provide the necessary tools for developing adequacy and equipment reliability. However, in the final analysis, flight test programs are required to prove performance and to demonstrate the full capability of the space vehicle. Furthermore, in order to ascertain the limits of his vehicle and the limits of his own performance, man must perform his various duties in the proper environment, i.e., in space.

<div align="center">

SECTION VI
FUTURE TEST REQUIREMENTS

</div>

The preceding sections on future missions in space, and of the vehicles needed to carry out those missions, have made it possible to hypothesize on some areas of major interest for future ground and flight testing. In the context of recognizing the limited information available on environmental data and technological exploitations forthcoming, a brief discussion on future ground and flight test requirements follows.

6.1 GROUND TEST FACILITIES REQUIRED

The numerous environmental tests contemplated for future facilities dictate that considerable knowledge must be made available prior to determining the simulation requirements. In many circumstances today the facility required to test the aerospace vehicle is as complex technically as the vehicle to be tested. Consequently, the major ground test facilities of tomorrow constitute what one might term long lead-time items. Facilities of this nature will normally take from 4 to 6 yr to design, procure, construct, and shake down. It is essential, therefore, that the design and construction of such facilities be started at an early date in the program cycle if the facilities are expected to be available for vehicle development.

6.1.1 Space Environmental Facility

Perhaps the largest and most expensive of these new facilities is a space environmental test chamber that will possess several unique characteristics not available in existing chambers.

48

6.1.1.1 Size

The chamber must be large enough to accommodate testing of complete future aerospace vehicles. For example, space station concepts, such as LORL, are in the order of 50 m in diameter. The unloading of a lunar surface vehicle from a lunar logistics vehicle and the subsequent operation of the surface vehicle for extended periods of time also dictate that the chamber possess a floor from 30 to 60 m in diameter. Vehicles for interplanetary travel may even be larger. Speculative estimates of from 60 to 120 m in length are sometimes mentioned (Ref. 43).

6.1.1.2 Vacuum Capability

Lunar landing vehicles, payloads, surface transportation, and base equipment, plus the interplanetary vehicles, will be exposed to an ultrahigh vacuum for long periods of time (months and years). Although much testing can be done in small facilities on components and subsystems, future facility capabilities should provide at least 10^{-10} or 10^{-12} torr under load. Any problems caused by surface cleansing, desorption of chemi-trapped gases, and release of interstitially trapped gases will be accelerated as a result of the simultaneous exposure of the surfaces to both high vacuum and the full spectrum of solar particulate and electromagnetic radiation.

6.1.1.3 Heat Sink

The present method of simulating the coldness of space (i.e., by circulating liquid nitrogen through specially designed wall panels) should remain basically adequate. However, the thermal loads produced in the chamber, particularly if operation of a nuclear power mechanism is conducted, will be extremely large. The facility may require an on-site air liquefaction and separation plant to meet its liquid nitrogen needs economically.

6.1.1.4 Solar Simulation

Solar radiation of satisfactory quality can be provided today over rather small irradiated areas in the order of 100 m^2. However, to irradiate a circular space station, a lunar base, or an interplanetary vehicle, a simulator capable of irradiating an area of a thousand square meters may be needed. Fortunately, the quality of radiation may be somewhat relaxed compared to the quality that must be furnished in space chambers in which the thermal performance of the smaller scientific and technological payloads has been evaluated (Ref. 44).

The major challenge facing facility design engineers is how to provide adequate solar simulation in an economical and reliable manner.

6.1.1.5 Lunar Surface Simulation

A space chamber designed specifically for lunar surface simulation will evolve as a specialized type of test facility if the projected benefits of a lunar laboratory and observatory to earth-bound man prove feasible. This facility will differ from a conventional space chamber in five major ways:

a. The floor will have to be heated and cooled to simulate the surface temperature variations that occur during the lunar day/night cycle.

b. The surface of the facility floor will require treatment to simulate the radiative characteristics (i. e. absorptance/emittance ratio) of the lunar surface.

c. Simulated solar radiation testing will require a variable incidence "sun" in order to simulate the thermal conditions at lunar dawn, dusk, and noon.

d. The floor surface should simulate the traction qualities of the lunar surface and, possibly, also the bearing qualities of the lunar surface if building assembly is to be attempted. Consideration should also be given to providing quantities of loose rubble that could be employed in the meteoroid shielding compartments both for protection against hypervelocity particles and as a thermal insulator (Ref. 21, p. 14).

e. If rock material is employed to simulate the terrain conditions, the pumping system of the chamber should be large enough to handle the tremendous outgassing load that will be encountered.

6.1.1.6 Man Rating

Since man is an integral part of most future concepts, the next generation of space chambers should provide safety provisions necessary for manned testing. Needless to say, the rapid repressurization of a chamber a few million cubic meters in volume is a vastly different proposition than that of repressurizing the existing smaller ones. Consideration should be given to providing rescue capsules as a possible way to eliminate the repressurizing action.

6.1.1.7 Penetrating Radiation

The radiation dose received by an astronaut from the ever-present, omni-directional cosmic ray radiation is close to or exceeds the

maximum permissible dose (MPD) of a radiation worker. However, this MPD is still small enough not to constitute a serious health threat for the proposed missions. The major source of concern is the very energetic proton radiation from solar flares. The solar flare hazard (as illustrated by the calculated dosages versus the MPD listed in Table VII) is representative of what the astronauts should expect in the APOLLO command module. On the lunar surface, the astronaut will be protected by a very small amount of shielding, and his accumulated dosage will be much higher unless precautionary measures are taken. On interplanetary trips, the astronauts will be exposed to a succession of flares since the astronauts' exposure time could be measured in years.

In the case of orbiting space stations, since the payload mass that can be placed in orbit with even the largest existing booster is finite, the designer is faced with a choice between a high altitude orbit that provides longevity at the cost of increased radiation dosage or a low altitude orbit implying the converse. Manned military space stations would almost certainly be required to be sufficiently "hardened" against artificially produced radiation that the station personnel could continue to perform their assigned tasks. Thus there is definitely a requirement to evaluate the actual shielding afforded an astronaut in the various crew positions by the basic spacecraft structure, its internal equipment, stores, fuel cells, and any necessary additional partial body shielding. Moreover, all components exposed to the full energetic proton flux, Van Allen radiation, and the less energetic plasma must be tested in a simulated radiation environment to ensure that they will function reliably over the long flight times. Materials must be tested to ensure that their mechanical and thermal radiative properties are known. The use of nuclear rockets and auxiliary power units also demands that the spacecraft and its human crew be protected against the neutron and gamma ray flux from the reactor.

6.1.2 Other Ground Facilities

In addition to tests in space environmental chambers, thorough testing of future vehicle systems will demand that many other types of advanced test facilities be available. For example, new large boosters will require larger engine test stands and vibrational test equipment. Furthermore, the emphasis that is likely to be placed on recoverable boosters and winged, maneuverable reentry vehicles returns aerodynamic, aerothermodynamic, and aerothermoelastic problems to prominence. Thus, hypersonic true temperature wind tunnels will be required to conduct the required extensive testing on materials, structures, and air-breathing propulsion units.

A typical hypersonic wind tunnel with such a capability is illustrated in an artist's sketch (Fig. 37). The basic operation of a facility of this nature might be as follows: High pressure air is passed through a pre-heated cored-brick heater to raise its enthalpy to the desired value. If necessary, this air is then mixed with the required amount of cold air for correct mass flow at a selected altitude and Mach number. The gas is accelerated through a nozzle into the test cell where it forms a free jet. This airflow is diffused and cooled before it enters the exhauster plant for ejection into the atmosphere. The exhaust from an operating propulsion unit is extracted from the test cell in a separate duct.

However, even tunnels using such techniques have bounded upper altitude and Mach number limits. To obtain still higher performance, the facility design engineer must turn to other means to increase the enthalpy of the gas flow. The prime methods on which research and development are under way include (Refs. 46 and 47):

1. Electric arc heating

2. Combustion of fuel/oxidizer mixture in the gas stream (vitiation heating)

3. Magnetohydrodynamic acceleration of an ionized gas stream.

In studies of reentry heating conditions at parabolic and hyperbolic velocities, various methods of simulating the radiant heat flux encountered by the vehicle by beaming in radiant energy generated in an arc furnace onto the test article are also being examined.

Much additional work also needs to be done on hypervelocity damage caused by meteoric impact on vehicle structures. Progress must await attempts to increase both the projectile mass and velocity capabilities of ballistic gun ranges and impact facilities.

6.2 FLIGHT TESTING REQUIREMENTS

Flight test objectives of the future will remain nearly the same as today. The need to qualify the various launch vehicles and spacecraft will still exist. The number of components and subsystems that will be tested in an earth-orbiting development laboratory should increase considerably in future missions. These orbital laboratories will be extremely valuable in studying medical and physical problems arising from weightlessness and in providing satisfactory methods for long-duration storage of propellants in space.

One major area in which considerable effort must be expended is that of flight testing nuclear-powered propulsion and electrical power

systems. Instrumentation to monitor the behavior and to measure the performance of these systems is relatively straightforward. The problem arises in carrying out the essential post-test inspection of the various components for wear, incompatibility, unforeseen degradation, and radiation damage. Such inspection is particularly important because the amount of radiation damage is known to be affected by the presence or absence of oxygen. For ground testing of such systems, special equipments with massive shielding and remotely controlled manipulators are used to disassemble and inspect the system components at close quarters. For in-flight inspection, a specially designed spacecraft may be required to conduct such an examination. Of course, should the reactor system fail during test, the residual radioactivity would be dispersed over all of the components and the radiation hazard would be correspondingly greater.

It has been assumed that man's usefulness in space and the man-machine interaction in space systems will have been affirmatively demonstrated early in the space station program. Therefore, as the astronaut work force available for lunar surface and deep space operations expands, it will be necessary to institute a training program whereby such men can receive flight and systems training in earth orbit, experience the sensations of weightlessness, and participate in the wide range of extra-vehicular activities now envisioned.

The test pilot will be required to prove the operational suitability of the design, development, and flight article. Pilots will operate maneuverable vehicles of the future that are required for safe reentry at parabolic and hyperbolic velocities and for landing at predetermined bases.

REFERENCES

1. Wyatt, D. D. "The National Program for the Exploration of Space." Conference on Space-Age Planning, NASA SP-40, May 1963, p. 9.

2. Carlson, Col. D. D. and MacFarlane, Sqn Ldr G. "Natural and Induced Environments of Near-Earth, Cislunar, and Deep Space, and their Effects on Spacecraft." AEDC-TR-65-108 (AD469015), August 1965.

3. O'Day, M. D. "Upper Air Research by Use of Rockets in the U.S. Air Force." Rocket Exploration of the Upper Atmosphere, edited by R. L. F. Boyd and M. J. Seaton. Pergamon Press, Ltd., London, 1954.

4. Pressly, E. C. "The Sounding Rocket as a Tool for College and University Research." NASA-University-Conference on the Science and Technology of Space Exploration, NASA SP-11, Vol. 1, November 1962, p. 107.

5. Newell, Dr. H. E. "Statement - 1965 NASA Authorization." Hearings Before the Committee on Aeronautical and Space Sciences, United States Senate, 88th Congress, Part 1. U.S. Government Printing Office, Washington, March 1964.

6. Pickering, W. H. "Solar System Exploration." Space Flight Report to the Nation, edited by J. Grey. Basic Books, Inc., New York, 1962.

7. Anon. Space Log, edited by T. L. Branigan. TRW Space Technology Laboratories, Vol. 4, No. 3, Fall 1964, p. 3.

8. Stambler, I. "The Orbiting Observatories." Space/Aeronautics, Vol. 42, No. 3, September 1964.

9. Nicks, O. W. "Space Sciences - Lunar and Planetary Exploration." Conference on Space Age Planning, NASA SP-40, May 1963, p. 27.

10. Stambler, I. "The Lunar Orbiter." Space/Aeronautics, Vol. 43, No. 2, February 1965, p. 32.

11. Hearth, D. P. "Voyager." Astronautics and Aeronautics, Vol. 3, No. 5, May 1965.

12. Anon. "The Soviet Space Program." Space Log, edited by Thomas L. Branigan. TRW Space Technology Laboratories, Vol. 5, No. 1, Spring 1965, p. 2.

13. Shea, J. F. "Manned Space Flight Program." Conference on Space-Age Planning, NASA SP-40, May 1963, p. 37.

14. Holmes, D. B. "Statement - NASA Authorization for Fiscal Year 1964." Hearings Before the Committee on Aeronautical and Space Sciences, U. S. Senate, 88th Congress, Part 1, Scientific and Technical Programs. U.S. Government Printing Office, Washington, April 1963, p. 501.

15. Houbolt, J. C. "Lunar Rendezvous." International Science and Technology, No. 14, February 1963, p. 62.

16. Staff Report. "MOL and Beyond." Space/Aeronautics, Vol. 41, No. 6, June 1964, p. 52.

17. Coulter, Col. J. M. and Loret, Maj. B. J. "Manned Orbiting Space Stations." Air University Review, Vol. 16, No. 4., May-June 1965, p. 33.

18. Vance, C. "Statement for the Record - 1966 NASA Authorization." Hearings Before the Committee on Aeronautical and Space Sciences, U. S. Senate, 89th Congress, Part 1. U. S. Government Printing Office, Washington, 1965, pp. 557-558, 566.

19. Mueller, Dr. G. E. "Statement for the Record - 1966 NASA Authorization." Hearings Before the Subcommittee on Manned Space Flight of the Committee on Science and Astronautics, U.S. House of Representatives, 89th Congress, Part 2. U.S. Government Printing Office, Washington, 1965.

20. Von Tiesenhausen, G. F. "Toward the Orbital Launch Facility." Astronautics and Aerospace Engineering, Vol. 1, No. 2, March 1963, p. 52.

21. Anon. "Initial Concept of Lunar Exploration Systems for APOLLO." NASA CR39, Vol. 5, Summary Digest, March 1964.

22. Waltz, D. M. "Problems Associated With the Design of Manned Lunar Surface Vehicles." Proceedings of AIAA/NASA Third Manned Space Flight Meeting, Houston, Texas, November 1964, p. 301.

23. Branigan, T. L. "MARINER IV Nears Mars." Engineering Opportunities, Vol. 3, No. 7, July 1965, p. 8.

24. Eggleston, J. M., Evans, D. E., and Harris, J. W. "Lunar and Planetary Environments." Towards Deeper Space Penetration, Science and Technology Series, Vol. 2, edited by E. R. Van Driest. Proceedings of American Astronautical Society Symposium, 131st Meeting of the American Association for the Advancement of Science, Montreal, Canada, December 1964, p. 7.

25. Newell, Dr. H. E. "Prepared Statement - NASA Authorization for Fiscal Year 1966." Hearings Before the Subcommittee on Space Science and Applications of the Committee on Science and Astronautics, U. S. House of Representatives, 89th Congress, Part 3. U.S. Government Printing Office, Washington, 1965, p. 273.

26. Pritchard, E. B. "Velocity Requirements and Re-Entry Flight Mechanics for Manned Mars Mission." Journal of Spacecraft and Rockets, Vol. 1, No. 6, November-December 1964.

27. Sohn, R. L. "Manned Mars Trips Using Venus Swingby Modes." Proceedings of AIAA/NASA Third Manned Space Flight Meeting, Houston, Texas, November 1964, p. 330.

28. Chapman, D. R. "An Analysis of the Corridor and Guidance Requirements for Supercircular Entry Into Planetary Atmospheres." NASA TR-R-55, 1959, p. 28.

29. Becker, J. V. "Entry Vehicles." Astronautics and Aerospace Engineering, Vol. 1, No. 10, November 1963, p. 70.

30. Sullivan, F. J. "Statement - 1966 NASA Authorization." Hearings Before the Subcommittee on Advanced Research Technology of the Committee on Science and Astronautics, U. S. House of Representatives, 89th Congress, Part 4. U. S. Government Printing Office, Washington, 1965, p. 649.

31. Sohn, R. L. "A Chance for an Early Manned Mars Mission." Astronautics and Aeronautics, Vol. 3, No. 5, May 1965, p. 28.

32. Bland, W. M., Jr., and Fisher, L. R. "Reliability Through Attention to Detail." Manned Spacecraft - Engineering Design and Operation, edited by P. E. Purser, M. A. Faget, and N. R. Smith. Fairchild Publications, Inc., New York, 1964, Ch. 38, p. 409.

33. Vickers, Dr. J.M.F. "Thermal Scale Modeling." Astronautics and Aeronautics, Vol. 3, No. 5, May 1965, p. 34.

34. Wainwright, J. B., Kelley, L. R., Wallace, D. A., and Keesee, T. H. "Modeling Criteria and Testing Techniques for the Simulation of Space Environments." AFFDL-TR-64-164 (AD 609858) October 1964, p. 125.

35. Shea, Dr. J. "Statement - 1966 NASA Authorizations." Hearings Before the Subcommittee on Manned Spaceflight of the Committee on Science and Astronautics, U. S. House of Representatives, 89th Congress, Part 2. U. S. Government Printing Office, Washington, 1965, p. 520.

36. Wolfers, C. V. "Integrating MERCURY Electronic Systems." Astronautics, Vol. 7, No. 5, May 1962, p. 32.

37. Barnes, L. T., and Goethert, B. H. "Testing Space Vehicles on the Ground." Astronautics and Aerospace Engineering, Vol. 1, No. 6, July 1963, p. 18.

38. Kotanchik, J. N. "Manned Spacecraft Material Problems." Astronautics and Aeronautics, Vol. 2, No. 7, July 1964, p. 12.

39. Boynton, J. H. "The Pilot's Role During MERCURY Systems Failures." AIAA Paper 64-222 (A64-20127), First AIAA Annual Meeting, Washington, June 29 to July 2, 1964.

40. Muckler, F. A. and Obermayer, R. W. "The Use of Man in Booster Guidance and Control." NASA CR-81, July 1964.

41. Hardy, G. H., West, J. V., and Gunderson, R. W. "Evaluation of Pilot's Ability to Stabilize a Flexible Launch Vehicle During First-Stage Boost." NASA TN D-2807, May 1955.

42. Trimpi, R. L., Grant, F. C., and Bohen, N. B. "Aerodynamic and Heating Problems of Advanced Reentry Vehicles." Proceedings of NASA-University Conference on the Science and Technology of Space Exploration. NASA SP-11, Vol. 2, November 1962, p. 235.

43. Finger, H. B. "Statement - 1966 NASA Authorization." Hearings Before the Subcommittee on Advanced Research and Technology of the Committee on Science and Astronautics, U. S. House of Representatives, 89th Congress, Part 4. U. S. Government Printing Office, Washington, 1965, p. 248.

44. MacFarlane, G. "Solar Simulation: Requirements for Large Aerospace Environmental Chambers." AEDC-TR-65-164 (AD469008) August 1965.

45. Mueller, Dr. G. E. "Statement - NASA Authorization for Fiscal Year 1966." Hearings Before the Committee on Aeronautical and Space Sciences, United States Senate, Part 1 - Program Detail. U.S. Government Printing Office, Washington, 1965, p. 200.

46. Cordero, J., Diederich, F. W., and Hurwicz, H. "Aerothermodynamic Test Techniques for Re-Entry Structures and Materials." Aerospace Engineering, Vol. 22, No. 1, January 1963, p. 166.

47. Kotanchik, J. N. and Greenshields, D. H. "Facilities for High-Temperature Flight Environment Simulation." Aerospace Engineering, Vol. 22, No. 1, January 1963, p. 192.

LUNAR PROBES

SOUNDING
ROCKETS

INTERPLANETARY
PROBES

SATELLITES

- GEOPHYSICS
- SOLAR PHYSICS
- LUNAR & PLANETARY
  SCIENCE

- INTERPLANETARY SCIENCE
- ASTRONOMY
- SPACE BIOSCIENCE
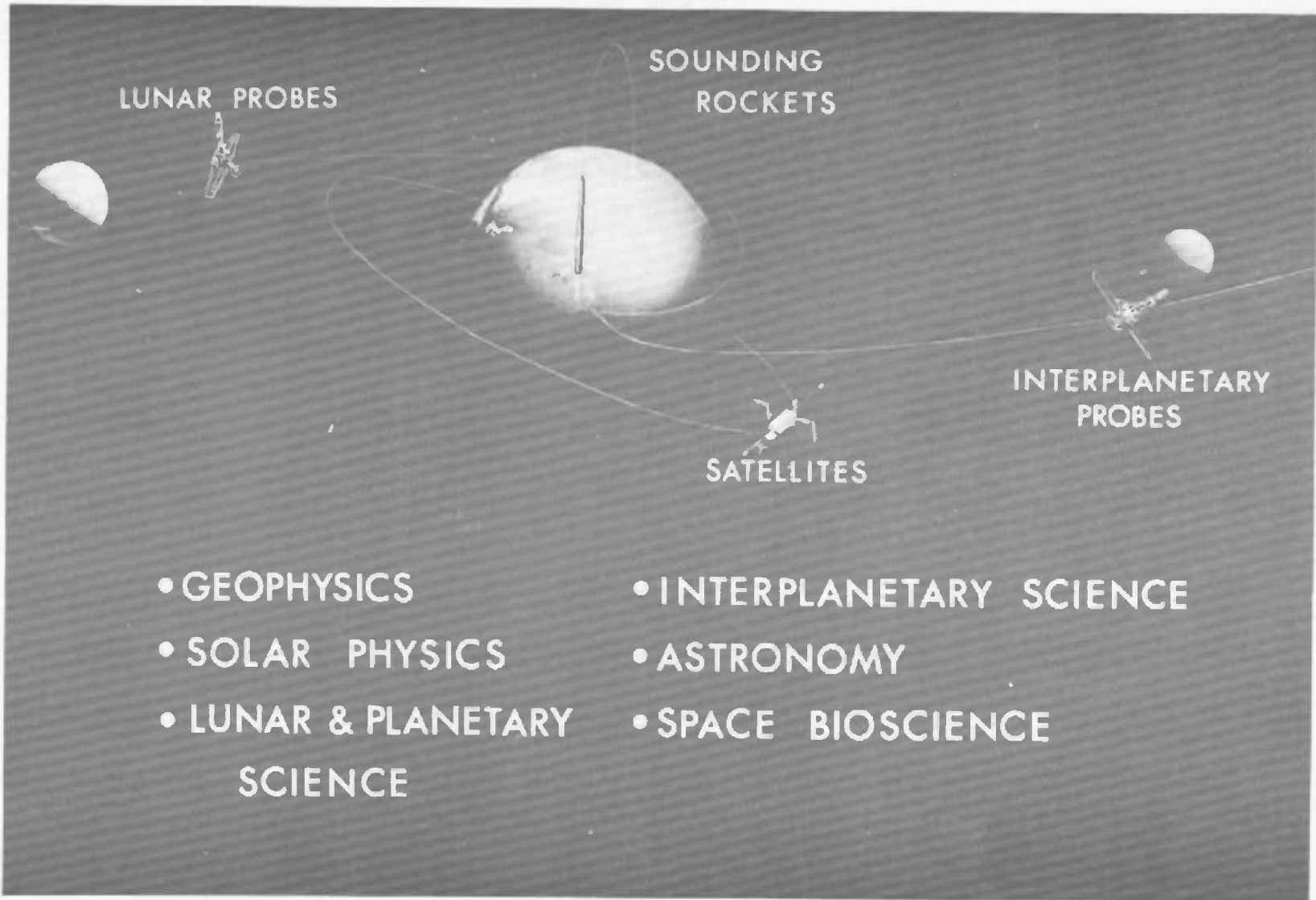
Fig. 1 Spacecraft Mission Profiles

NIKE-CAJUN   NIKE-APACHE   AEROBEE
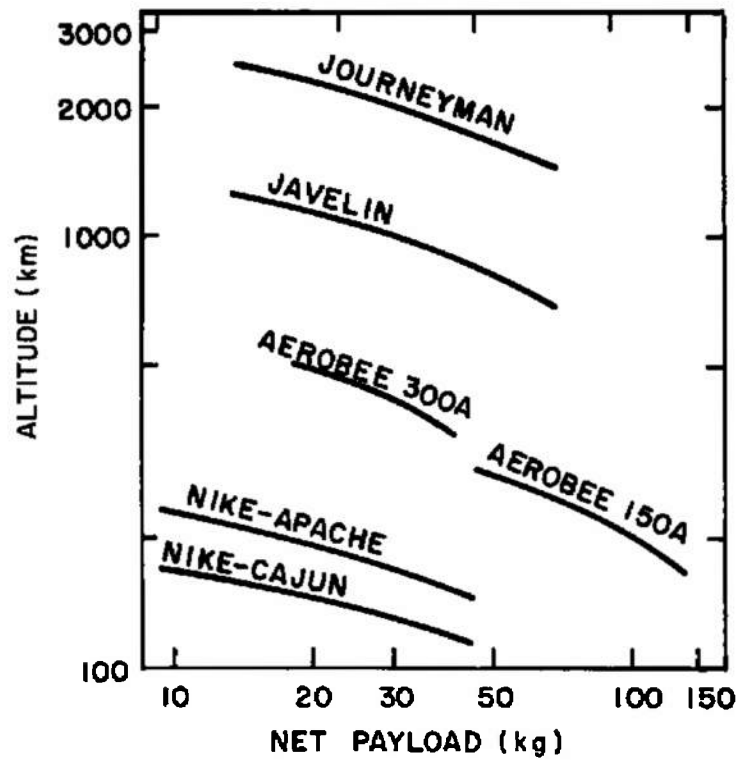150
AEROBEE
150A
AEROBEE
300 A
JAVELIN   JOURNEYMAN
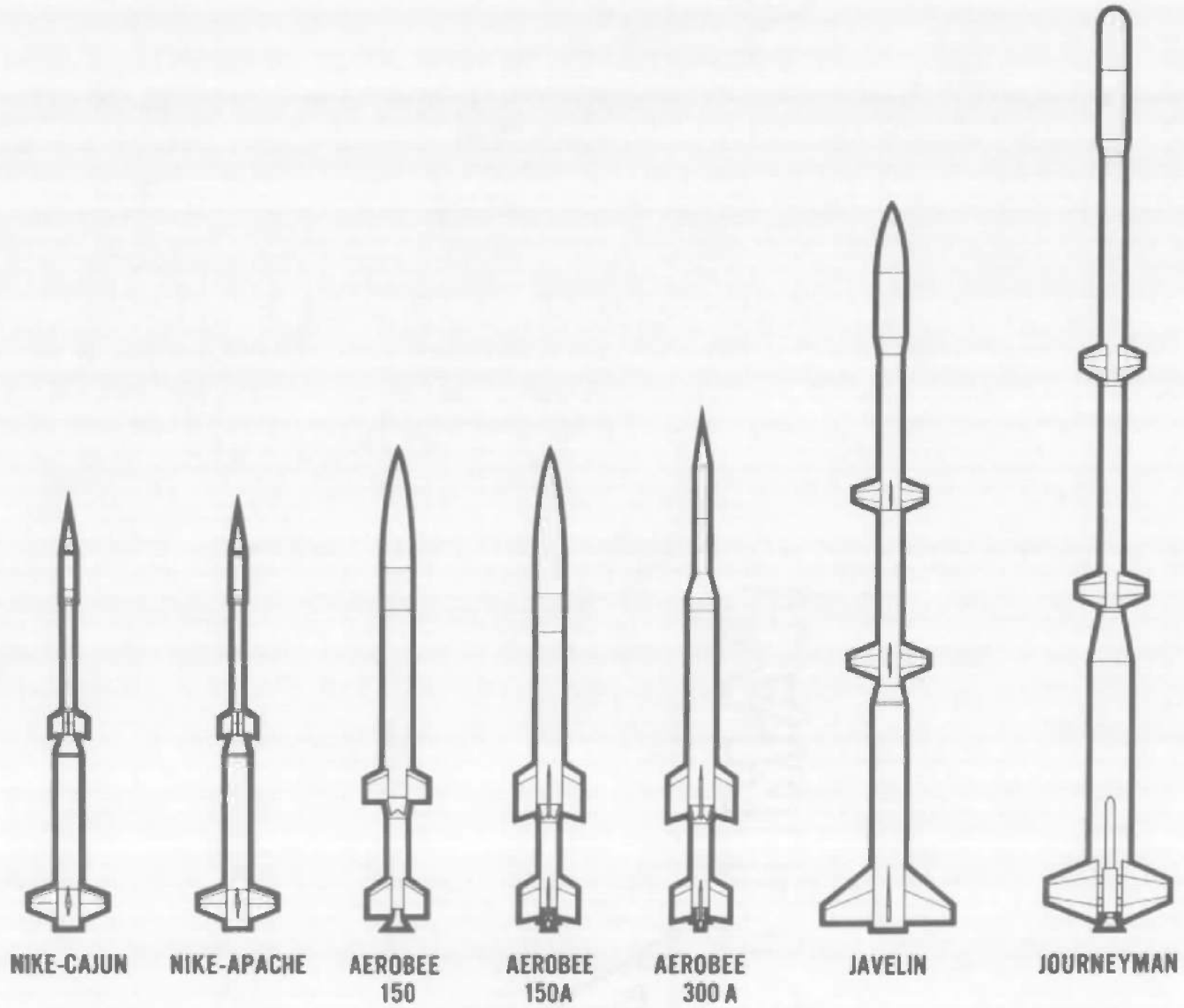
Fig. 2 Sounding Rockets

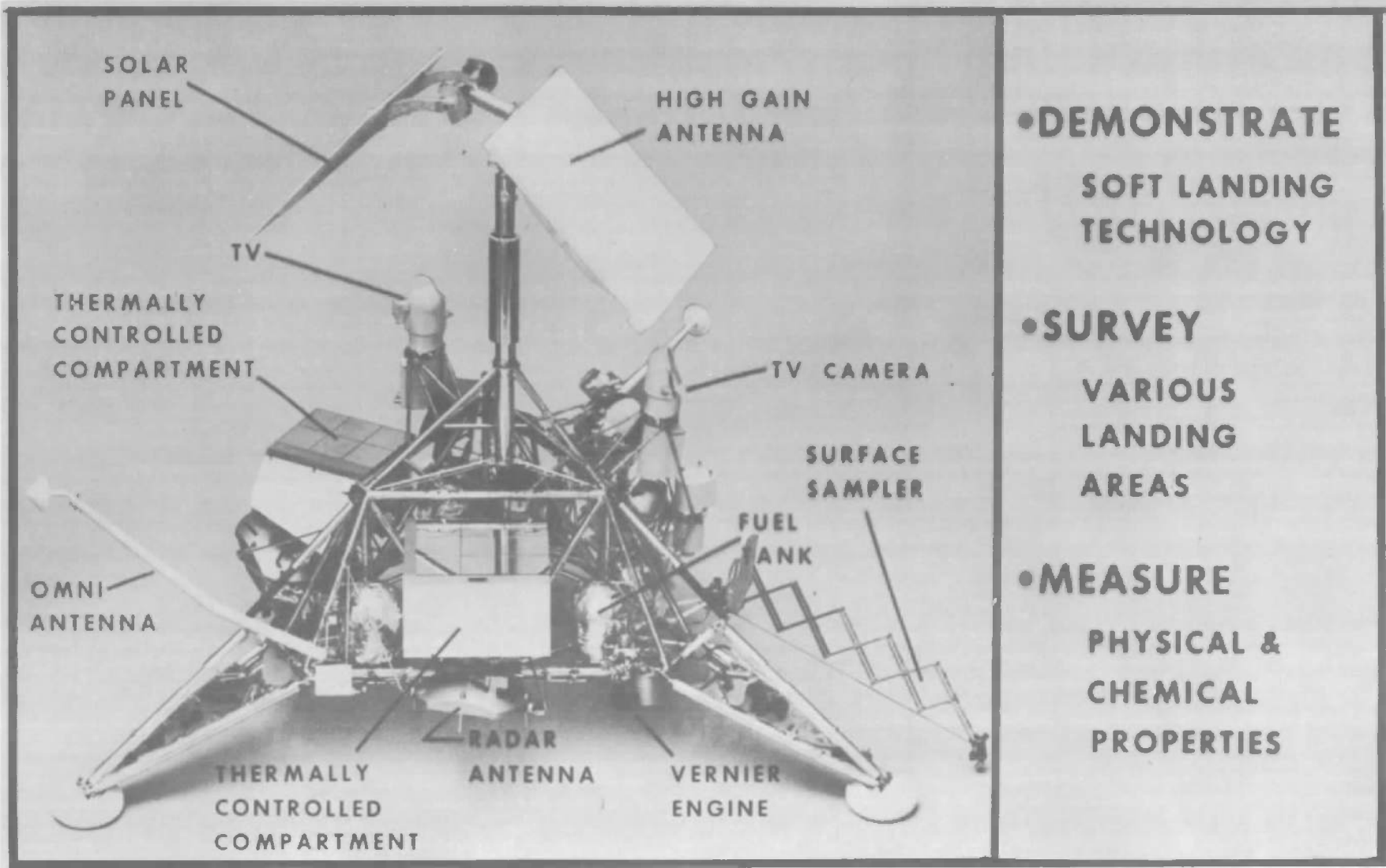Fig. 3 Sounding Rocket Peak Altitude in Terms of Payloads

OMNI–ANTENNA

HIGH GAIN ANTENNA

MAGNETOMETER

EXPERIMENT VIEWING

SOLAR CELLS

Fig. 4 Typical Scientific Probe – PIONEER Series
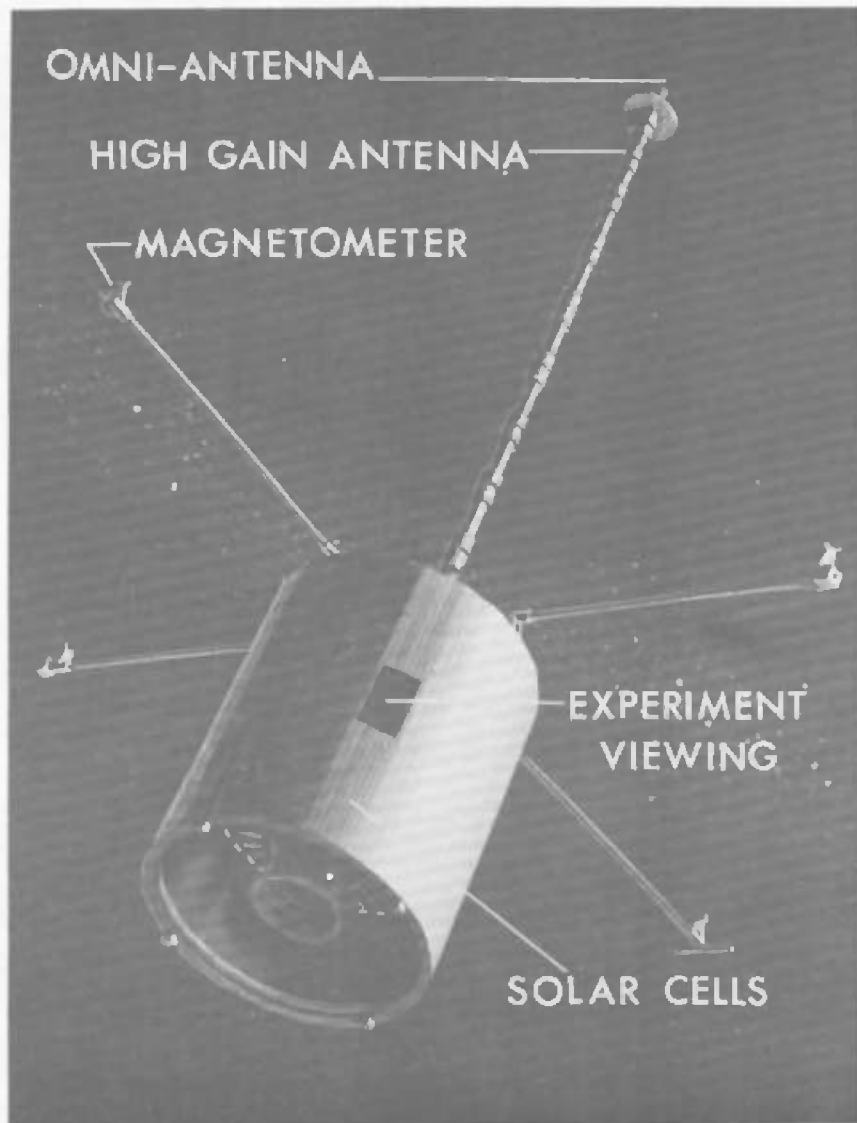
GROSS WEIGHT	64 kg
INSTRUMENT WT.	13. 6 kg
INVESTIGATIONS	PARTICLES & FIELDS
POWER	50 w
STABILIZATION	SPIN
DESIGN LIFE	6 MONTHS
LAUNCH VEHICLE	THRUST-AUGMENTED DELTA
MISSION	96×10^6 km FROM EARTH, 0. 8 AU TO 1. 2 AU FROM SUN
STATUS	DESIGN COMPLETED, FIRST LAUNCH LATE 1965
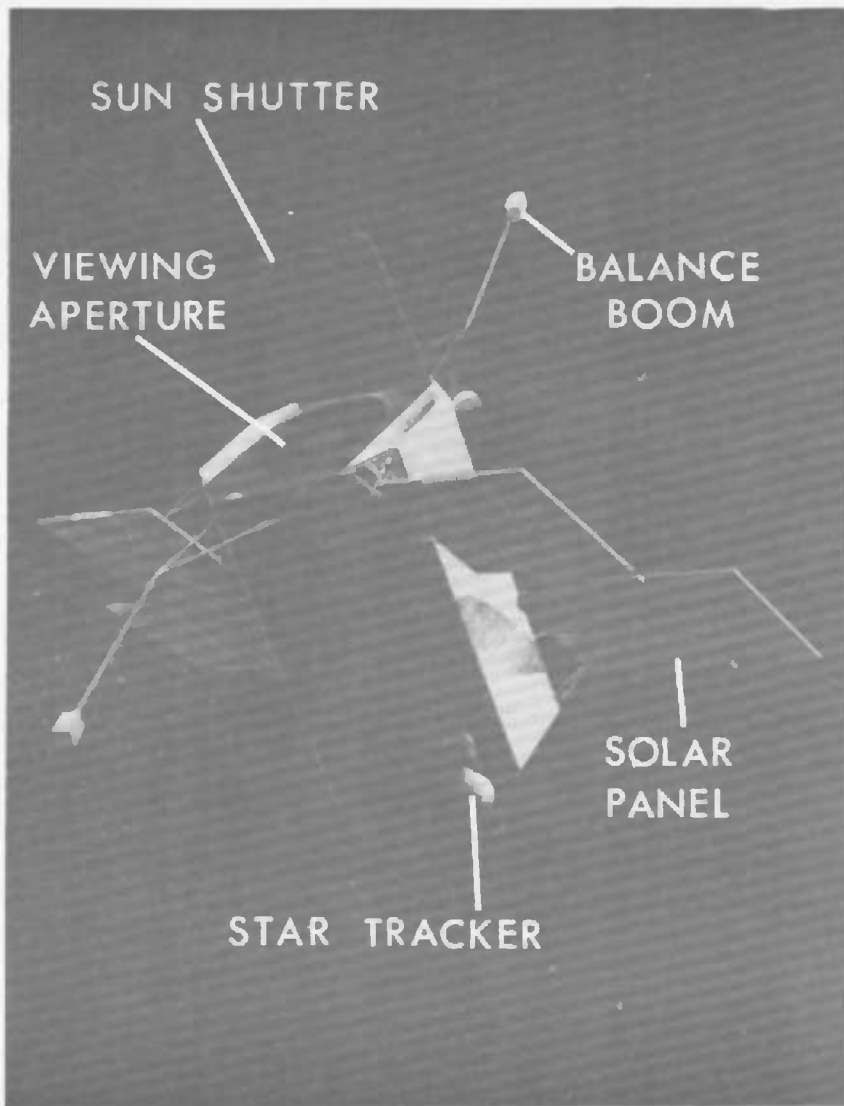
GROSS WEIGHT	1, 820 kg
INSTRUMENT WEIGHT	454 kg
INVESTIGATIONS	SEVERAL PER SPACECRAFT
STABILIZATION	ACTIVE, 3-AXIS
DESIGN LIFE	1 YR
LAUNCH VEHICLE	ATLAS-AGENA
ORBIT	CIRCULAR, 800 km, INCLINATION 35 deg
PLAN	FIRST FLIGHT 1966

Fig. 5 Typical Scientific Earth Satellite — ORBITING ASTRONOMICAL OBSERVATORY

Fig. 6 Lunar Soft Lander — SURVEYOR

SOLAR PANEL

HIGH GAIN ANTENNA

TV

THERMALLY CONTROLLED COMPARTMENT

TV CAMERA

SURFACE SAMPLER

FUEL TANK

OMNI ANTENNA

THERMALLY CONTROLLED COMPARTMENT

RADAR ANTENNA

VERNIER ENGINE

•DEMONSTRATE

SOFT LANDING TECHNOLOGY

•SURVEY

VARIOUS LANDING AREAS

•MEASURE

PHYSICAL & CHEMICAL PROPERTIES

1   Landing Zone of SURVEYOR
2   APOLLO Landing Area
3   Initial Lunar Photograph
4   Injection into Initial Elliptical Lunar Orbit
5   Second Set of Photographs of SURVEYOR
        Landing Site
6   Injection into Final Elliptical Lunar Orbit
7   Low Altitude Photographic Coverage of SURVEYOR
        and APOLLO Landing Area
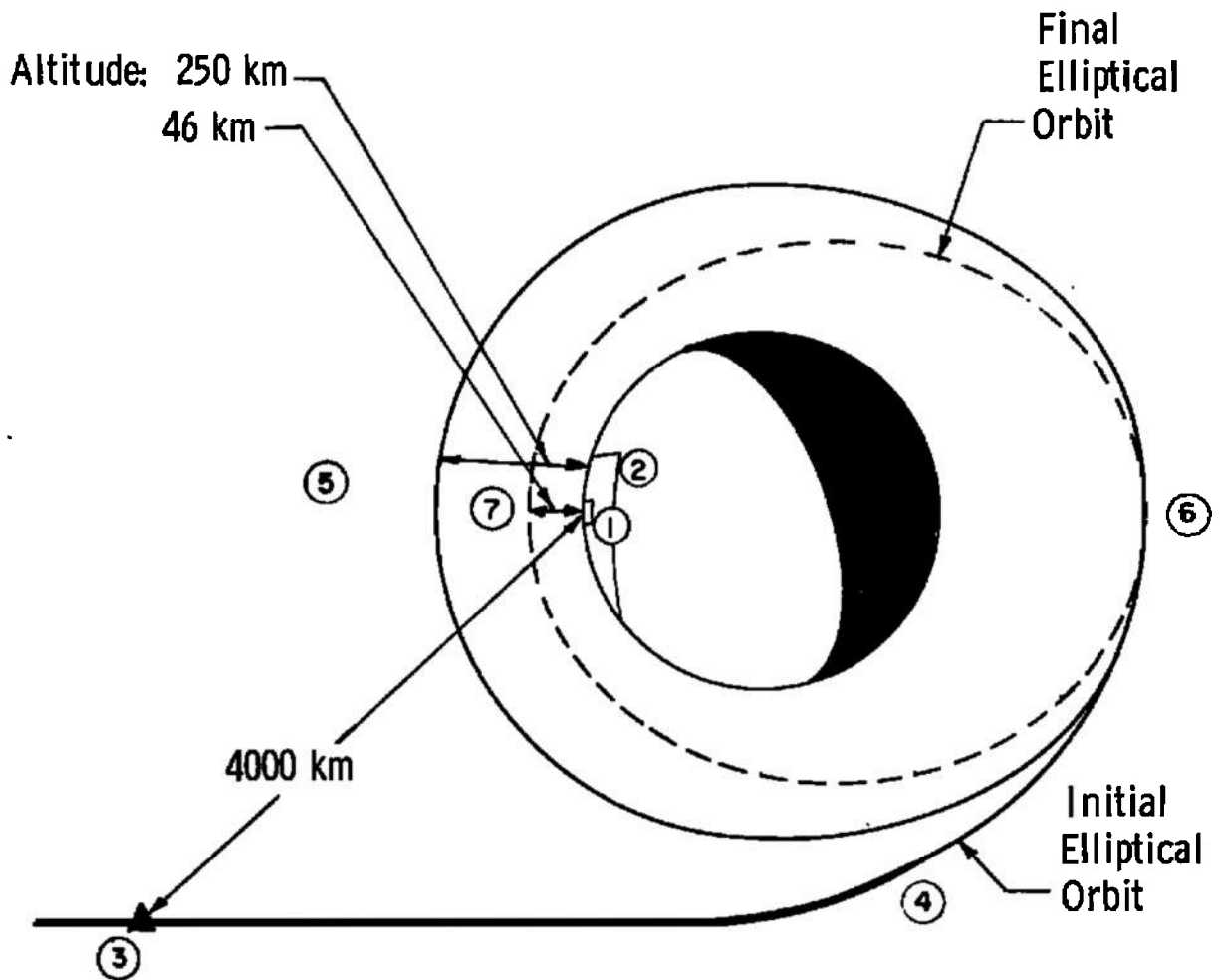
(Not To Scale)

Fig. 7 LUNAR ORBITER Photographic Mission
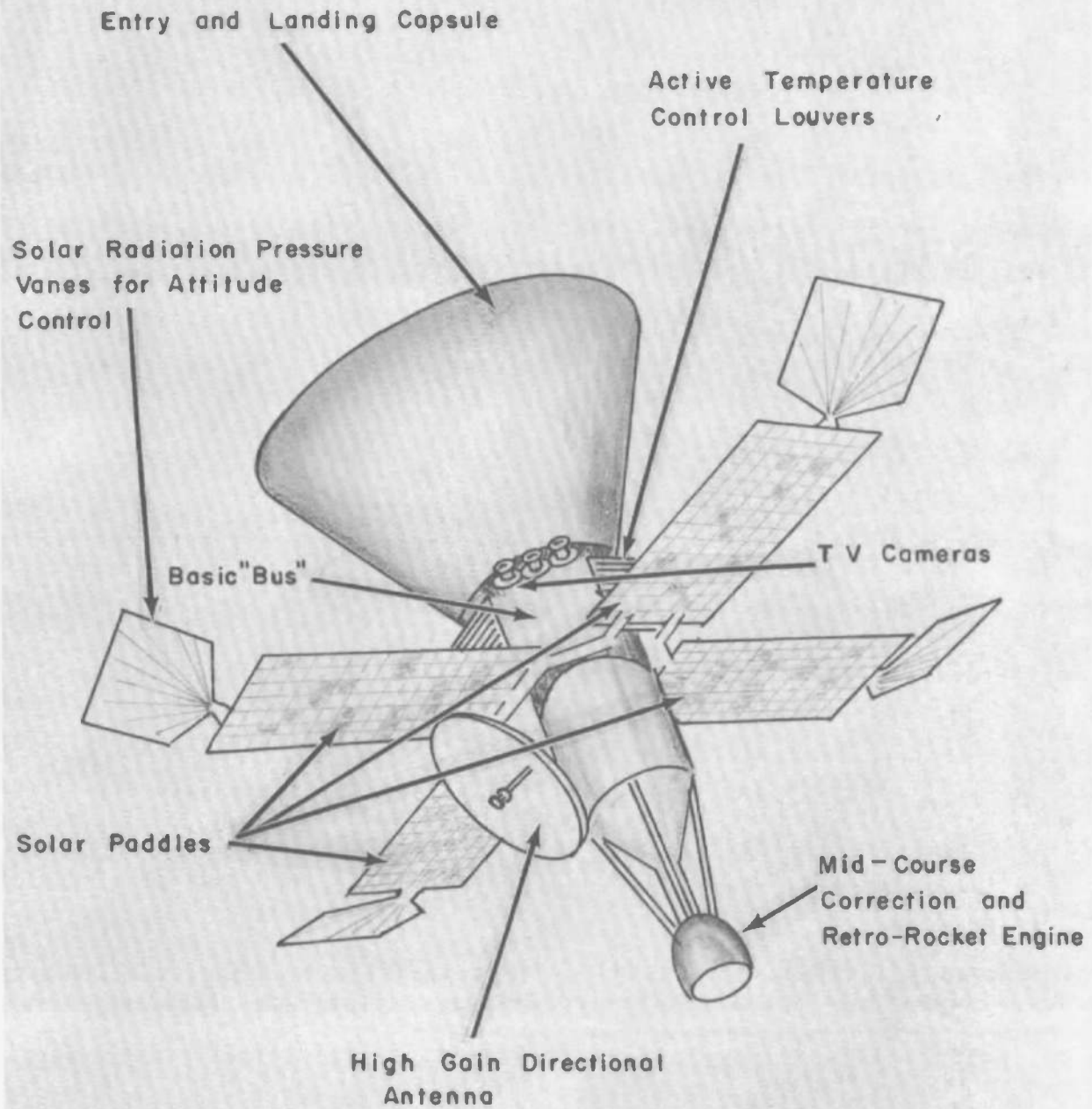
Fig. 8 VOYAGER Concept

Fig. 9 MERCURY Vehicle

ADAPTER
EQUIPMENT
SECTION

EJECTION SEAT

RENDEZVOUS AND
RECOVERY SECTION

UNITED
STATES

REACTION
CONTROL
SYSTEM
SECTION

CABIN

RETROGRADE
SECTION
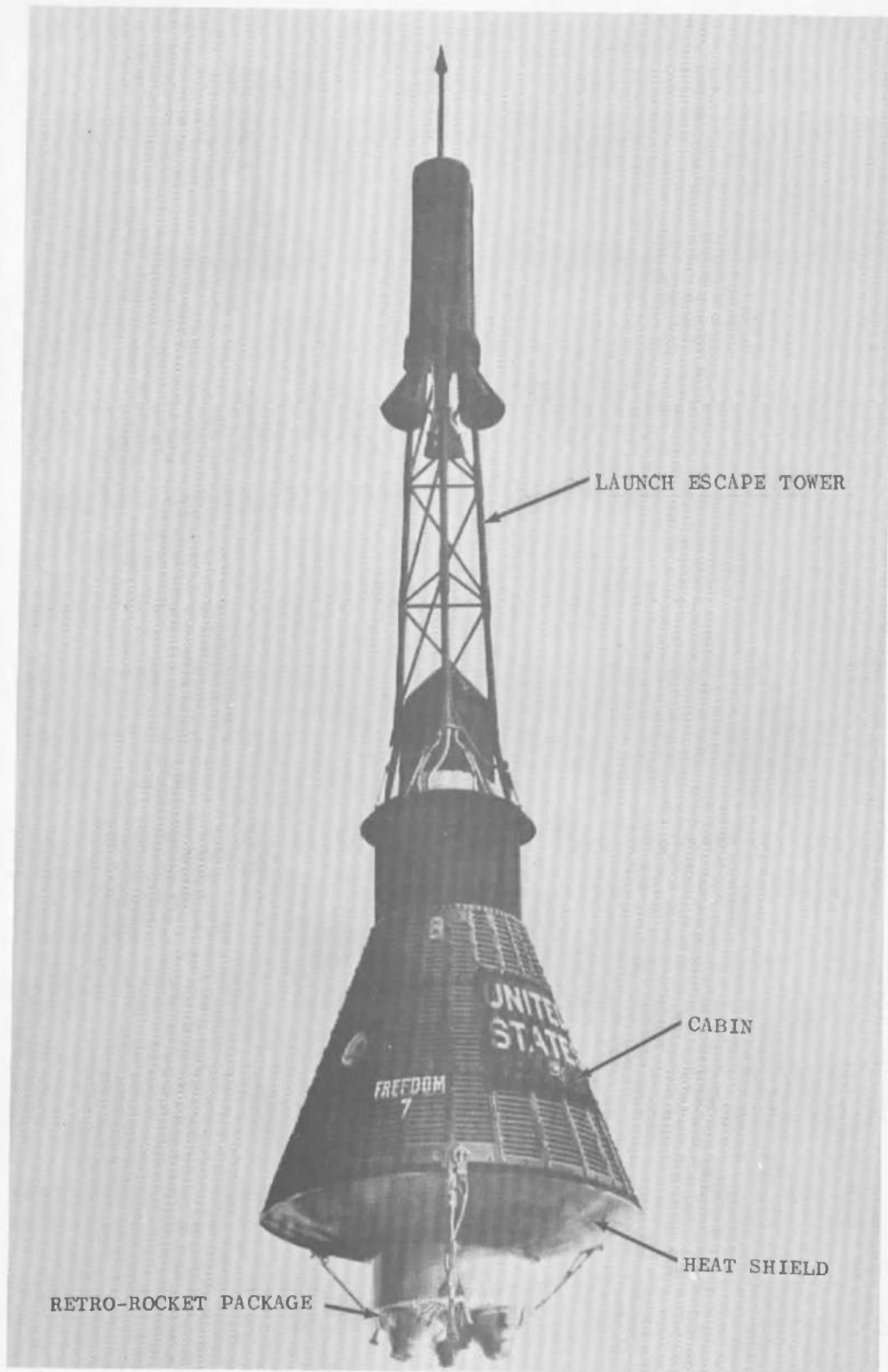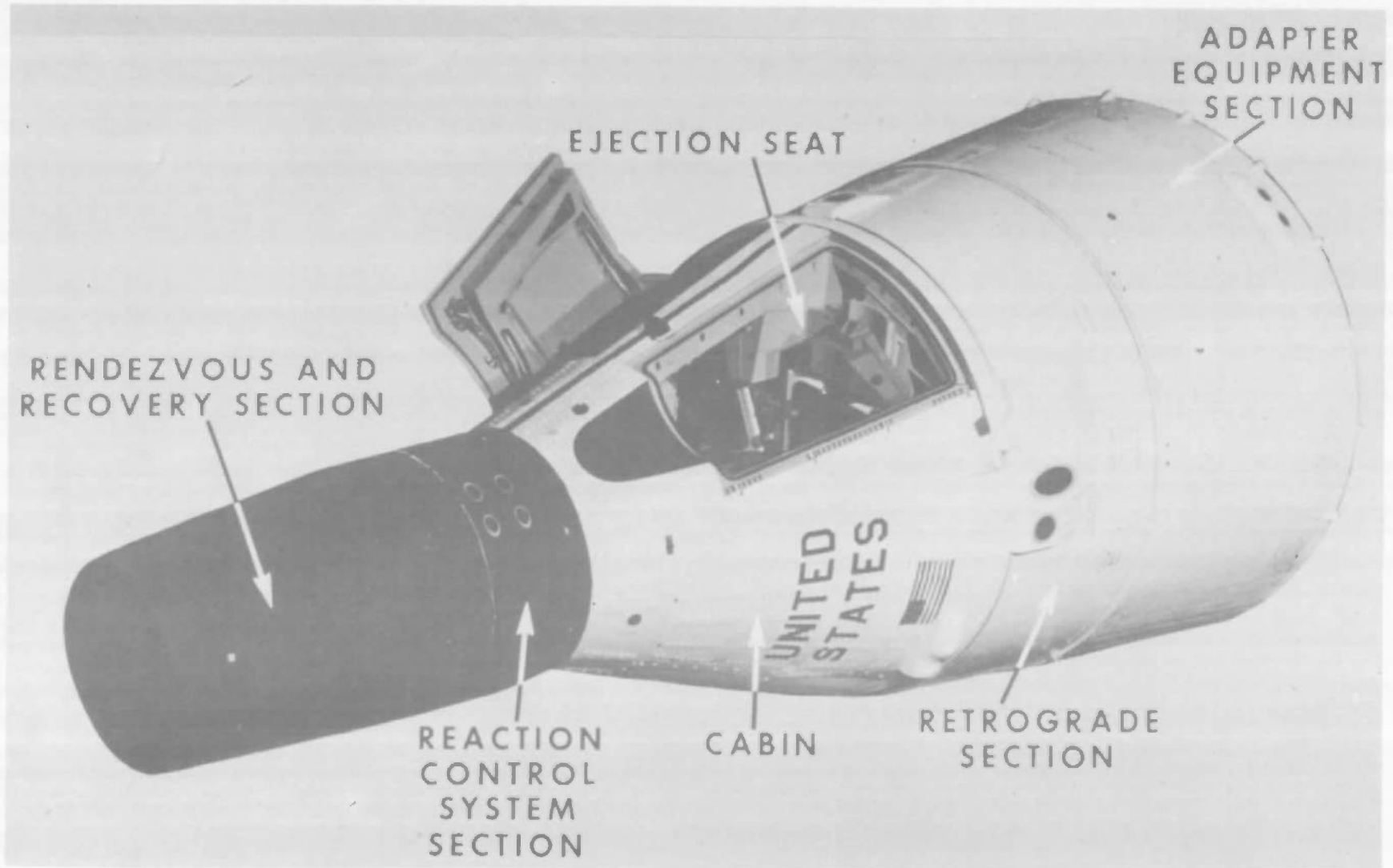
Fig. 10 GEMINI Vehicle
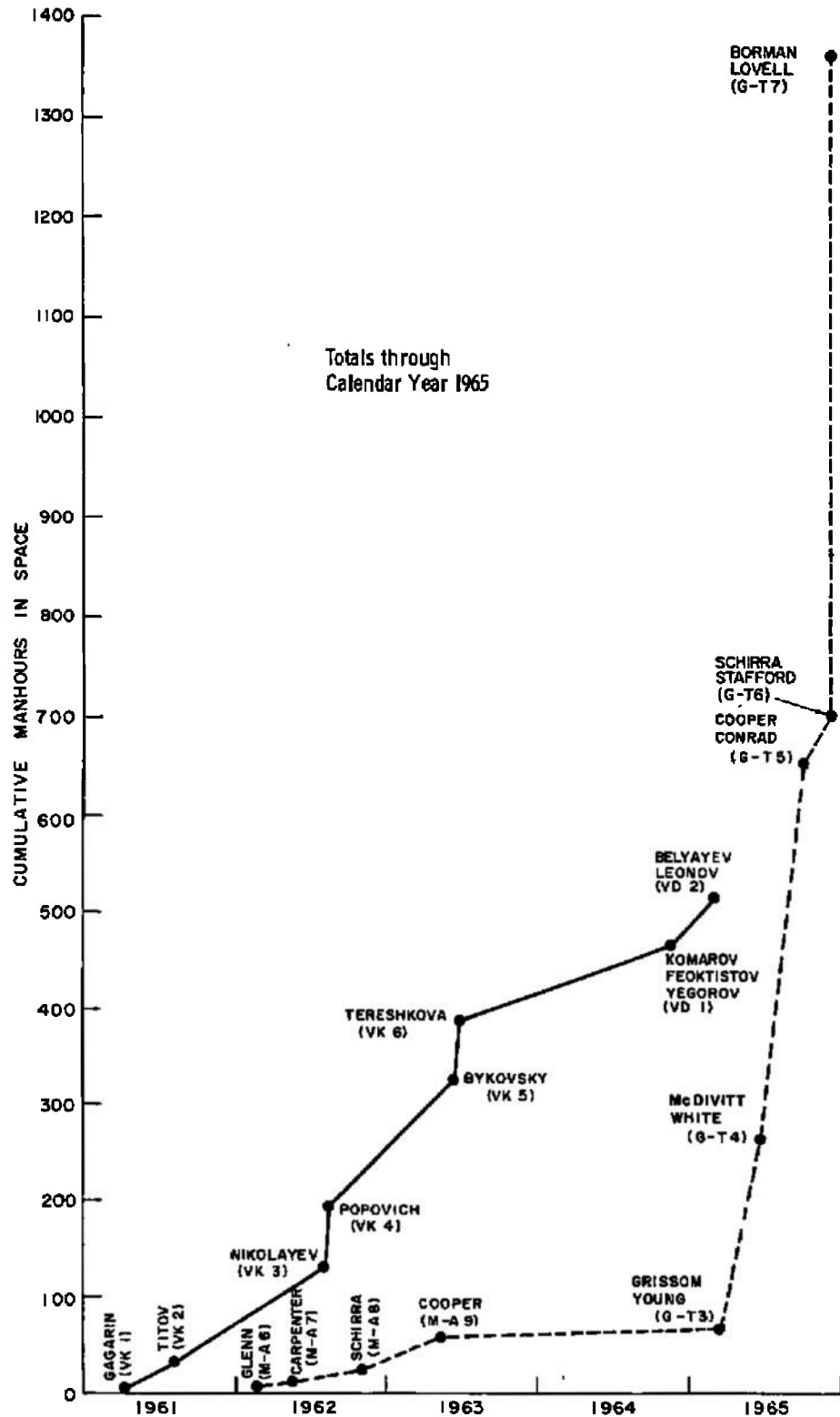
Fig. 11 Manned Space Flight Experience, US and USSR

LAUNCH ESCAPE SYSTEM

COMMAND MODULE

SERVICE MODULE

24.4m

LUNAR EXCURSION MODULE

TOTAL WEIGHT FUELED

41,000 kg
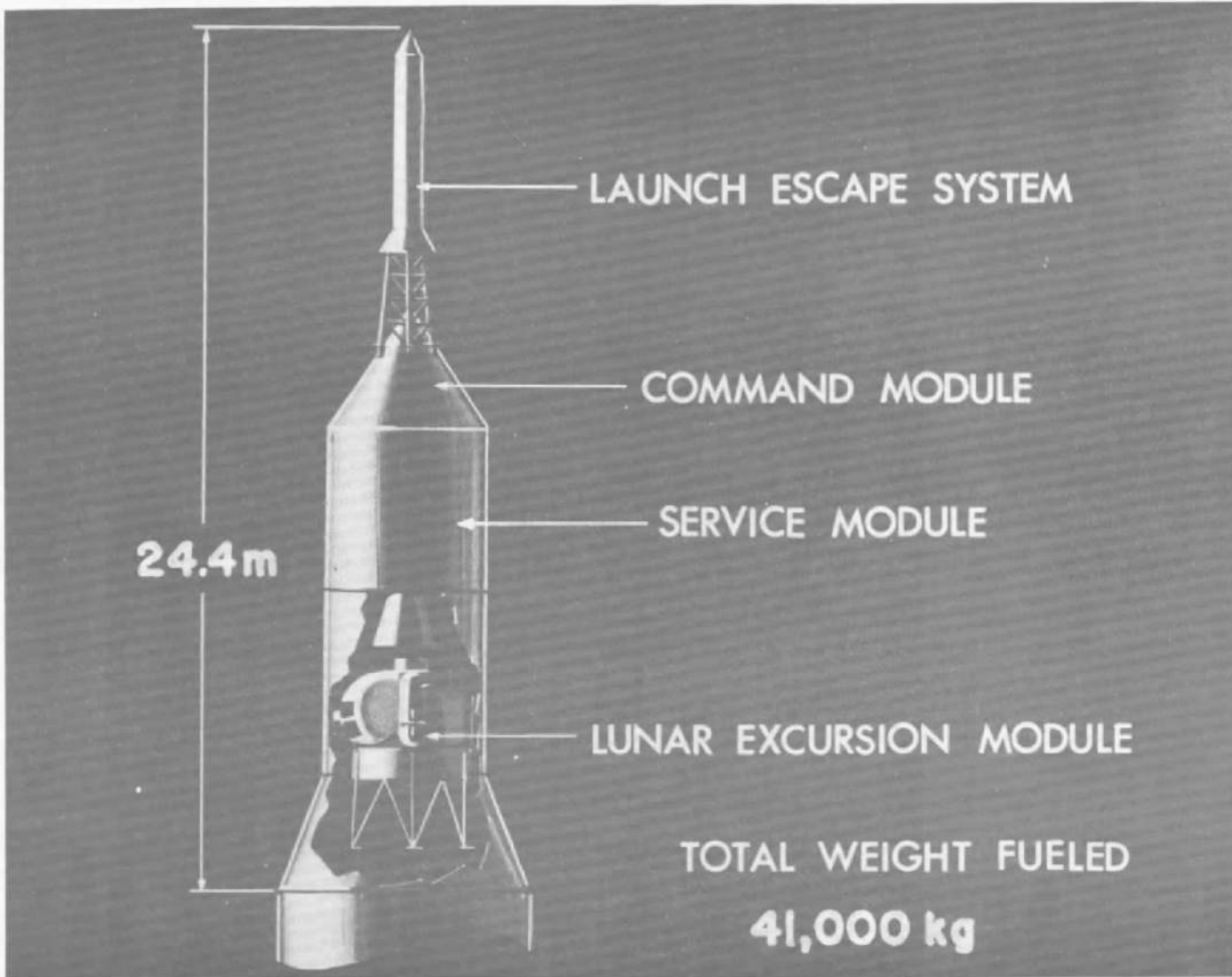
Fig. 12 APOLLO Vehicle
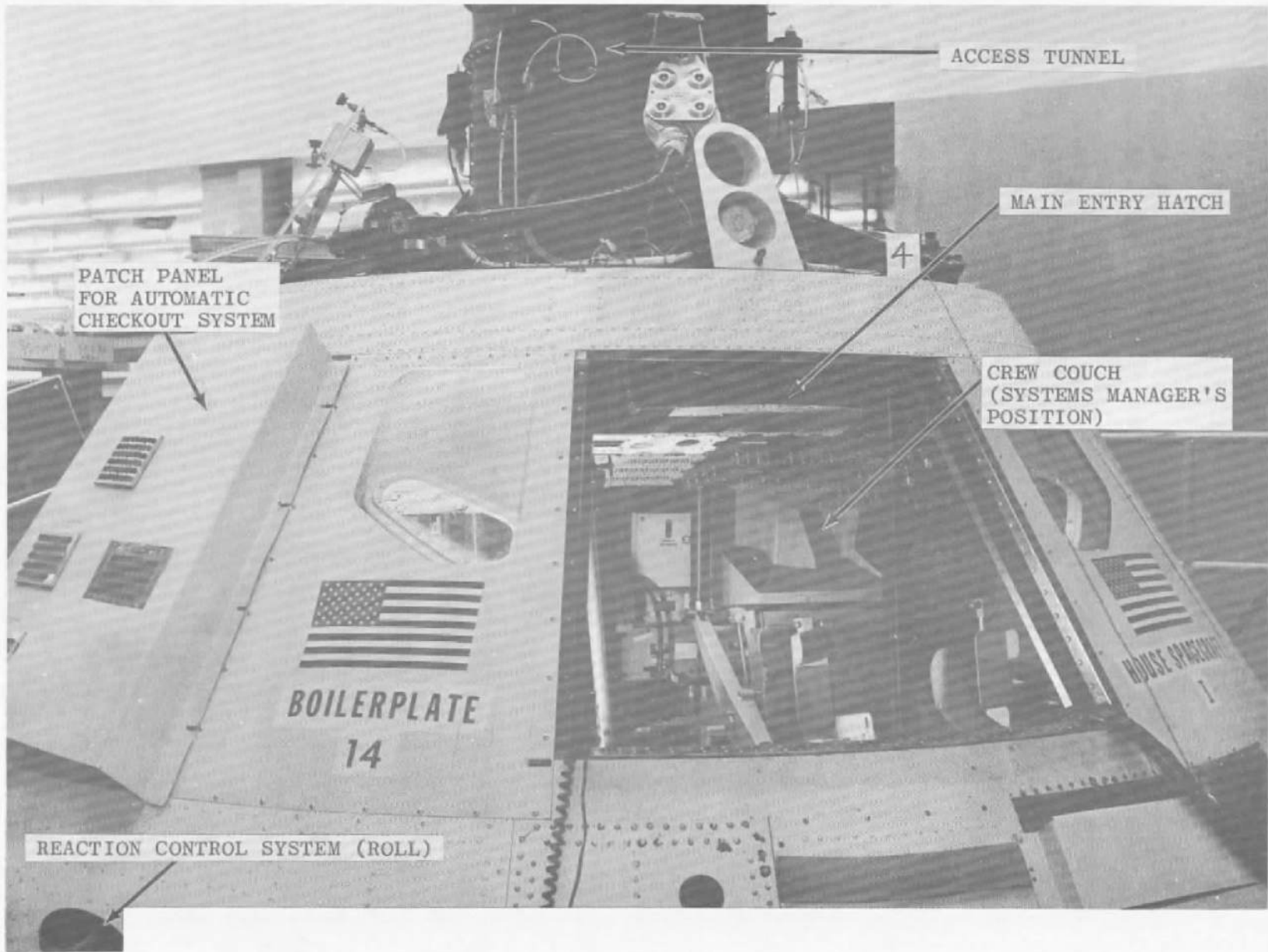
Fig. 13 Boilerplate APOLLO Command Module
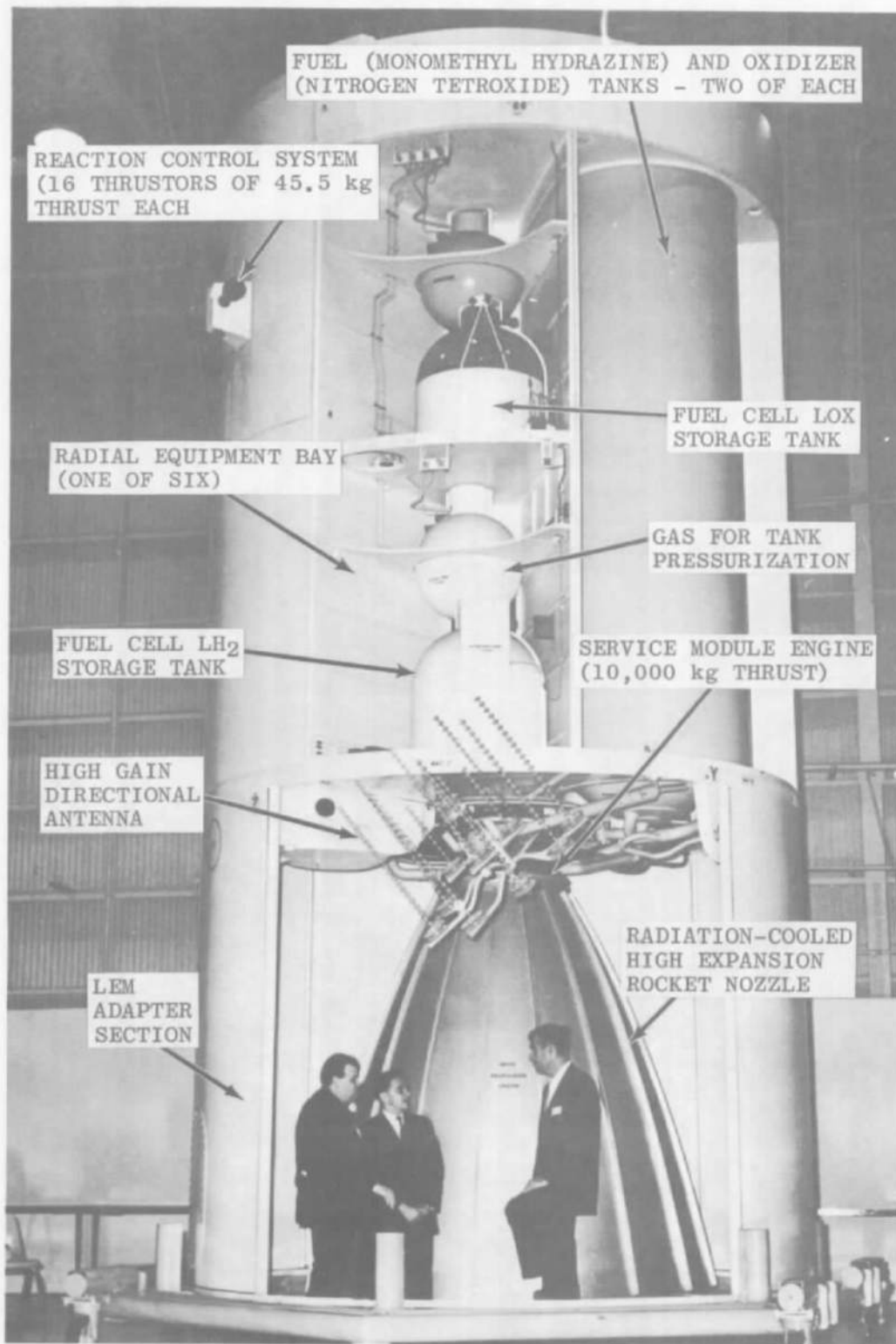
Fig. 14 Mockup of APOLLO Service Module
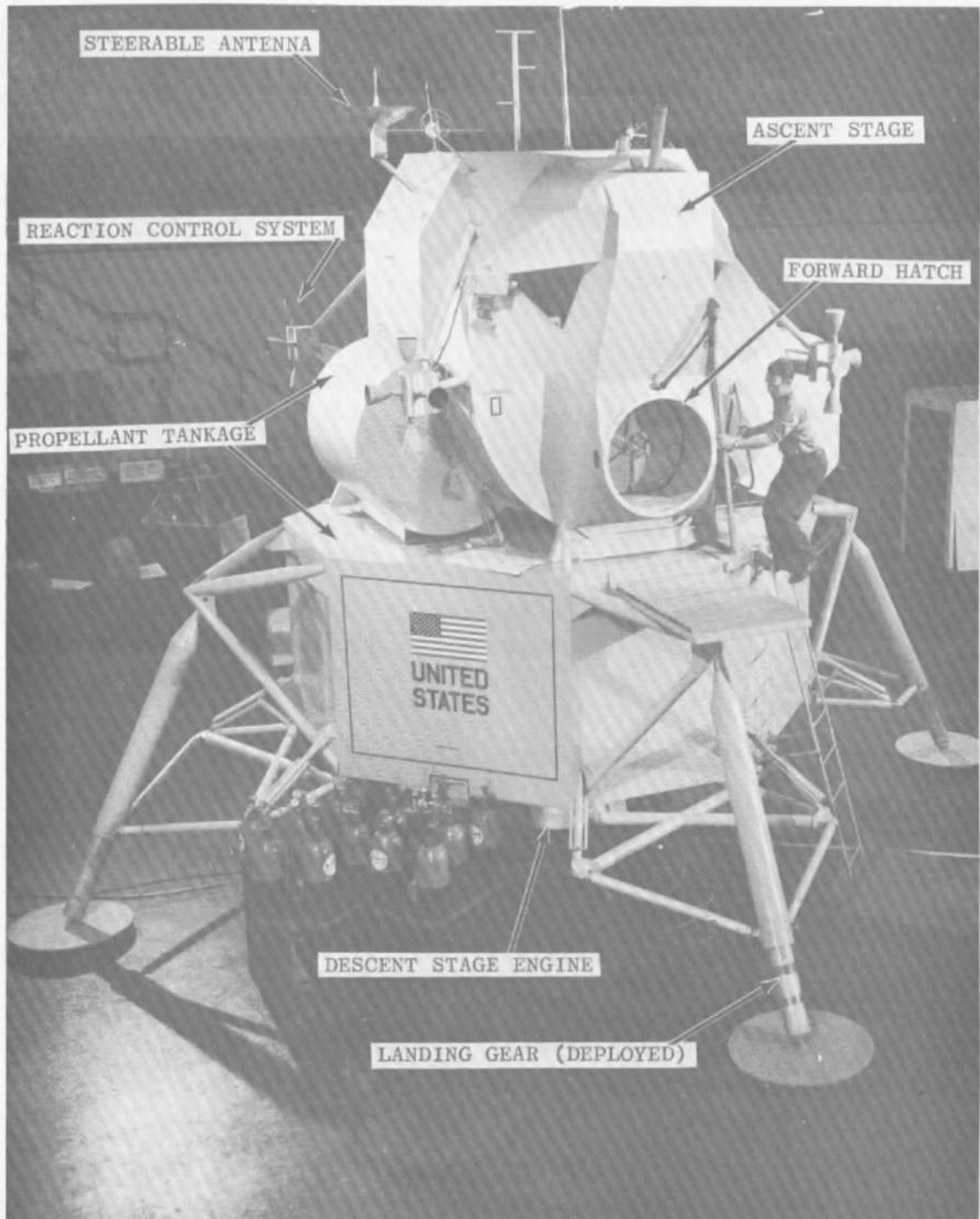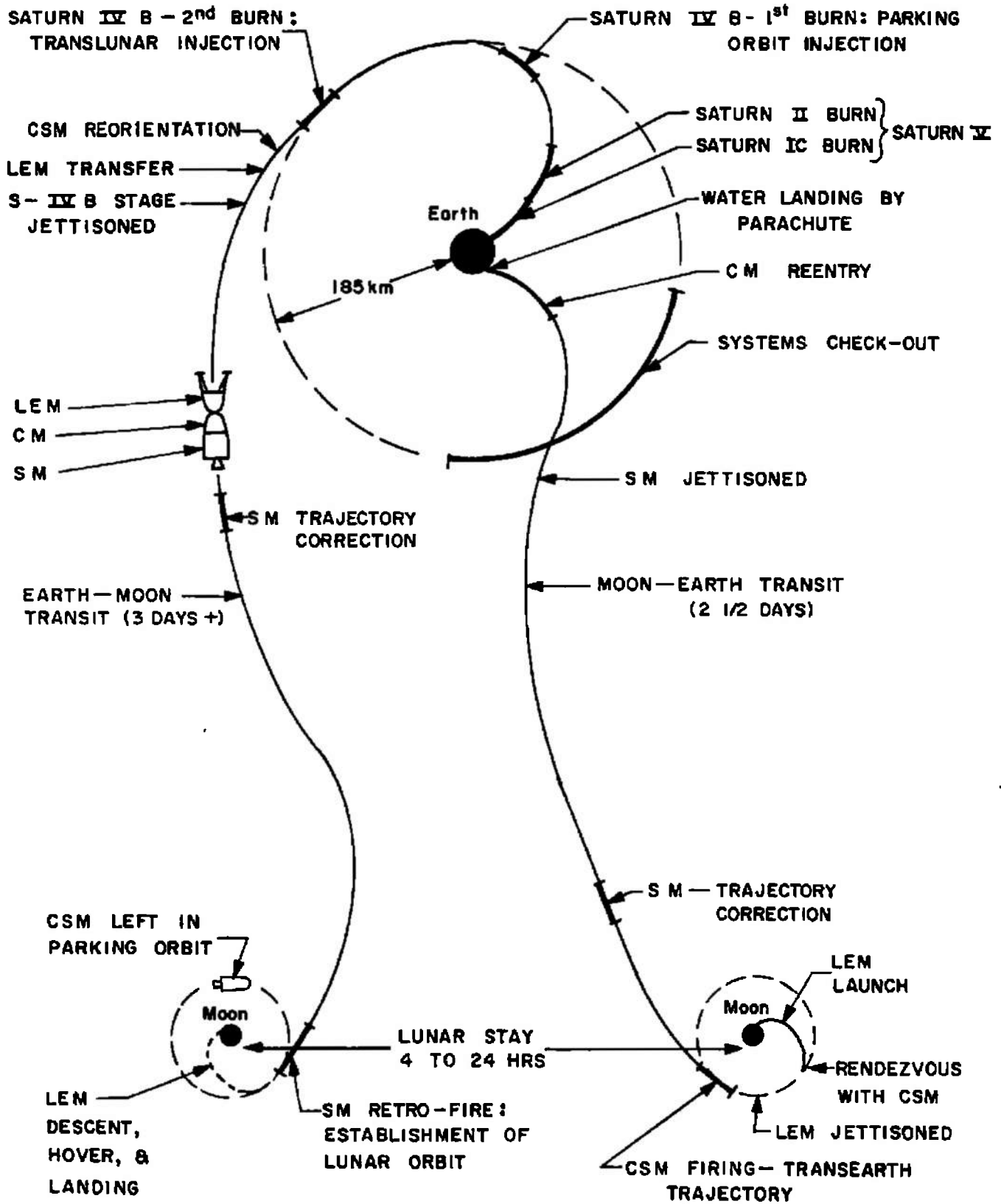
Fig. 15 Mockup of LUNAR EXCURSION MODULE

SATURN IV B — 2nd BURN :
TRANSLUNAR INJECTION

SATURN IV B - 1st BURN : PARKING
ORBIT INJECTION

CSM REORIENTATION

SATURN II BURN
SATURN IC BURN } SATURN V

LEM TRANSFER

S- IV B STAGE
JETTISONED

WATER LANDING BY
PARACHUTE

Earth

C M REENTRY

185 km

SYSTEMS CHECK-OUT

LEM
CM
SM

SM JETTISONED

S M TRAJECTORY
CORRECTION

MOON — EARTH TRANSIT
(2 1/2 DAYS)

EARTH — MOON
TRANSIT (3 DAYS +)

S M — TRAJECTORY
CORRECTION

CSM LEFT IN
PARKING ORBIT

LEM
LAUNCH

Moon

Moon

LUNAR STAY
4 TO 24 HRS

RENDEZVOUS
WITH CSM

LEM
DESCENT,
HOVER, &
LANDING

SM RETRO - FIRE :
ESTABLISHMENT OF
LUNAR ORBIT

LEM JETTISONED

CSM FIRING— TRANSEARTH
TRAJECTORY

Fig. 16 Highlights of APOLLO Lunar Mission

APOLLO COMMAND—SERVICE MODULE FERRY

DEPLOYED SOLAR PANELS
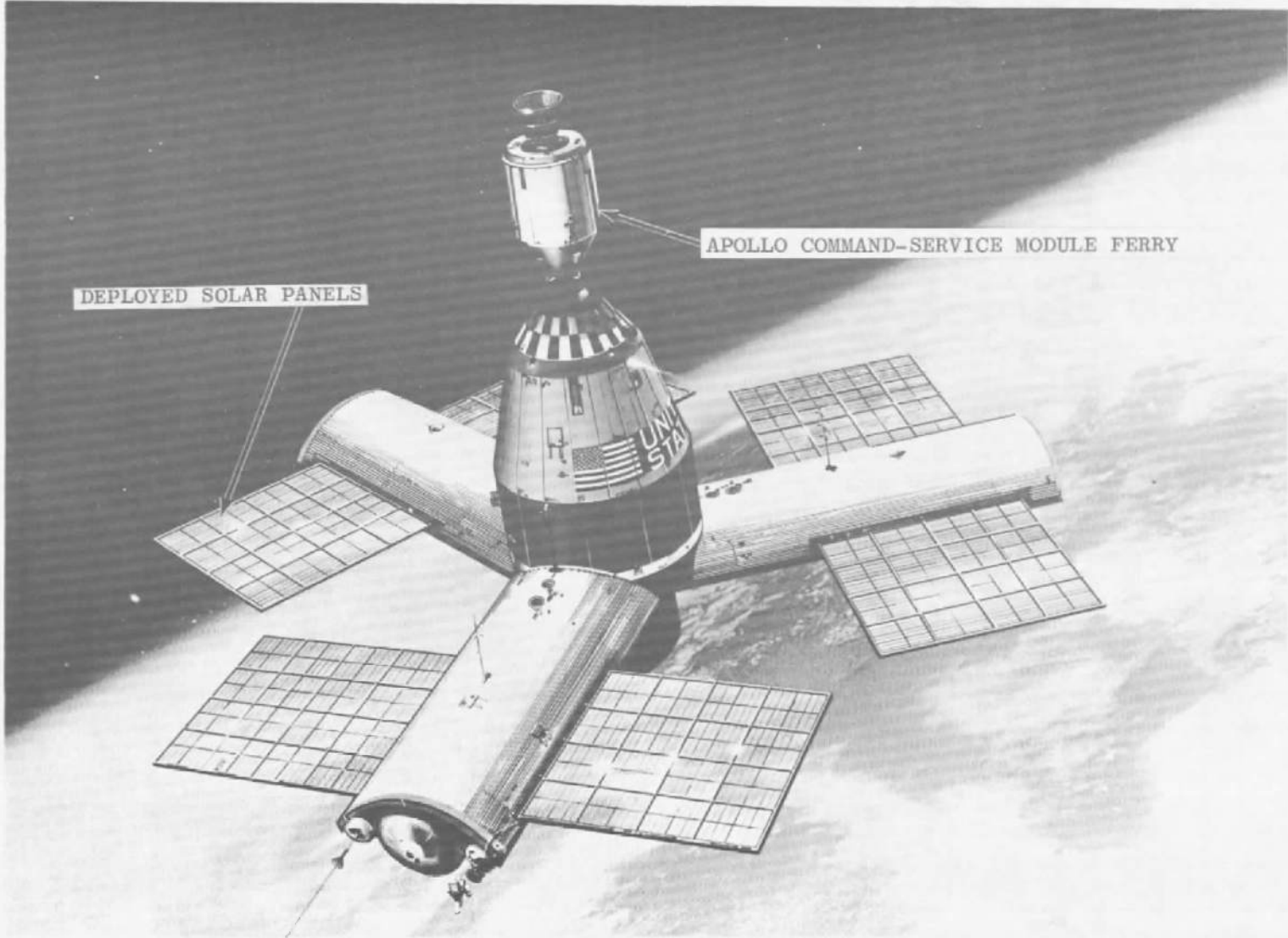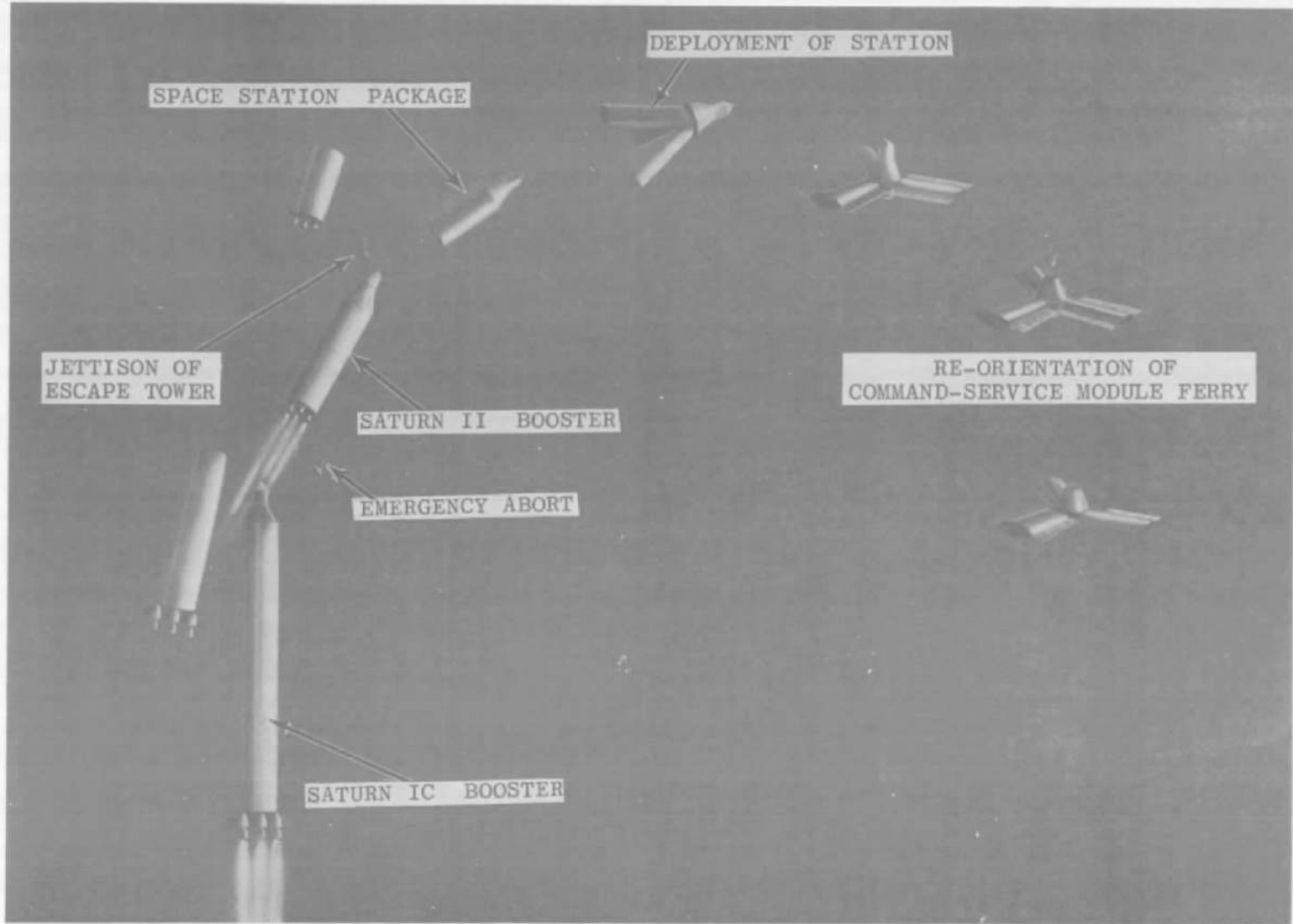
Fig. 17 An Artist's Sketch of an LORL

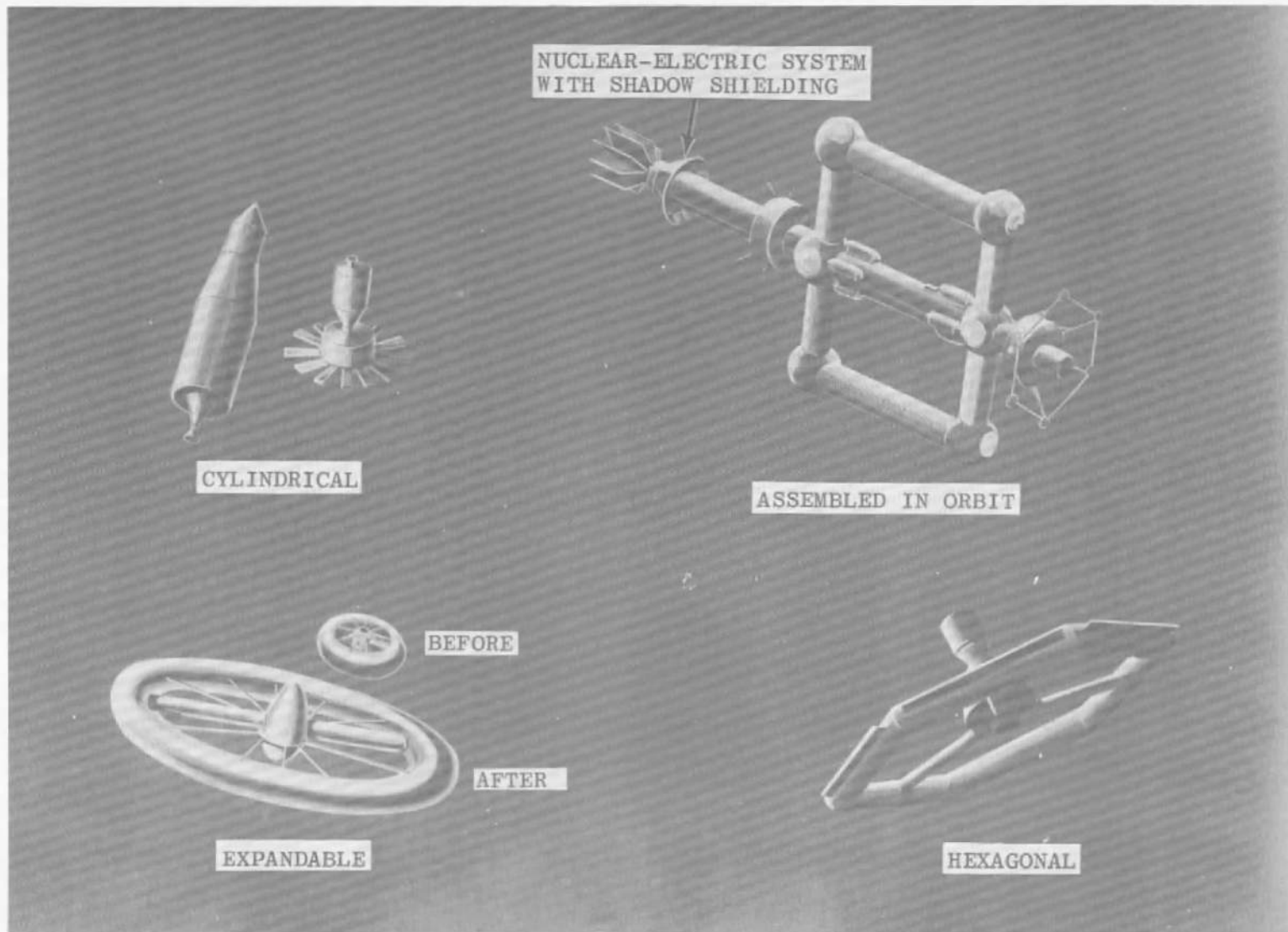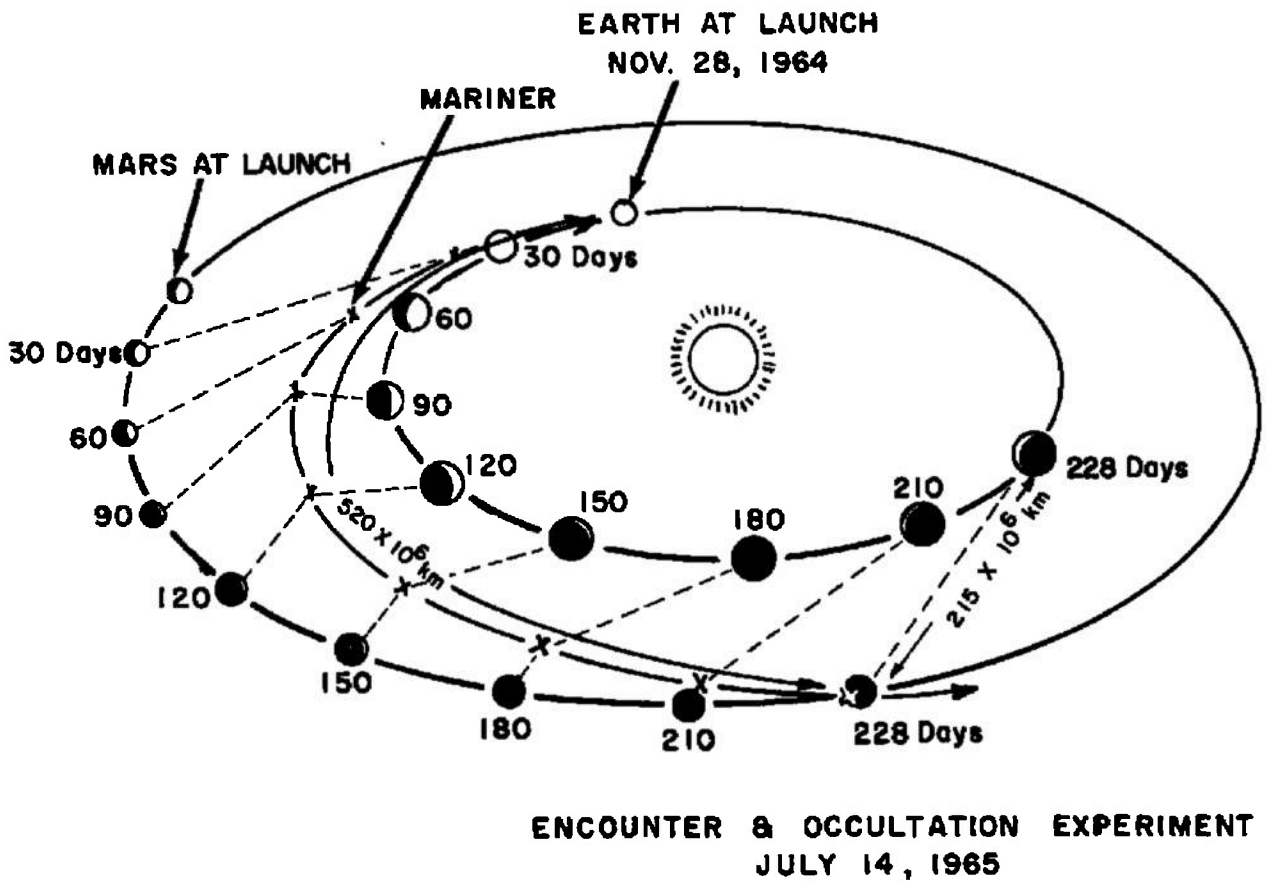Fig. 18 Launch and Deployment Sequence of an LORL-Type Space Station

Fig. 19 Other LORL Concepts

ANTENNA IN
OPERATIONAL POSITION

TRAILER MODULE

DOCKING ADAPTER

(RADIATOR)

CABIN MODULE

LIQUID
HYDROGEN

FUEL CELLS

LIQUID
OXYGEN

AIRLOCK

WHIP ANTENNA
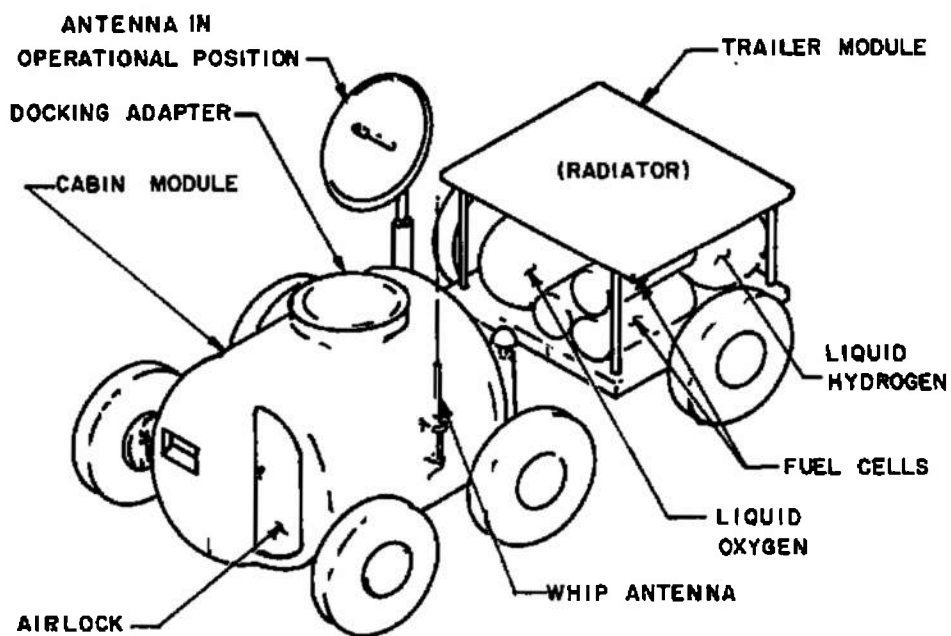
(From Ref. 22)

Fig. 20 Concept of a Manned Lunar Surface Vehicle

EARTH AT LAUNCH
NOV. 28, 1964

MARINER

MARS AT LAUNCH

30 Days

30 Days

60

90

120

150

180

210

228 Days

30 Days

60

90

120

150

180

210

228 Days

520 X 10⁶ km

215 X 10⁶ km

ENCOUNTER & OCCULTATION EXPERIMENT
JULY 14, 1965

(From Ref. 23)

Fig. 21 Trajectory of MARINER IV to Mars

The Figure Background is the Plane
NORMAL To the Ecliptic Plane, with
the Projections of the Included Planetary
Orbits Shown.
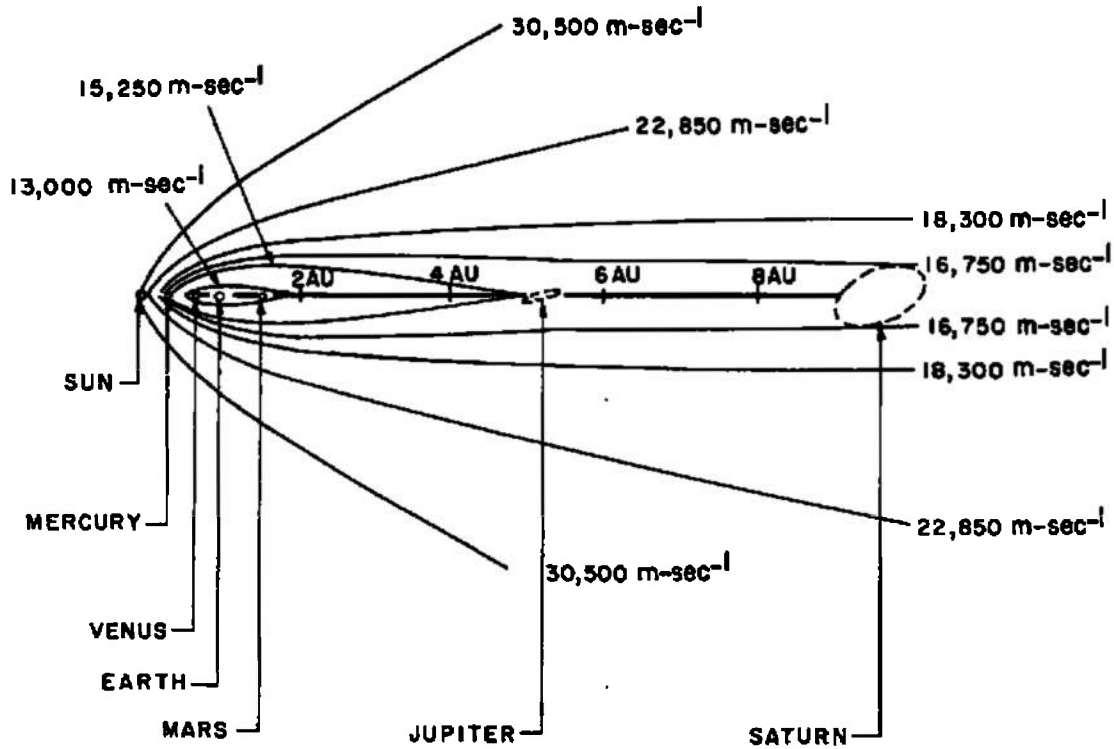
(Drawn To Scale)

(Adapted From Ref. 25)

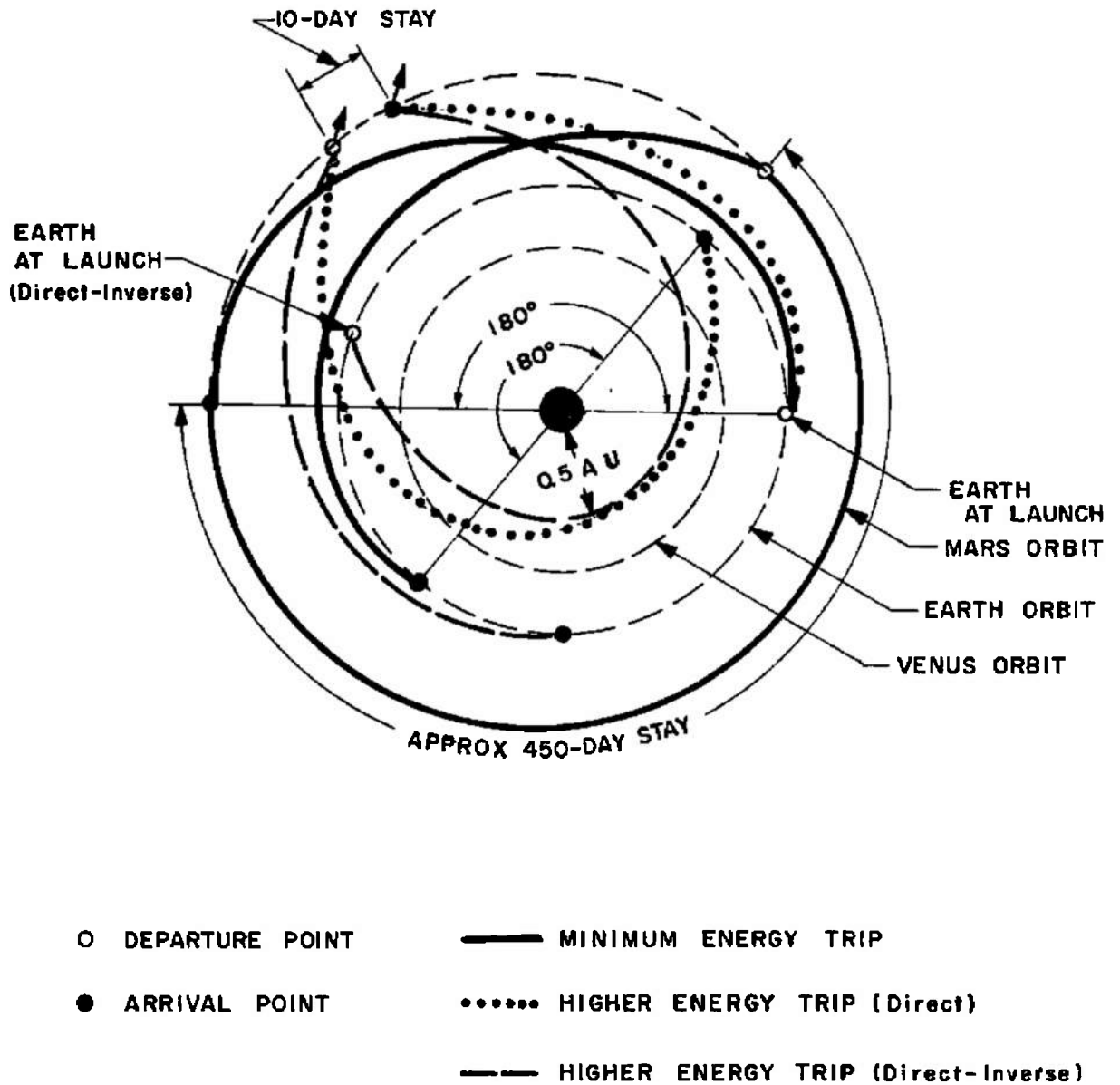Fig. 22 Ideal Velocity Contours of Minimum Energy Ballistic Trajectories

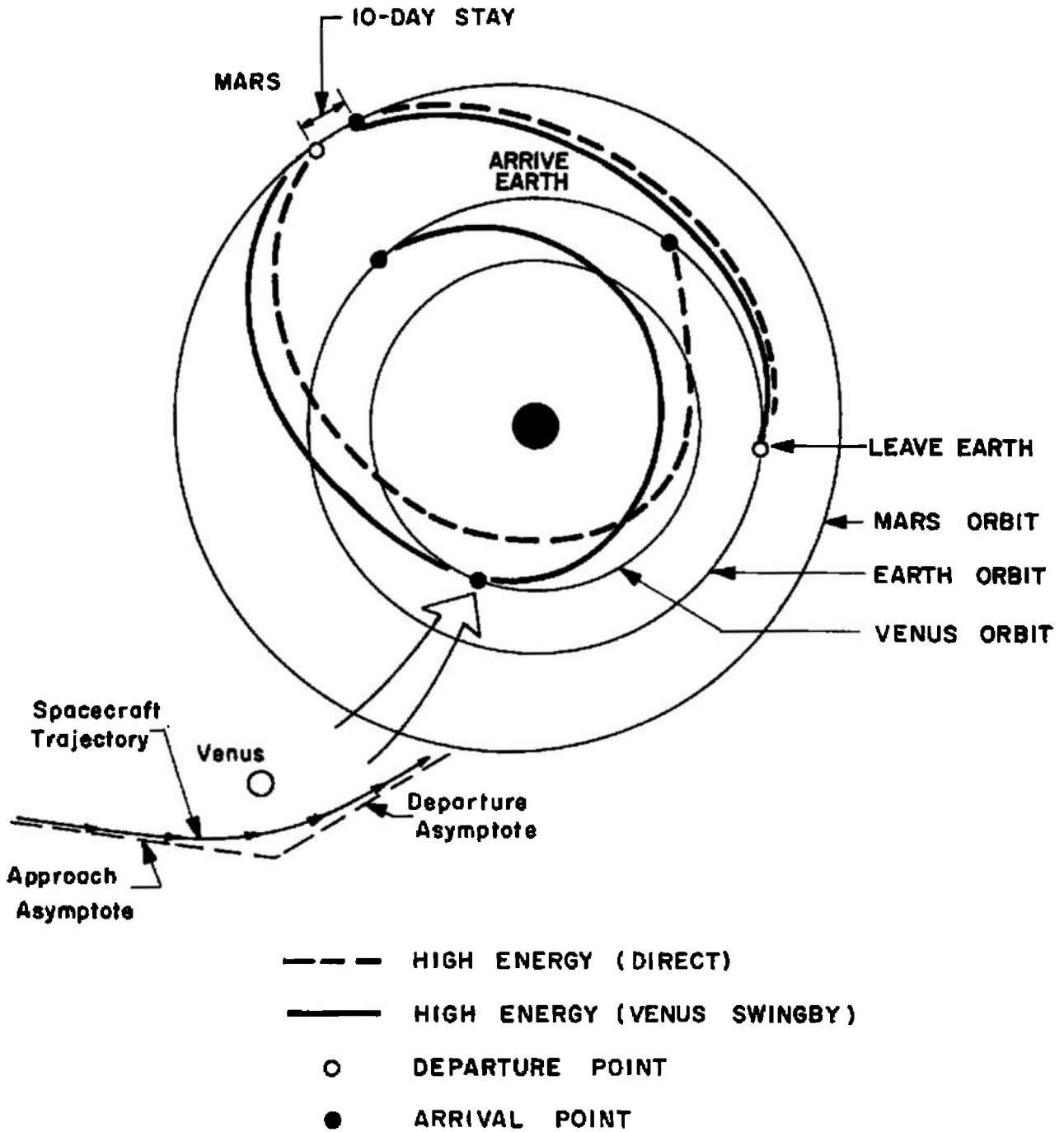Fig. 23 Comparison of Minimum Energy and Higher Energy Trajectories for Martian Visit
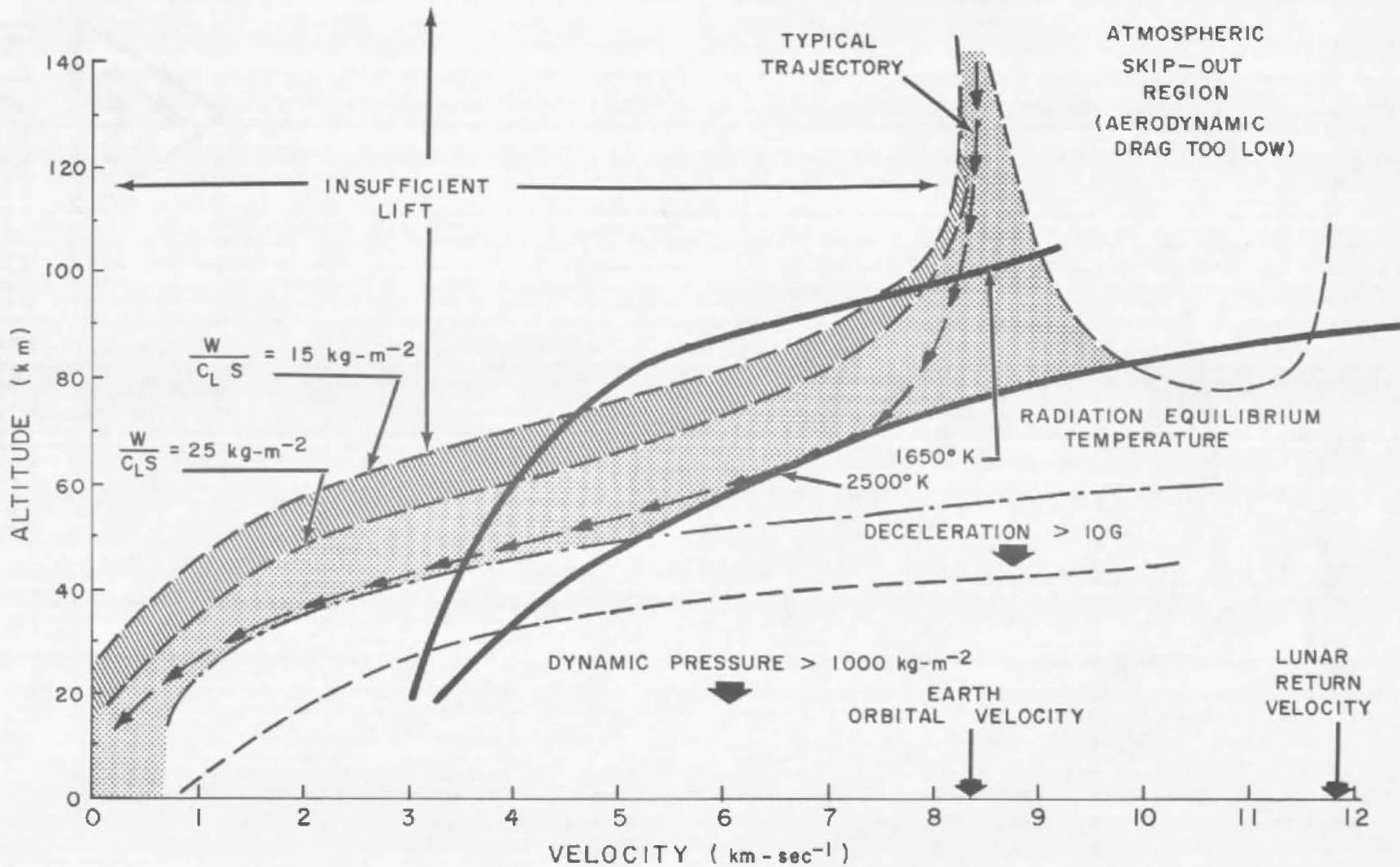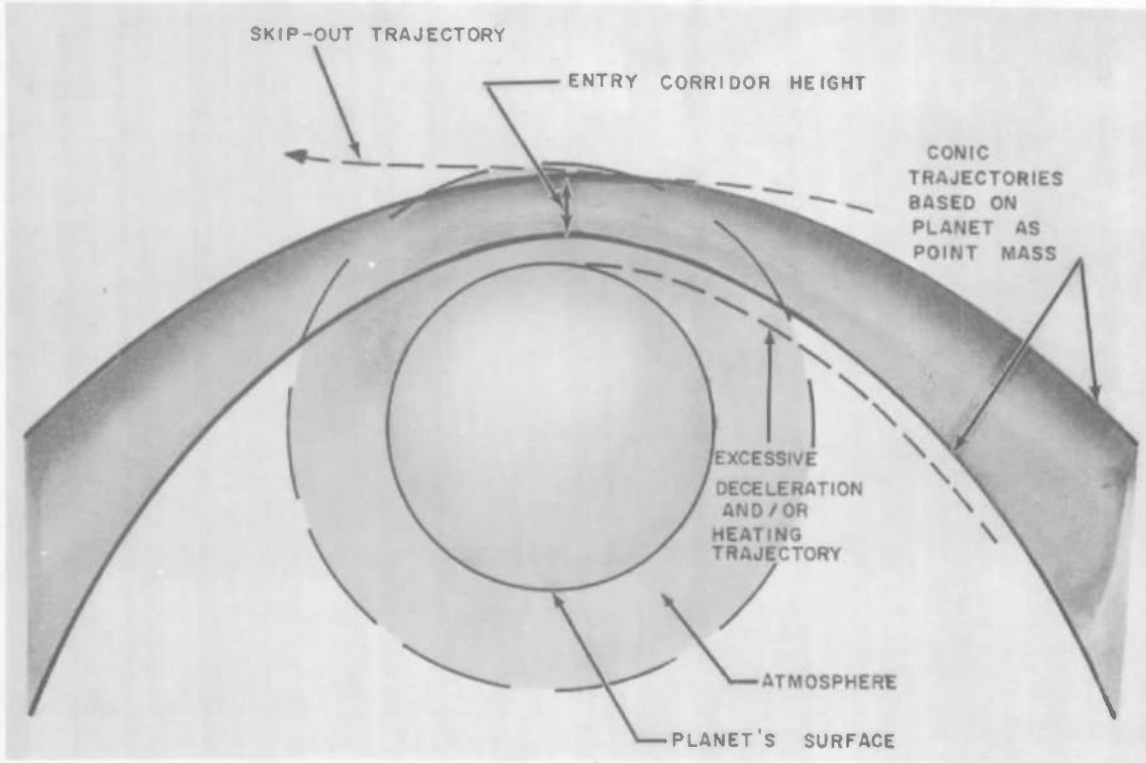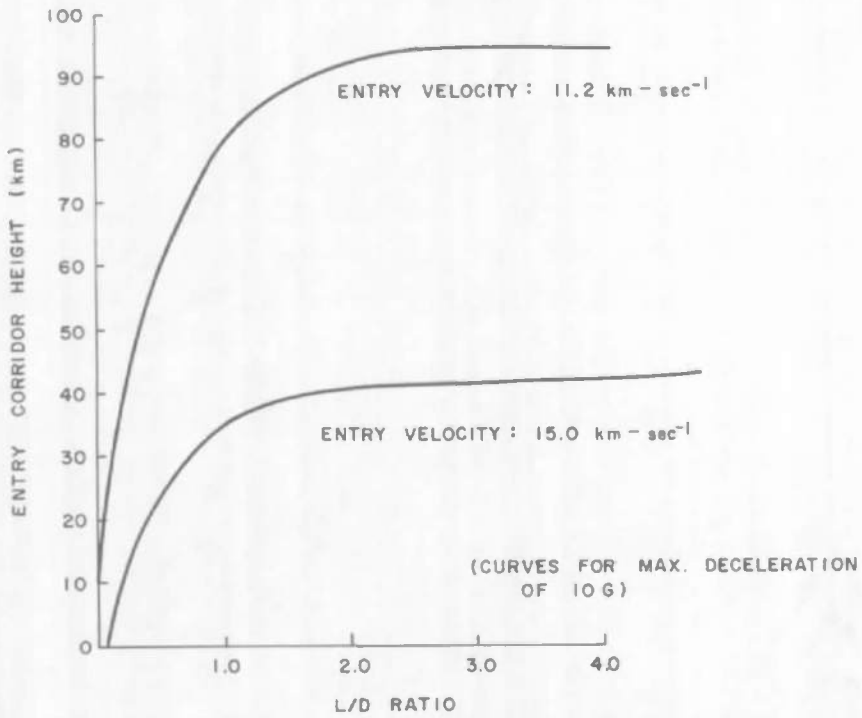
Fig. 24 Venus Swingby Return Mode

Fig. 25 Typical Continuous Flight Corridor with Constraints

a. Entry Corridor Limitations

b. Variation in Reentry Corridor Height with Lift/Drag Ratio and Entry Velocity

Fig. 26 The Reentry Corridor

FROM MATERIAL AND COMPONENT TESTS   TO SUBSYSTEMS TESTS   TO   INTEGRATED SYSTEMS TESTS

AND FINALLY, OPERATIONAL SUITABILITY   TESTS   TO   FLIGHT   TESTS

Fig. 27 Test Sequence in Vehicle Development Program

CHAMBER LOADING HATCH

OIL DIFFUSION PUMP

TEST SUBJECT

DOUBLE COMPARTMENTED MANLOCK

SOLAR SIMULATOR

STANDBY RESCUE MAN

REPRESSURIZATION SYSTEM
MANIFOLD PIPING

TEST FLOOR

ELEVATOR
(FOR ACCESS TO
SOLAR SIMULATOR MODULES)

PREPARATION ROOM AND BIOMEDICAL
MONITORING FACILITIES
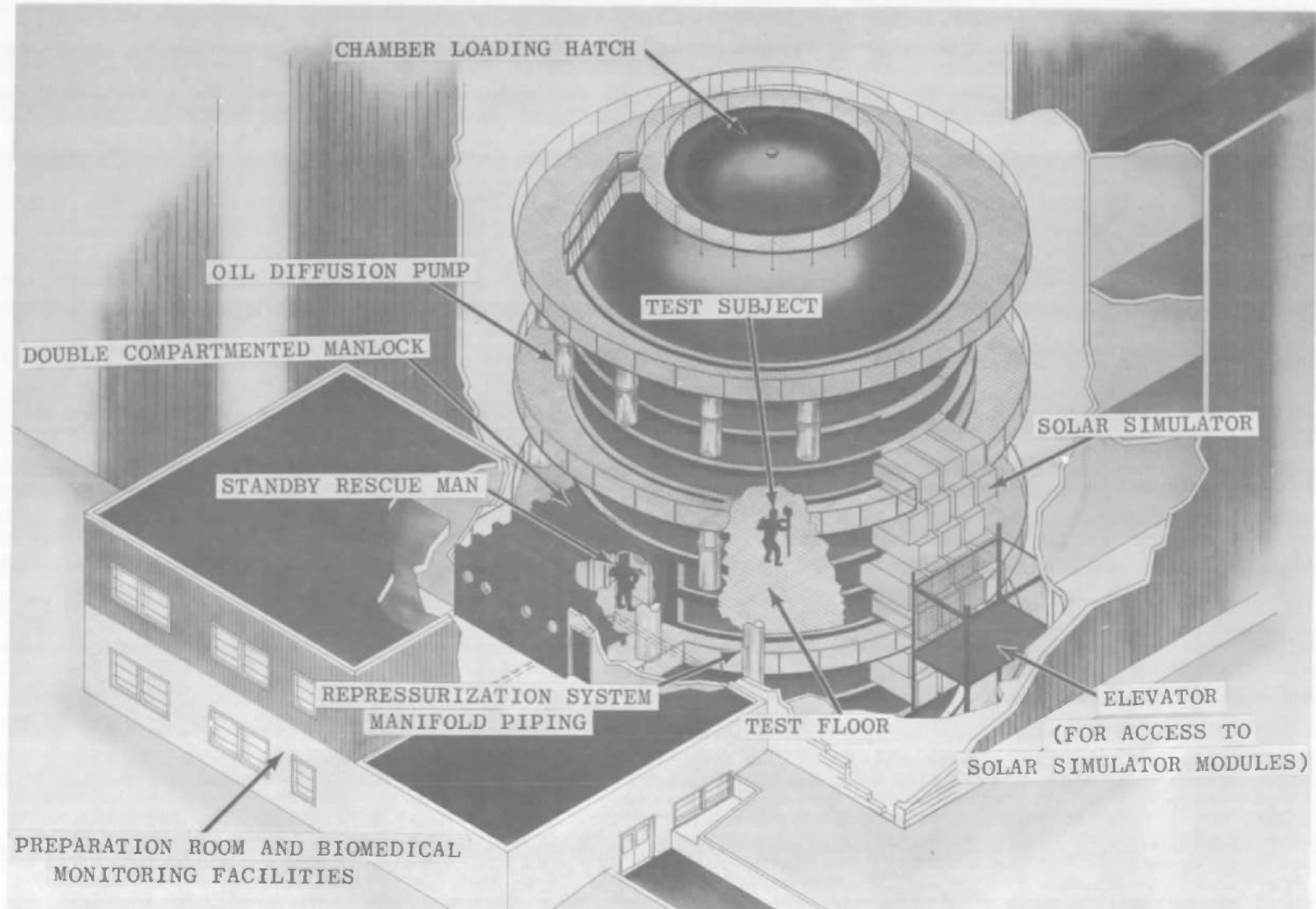
Fig. 28 Proposed Manrating Provisions in Mark I Aerospace Environmental Chamber
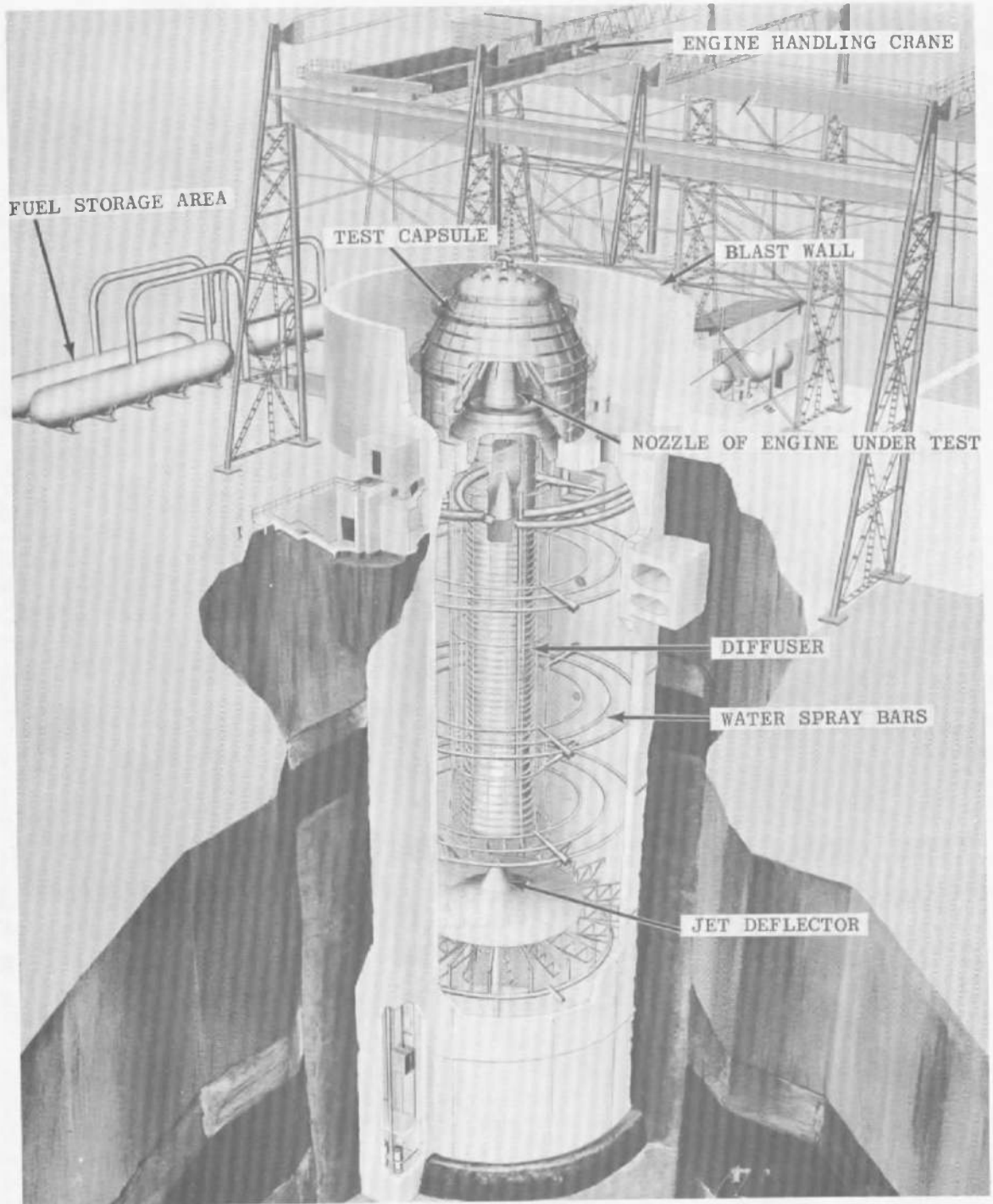
Fig. 29 High Altitude Rocket Test Cell (J-4)

FROM J-4
(UNDERGROUND)

ATMOSPHERIC STACK

TO EXHAUSTER
(UNDERGROUND)

ISOLATION
DOORS

EXHAUST GAS
COOLER

ACCESS HATCH
DOORS

COOLING WATER
SYSTEM

WEATHER
COVER

ROCKET MOTOR

MECHANICAL
SERVICE HUT
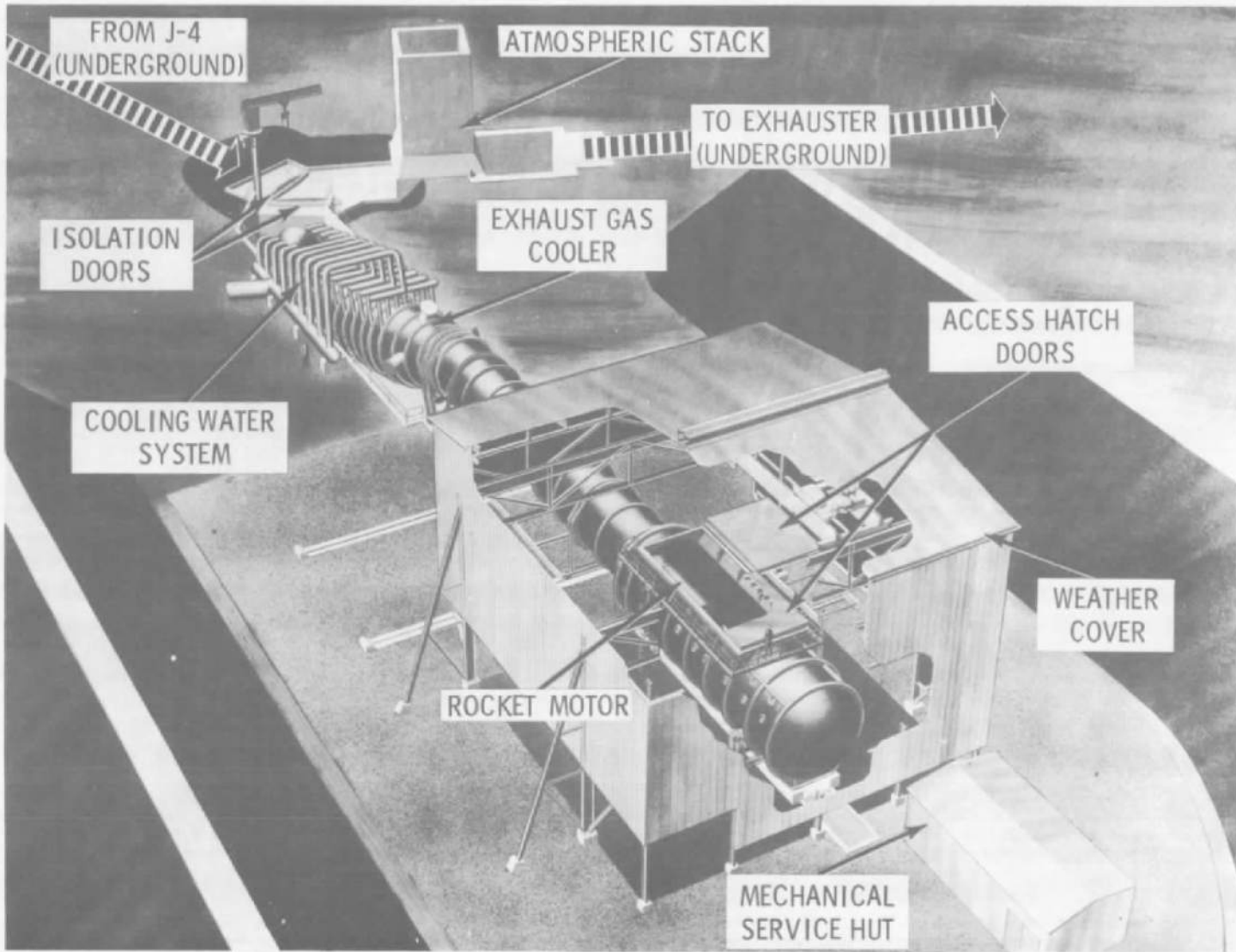
Fig. 30 High Altitude Solid Propellant Rocket Test Cell (J-5)

TO VACUUM PUMPS

TWO–STAGE LAUNCHER

26 METERS

BLAST CHAMBER

VELOCITY MEASURING STATIONS

TARGET CHAMBER

RECOVERY BED
(BEHIND ACCESS DOOR)

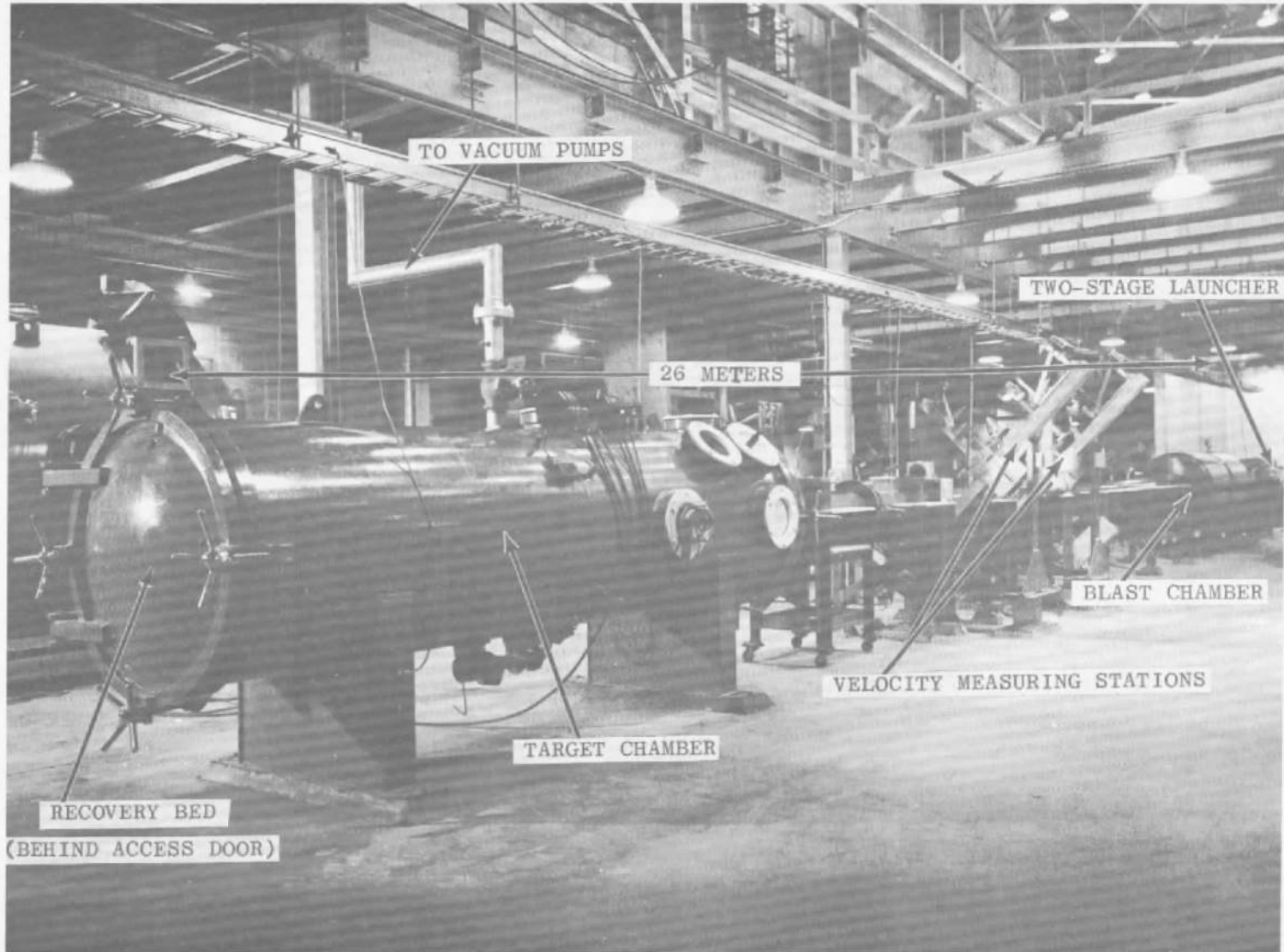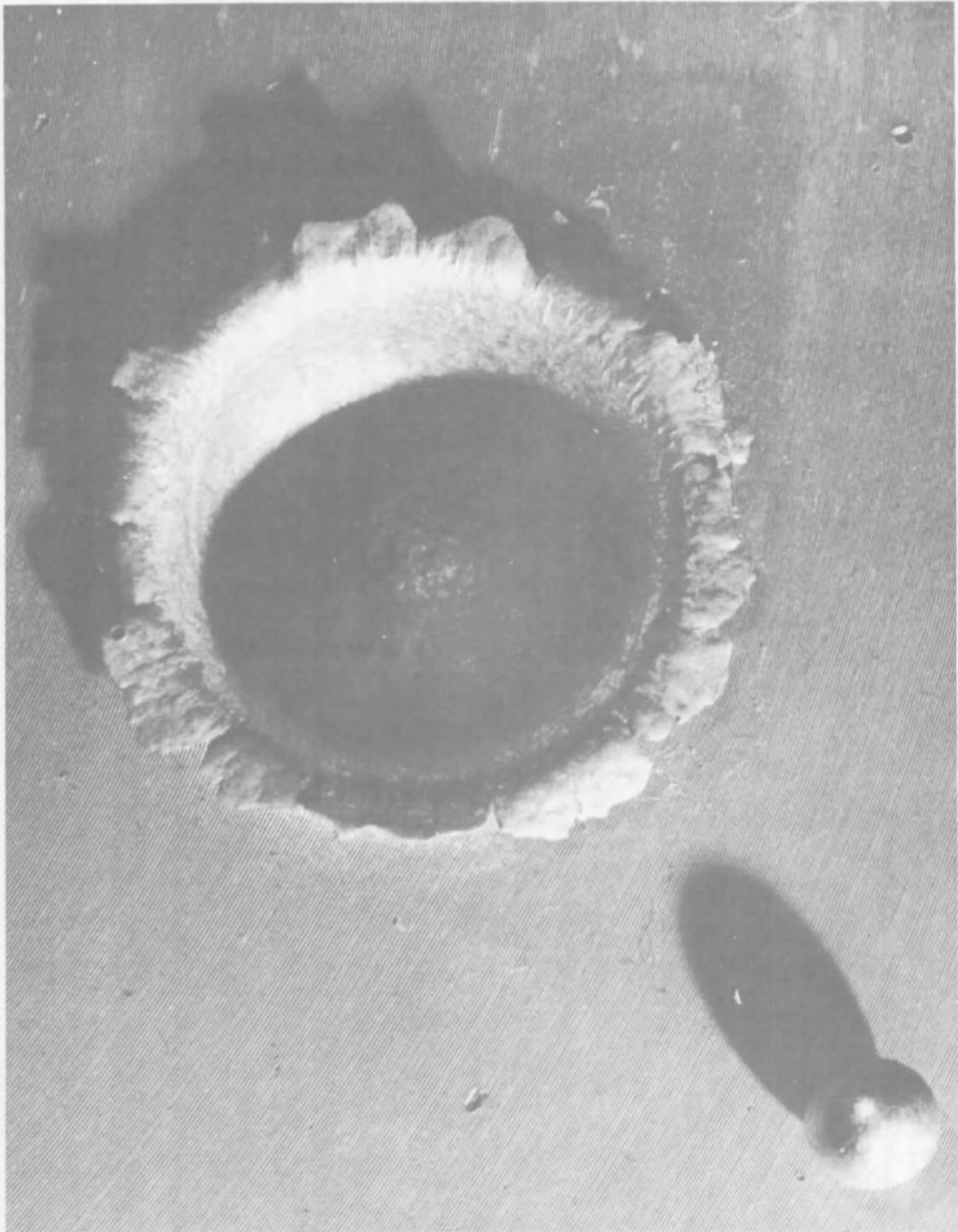Fig. 31 S-1 Hypervelocity Impact Range at Arnold Engineering Development Center

Fig. 32 Crater Produced by Hypervelocity Impact
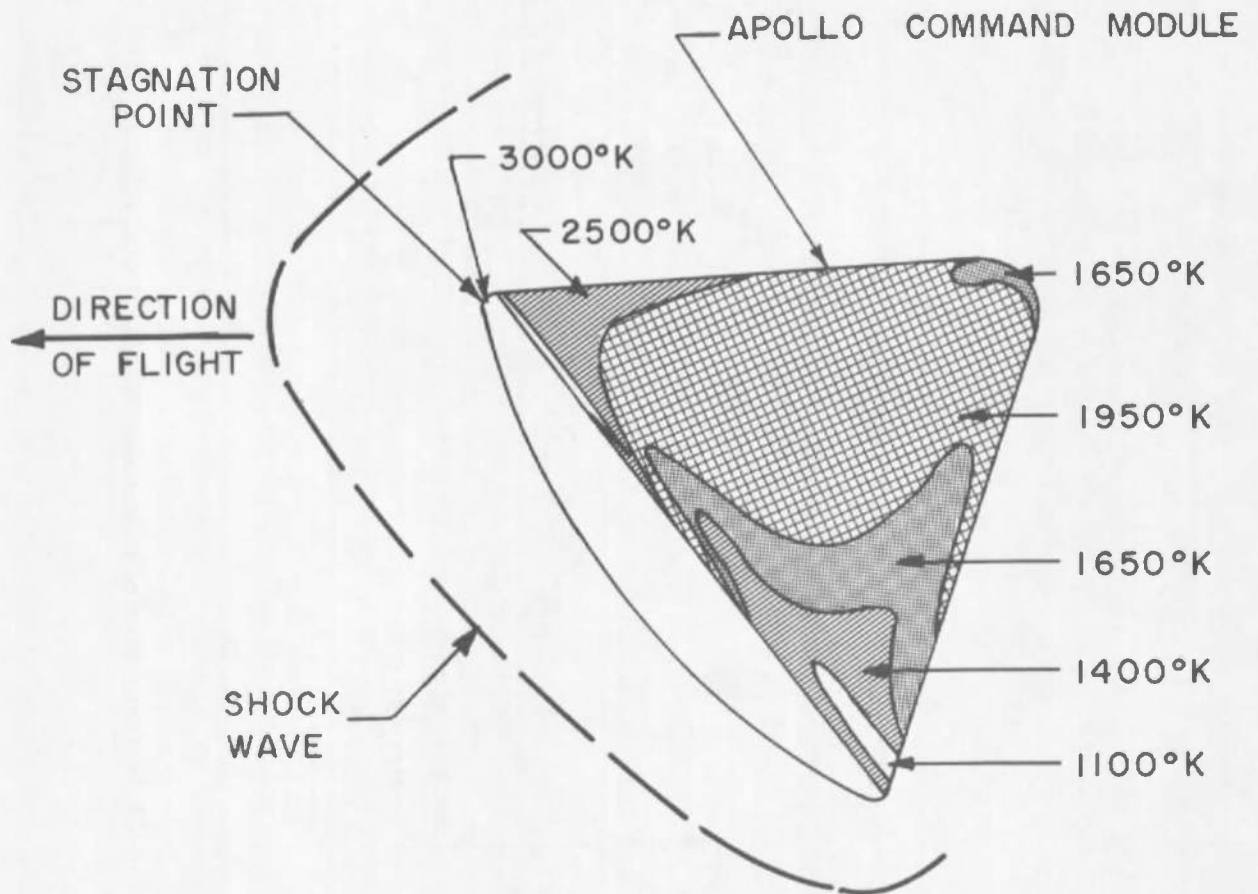
(From Ref. 38)

Fig. 33 Maximum Temperatures Developed during 20-G Emergency Reentry
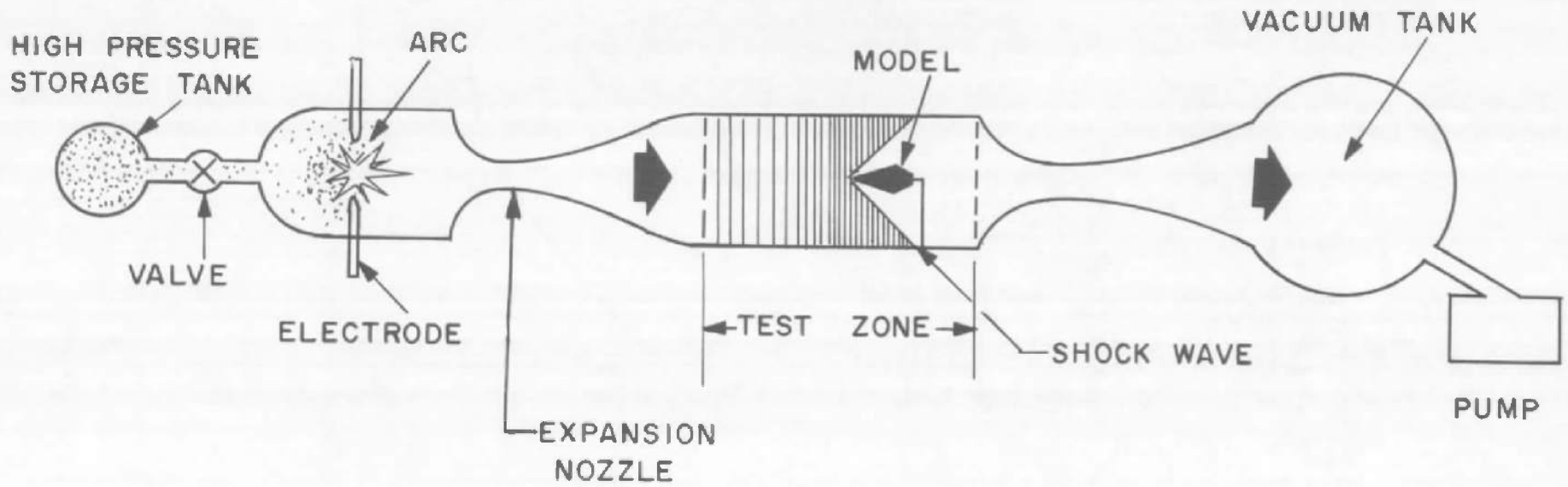
Fig. 34 Schematic Arc-Heater (Hotshot) Hypersonic Wind Tunnel

Fig. 35 Schematic of Typical Hypervelocity Ballistic Range
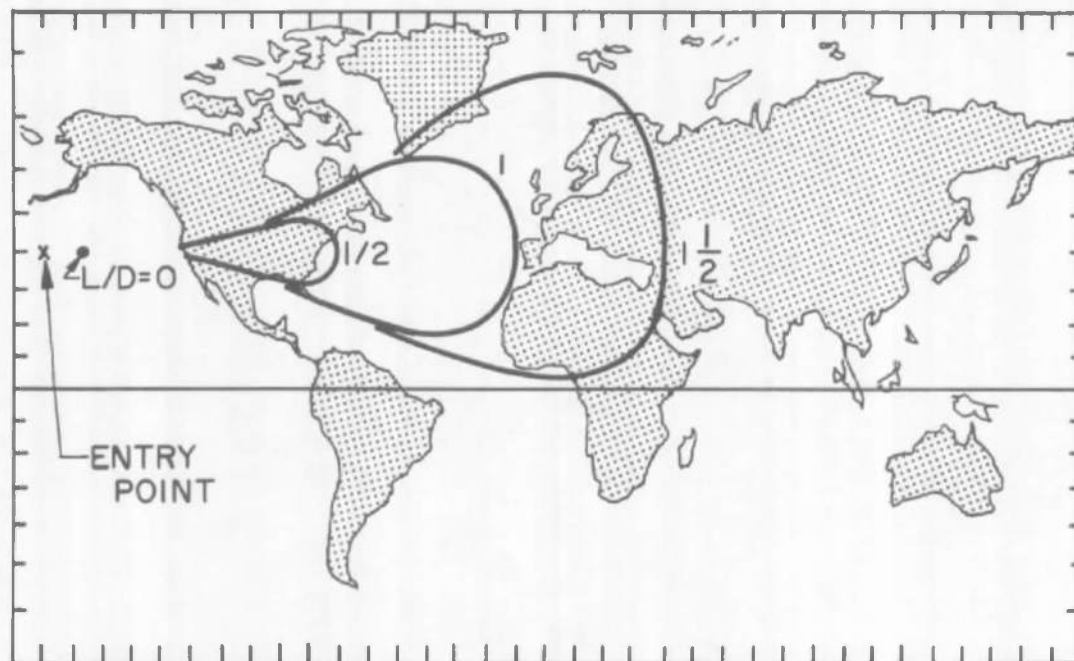
(From Ref. 42)

Fig. 36 "Footprint" of Lifting Reentry Vehicle

EXHAUST GAS COOLER

PROPULSION SYSTEM EXHAUST

TEST CELL

CERAMIC CORED-BRICK HEATER

HIGH PRESSURE AIR STORAGE BOTTLES

SUPERSONIC DIFFUSER

TO EXHAUSTER PLANT OR ATMOSPHERE

HYPERSONIC NOZZLE

FUEL STORAGE

CONTROL BUILDING
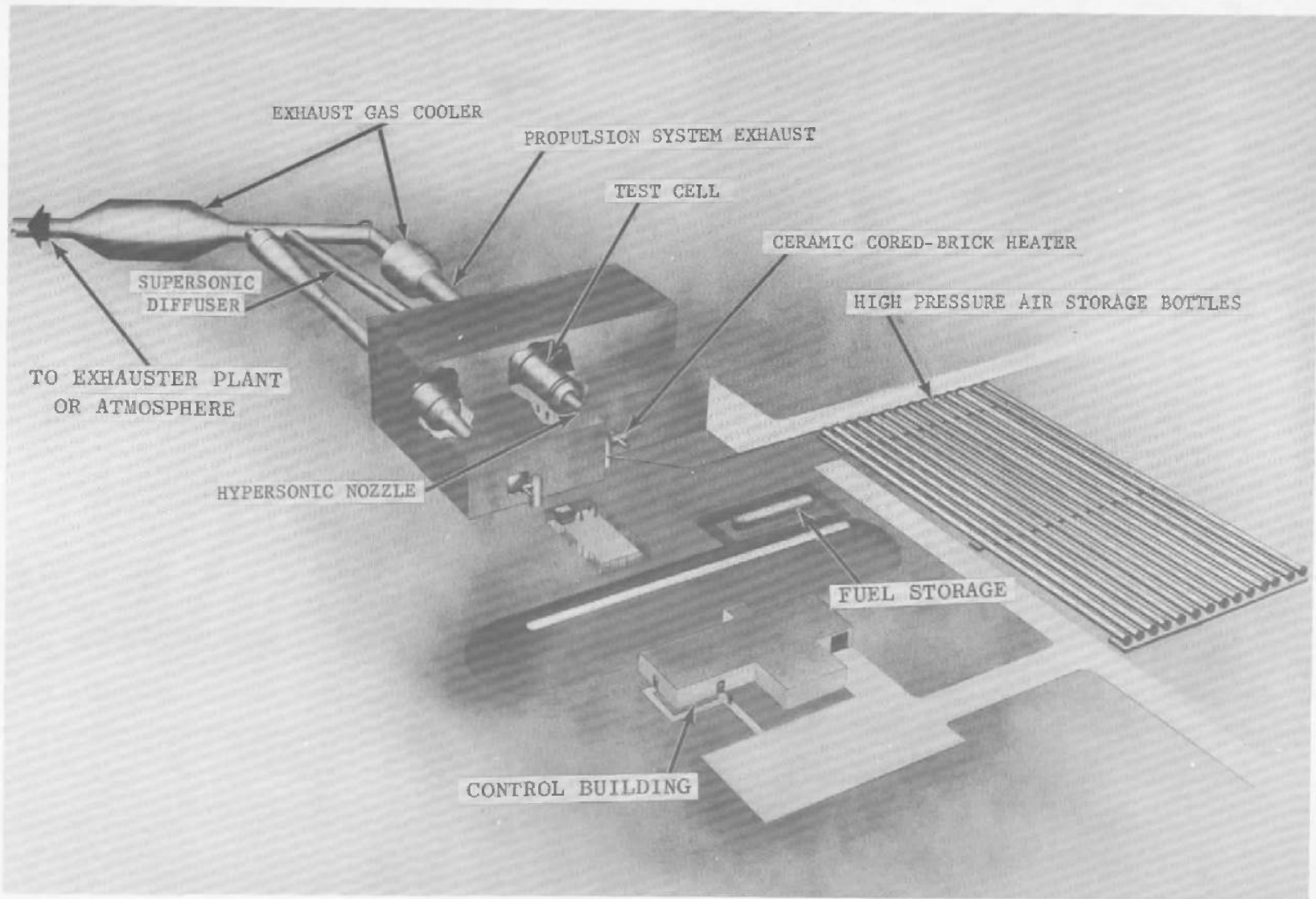
Fig. 37 Hypersonic Wind Tunnel (True Temperature)

TABLE I
EARTH SATELLITES*

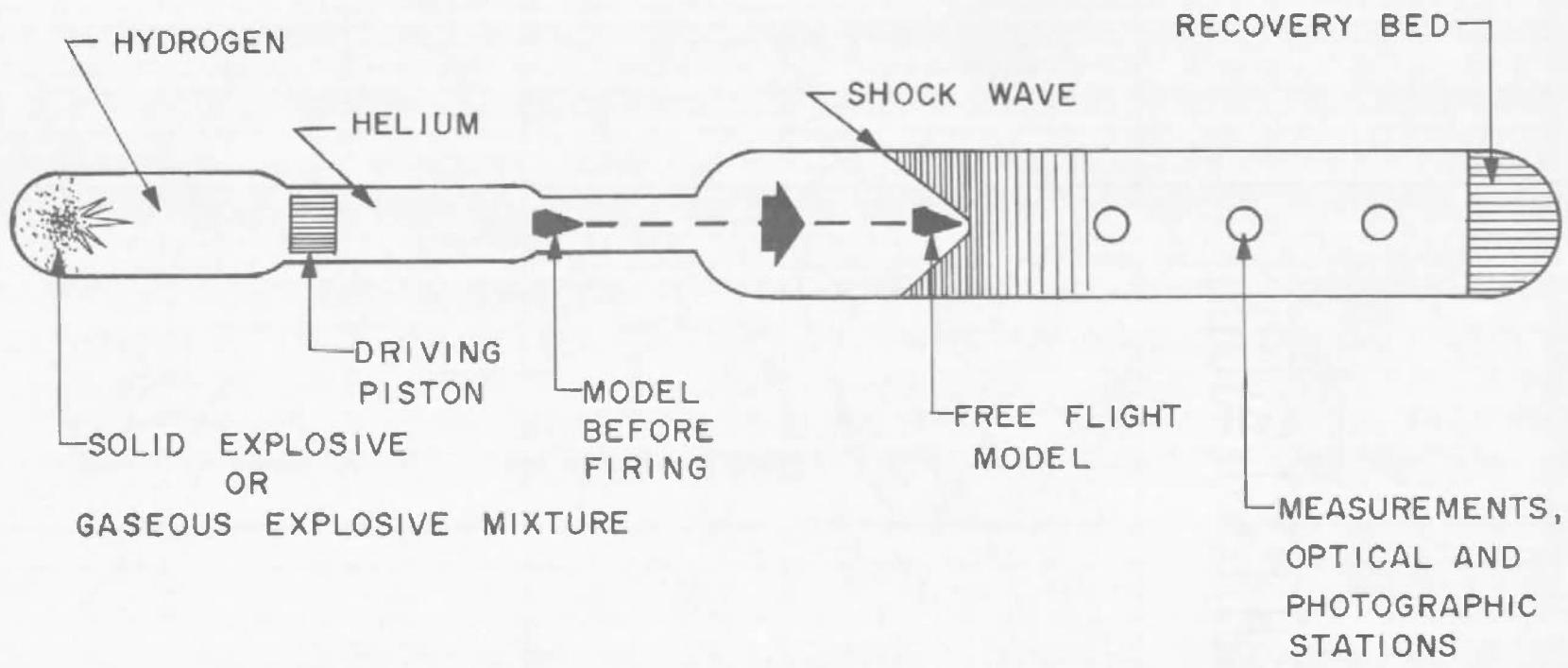| Type	Specific Name	General Instrumentation
Meteorological	TIROS, NIMBUS	Television and Photographic Equipment (Visible and Infrared)
Communication	SYNCOM, RELAY, TELSTAR, EARLY BIRD	Receivers, Telemetry Transmitters, Telephone and Teletype Channels
Navigational	NAVIGATION SATELLITE	Maritime and Aerial Navigation Equipment, Geodetic Data-Gathering Instrumentation
Observatory		
Geophysical	ELEKTRON (USSR)	Radiation Measurements, Geomagnetic Fields
	OGO	Radiation, Dust, Magnetic Fields
	ALOUETTE (Canada)	Ionosphere Data
	ARIEL (U.K.)	Micrometeoritic Information
Solar	OSO	Electromagnetic Radiation
Astronomical	OAO	Stellar UV, Visible, and IR Observations

*From Ref. 7

TABLE II
VENUS AND MARS DATA*

Parameter	Venus	Mars
Mass ($M_\oplus = 1$)	0.8137	0.1071
Mean Diameter ($Diam_\oplus = 1$)	0.957	0.532
Mean Density ($\rho_{H_2O} = 1$)	5.11	4.16
Surface Gravity ($G_\oplus = 1$)	0.89	0.39
Escape Velocity ($km\text{-}sec^{-1}$)	10.3	5.06
Length of Year ($Yr_\oplus = 1$)	0.615	1.88
Length of Day	225 days	24 hr 37 min
Atmospheric Composition	CO_2	CO_2
	N_2	N_2
	H_2O	Ar
Surface Pressure ($P_\oplus = 1$)	5 to 40	0.02 to 0.04
Surface Temperature	600-650°K (Radio-telescope)	Min: 178°K Max: 298°K
	700°K (MARINER II)	Mean: 253°K

*From Ref. 24

TABLE III
VELOCITY INCREMENTS REQUIRED FOR MARS RETURN FLIGHT*

Trajectory	1970-71	1979-80
Minimum Energy (slow) Transfer	12.4 km-sec^{-1}	12.4 km-sec^{-1}
Fast Transfer	19.2 km-sec^{-1}	25.6 km-sec^{-1}

*From Fig. 2, Ref. 26, p. 605

TABLE IV
COMPARISON OF VARIOUS TRAJECTORY MODES

Parameter	Hohman Transfer (minimum energy)	Direct	Direct-Inverse	Venus Swingby
ΔV (Earth orbit to Mars orbit), km-sec^{-1}	4.27	4.57	12.80	4.27
ΔV (Mars orbit to Earth orbit), km-sec^{-1}	2.59	5.03	5.03	5.03
ΔV Course Correction Allowance, km-sec^{-1}	0.18	0.18	0.18	0.27
Total ΔV Requirements, km-sec^{-1}	7.04	9.78	18.01	9.57
Earth Reentry Velocity, km-sec^{-1}	11.9	20.10	12.20	13.40
Stay Time on Mars, days	480	10	10	10
Trip Duration, days	920	430	430	430

From Refs. 27 and 19, p. 288

TABLE V
REDUCTION IN VELOCITY INCREMENT MADE POSSIBLE BY AERODYNAMIC BRAKING*

Trajectory	Required Velocity Increment 1970-71 (km-sec^{-1})	ΔV Savings (km-sec^{-1})	Required Velocity Increment 1979-80 (km-sec^{-1})	ΔV Savings (km-sec^{-1})
Hohman Transfer				
a. All Propulsive	12.4	---	12.4	---
b. Atmospheric Braking at Earth	8.7	3.7	8.1	4.3
c. Atmospheric Braking at Mars and Earth	6.1	6.3	5.7	6.7
Direct Flight				
a. All Propulsive	19.2	---	25.6	---
b. Atmospheric Braking at Earth	11.1	8.1	12.8	12.8
c. Atmospheric Braking at Mars and Earth	8.5	10.7	8.9	16.7

*From Fig. 2, Ref. 26, p. 606

TABLE VI
ELECTRONIC RESEARCH REQUIREMENTS

System	Capability in 1965	Future Capability
1. Reliable Operating Life		
a. Manned Missions	14 days	400 days
b. Unmanned Missions	6 to 7 months	3 yr
2. Component Technology	Adequate for Short-Duration Missions	New concepts such as: Failure Predicting Self-Repairing Self-Checking
3. Spacecraft Transmitter Power	10 w	1000 w
4. Data Rate from Mars	8 bits-sec^{-1}	5×10^6 bits-sec^{-1}
5. Guidance System Accuracy for Mars Landing	Nonexistent	10-km entry corridor
6. LASER Pointing Accuracy	3×10^{-4} deg	3×10^{-6} deg

TABLE VII
SOLAR PROTON RADIATION DOSAGE INSIDE APOLLO COMMAND MODULE

Proton Event	Flux of H^+, $N(E > 30 \text{ Mev}) \times (10^9 \text{ cm}^{-2})$	Skin Dose, Rads	Blood-Forming Organ Dose, Rads
Feb 56	1	40	15
May 59	1	25	2
Jul 59	3	90	10
Nov 60	2	70	15
Safe Limit (From Ref. 45)		700	200

DOCUMENT CONTROL DATA - R&D

(Security classification of title, body of abstract and indexing annotation must be entered when the overall report is classified)

1. ORIGINATING ACTIVITY (Corporate author)	2a. REPORT SECURITY CLASSIFICATION
Arnold Engineering Development Center Air Force Systems Command Arnold AF Station, Tennessee	UNCLASSIFIED
	2b GROUP N/A

3. REPORT TITLE

DEVELOPMENT TESTING OF SPACE VEHICLES

4. DESCRIPTIVE NOTES (Type of report and inclusive dates)

N/A

5. AUTHOR(S) (Last name, first name, initial)

Carlson, Col. Donald D. and MacFarlane, Sqn Ldr George

6. REPORT DATE	7a. TOTAL NO. OF PAGES	7b. NO. OF REFS
February 1966	111	47

8a. CONTRACT OR GRANT NO.	9a. ORIGINATOR'S REPORT NUMBER(S)
b. XXXXXXXX. Program Element 65402234	AEDC-TR-65-199
c.	9b. OTHER REPORT NO(S) (Any other numbers that may be assigned this report) N/A
d.	

10. AVAILABILITY/LIMITATION NOTICES

Qualified users may obtain copies of this report from DDC.

11. SUPPLEMENTARY NOTES	12. SPONSORING MILITARY ACTIVITY
N/A	Arnold Engineering Development Center Air Force Systems Command Arnold AF Station, Tennessee

13 ABSTRACT This report reviews the salient characteristics of various classes of spacecraft, unmanned and manned, present and future. The test objectives and testing techniques used for development of aerospace vehicles are discussed. This portion includes the range of required ground test facilities and the role that flight test programs play. The report also discusses the hypothetical ground test facilities that will be needed to test adequately the conceptual systems of the future, and concludes with the role of earth-orbiting research laboratories in the overall spectrum of flight testing. (U)

DD FORM 1 JAN 64 1473

14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
space vehicles development testing ground testing flight testing						

INSTRUCTIONS

1. **ORIGINATING ACTIVITY:** Enter the name and address of the contractor, subcontractor, grantee, Department of Defense activity or other organization (*corporate author*) issuing the report.

2a. **REPORT SECURITY CLASSIFICATION:** Enter the overall security classification of the report. Indicate whether "Restricted Data" is included. Marking is to be in accordance with appropriate security regulations.

2b. **GROUP:** Automatic downgrading is specified in DoD Directive 5200.10 and Armed Forces Industrial Manual. Enter the group number. Also, when applicable, show that optional markings have been used for Group 3 and Group 4 as authorized.

3. **REPORT TITLE:** Enter the complete report title in all capital letters. Titles in all cases should be unclassified. If a meaningful title cannot be selected without classification, show title classification in all capitals in parenthesis immediately following the title.

4. **DESCRIPTIVE NOTES:** If appropriate, enter the type of report, e.g., interim, progress, summary, annual, or final. Give the inclusive dates when a specific reporting period is covered.

5. **AUTHOR(S):** Enter the name(s) of author(s) as shown on or in the report. Enter last name, first name, middle initial. If military, show rank and branch of service. The name of the principal author is an absolute minimum requirement.

6. **REPORT DATE:** Enter the date of the report as day, month, year; or month, year. If more than one date appears on the report, use date of publication.

7a. **TOTAL NUMBER OF PAGES:** The total page count should follow normal pagination procedures, i.e., enter the number of pages containing information.

7b. **NUMBER OF REFERENCES** Enter the total number of references cited in the report.

8a. **CONTRACT OR GRANT NUMBER:** If appropriate, enter the applicable number of the contract or grant under which the report was written.

8b, 8c, & 8d. **PROJECT NUMBER:** Enter the appropriate military department identification, such as project number, subproject number, system numbers, task number, etc.

9a. **ORIGINATOR'S REPORT NUMBER(S):** Enter the official report number by which the document will be identified and controlled by the originating activity. This number must be unique to this report.

9b. **OTHER REPORT NUMBER(S):** If the report has been assigned any other report numbers (*either by the originator or by the sponsor*), also enter this number(s).

10. **AVAILABILITY/LIMITATION NOTICES:** Enter any limitations on further dissemination of the report, other than those imposed by security classification, using standard statements such as:

(1) "Qualified requesters may obtain copies of this report from DDC."

(2) "Foreign announcement and dissemination of this report by DDC is not authorized."

(3) "U. S. Government agencies may obtain copies of this report directly from DDC. Other qualified DDC users shall request through

_____."

(4) "U. S. military agencies may obtain copies of this report directly from DDC. Other qualified users shall request through

_____."

(5) "All distribution of this report is controlled. Qualified DDC users shall request through

_____."

If the report has been furnished to the Office of Technical Services, Department of Commerce, for sale to the public, indicate this fact and enter the price, if known.

11. **SUPPLEMENTARY NOTES:** Use for additional explanatory notes.

12. **SPONSORING MILITARY ACTIVITY:** Enter the name of the departmental project office or laboratory sponsoring (*paying for*) the research and development. Include address.

13. **ABSTRACT:** Enter an abstract giving a brief and factual summary of the document indicative of the report, even though it may also appear elsewhere in the body of the technical report. If additional space is required, a continuation sheet shall be attached.

It is highly desirable that the abstract of classified reports be unclassified. Each paragraph of the abstract shall end with an indication of the military security classification of the information in the paragraph, represented as (TS), (S), (C), or (U).

There is no limitation on the length of the abstract. However, the suggested length is from 150 to 225 words.

14. **KEY WORDS:** Key words are technically meaningful terms or short phrases that characterize a report and may be used as index entries for cataloging the report. Key words must be selected so that no security classification is required. Identifiers, such as equipment model designation, trade name, military project code name, geographic location, may be used as key words but will be followed by an indication of technical context. The assignment of links, rules, and weights is optional.