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AIR UNIVERSITY REVIEW

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SPACE OPERATIONS ISSUE

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the cover

As a strong bird on pinions free,
Joyous, the amplest spaces
heavenward cleaving . . .
Walt Whitman

The "strong bird" seems as evocative an emblem of space aspirations and achievements as it traditionally has been for winged flight. In this Space Operations Issue the *Air University Review* presents a survey of man's present knowledge of the space environment, his role in it, and the theories and emerging equipments that will enable him to maneuver there.

ACKNOWLEDGMENT

The Editors express particular appreciation to Colonel Francis X. Kane, former Chief, Space and Ballistic Missile Planning Division, DCS/Plans, Headquarters Air Force Systems Command, who first suggested that an issue of *Air University Review* be devoted to the role of the United States Air Force in space operations. He not only drew up the syllabus, found the right authors, and reviewed the articles for this Space Operations Issue but also coauthored one of the articles. Major William C. Ross, Hq AFSC, ably assisted both in authorship and in reviewing and coordinating this interrelated series of expositions. We are also indebted to the Director of Information, Hq AFSC, and his staff for their ready attention to our every call for help. To the authors we extend our admiration—and our thanks for their patience with our numerous queries.

THE SPACE CHALLENGE

General Bernard A. Schriever

SPACE is an area of vital concern to the military strategist. It is a new medium of operations where the actions of our opponents must be closely observed by those of us concerned with national security. It is also a region where our own activities could enhance our security against both earth-based and space-based threats. Thus space adds a new dimension to military thinking. For 6000 years land has been a military medium. The sea has been a military medium for some 4700 years. By contrast, the atmosphere has been a military medium for less than 60 years, and space has been a potential military medium for about 8 years.

We have two specific reasons to be concerned with space. First, space flight offers certain advantages for military operations through its four unique characteristics: extreme altitude, very high speeds, long flight duration, and extremely accurate predictability of flight path. Such characteristics make practical the development of a number of space systems to support military land, sea, and air operations. These include systems for reconnaissance, surveillance, communications relay, command and control, weather prediction, navigation, and geodetic measurement. Space also offers attractive possibilities for development of defense systems against hostile missiles and satellites.

A second reason to be concerned with space is that space progress contributes directly to a nation's leadership in technology and to its national prestige. These factors, like the purely operational aspects of space, have a vital impact on national security.

Both of these reasons compel the military professional to take a serious and continuing interest in space. He also has reason to be concerned about the accomplishments and the potentials of our competition. To date, the Soviets' timetable has always put them one step ahead of us in space. They have orbited the first satellite, the first living creature, the first man, the first woman, and the first multiman space vehicle. They hold world records for manned time in orbit, orbital distance, orbital weight lifted, and highest manned orbital altitude.

This demonstrated technological capability, combined with their openly avowed intention of ruling the world, leaves no room for complacency on our part. Regardless of the debate over the size or direction of our space efforts, the fact remains that we are already involved in space. During the past ten years of ballistic missile development, the Air Force has made fundamental and indispensable contributions to the development and operation of space hardware, including facilities, boosters, and payloads. Air Force contributions to space medicine extend over an even longer span of time. And over the years we have supplied a continuous stream of people with the unique training and experience required in the space effort. These Air Force contributions laid the foundation for U.S. progress in space.

In the years ahead, space shows every sign of becoming even more important to our national security. To cope with this situation, one of our most pressing requirements is to prepare Air Force people to discuss space intelligently. We cannot just extrapolate from past experience; the unique characteristics of space mean that we must do our homework carefully in planning for the exploration and utilization of this new environment.

This Space Operations Issue of the *Air University Review* is designed to help meet the need. It was conceived as an effort to assist a wide Air Force professional audience in preparing for possible space operations. The articles were selected to give some idea of the broad scope of the activities which must be covered—from theory to practical operations. The aim of the issue is to spur greater interest in space and to stimulate individual study and thought, both of which are essential to the improvement of professional competence.

The book is divided into four sections, each of which presents a distinctive aspect of space: Part I—Manned Space Operations; Part II—Exploring the Space Environment; Part III—Theory of Space Operations; and Part IV—Space Applications. Since the authors come from a wide variety of activities in the Air Force and in the National Aeronautics and Space Administration, the coverage of these areas is broad and authoritative.

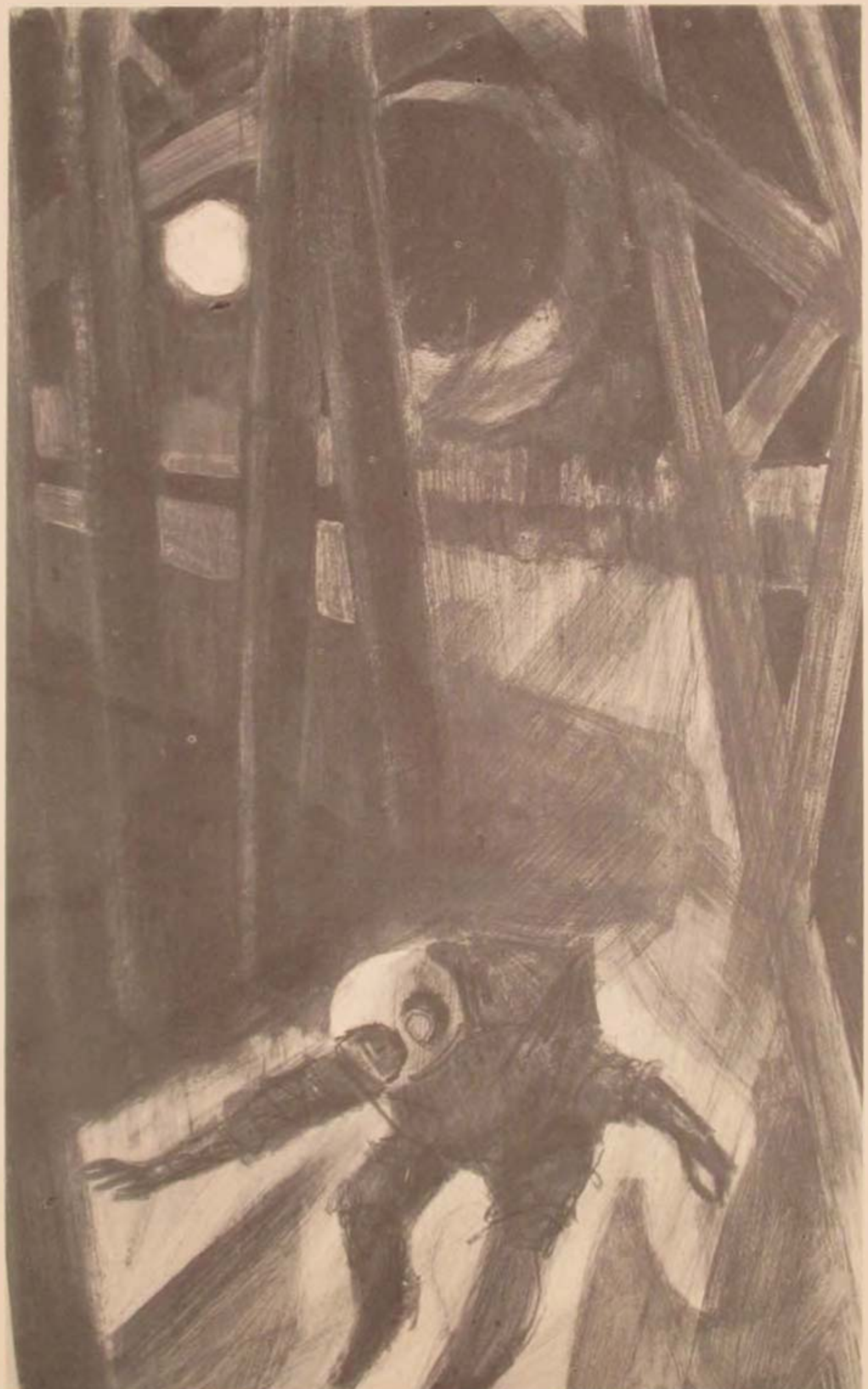
Air Force officers should find these articles both interesting and relevant. As we near the end of the first decade of the Space Age, we have acquired the capabilities which meet our current security needs. During the first half of the next decade we must expand our competence. It remains for us to be ready to apply that competence to the conduct of military operations when necessary to meet future threats.

Air Force Systems Command

MANNED SPACE OPERATIONS

Thou pulse—thou motive of the stars,
suns, systems,
That, circling, move in order, safe, harmonious,
Athwart the shapeless vastnesses of space,
How should I think, how breathe a
single breath, how speak, if, out
of myself,
I could not launch, to those, superior
universes?

Walt Whitman, "Passage to India"



THE GEMINI PROGRAM

Colonel Daniel D. McKee

THE Department of Defense is taking part in the National Aeronautics and Space Administration's Gemini program in a great many ways. The Army and Navy are involved in certain aspects of the program, but the Air Force is actively participating in each phase of development, test, and operations. The purpose of this article is to discuss briefly the history related to the military participation in Gemini and to point out the areas of and reasons for Air Force interest in the program.

The National Aeronautics and Space Administration initiated the Gemini program in December 1961. Prior to any contractual action with industry, the Air Force was called upon to assist in the preparation of a NASA/DOD operational and management plan for the program. This initial coordination between NASA and the Air Force was attributable to the following planning factors:

(1) The Air Force Titan II with minimum modifications could provide the performance capability required to place the Gemini spacecraft in the desired earth orbit.

(2) The Air Force Atlas-Agena combination offered the best possibility of a target vehicle to be used for the achievement of orbital rendezvous and docking.

(3) Existing Air Force launch pads for the Titan II and the Atlas were the logical facilities for launch operations.

(4) Air Force launch crews with previous experience on the Titan II and the Atlas-Agena could be available to supervise the contractors' launch teams.

(5) Air Force procurement contracts for the Titan II, Atlas, and Agena were still active and projected to continue.

(6) The DOD had been responsible for range, worldwide network, and recovery support to NASA in the Mercury program, and a similar role for DOD agencies was envisioned for Project Gemini.

The net result of the joint consideration of these factors was an operational and management plan which made the DOD, and in turn the Air Force, responsible for Titan, Atlas, and Agena procurement for NASA; technical supervision, under a NASA operations director, for the launch of both vehicles; and range and recovery support. These arrangements were consummated in order to facilitate the Gemini program and to ensure that DOD organizations would acquire additional design, development, and operational experience relative to manned space flight. Simultaneously, the Air Force was participating in the development of Gemini by detailing Air Force officers to NASA assignments. These officers worked directly for NASA in various positions, with authority and responsibility comparable to that of full-time NASA employees.

The year of 1962 was devoted to the establishment of industrial contracts, refinement of technical and organizational interfaces, system and subsystem design, and the early stages of hardware fabrication. In January 1963 the

Secretary of Defense and the Administrator of NASA signed an agreement calling for additional DOD participation in Gemini. The intent of this agreement was to ensure that the scientific and operational experiments which constitute the Gemini flight missions would be directed toward achievement of the manned space flight objectives of both DOD and NASA. As a result the Gemini Program Planning Board was established, composed of these top management officials of NASA, DOD, and the Air Force:

Deputy Director of Defense Research and Engineering (Space)
Under Secretary of the Air Force
Commander, Air Force Systems Command
Associate Administrator, NASA
Director of Manned Space Flight, NASA
Deputy Associate Administrator, NASA, for Defense Affairs.

In March 1963 the Gemini Program Planning Board directed a joint Ad Hoc Study Group to confer and recommend the extent and method of additional Air Force participation in the Gemini program. Upon completion of the Study Group effort, the board approved a slate of DOD experiments for Gemini flights and the establishment of an Air Force field office at the NASA Manned Spacecraft Center in Houston, Texas, to be directly responsible for implementation of the experiments. The action of the board was reviewed and approved by the Secretary of Defense and the Administrator of NASA.

From the chain of events that has been briefly described it is evident that the DOD and the Air Force are deeply involved in the Gemini program. In the period of two and a half years that Gemini has been under development the Department of Defense has committed manpower, facilities, and funds to assist NASA in achieving the Gemini objectives. The subsequent discussion will cover some of the reasons for this high level of military interest in the second United States manned space flight program.

advances in Gemini

In the first place, what does Gemini offer that could not be accomplished with Mercury? Although the Gemini design is firmly based on Mercury technology, the differences in capability are multitudinous. Since we are primarily interested in the contribution of man to space flight, the effectiveness of Gemini compared with Mercury was essentially doubled as soon as the crew was increased to two.

Other major steps forward in the Gemini basic design which not only contribute to the NASA manned lunar landing program but also are of vital interest to the Air Force include the following:

- Astronaut control of abort modes while on the launch pad and during the boost phase of flight

Throughout the history of military flight the Air Force has traditionally depended primarily upon pilot judgment in emergency conditions. We in the Air Force have flown for thousands of hours in ejection seats similar to those in the Gemini spacecraft, and the decision to trigger the escape system has always been reserved for pilot discretion. It can be argued that rocket flight is different, that failures are more catastrophic, and that the pilot no longer has the alternative of gliding to a safe landing. While these contentions, per se, are true, the implication that control can no longer be a pilot function does not necessarily follow. The questions of when and how to initiate an abort mode, or escape system, are still a matter of timing and judgment. The pilot must be presented with adequate information to judge the situation and must have available sufficient time to react. In the case of Gemini, extensive analysis and simulation have substantiated the pilot's effectiveness under various pad and launch abort conditions. This is a significant departure and, from a pilot's standpoint, a major improvement in comparison with the Mercury abort system, which was dependent upon automatic sensing and implementation. Of course, even in Gemini and the Titan II, automatic malfunction detection is necessary in order to provide the pilot with emergency warning information. A related capability during the boost phase of flight is provided by an alternate guidance system for the control of the

Titan II launch vehicle. In the event of primary launch vehicle guidance failure, automatic switchover to the backup system is provided. The backup system may also be selected manually by the astronaut. These two trends toward more pilot employment in the control loop are interesting to the Air Force because of the predicted increase in system reliability as well as greater pilot confidence. Continued exploration of pilot control functions during the launch phase also contributes to the eventual development of recoverable boosters, which are of operational interest to the Air Force.

- The capability to perform translation maneuvers in space, in addition to the attitude control which was possible in Mercury

The operational application of manned space flight really begins with maneuverability. A man's ability to perform a military mission in space is severely limited unless he can induce directional and altitude changes. The translation capability of Gemini is very small, but at least it is the first step toward manned orbit control. Whereas Mercury had a hand controller for attitude, Gemini has an additional hand controller for translation maneuvers. The actual maneuvers to be conducted are associated with rendezvous and docking, but the technique developed will be applicable to future military spacecraft where changes in orbit inclination and/or altitude are necessary in order to meet mission requirements involving the spacecraft position relative to the ground or to another orbiting object.

- The ability to rendezvous and dock rigidly with another vehicle in space by the use of radar tracking, astronaut judgment, and spacecraft maneuverability

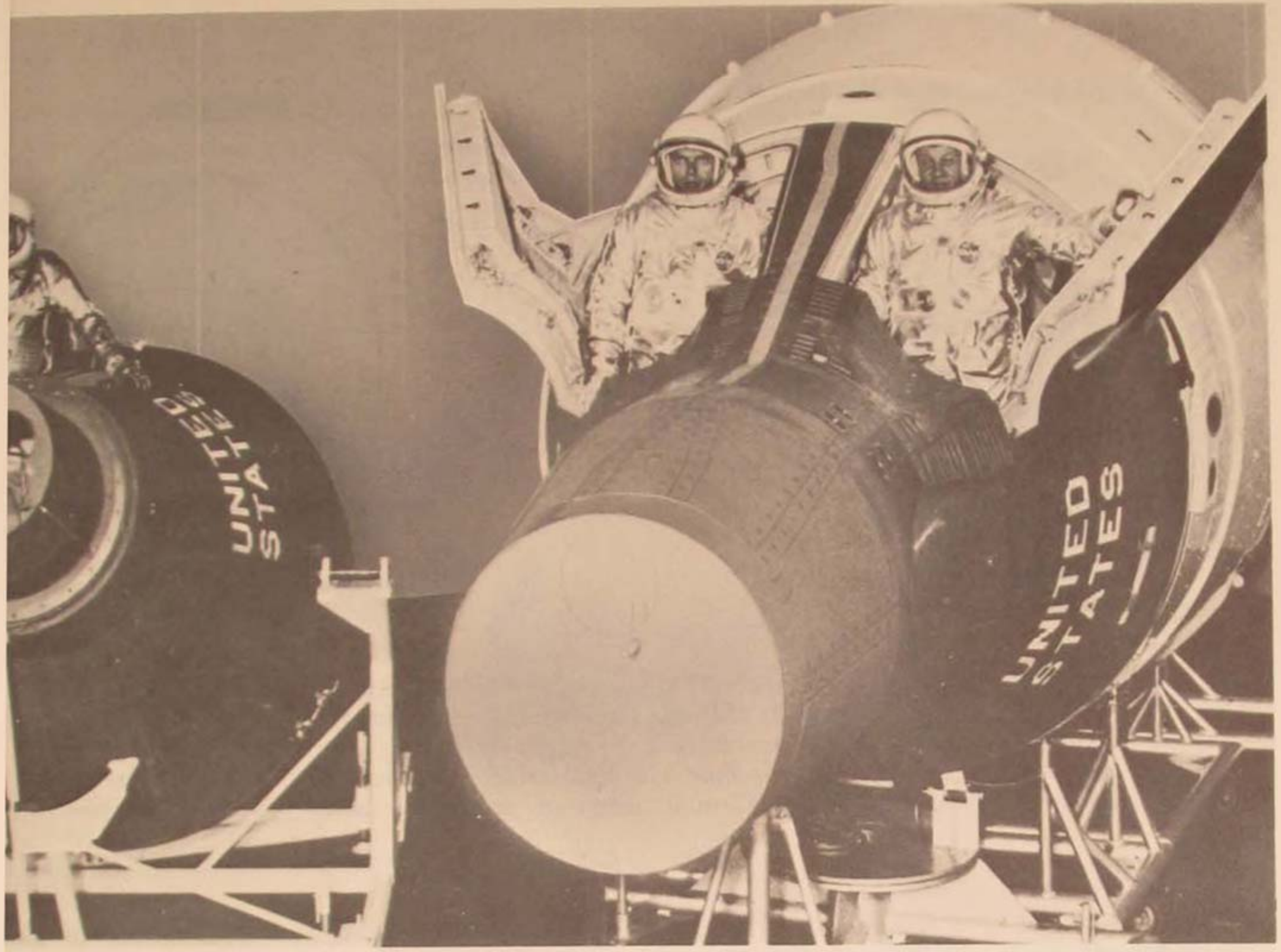
NASA will evaluate two or more techniques for accomplishing the rendezvous of two vehicles in earth orbit. The Gemini rendezvous mission will be conducted with a cooperative target, but in all other respects the methods developed will provide data for application to the potential military requirement for satellite interception and inspection. Rendezvous and docking are also of military interest in the areas of space maintenance, assembly, rescue, logistic support, and personnel transfer. In addition, docking Gemini with the Agena will allow exploration of the problems of manned checkout and firing of a large rocket engine in space. This is another step in the development of the translation maneuverability so important to the Air Force.

- Adequate electrical power, attitude control fuel, and life-support equipment to remain in orbit for long duration (up to 14 days)

Extended time periods in space are naturally of scientific and military interest. A prime advantage of flight above the sensible atmosphere is the prolonged duration that is possible without propulsive power. To exploit this advantage we must determine the psychological and physiological limitations of the crew and the electrical and mechanical duration capabilities of the spacecraft equipment. Such information is essential to the Manned Orbiting Laboratory Program of the Air Force as well as all future manned space missions. If man cannot remain under zerogravity conditions for extended periods without adverse effects, a practical method of providing artificial gravity must be devised. The Gemini spacecraft offers the earliest possibility of flight durations beyond the period experienced by the Russians. Although the Gemini maximum duration is slightly less than half of that projected for the Manned Orbiting Laboratory, the two-week capability should provide either positive or negative answers that will have a marked effect on the design of the Manned Orbiting Laboratory.

- Equipment and procedures to allow an astronaut to engage in extravehicular activity, which includes egress from the spacecraft during orbital flight, demonstration of the ability to perform useful tasks while outside the pressure capsule, and spacecraft ingress

Most of the military missions associated with an orbit rendezvous will also benefit from

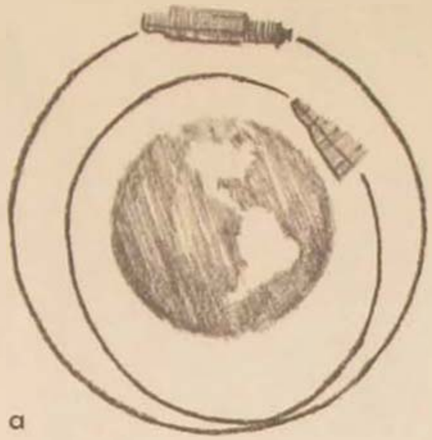


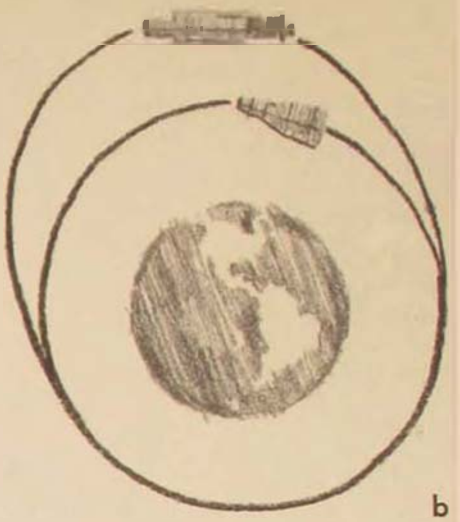
Mockup comparison of Mercury and Gemini spacecraft

the ability of man to operate outside the protected environment of a spacecraft. There are ways to accomplish satellite inspection, maintenance, assembly, logistic support, and personnel transfer and rescue without extravehicular activity, but extravehicular activity may prove to be the most effective means. One of the functions of Gemini is to demonstrate the feasibility and usefulness of extravehicular activity. The Air Force is expanding upon this basic Gemini objective by developing an astronaut maneuvering unit, which will be discussed under DOD experiments.

- Controlled atmospheric re-entry to improve recovery accuracy

One of the main reasons that the Air Force initiated the X-20 (Dyna-Soar) program was to prove a re-entry technique that would make possible sufficient maneuver control to navigate to a selected landing site. The X-20 spacecraft was designed with a lift-to-drag ratio that could be used to alter its re-entry trajectory and atmospheric flight path to reach a landing site almost anywhere in the continental United States. Mercury, in contrast, had no re-entry or atmospheric flight control. The Mercury landing site was determined entirely by timing, attitude, and duration of retrofire. The Gemini is in between. It is not as controllable as the X-20 was projected to be, but it does have sufficient lift to correct its trajectory within a footprint area approximately 450 miles long and 100 miles wide. Gemini uses an offset center of gravity and modulated roll control to apply lift in the direction desired. Theoretically this degree of control will produce a circular accuracy of approximately ten miles' radius, prior





Gemini rendezvous technique. a. *Gemini adjusts to same orbital plane as Agena — Gemini adjusts to 87/161-mile elliptical orbit — Agena in 161-mile circular orbit — Agena orbital period greater — Gemini catch-up to Agena after* n *revolutions.* b. *Agena adjusts to same orbital plane as Gemini — Agena adjusts to 161/161+ + mile elliptical orbit — Gemini in 161-mile circular orbit — Agena orbital period greater — Gemini catch-up to Agena after* n *revolutions.*

to the deployment of a parachute. While this is not exactly the capability the Air Force is seeking, it does offer some choice of landing site and the additional assurance of completing the re-entry in close proximity to the site selected.

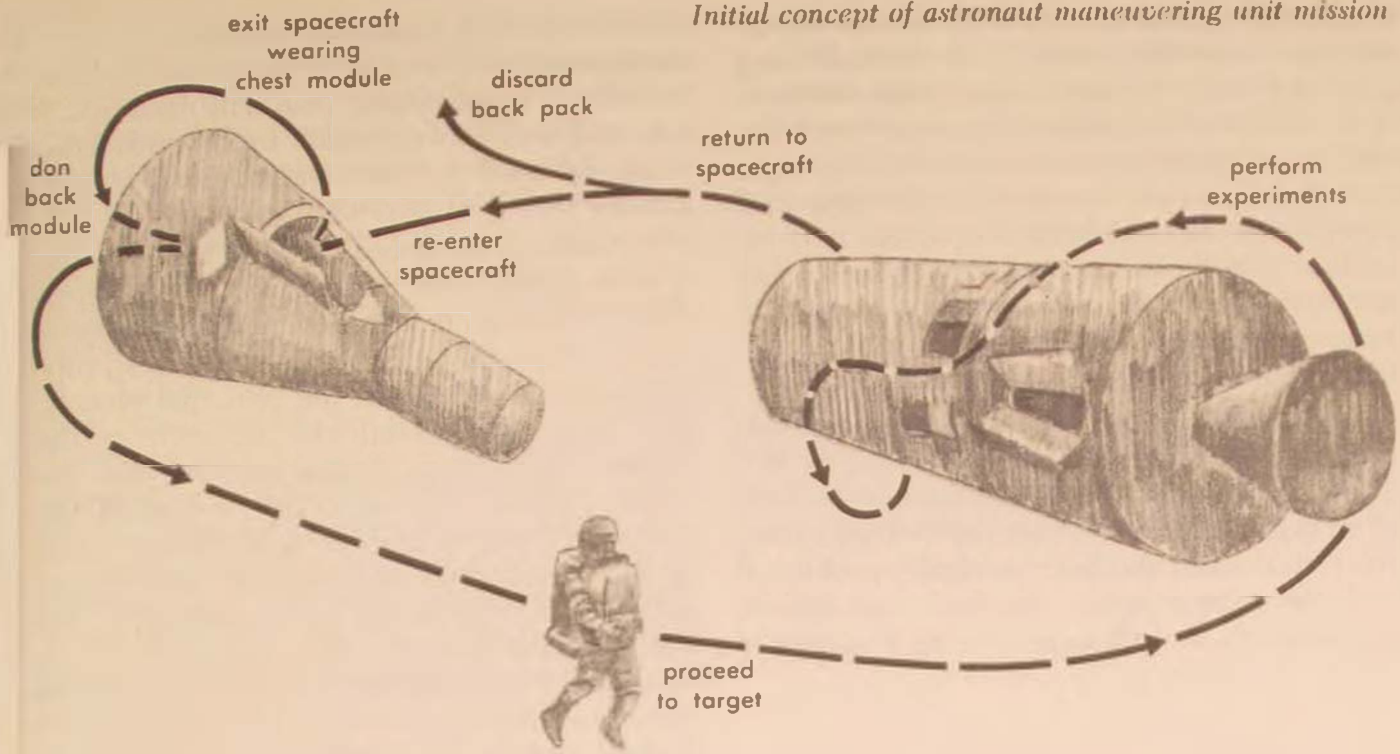
The Air Force Systems Command Field Office (Detachment 2, Space Systems Division) at the NASA Manned Spacecraft Center in Houston is monitoring every aspect of the Gemini spacecraft development and flight mission planning. The primary functions of the Field Office are to gain experience in manned space flight development, test, and operations; ensure the free flow of information on Gemini to the Air Force; and manage the DOD experiments program that will be conducted on NASA Gemini flights.

DOD experiments

The DOD experiments program consists of sixteen experiments, thirteen of them Air Force and three Navy. These experiments are designed to investigate techniques and equipments as additions to the basic Gemini objectives. Although the experiments will have no real-time operational capability, they will add significantly to the total reservoir of understanding related to the potential military application of manned space flight. Instead of discussing each experiment separately, I have grouped them into eight areas.

Photographic and Visual Observation. Three photographic experiments are scheduled for early manned flights of Gemini. The purposes of these experiments are to investigate man's ability to observe, evaluate, and photograph objects in space; to demonstrate human proficiency and spacecraft functional compatibility for observations in space while maneuvering and station-keeping; and to investigate the technical problems associated with man's ability to acquire, track, and photograph terrestrial objects. The equipment to be used is a 35-mm photographic system with interchangeable telephoto lenses, periscopic reflex viewer, telescopic sight, and photographic event-timer for time correlation. The photographic system

Initial concept of astronaut maneuvering unit mission



would be stored in the crew compartment when not in use, and the right-hand astronaut would be required to attach it to the window mount and position the necessary lens and camera body. The motion of the object to be photographed with respect to the mounted camera system would be compensated by the astronaut's maneuvering the spacecraft to maintain the proper object/camera orientation. A fourth experiment in this general area is jointly sponsored by the Navy Bureau of Naval Weapons and the Manned Spacecraft Center, NASA. It involves the measurement of the ability of astronauts in earth orbit to identify ground objects with the naked eye. A prepared sequence of targets with the required visual characteristics will be laid out on the ground in an area of the

Representation of astronaut using minimum-reaction space tool continental United States that is suitable in relation to the Gemini orbital inclination. During passage over the targets the command astronaut will maintain the proper spacecraft attitude, while the second astronaut observes the target and makes verbal comments to the principal investigator at the target site. These four experiments will provide data applicable to studies of the potential military requirements for manned space observation and satellite inspection missions.

Mass Determination. The purpose of this experiment is to investigate the feasibility of a direct contact method of determining the mass of an orbiting vehicle. After the Gemini spacecraft has docked and become rigidly positioned with the Agena target, the dual craft will be propelled by the aft maneuvering thrusters of the Gemini. The average acceleration will be determined from measured incremental velocity change and thrusting time. The mass determination will be made both by the astronaut, using onboard instrumentation, and on the ground by use of telemetry data.

Astronaut checks operation of stadimeter mock-up for manual spacecraft navigation experiment.

Radiometric Measurement. The Air Force is supporting two radiometric experiments. These experiments will provide information on the spectral analysis of regions of interest, supplied by the star fields, the principal planets, the earth, and orbital objects, such as the Gemini rendezvous evaluation pod and the Agena vehicle. The astronaut will point the spacecraft at the objects or background regions of interest and will record spectral data. If possible, visual correlation by photography will also be accomplished. The spacecraft will be equipped with radiometric measuring devices using common mirror optics that can measure radiant intensity from the ultraviolet through the infrared as a function of wave length. One of the prime values of these experiments will be the use of input by a human operator for pointing accuracy, ability to change sensitivity levels, and efficient performance of basic control functions. In addition, the threshold of sensitivity values in absolute numbers for earth and sky background radiation as well as the separation of the irradiance of objects in space from background returns is required for Department of Defense programs. The data being collected are also of significant interest to weather system groups.

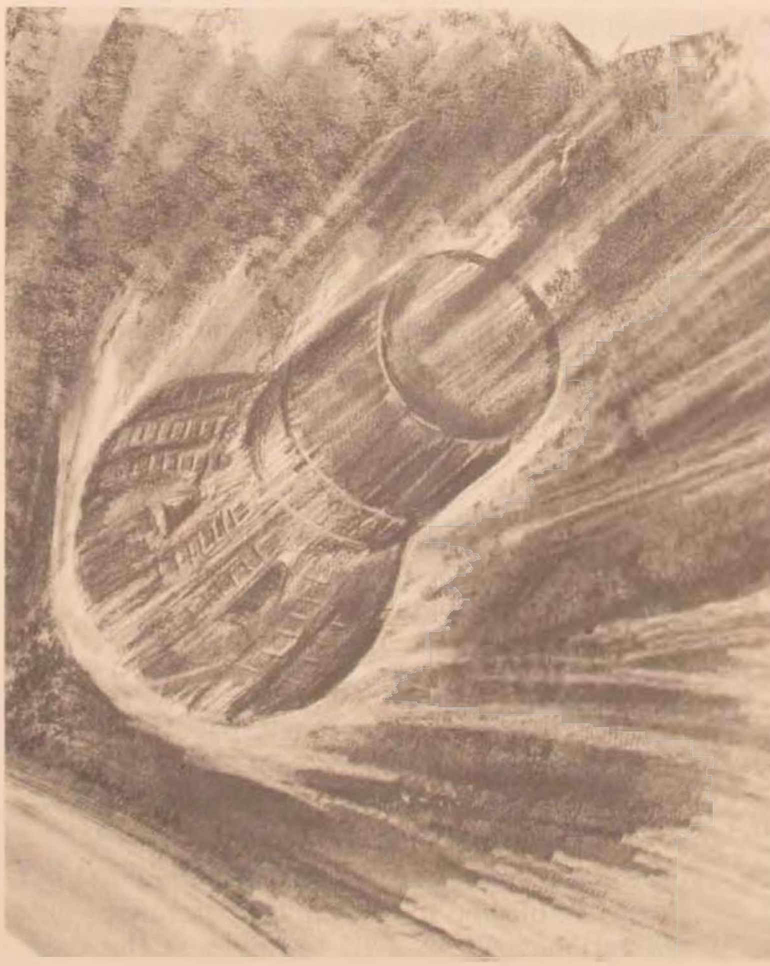
Navigation. In the DOD experiments on Gemini the Air Force is concentrating upon the development of manual techniques for onboard navigation systems. The astronaut will use simple stadimetric measuring devices to make visual sightings and measurements on the horizon and the stars. Data from the sightings will be used for postflight computations with manual computers to determine the orbital parameters. This capability is desired as a backup or emergency navigation procedure for military spacecraft operations.

Another navigation experiment will investigate the feasibility and accuracy of ion sensing as a means of determining yaw and pitch attitude. Ion sensors will be extended in orbit and one of the astronauts will visually align the spacecraft to test for sensitivity and null point. Programed yaw and pitch maneuvers on each side of the null point will be accomplished, and the output meter variations will be recorded. This experiment is part of the navigation group because precise attitude information is required for effective position determination using stellar angle references.

A third experiment involves star occultation measurements for spacecraft navigation. For many years astronomers have used the disappearance of stars behind the planets and the moon to make accurate celestial calculations. The same phenomenon occurs repeatedly as viewed from earth orbit, where stars rapidly occult below the earth's horizon. Extensive analytical work has been accomplished on star occultations with the earth's horizon, and the technique has shown considerable promise in meeting long-term military navigation and guidance requirements. The orbit of an earth satellite would be determined by measuring the time it takes six stars to dip behind an established horizon. The astronaut would only point a helmet-mounted or hand-held photoelectric occultation telescope at the stars when they are occulted by the edge of the earth's atmosphere. Time and attenuation of the stars would be fed into a computer for computation of spacecraft position and velocity. The eventual navigational system thus derived could be regarded as a primary mode of navigation if the method of position fixing must be self-contained; alternatively, it may be regarded as a backup system in the event that communication with the ground is feasible under the normal mode of system operation. The occultation technique appears most practical for application to manned earth satellite systems because of the simplicity of the observational instrumentation and procedures. There are about 50 standard navigational stars in the sky, and since no more than six of these are needed for the general orbit determination, there should be no difficulty in navigating by this method from satellites at altitudes as high as 22,000 miles. As the satellite altitude is increased from 200 to 22,000 miles, the loss of position accuracy is very slight, and thus the occultation technique becomes particularly attractive for a high-altitude manned earth satellite.

In the Gemini experiment test system the raw data from the occultation measurements will be processed on the ground by use of a general-purpose digital computer. The raw data, combined with the reference orbit data, can be used in three ways: to obtain the printout of the position and velocity errors of the occultation-based navigation system, the printout of the atmospheric density profile, and the evaluation of the astronaut's capability to record occultation times manually.

Gemini re-entry



Radiation. The Air Force program to measure radiation dosage encountered in Gemini flights complements the NASA program. The purposes of the Air Force program are to make highly accurate measurements of absorbed dose rate and total dose inside the Gemini spacecraft, to ascertain the accuracy and suitability of small passive devices as dosimeters of space radiation, and to study the spatial distribution of dose levels inside the Gemini cabin and at various locations on the astronaut's body. Two tissue-equivalent, current-mode ionization chambers will measure the dose rate inside the spacecraft as a function of time, as the spacecraft passes into the South Atlantic geomagnetic anomaly. These measurements will be correlated with NASA external measurements and should provide a determination of whether the chief contributor to measured dose is from electrons or protons. One ionization chamber will be portable so that the astronaut may place the radiation-sensitive head at preplanned positions in the spacecraft and on his body. Five small passive dosimeters will be placed in various locations around the crew compartment to cross-correlate dose readings.

Extravehicular Activity. Since both NASA and DOD have planned extravehicular operations, it became apparent at an early date that an integrated program to meet the objectives of the two agencies would be advantageous. Therefore the AFSC Field Office has been working with the Manned Spacecraft Center to plan a controlled sequence of developments and tests to accomplish the total requirements for extravehicular activity within the Gemini flight schedule. The objectives are to evaluate man's capability to perform useful tasks in a space environment, to employ extravehicular operation to augment the basic capability of the Gemini spacecraft, and to provide the capability to evaluate advanced extravehicular equipment in support of manned space flight and other national space programs. The NASA program will encompass the initial steps to evaluate man's capability to perform outside the spacecraft environment. After feasibility has been established in early flights, extravehicular operations will be used to augment the basic capability of the Gemini spacecraft and to evaluate advanced equipment such as the Astronaut Maneuvering Unit which is being developed by the Air Force. In particular, NASA's objective is to have a man leave the cockpit of the Gemini spacecraft, proceed to the interior of the Gemini adapter (large rear section of the spacecraft), and in so doing remain outside the spacecraft for 30 minutes. The objective of the DOD Gemini experiment is to prove the feasibility of man to maneuver and perform useful functions in free space. Since the maneuvering unit is stored externally, the DOD experiment requires the NASA extravehicular equipment to support the astronaut while he retrieves and dons the maneuvering unit. The current plan envisions the use of an umbilical line and a chest pack for initial extravehicular operations. The chest pack will contain a semiopen-loop life-support system as well as the abort alarm and displays necessary for use of the maneuvering unit. The maneuvering unit is designed as a pack to be strapped to the astronaut's back. The back pack will supply the primary oxygen for life support after the extravehicular umbilical is disconnected. The back pack will also contain the propulsion system, stabilization and control system, and equipment for voice and telemetry communications. It is, in effect, a miniature spacecraft which the astronaut uses to control his attitude and to provide translation maneuver capability in free flight. The NASA and DOD development efforts for extravehicular activity are completely integrated.

An additional extravehicular experiment is the evaluation of a minimum-reaction power tool. In order to perform maintenance in a space environment, man will require the ability to overcome the reaction of torques and forces transmitted to him in the performance of maintenance tasks under zerogravity conditions. A power tool which for all practical purposes transmits all the reactive torque away from the operator has been tested in a 5-degree-of-freedom simulator and in a zero-g aircraft. These tests verified the principles under which the tool operates. The Aero Propulsion Laboratory at Wright-Patterson Air Force Base conducted the study and test program in this area. The same principles will be used in designing and fabricating the power tool to be evaluated in this space demonstration. It is currently planned that the tool will be carried in the adapter section of the Gemini spacecraft. During the Gemini mission the astronaut will traverse to the adapter, remove the tool from stowage, perform predesignated tasks on a special work panel, and then return to the Gemini cockpit. The data derived will be applicable to potential military requirements for space maintenance and assembly.

Communications. The second Navy experiment in the DOD experiments program is concerned with polarization measurements using UHF and VHF transmissions. The objective of the experiment is to obtain precise measurements of the electron content of the ionosphere below the satellite and the horizontal gradients of the electron content as functions of time. These factors have a bearing on the design and effectiveness of spacecraft communication and control systems operating through the ionosphere. New polarimeters recently developed by the Naval Research Laboratory (NRL) are particularly applicable to the problem. Nothing new or beyond known techniques is required. The spacecraft attitude may either be recorded in the vehicle or telemetered to the ground, depending on the availability of telemetering channels. The principal ground equipment will be located at Kauai, Hawaii.

Television. The third Navy experiment is a low-light-level television system. Equipment and means are needed whereby nighttime observation of the sea and other terrestrial features, including cloud cover, can be accomplished. To partially meet this need, an image orthicon television camera system employing an image intensifier has been designed by a contractor for the Naval Air Development Center, Johnsville, Pennsylvania. All the components needed have been flown in aircraft individually and have operated satisfactorily. For the Gemini experiment there will be a monitor screen in the cockpit for astronaut viewing and an additional monitor for recording purposes. No transmission to the ground is presently planned, since test pictures will be recorded on film in the satellite. These data will be correlated with all weather data obtainable during the time of the flight, including data from Tiros and/or Nimbus. It will be necessary to relate pictures obtained to the orbital position of the satellite and, thence, to the geographical area view.

EVERY EFFORT is being made to derive the maximum military benefit from the NASA Gemini program. The DOD experiments are being managed by an Air Force Space Systems Division unit colocated with the NASA Gemini Program Office. Such management requires Air Force engineering and operational participation in all phases of this manned space flight program. The experiment equipment must be designed to be compatible with the Gemini spacecraft in size, weight, power requirements, and operating procedures. The equipment must meet special manned flight qualification specifications and be available for periods of flight simulation and astronaut training. Finally, Air Force and Navy engineers and their equipment contractors must be available to support NASA during actual flight missions in the same manner that major subsystems of the spacecraft and launch vehicle are supported. The experience gained in this step-by-step participation is not limited to the Air Force Systems Command Field Office at Houston. It extends into the Air Force laboratories at Wright-Patterson, Cambridge, and Albuquerque as well as the Navy laboratories at Washington and Johnsville. This expanded participation in Gemini, plus the DOD role in the Gemini launch vehicle, the Gemini target vehicle, launch operations, worldwide network, and recovery support, will provide much of the basic knowledge for the definition of future manned military missions in space.

Det. 2, Space Systems Division, AFSC

THE APOLLO PROGRAM

Colonel C. C. Lutman

THE ULTIMATE objective of the manned space flight program is to provide a broad capability for exploration which will achieve and maintain a position of space leadership for the United States. A specific goal in acquiring this capability is the landing of men on the moon and returning them safely to earth, a goal that will be realized through the Apollo program.

To realize the goal, the entire spectrum of the national space effort contributes in some degree, either directly or indirectly, to the lunar landing mission. Each element of the national space program, then, also contributes to U.S. pre-eminence in space. While this article concerns itself only with the lunar landing mission, it should be understood that other projects —such as the bioscience programs, chemical propulsion development, Gemini, orbiting astronomical observatories, orbiting geophysical observatories, atmospheric structure satellites, air density explorers, lunar orbiter, Pioneer, Ranger, Surveyor, and others, as well as the supporting research and technological facilities —contribute to Apollo also. Before we describe what Apollo is, it would be well to say what Apollo is not. It is not a dead-end program culminating in the landing of man on the moon and returning him to earth. Rather it is a beginning, a steppingstone to the more advanced missions such as the NASA manned orbiting research laboratories, the Apollo logistic support system, more extensive manned lunar exploration, and manned interplanetary flight.

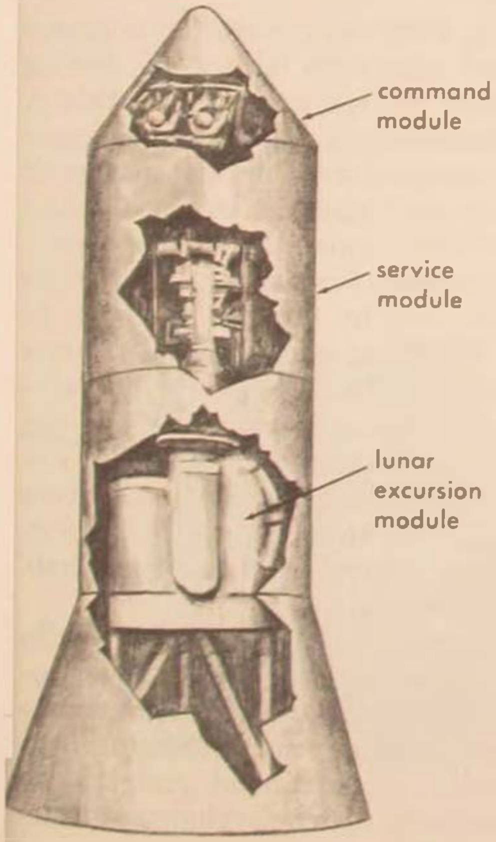
The Apollo spacecraft is composed of separate modules, each designed to fulfill specific mission requirements. The command module houses the three-man crew, serves as the control center for spacecraft operation, and is designed to safely re-enter the earth's atmosphere at a velocity of about 25,000 miles per hour upon return from the moon. The service module houses many of the spacecraft support systems and the major propulsion system for mission abort, mid-course corrections, and injection into and out of lunar orbit. The lunar excursion module (LEM) is a special-purpose shuttle or space ferry for the two men who make the lunar landing. It contains the necessary systems for descending from lunar orbit, performing the lunar landing and take-off, and accomplishing the lunar orbit rendezvous with the command and service modules.

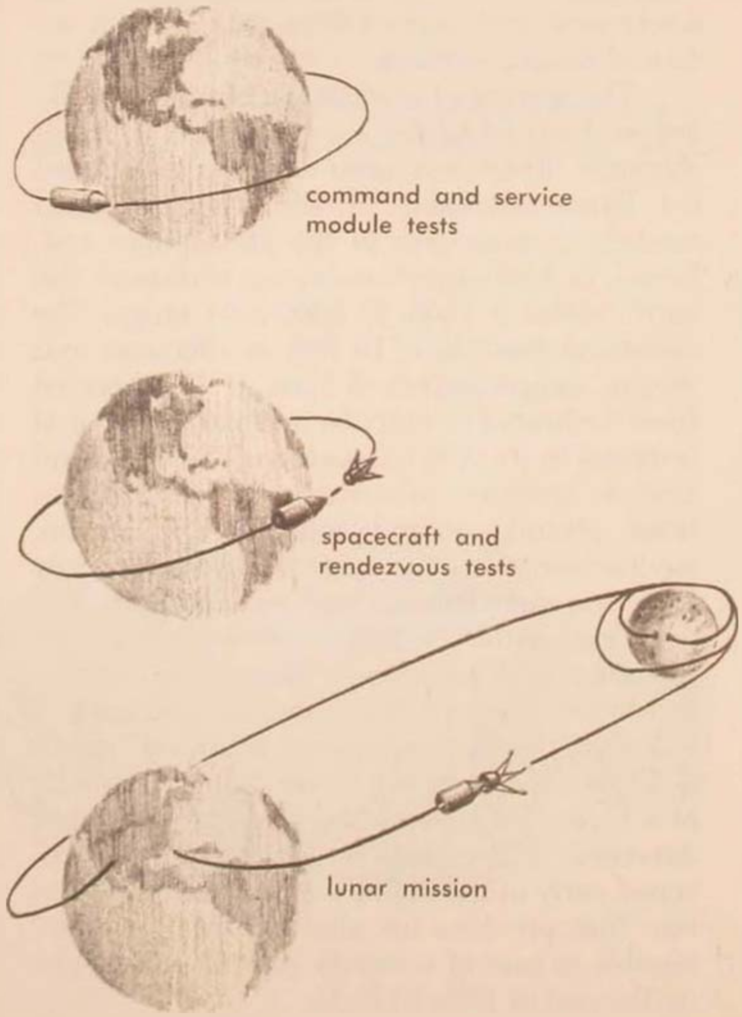
The lunar landing mission is to be performed by use of the lunar orbit rendezvous method, a technique utilizing a single Saturn V launch vehicle. The Saturn V launches the three-module spacecraft into earth orbit and then into the translunar trajectory. Upon approaching the moon, the spacecraft is placed into lunar orbit by the service module propulsion system. Once in lunar orbit, two of the three men transfer to the lunar excursion module, separate from the command and service modules, and descend and land on the lunar surface. A 24-hour stay, with capability for an additional 24 hours, is provided for exploration of the lunar surface near the landing site. After take-off from the moon, a rendezvous with the command and service modules is accomplished in lunar orbit. After the crew of two has completed the transfer from the lunar excursion module to the command module, the service module propels itself and the command module on a transearth trajectory. The lunar excursion module is left in lunar orbit. The service module is jettisoned prior to atmosphere re-entry, and the command module serves as the re-entry vehicle.

The overall scheduling of flight missions for the Apollo spacecraft program progresses in three phases: (1) suborbital and earth orbital flights, (2) circumlunar and lunar orbital flights, and (3) lunar landing flights.

The spacecraft modules will be qualified in suborbital and earth-orbital missions. Suborbital flights using Little Joe II solid-propellant rocket launch vehicles will be made to qualify the abort propulsion system and the spacecraft landing system. Manned long-duration earth-orbital flights of the command and service modules, using Saturn launch vehicles, will develop reliable spacecraft systems and train the flight crews. Manned earth-orbital rendezvous flights of command module, service module, and partially fueled lunar excursion module are scheduled, using the more powerful Saturn I-B launch vehicles. The Saturn I-B vehicle

Apollo and the lunar landing mission





will permit the development of the lunar excursion module rendezvous technique using lunar mission Apollo hardware.

The entire Apollo spacecraft, together with the Saturn V launch vehicle, will be used for manned circumlunar and/or lunar-orbital flights to develop operational techniques in the lunar environment and to conduct scientific experiments in cislunar space.

Lunar landing missions using the entire Apollo spacecraft, launched by the Saturn V launch vehicle, will be made to explore the lunar surface and to conduct scientific experiments in the lunar environment.

command and service modules

The command module houses the three-man crew and is the only part of the spacecraft to return to earth upon the completion of a mission. The service module contains the spacecraft propulsion system and houses spacecraft equipment and expendables that are not required during re-entry.

The command module is a blunt, conically shaped body which has a nominal re-entry aerodynamic lift-to-drag ratio of between 0.3 and 0.4. This aerodynamic lift allows the command module to maneuver in the atmosphere and, hence, to land anywhere on the surface of the earth within a 1500- to 2500-mile range. The command module is 13 feet in diameter and weighs approximately 5 tons. It is protected from the heat of re-entry by an ablative material fastened to its external surface. The command module contains subsystems for communications, attitude control, attitude stabilization, environmental control, electrical power supply batteries, earth landing, and crew support. Also contained within the command module are the guidance and navigation system and instrumentation displays. The internal volume provided for the three-man crew is approximately 300 cubic feet, slightly larger than the interior of a typical office-building passenger elevator. Attached to the command module, but jettisoned early in the flight, is a launch escape system that provides for abort of the command module in case of a launch vehicle catastrophe on the pad or early in flight.

The service module contains a 22,000-pound-thrust, hypergolic-fueled propulsion system, an attitude-control system, hydrogen/oxygen fuel cells for electrical power supply, radiators for the spacecraft cooling, radar, and the major supply of expendables for the life-support system and the electrical power supply. The service module weighs approximately five tons empty. Its diameter is 13 feet, and the overall length is approximately 20 feet.

Preliminary design of the command and service modules has been completed, and the detailed design and development effort is under way. All major subsystems have been placed under subcontract, and ten development spacecraft are being fabricated. The first developmental spacecraft was delivered in August 1962 and has completed water-flotation and seaworthiness testing. Early developmental impact tests, on both land and water, and parachute and landing system development have also been completed. Spacecraft and launch-vehicle dynamics tests have been conducted in the Vertical Dynamics Test Facility at the Marshall Space Flight Center, using an Apollo spacecraft and a flight prototype Saturn launch vehicle.

Fiscal year 1964 was devoted to intensive component and subsystem fabrication, testing and qualification, and spacecraft fabrication. A total of six development spacecraft was constructed, and manufacture of the first spacecraft for manned orbital flight was begun. Ground qualification of all command module and service module subsystems for manned orbital flight was well under way by the end of FY 64. In addition, the abort and earth-landing systems were tested under a simulated off-the-pad mission abort and a transonic abort flight at the White Sands Missile Range in New Mexico. Three Saturn launches were conducted from the Kennedy Space Center to develop the Saturn I launch vehicle and to obtain spacecraft launch environment data.

The lunar excursion module is the Apollo spacecraft module that transports the astronauts and scientific payload from lunar orbit to the lunar surface and returns to lunar orbit to rendezvous with the command and service modules. It has been under development since

December 1962. This module must have the capability of performing separation, lunar descent, landing, lunar ascent, rendezvous, and docking, independent of the mother spacecraft. It is essentially a self-contained vehicle, weighing about 15 tons, with its own electrical power, guidance and control, communications, propulsion, and life-support systems. Because of the moon's relatively weak gravitational field and lack of an atmosphere, the LEM does not need the structural and heat-resistant provisions that are required in the command module for safe re-entry into the earth's atmosphere and recovery on the earth's surface. Its design can therefore accentuate the features essential for lunar landing, take-off, and rendezvous. Windows will provide the astronauts visual reference during these critical maneuvers.

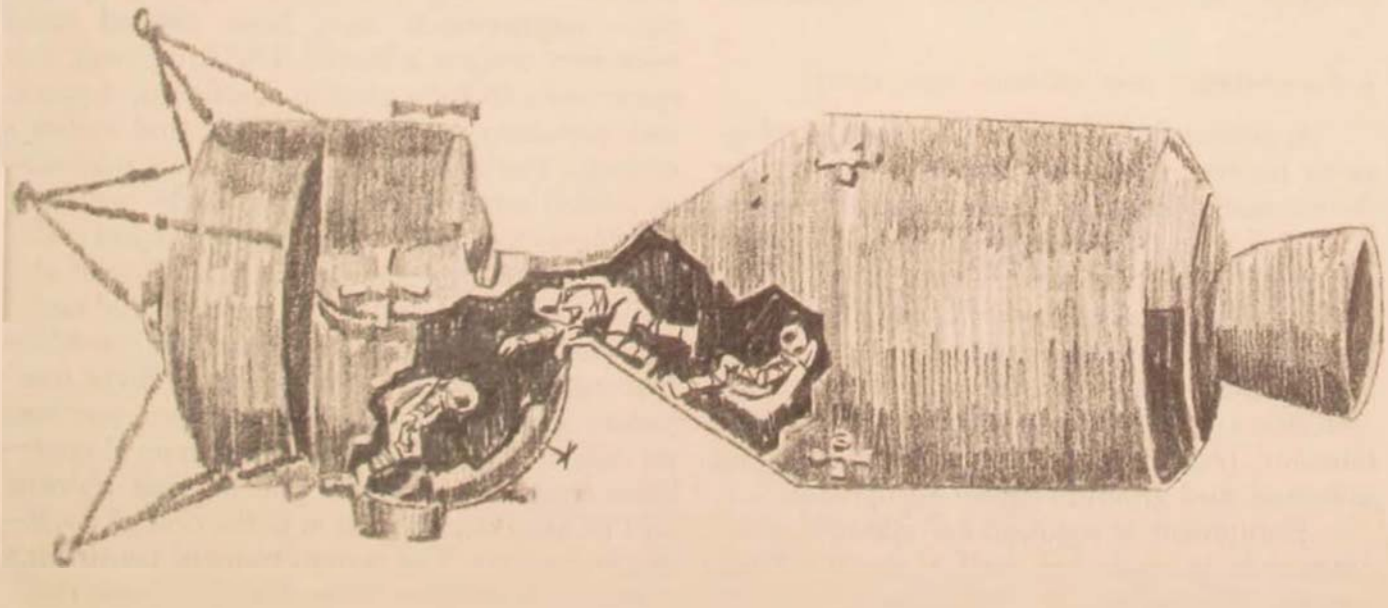
The propulsion system will utilize earth-storable, hypergolic bipropellants and will have a pressurized propellant feed system. It is planned that the LEM will be a staged configuration: the propulsion system (engine and tanks) used to land on the moon will be left there, and a separate propulsion system will be used to take off from the lunar surface and rendezvous with the mother spacecraft. Major effort will be expended to develop the extremely high reliability required for these propulsion systems and the spacecraft stabilization and control systems.

Ground test articles for propulsion tests will be fabricated and tests will be conducted at the White Sands Missile Range. The descent engine will be a throttleable rocket engine that will permit the spacecraft to hover above the lunar surface while a landing site is being selected by the astronauts. In addition, a test article will be constructed to test the basic structure design and to investigate provisions required to protect the astronaut from the hazards of radiation and meteoroids. Eight other LEM ground test articles will be fabricated, and they will be used for integration tests, dynamic tests, environmental tests, vibration tests, and compatibility tests of command and service modules.

guidance and navigation system

The functions of the Apollo guidance and navigation system are to determine the posi-

Transfer to the lunar excursion module (LEM)



tion, velocity, and trajectory of the spacecraft and to control the spacecraft's engines and re-entry lift for the precise maneuvers necessary for the flight to the moon, lunar orbit, lunar landing and take-off, lunar orbit rendezvous, and the return to earth at a preselected landing site.

The guidance and navigation system consists of three major components: the inertial subsystem, the guidance computer, and the optical subsystem. The inertial subsystem consists of a gimbaled platform, gyroscopes, accelerometers, and associated electronics. It is used to establish a reference in space and to measure velocity and course corrections applied by the propulsion systems or by aerodynamic maneuvers. The guidance computer is used to determine the trajectory, to compute required velocity changes, and to send engine start and cutoff signals to the propulsion systems or to signal for proper orientation of the aerodynamic lift vector during atmospheric re-entry. The optical subsystem, consisting of a scanning telescope and a space sextant, is used to find the craft's position in space and to align the inertial platform prior to each maneuver.

Broad guidance and navigation concepts have been established, and command module equipment designs have been finalized. Experimental and development models will undergo reliability and environmental testing. In fiscal year 1964, prototype guidance systems were developed and intensive ground testing started.

instrumentation and scientific equipment

Specialized flight research and development instrumentation is required during the development phase of flight-testing the spacecraft and the scientific equipment for in-space and lunar scientific experiments. This instrumentation is required to obtain the detailed engineering test data for evaluation of spacecraft system performance under critical operating conditions. Included are sensors, transducers, telemetry transmitters, appropriate transmitting antennas, and ground support equipment.

Equipment is required for scientific measurements in space, as well as on the lunar surface. Typical scientific equipment includes special cameras, magnetometers, seismographs, and radiation-measuring devices.

operations

Operational support required for the conduct of flight missions can be classified in three major categories: flight, recovery, and crew. Flight operations include operation planning, preflight, checkout of the spacecraft, and control of flight execution from lift-off to landing. Personnel must be trained to operate the Mission Control Center and outlying stations to control the various phases of the Apollo flights.

Recovery operations include the efforts required to effect either a land or sea recovery of the Apollo spacecraft and crew at any time during the flight. Adequate planning and study must be accomplished to provide an operational recovery capability for the Apollo missions. Recovery forces must be equipped with direction-finding and location aids. Determination of the sites for deployment of planned and contingency forces is necessary. These efforts include evaluation of recovery operation techniques, tests of the spacecraft landing and recovery systems, development and procurement of electronic and visual locating systems, and procurement of handling and retrieval equipment.

Crew operations provide for spacecrew training and integration of their activities with the engineering development, mission planning, and the flight missions. Training equipment requirements have been defined and necessary designs initiated. The equipment requirements include mission simulators, a part-task simulator, a docking trainer, and system trainers. The mission simulators allow manned simulated missions to be flown under realistic conditions to evaluate and improve spacecraft design, allow the planning of primary and alternate missions, and allow training for each specific flight. The part-task simulator provides training in a selected portion of the flight trajectory, thereby allowing concentrated practice on difficult flight tasks with a minimum of operation time and expense. The docking trainer will be used for instruction in the critical docking maneuvers. The system trainers, consisting of animated displays of spacecraft systems, provide rapid and dynamic instruction in system operation details.

IT CAN BE SEEN that the Apollo program is not aimed solely at the successful completion of a lunar landing but rather is a tool employed to obtain and keep U.S. supremacy in space. Apollo is a beginning, not an end, of a program for further lunar exploration. Apollo is a huge network of tracking and data-acquisition stations girdling the globe to provide that vital link between the spaceborne and earthbound. Apollo is the discovery and perfection of technologies for future use in the Nation's industries and improvements in standards of living. Apollo is financial: the effects reach almost all areas of the country and provide a stimulation to the national economy. Finally, but most importantly, Apollo is people—medical doctors, professors, scientists, engineers, designers, technicians, managers, plant operators, astronauts, pilots, and artisans—all welded together as a Government/science/industry team dedicated to a common cause with a singleness of purpose —success.

Hq Air Force Systems Command

PILOT RELIABILITY AND SKILL RETENTION FOR SPACE FLIGHT MISSIONS

Dr. Milton A. Grodsky and
Colonel C. C. Lutman

One of the more important questions in the conceptual design of long-duration space flight systems is the utility of the pilot in the control of the system and in the performance of the associated mission tasks. The question of the pilot's utility is based not upon an estimated lack of his ability to perform the assigned tasks after appropriate training but upon an estimate of his reactivity to the stresses imposed upon him during the flight. These stresses include both the physiological (reduced pressure, weightlessness, etc.) and psychological factors (confinement, workload, etc.) involved in such flight.

One recent emphasis in the design of large weapon systems is to determine the effectiveness of the system by estimating its reliability. It appears that this same approach would be appropriate and desirable for long-duration manned space flight. In the design of conventional high-performance systems, the coupling of man with the machine, though imperfect, is usually successful as a result of past experience with the stress of such flights, the duration of the missions, and the type of task required of the pilot. The use of proper crew-station design criteria based upon this past experience produces a reliable overall system with extremely high pilot reliability. Unfortunately, insufficient experience in manned space flight makes such estimates of overall man/machine system reliability difficult. In particular, the durations of the flights contemplated make reliability estimates based upon past experience highly controversial.

The purpose of this article will be to discuss recent studies concerned with experimentally determining the reliability of pilots in an integrated mission simulation of a long-duration space flight mission. Two aspects of pilot reliability were investigated. First, the reliability of groups of pilots during a 7-day simulated lunar mission after a period of training and assurance of a stable level of performance prior to this mission. Second, the retest of the same groups of pilots, one group tested 30 days after the mission, the other 60 days after, to ascertain skill retention in selected tasks. Though we shall be primarily concerned with the reliability of task performance after long periods without practice (30 and 60 days), sufficient discussion of the initial reliability study (7-day mission) will be presented to provide a proper background.

the simulation method

The basic purpose of the study dictated the type of simulation method to be used. It was initially determined that only the task sequence, task difficulty, mission duration, and confinement in a reduced volume would be investigated. Though physiological factors are also of importance in ascertaining the pilot's reliability, it was decided that this initial study would concentrate on the nonphysiological stress factors. To achieve valid data in this area, the integrated mission simulation technique was used. This technique is designed to provide the pilot with tasks that are performed in the appropriate time sequence under conditions approaching the actual flight.[1] Further, the tasks are as realistic as possible, with the associated mission sequences and displays. This simulation technique performed in real time differs from the part-task simulation in that it provides an estimate of the influence of the mission duration, mission sequence, and task difficulty on pilot performance. This technique also provides the pilot realistic tasks related to actual or conceptualized systems not obtainable in many instances in the more basic laboratory studies.

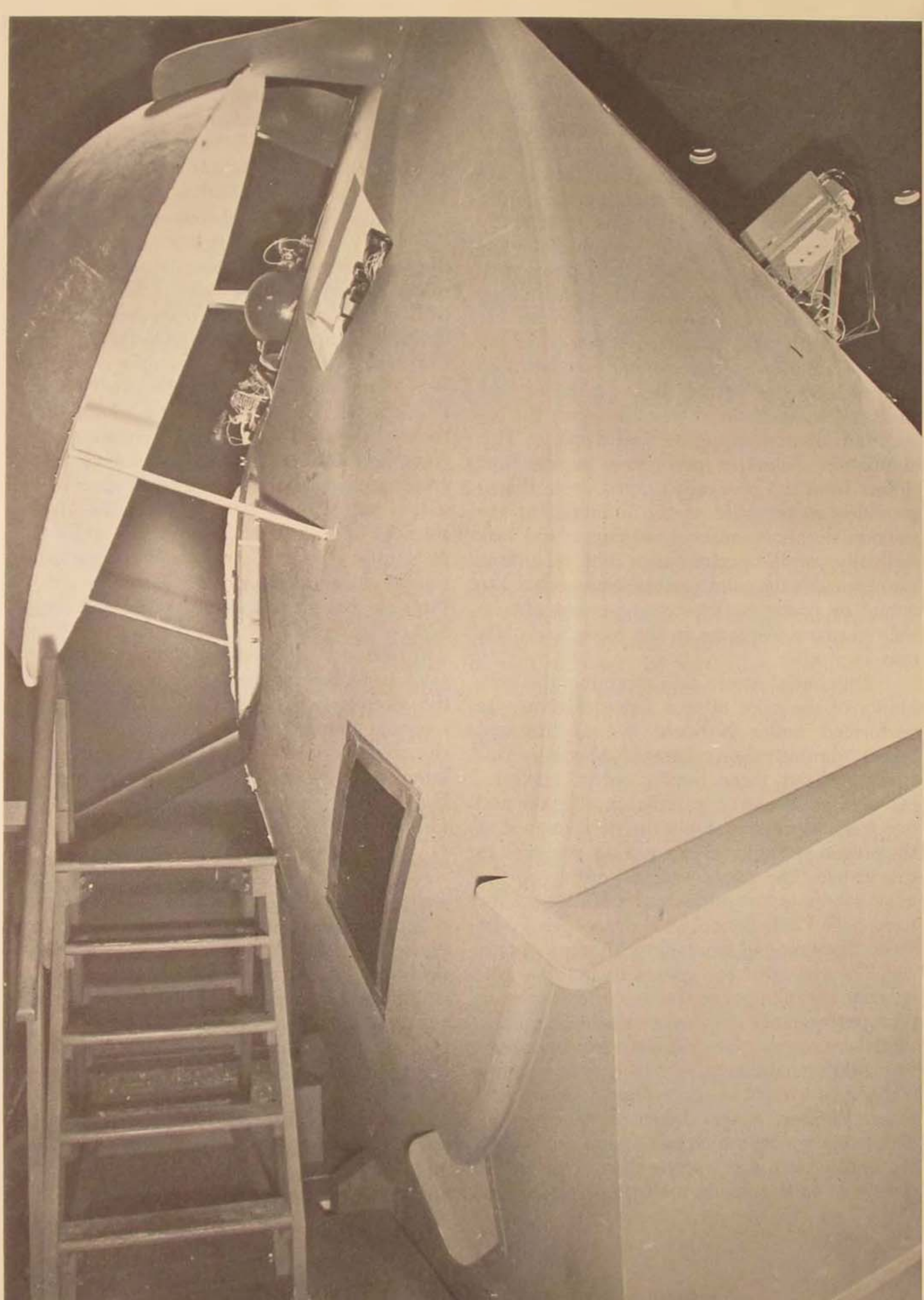
The initial study to determine the reliability of the pilot after a 7-day mission was performed under National Aeronautics and Space Administration contract NASw-833. This study utilized three 3-man crews, each receiving five weeks of training on all tasks and one week of mission. Only the performance of the second and third crews will be discussed in this article. The pilots were all officers of the USAF, graduate test pilots, and graduates of the Aerospace Pilots School at Edwards Air Force Base. They ranged from 30 to 37 years of age.

The choice of the system to simulate was based upon an initial goal of the study to obtain pilot performance data which could be manipulated by appropriate reliability formulations into pilot reliability and which was generalizable to a variety of space flight missions and tasks. Further, it was believed necessary to simulate a system which had operational validity and which was sufficiently advanced in design so as to include realistic displays, controls, and vehicular dynamics. The choice of the system was the Apollo lunar landing system.

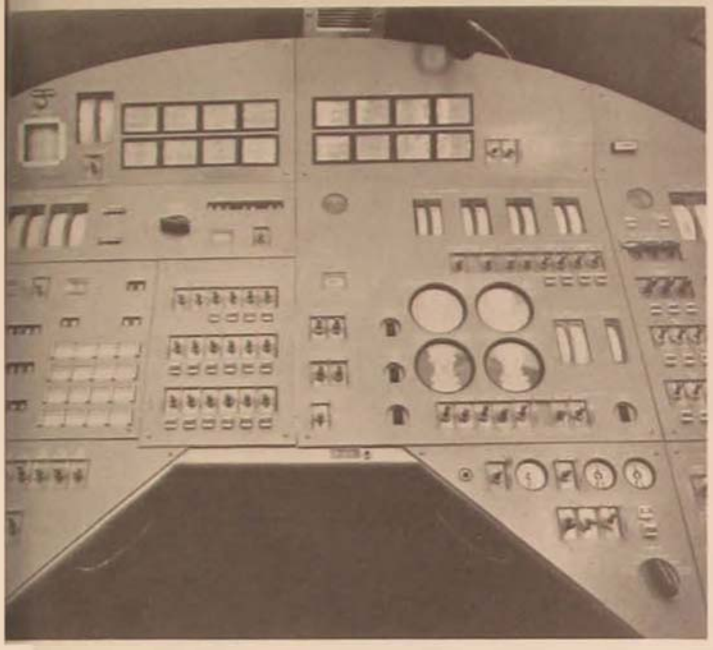
The system was simulated with as much fidelity as the available design data allowed in October and November 1963. Though the design parameters of the system have since changed somewhat, the utility of the collected data still appears applicable.

Two vehicle simulators were developed for this study, with appropriate out-the-window displays for each. The simulator of the command module, the vehicle in which the astronauts will travel to and from the moon, contained approximately 350 cubic feet. It had a forward display panel in which the crew members would be seated three abreast. The simulated display panel was closely configured to the NASA panel with minor deviations. The command module simulator also included an enclosed sleeping area, an off-duty area, a navigation area with an associated out-the-window starfield display, a sanitation area, and a food preparation area (Figure 1). The lunar excursion module simulator also was configured like the NASA design and contained approximately 170 cubic feet. Above the display panel were two triangular-shaped windows that gave the pilot an external view. There were no seats in the excursion module but harness arrangements in which the pilots were confined. While the command module was a fixed-base simulator, the lunar excursion module was a moving base with three degrees of freedom in rotation (Figure 2).

Also associated with the excursion module were various out-the-window displays which were utilized during the lunar landing operation in order to increase the realism of this phase. Directly facing the two triangular windows of the excursion module was a 24-foot-diameter hemispherical screen capable of splitting and moving out of the pilot's view through a hydraulically controlled mechanism. During the lunar landing phase the screen was maintained whole, and upon it was projected a lunar horizon and starfield from a projection system located above the excursion module. Apparent motion of the excursion module relative to the stars and lunar horizon was obtained



gure 1. External and inter-
l views of command module



split. The pilot then controlled the translational rate and direction of the command module with a translation control within the excursion module and controlled the attitude of the excursion module with the attitude control stick (Figure 3). The task was to dock the command module model with the excursion module. Crew transfer from one simulator to another prior to lunar landing and after docking is accomplished by hatchways associated with each simulator.

All dynamic flight phases were controlled by a 360 amplifier analog facility, and the nonflight phases were controlled from adjacent control room systems. The control room contains a mission controller's console and flight director's console, from which the mission and tasks were managed. Also located in the control room are 590 channels of recording equipment, which were utilized for data collection on switches and meters in both the simulators (Figure 4). The analog has an additional 100 channels of recording equipment for the dynamic flight data, including some for automatic data processing of flight performance measures.

by an analog computer tie-in between the attitude stick in the excursion module and the projector. When the pilot pitched the excursion vehicle away from the lunar surface, the horizon disappeared and more stars came into view.

During the lunar ascent phase the pilot in the excursion module searched the starfield on the screen to locate a blinking light that was the simulated beacon of the command module in lunar orbit. The pilot, once he acquired the blinking light, was required to control the excursion module in such a manner that the range between the two modules decreased. As he approached a distance of 2000 feet from the beacon, it began to grow in size perceptually, reaching finally a diameter of 8 feet. At this point the screen split and a full-size model of the command module suspended from a boom moved toward the excursion module at the rate at which the excursion module was moving toward the circle of light before the screen

pilot tasks

The lunar mission tasks which were simulated may be divided into the following categories:

- flight control
- switching
- information handling
- procedural tasks
- navigation.

The flight control tasks were those concerned with the control of the vehicle during the dynamic phases of the lunar mission. In the current simulation, the dynamic phases were the translunar insertion, excursion module transposition, translunar mid-course corrections, lunar orbit insertion, excursion module separation and deorbit, excursion module coast descent, excursion module brake and hover, excursion module landing, excursion module ascent, rendezvous, docking, transearth insertion, transearth mid-course corrections, and earth entry. The tasks varied from the simple attitude control and energy management in the insertion phases to the complex maneuvers during the excursion module brake and hover phases. In all dynamic situations except earth launch, the pilot was given direct manual control of the vehicle. The measures utilized in the evaluation of the pilot's performance were designed to cover the gamut of pilot performance during any particular phase. These included terminal conditions as well as measures of pilot variability and error.

The switching tasks were those concerned with the "setting up" or returning to normal of onboard systems prior to or after a flight control or navigation task and the conduct of system checks or malfunction detection. The switching tasks were evaluated on the basis of errors made in a switching act. These errors were differentiated into three categories: failure to throw a switch, or throwing the wrong switch, when a switching act was required; false-alarm switching, which was the throwing of a switch when no switching act was required; and other inadvertent switching errors.

The information-handling tasks were those concerned with the pilot's ability to place information into the computer and receive information from it, perform log checks of pertinent displays, etc. The measure of this performance was simply the number of digits of information handled correctly divided by the number of possible digits of information that could be handled.

The procedural tasks were those concerned with the observation of communication blackouts, conformance to the duty cycle, etc. The measure utilized is a simple count of the number of deviations from established procedures.

The navigation activities are partially accounted for by the switching and procedural tasks. However, one additional estimate of the pilot's ability is indicated during position determination by the difference in degrees between a particular star's absolute position and its position as determined by the pilot.

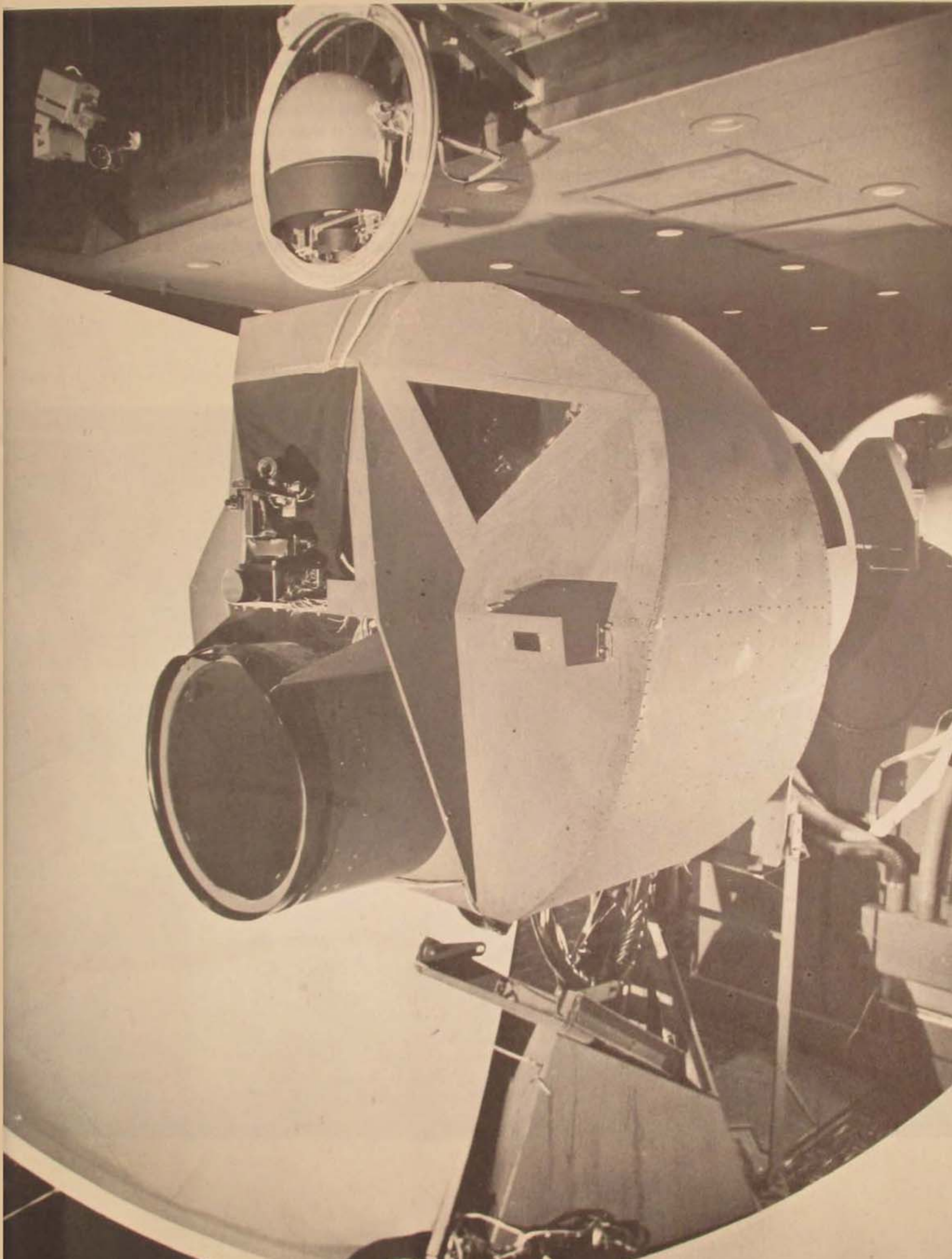
During the 5-week training period of the initial reliability study, all pilots received trials on each task associated with each mission phase. The procedure in training progressed from lecture to part-task training to whole-

Figure 2. Internal and external views of lunar excursion module

mission phase training to an integrated fast-time mission in which the coast periods of the mission were deleted. The premise upon which the reliability of pilot during the 7-day mission was to be determined was based on premission performance yielding statistical evidence that the pilot had learned the task and that his performance level had stabilized. In the two groups of pilots, this was indeed the case. Performance during the mission was then compared with the base-line performance.

The missions were conducted in real time, the pilot performing the appropriate tasks during each phase. Table I presents in abbreviated form the tasks by mission phase based upon total elapsed mission time as simulated in the 7-day study.

Those familiar with the Apollo mission profile will note in the table some differences in the mission-time history. The differences occurred primarily because of the experimental requirement to collect as much data as possible

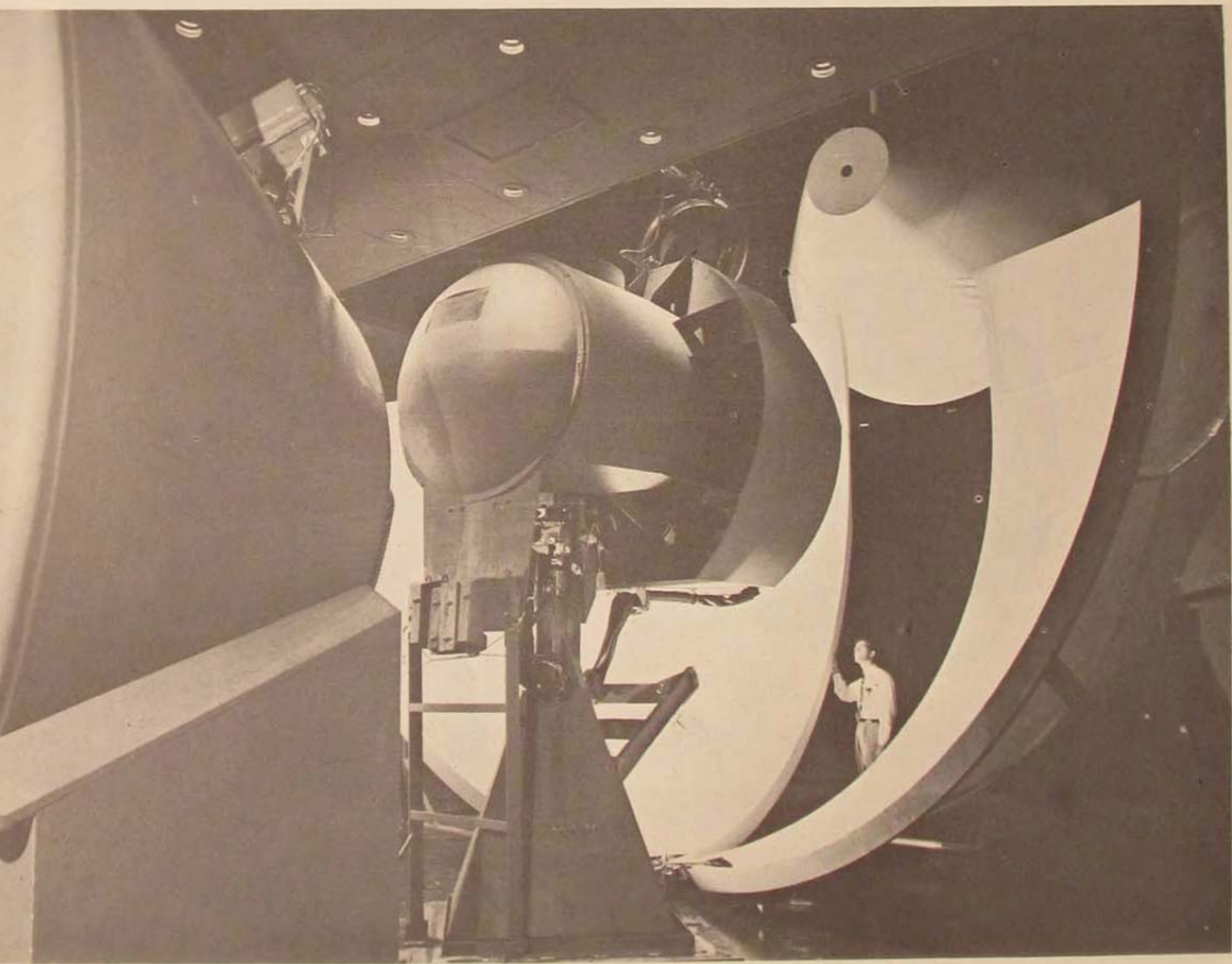


during each mission. It was necessary to deviate somewhat from mission realism so that each pilot would have an opportunity to perform each flight control phase.

retest for skill retention

After completion of the initial training and the 7-day mission, the two groups of pilots went back to a variety of flying tasks associated with their assignments as USAF test pilots. When the two groups returned for retest, after 30 and 60 days respectively, they had been without any specific related training since their last earth entry during the 7-day simulated mission. Upon interview prior to the retest for skill retention, it was determined that no formal review or discussion of test tasks or procedures occurred prior to the retest period with the exception of some individual thought on the mission, tasks, etc., which occurred immediately prior to their return to the simulation

Figure 3. Screen and command module model



facility for skill-retention retesting.

Since this portion of the study was concerned with the effects of intervening activity between the practice of tasks and their actual mission performance, the situation presented in the present study seemed ideal. First, the pilots had achieved statistical base lines on all tasks prior to the 7-day mission, and their performance during the mission was known. Second, the tasks presented to the pilots were varied in complexity and difficulty as well as representative of the gamut of tasks to be performed in space flight. Third, the tasks and simulation technique were based upon an actual system and were as realistic as possible within the constraints of the program. Fourth, it was possible to present the tasks and the dynamics of the simulation in exactly the same form as previously presented.

This study, then, was capable of evaluating skill retention and the reliability of performance in trained test pilots on tasks related

Figure 4. View of control room

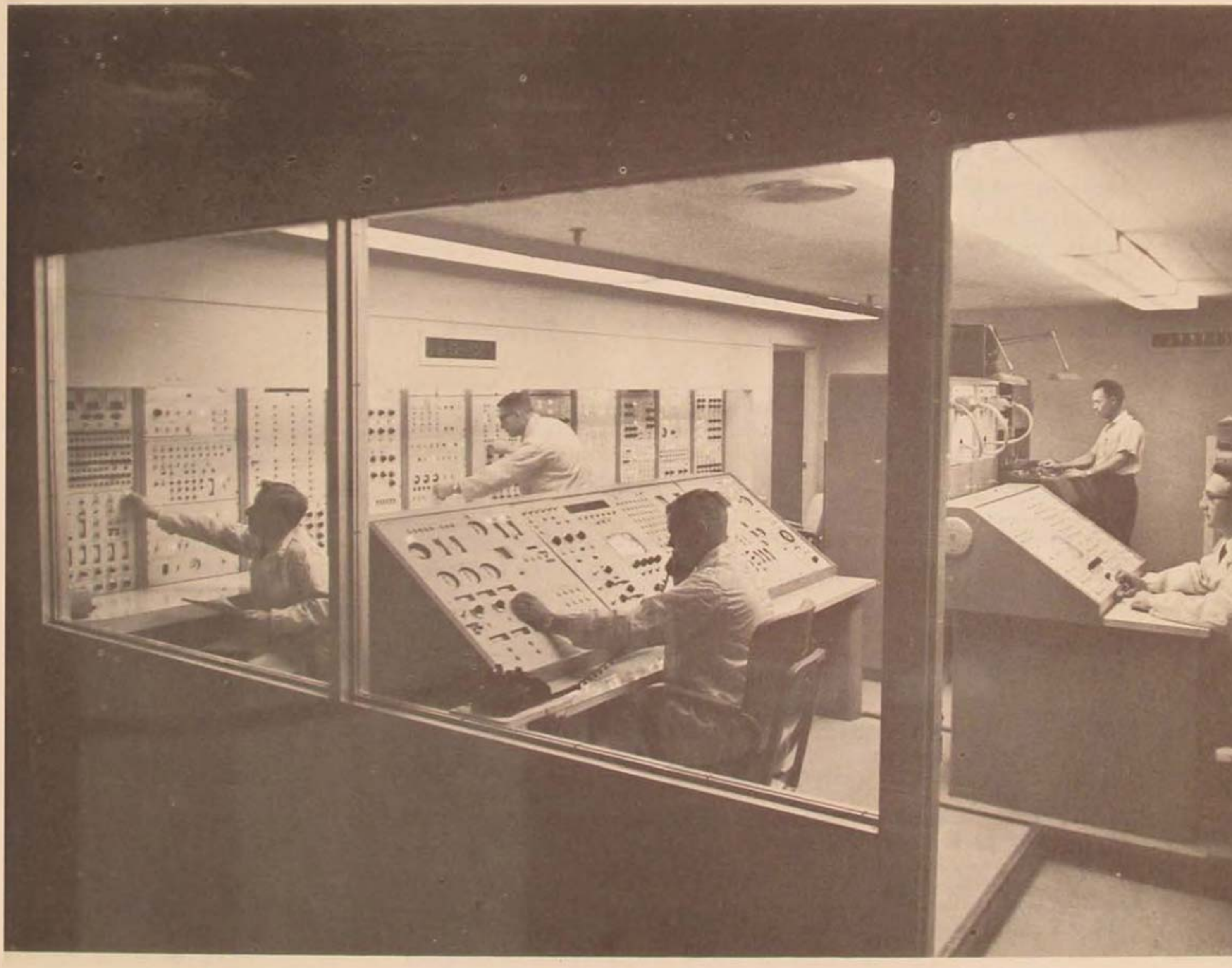


Table I. Sample Simulated Mission-Task Sequences

Total Elapsed Mission Time	Phase	Task	Crew Member Involved Pilot	 Engineer	 Navigator
11 min to 2 hr	Earth parking orbit	System checks & prep. for translunar insertion	On-duty	On-duty	On-duty
2 hr 20 min	Translunar insertion	Perform insertion including control, switching, & monitoring tasks	On-duty	On-duty	On-duty
3 hr 30 min	Lunar excursion module transposition	Perform transposition including control, switching, & monitoring tasks	On-duty	On-duty	Off-duty
5 to 11 hr	System checks & position determination	Perform switching, information handling, procedural & navigation tasks		On-duty (alternated between the 3 crew members)	
11 to 12 hr	Platform alignment & mid-course corrections	Perform switching, information handling, procedural and navigation tasks	On-duty	On-duty	Off-duty
12 to 73 hr	Same as 5 to 12 hours			On-duty (alternated between the 3 crew members)	
73 hr	Lunar orbit insertion	Perform insertion including control, switching & monitoring tasks	On-duty	On-duty	On-duty
76 hr	LEM deorbit to lunar touchdown	Perform control, switching & monitoring tasks in the lunar excursion module & monitoring in the command module	On-duty (lunar excursion module)	On-duty (lunar excursion module)	On-duty (command module)
82 hr	Lunar ascent to docking	Perform control, switching & monitoring tasks in the lunar excursion module & monitoring in the command module	On-duty (lunar excursion module)	On-duty (lunar excursion module)	On-duty (command module)
83 to 101 hr	Two complete lunar phases as performed between 76 and 82 hours				
103 hr	Transearth insertion	Perform insertion including control, switching, & monitoring tasks	On-duty	On-duty	On-duty
104 to 165 hr	Same as 12 to 73 hours				
165 hr	Earth entry	Perform entry including control, switching, & monitoring tasks	On-duty	On-duty	On-duty

to space flight which were sufficiently realistic so as to motivate the pilot to perform as well as he could. Further, it was possible to determine to some extent the amount of interference or facilitation in task performance resulting from the activities in which the pilots participated during their period away from the simulation. This then would provide a preliminary estimate of pilot reliability after periods of 30 to 60 days without practice.

The approach used to determine the retention of pilot skills was to subject each crew of pilots to two days of testing. Day 1 was an integrated fast-time mission similar to the one conducted during the later phases of each crew's training. Because of time limitations, not all phases were evaluated, but a sufficient range of task difficulty was tested so that skill retention could be estimated on this basis. Table II shows phases evaluated on Day 1.

Again there was deviation from the operational realism of the mission to provide each pilot an opportunity to perform each dynamic phase. The only refamiliarization the pilots received with the tasks was a half-hour briefing and a half-hour review of the check list for the 30-day crew and a two-hour review of the check list for the 60-day crew.

Day 2 was composed of part-task trials on the phases shown in Table III.

A debriefing at the end of the second day's testing terminated the study for each group of pilots.

results

The data obtained from the skill retention program are still in the process of being analyzed. However, a preliminary review of the data indicates a number of interesting items:

- For the 30-day group there appeared to be a minimal loss in pilot skill retention in all task categories during the integrated fast-time mission. Losses which did occur appeared to be in the switching area, but inspection of these errors also indicates that the majority of switching errors made may be considered noncritical to both mission success and pilot safety.

- For the 60-day group there appeared to be a loss in pilot retention of tasks in the complex phase of braking and hovering and in

Table II. Day 1—Fast-Time Mission Phases

Time	Phase
0830 to 0900	Pilot insertion and prelaunch check
0900 to 0930	Launch to parking orbit to system check
0930 to 1030	Preparation for translunar insertion to post-insertion check
1030 to 1145	Preparation for transposition to transposition
1145 to 1230	Lunar excursion module status check
1230 to 1330	Preparation for lunar orbit insertion to post-insertion check
1330 to 2000	Crew transfer to excursion module to landing to lunar ascent to docking
2000 to 2130	Preparation for earth entry to touchdown

Table III. Day 2—Trials

Phase	Number of Trials
Brake & hover of lunar landing	8
Docking	6
Earth entry	4
Lunar excursion module transposition	2
Transearth insertion	2

the switching tasks during the integrated fast-time mission.

- For both groups the results of the second day's testing indicated a very rapid increase in proficiency of performance in the trials presented to each pilot.

These preliminary results are based upon a cursory review of the available data. Work currently in progress will attempt to quantify more precisely the results of the skill retention in terms of the following factors:

(1) Changes in reliability of performance for both groups on each task, comparing performance during the mission phase of their respective training periods with performance during the base line.

(2) Errors in tasks will be compared on the basis of mission criticality and pilot safety, to provide an indication of the type of information and procedural loss exhibited by the crews.

A PRELIMINARY examination of the test data indicates minimal loss in the 30-day group and a more severe loss in the 60-day group. Though these results are preliminary and the number of subjects available for testing was small, the importance of these data to long-duration flights is substantial.

If these data are borne out by future testing, it would indicate that the reliability of pilots during long-term space missions is adequate when based only upon the retention of skills without practice for 30-day periods. This, then, might be a factor in the elimination of an onboard simulator for such long-duration missions. The 60-day group's apparent loss of skill, if later substantiated, puts an outer bound on the retention capability for this type of tasks.

Continued testing in a simulator situation for 30- or 60-day periods using an integrated mission technique is required for substantiation. The current data were obtained under conditions incorporating the effect of such important components of the environment as confinement in reduced volumes, artificial duty cycles, etc., which can be simulated. Those aspects of the environment which cannot be simulated, such as actual flight motivation and weightlessness, must await in-flight testing. However, continued simulation testing would provide a basis for many of the design decisions which must be made for future long-duration space flights.

Hq Air Force Systems Command

Note

1. M. A. Grodsky and J. P. Bryant, "Integrated Mission Simulation for Long Term Space Flight," *Proceedings of AIAA Simulation for Aerospace Flight Conference*, August 1963.

MANNED ORBITING SPACE STATIONS

Colonel John M. Coulter and
Major Benjamin J. Loret

IN RECENT months there has been an increasing interest in the initiation of a manned space station program as a logical next step in the exploration of space. This interest has been manifested both within the National Aeronautics and Space Administration in the initiation of various space station study programs and within the Department of Defense as evidenced by the announcement of the Secretary of Defense in December 1963 of assignment to the Air Force of development responsibility for a Manned Orbiting Laboratory (MOL) System.

This increasing tempo of space station planning activity clearly indicates that initiation of some form of a space station "hardware" program may lie in the not too distant future. The importance of such a program to the national space effort and, more specifically, its implications to the possible requirement for manned military operations in space will be vital in determining the future of the Air Force and its operational concepts. For this reason it is important to every forward-looking Air Force officer to keep informed on space station activities and to understand the role of a space station in DOD planning for possible future national defense requirements in space.

The purpose of this article is to discuss the contribution potential of a space station to the overall national space effort and to describe current NASA and DOD activities in space station planning; this background to serve as a frame of reference which will facilitate assessment of past and future events and decisions related to the subject.

where we stand

It has been more than six years since Congress created the National Aeronautics and Space Administration on 1 October 1958. In the Space Act Congress declared: "It is the policy of the United States that activities in space should be devoted to peaceful purposes for the benefit of all mankind." The national objectives established were four in number: (1) to conduct the scientific exploration of space for the United States, (2) to begin the exploration of space and the solar system by man himself, (3) to apply space science and technology to the development of earth satellites for peaceful purposes to promote human welfare, and (4) to apply space science and technology to military purposes for national defense and security. Of these tasks, NASA was charged with the first three, and the last was assigned to the Department of Defense. In his historic address to Congress in May 1961, President Kennedy added to these original objectives the specific goal of landing a man on the moon and returning him safely. It was during this address that he established space exploration as a major instrument of national policy.

The progress made toward these goals in the last six years has been impressive. The Mercury program the first step in the U.S. manned exploration of space, was highly successful in demonstrating that this nation could place a man in earth orbit, sustain him there for a period of time, and effect his safe recovery. In the course of that program a significant and comprehensive groundwork was laid and experience gained in manned earth-orbital operations, thus providing a sound technological and operational basis for follow-on programs. The Gemini program, discussed elsewhere in this issue, is an essential next step in extending man's duration in orbit and in demonstrating the capability for in-space rendezvous and docking. Both are prerequisite objectives to successful accomplishment of the Manned Lunar Landing Program (MLLP).

In view of the broad and relatively unlimited goals of the national space program established in the Space Act, the Manned Lunar Landing Program may be considered a limited, although very important, objective. Undoubtedly the MLLP will contribute to these goals in the form of fallout technology and knowledge of benefit to national defense as well as to pure scientific progress.

space station program

It is in looking ahead to progressive accomplishment of the fourfold objectives that a manned orbiting space station appears to be ideally suited. Envisioned as an experimental laboratory, a space station could make necessary and valuable contributions to peaceful exploration of space as well as to military space technology vital to national defense interests.

A space station would provide an opportunity to investigate the ability of man to withstand the stresses involved under prolonged exposure to the space environment, e.g., his ability to withstand the effects of weightlessness, long periods of confined isolation, and radiation and meteorite hazards. It would permit investigation of the effects of space environmental phenomena on metals, materials, fluids, and lubricants, while under man's direct observation and control. Of utmost importance and benefit here would be the ability to observe the integrated effect on equipment and componentry of the dynamic interplay of all the environmental phenomena simultaneously applied in true space. This capability, which cannot be identically duplicated on earth, would be of inestimable value in the design of future space hardware. A space station would facilitate the conduct of scientific observations and experiments in the fields of astronomy, geodesy, bioscience, etc., which also cannot be duplicated on earth. Various space operational techniques could be investigated by incorporation of provision for logistics resupply through rendezvous, docking, and transfer and for performance of in-space maintenance and repair.

Aside from the immediate benefits to the Nation and to the world which could be derived by including a capability for selective real-time weather reporting, for communications relay, etc., the broad spectrum of activities which could be conducted would provide the technology and experience prerequisite to further manned exploration of the solar system and beyond and to the formulation of sound decisions concerning the nature of possible future military space systems.

Although a space station program would involve a major expenditure of resources, the economic attractiveness of a manned orbiting laboratory is worthy of mention. The multipurpose use of such a laboratory in providing a universal test-bed to accommodate the wide variety of activities contemplated may well prove to be a cost-effective approach in obtaining the broad technology and experience required for future pursuit of national space objectives.

The potential role of the manned space station in furthering those objectives, as outlined above, has led to the initiation of space station study efforts both by NASA and by DOD, in coordination with each other.

NASA Orbital Research Laboratory Study Program

In pursuit of its assigned mission of peaceful exploration of space, NASA has been engaged in the past few years in conceptual design studies for several space station programs. Of most immediate interest to NASA for the near term is the Extended Apollo Program, which envisions the use of the Apollo spacecraft, modified to extend its capability to permit duration in orbit beyond that presently planned in the early earth-orbital flights of the Manned Lunar Landing Program. In addition, three other candidate programs are under study for possible follow-on to the proposed Extended Apollo Program: the Apollo Orbital Research Laboratory, the Medium Orbital Research Laboratory, and the Large Orbital Research Laboratory. These are listed in order of increasing capability as well as of increasing technological complexity and cost.

The Extended Apollo concept is one which would provide an early, although relatively limited, capability through minor modification of the Apollo spacecraft to permit its functioning as a space laboratory. The present lunar-configured Apollo spacecraft provides for a crew of three with a life-support capability of 14 days. Although satisfactory for the lunar mission for which the spacecraft was designed, the relatively small pressurized volume (360 cubic feet) limits its capability to perform a space station mission. The Extended Apollo, schematically depicted in Figure 1, envisions modification to permit extending its duration in orbit to 45 days. This could be accomplished by making available about 190 cubic feet of usable pressurized volume through elimination of one crew member. Removal of lunar-mission-peculiar tankage and propellants would provide 1000 cubic feet of unpressurized volume in the service module for experimental payloads. Feasibility studies to date indicate that with these modifications the Extended Apollo could perform as a zerogravity space station, launched into a 150–200-nautical-mile orbit using a Saturn I-B booster, with an experimental payload of 5000 pounds. Duration in orbit of the Extended Apollo could be further extended to 120 days with additional modification in the form of replacement of some subsystems, including substitution of a solar cell electrical power source for the present fuel cell system.

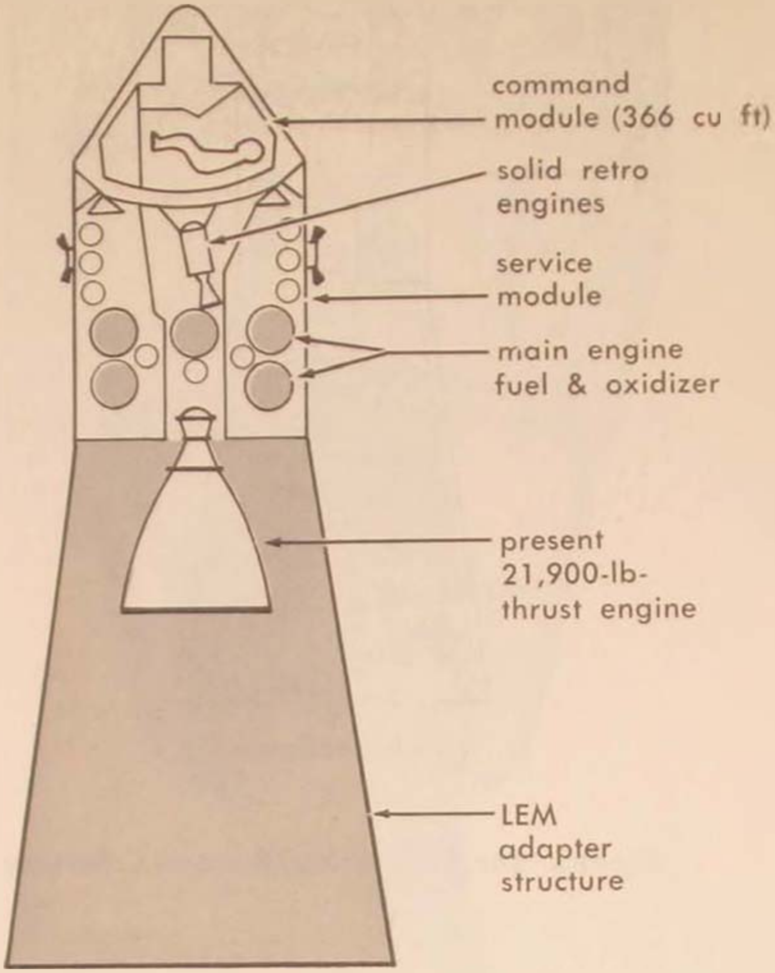
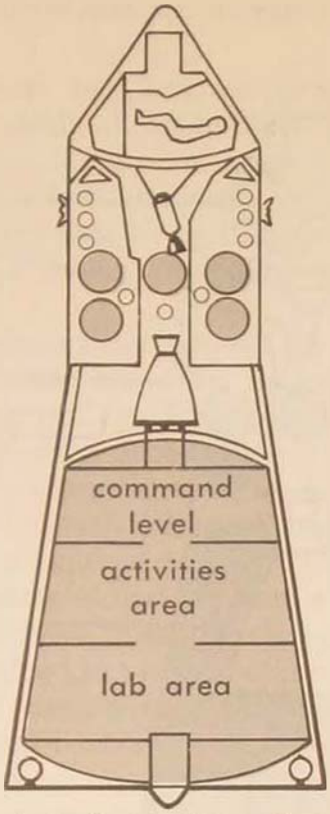


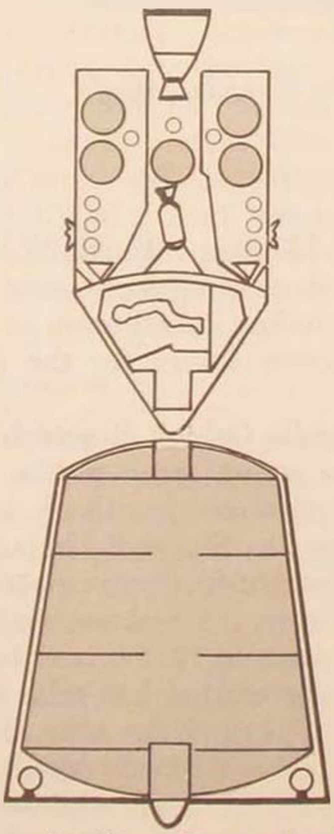
Figure 1. The Extended Apollo

The Apollo Orbital Research Laboratory (AORL) is a zero-g space station concept reflecting a logical outgrowth of the Extended Apollo. Shown in Figure 2, the AORL incorporates a 5600-cubic-foot pressurized laboratory. The greatly increased volume available would permit the AORL to be manned by a crew of six, three to be carried into orbit in the initial launch, three to enter the AORL after effecting rendezvous in a second Apollo spacecraft which would also carry additional supplies. As shown in the figure, after achieving orbit



launch configuration

Figure 2. The Apollo Orbital Research Laboratory



orbital configuration

the Apollo command module would be turned around and reattached to the laboratory to permit direct access by the crew between the Apollo command module and the laboratory. The second Apollo would dock at the other end of the laboratory. It is envisioned that the AORL could function for at least a year, with resupply and exchange of crew members effected every three months by use of additional Apollo ferry vehicles.

The Medium Orbital Research Laboratory (MORL) is similar to the AORL, the main difference being that the MORL would be launched unmanned. Without having to boost the Apollo spacecraft, the Saturn I-B could launch a larger laboratory carrying more elaborate equipment, with increased radiation protection and with provision for a hangar for docking, repair, and unloading of ferry vehicles. The six-man crew would be ferried to the space station by either Apollo or Gemini vehicles. Figure 3 shows the laboratory as configured in orbit with one Gemini vehicle in a docked position. As can be seen, the laboratory is of spherical shape. The 22-foot diameter provides for two compartments, the top one to be used as living quarters, the lower one to function as the laboratory. Although the MORL is conceived as a zero-g system, artificial gravity could be incorporated by rotating the space station with a system of cables connected to a countermass consisting of the expended upper stage of the launch vehicle, as shown in Figure 4. Also as shown, ferry vehicles would be docked at the laboratory at all times to permit abandonment of the space station in case of emergency.

The Large Orbital Research Laboratory (LORL) is designed to maintain a crew of 24 astronauts in a 260-nautical-mile orbit for periods of five years; as such it may be considered more or less as an ultimate, permanent-type earth-orbital space station (Figure 5). It is gigantic in size, with a diameter of 150 feet, a total volume of 67,000 cubic feet, and a weight of a quarter of a million pounds, thus requiring a two-stage Saturn V booster as the launch vehicle. The large size would require that the space station be launched in a "folded" configuration and, once in orbit, extended to the shape shown. The radial spoke configuration shown is one of several similar ones being investigated. The entire station could be rotated to provide artificial gravity in the spoke-like compartments; the hub would contain a large zero-g laboratory. The LORL would be logistically resupplied at periodic intervals by enlarged Apollo-like vehicles carrying from six to twelve personnel or by use of ferry vehicles of lifting-body configuration, as shown in the figure.

Taken in total, it is evident that the NASA space station study program encompasses the entire spectrum of space station capabilities from the small, with limited capability and orbital lifetime, to the large, with extensive capability and lifetime. As such, the results of the NASA studies under way will provide a range of alternatives to support any future decision that may be made concerning development of a space station system in furtherance of the multifaceted goals of NASA under its responsibility for peaceful exploration of space.

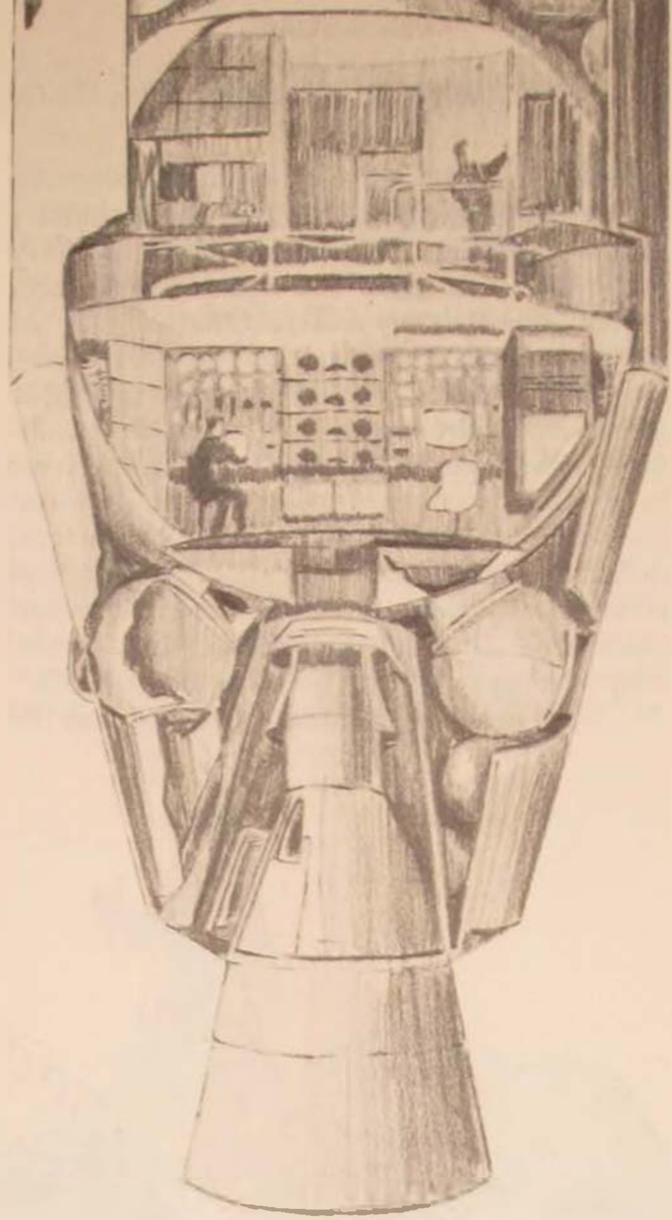


Figure 3. The Medium Orbital Research Laboratory

Department of Defense space station program

The military manned space station program had its genesis in space station planning studies initiated as early as 1958 as part of the then-proposed USAF space program. The intervening years witnessed a continuation, at a relatively low level of effort, of conceptual studies investigating various space station configurations and military uses to which a space station could be put. The slow progress made toward initiation of such a program can be attributed primarily to the lack of a validated requirement for the presence of military man in space, particularly in view of the Nation's dedication to peaceful use of space. Although a firm requirement for a manned military space capability has still not been conclusively demonstrated, the rapid technological strides made, the expanding capabilities demonstrated in space, and the increasing tempo of manned activities in space, both by the United States and the Soviets, have led to a reconsideration of possible future national requirements which may arise in the area of military defense operations in space. The renewed interest raises the question of the role which man can or will play in these operations, should they materialize. It is in the early assessment of man's military role in space that a military-sponsored space station is eminently suited.

These considerations, i.e., provision for national defense preparedness in the area of space operations and, specifically, investigation of man's utility in these operations, are among those which contributed to the Secretary of Defense's decision in December 1963 to have the Air Force initiate a space station program.

The role of a space station in the military space program is one of evaluating military man's utility in space and measuring his utility both qualitatively and quantitatively. There are several questions that must be answered. Can man perform military tasks effectively in the space environment? What are these military tasks? Exactly how well can man perform in space as compared to his known performance on earth and in aircraft? Can he perform military tasks in space more effectively and economically than could be done by use of automated equipment launched into space, controlled remotely by man on the ground?

The answers to these questions can be determined in a three-step approach. First, military-type experiments must be devised which insofar as is practicable can be conducted initially on the ground and in aircraft, to establish base-line data against which to compare in-orbit test results. These experiments must be oriented around the man to ensure that man's performance rather than equipment performance is the measured variable. Therefore, this approach logically emphasizes the use of already developed and proven equipment rather than newly developed and untested hardware. Second, the experiments must be conducted in the space environment in accordance with predetermined test procedures and specific test objectives. Finally, the

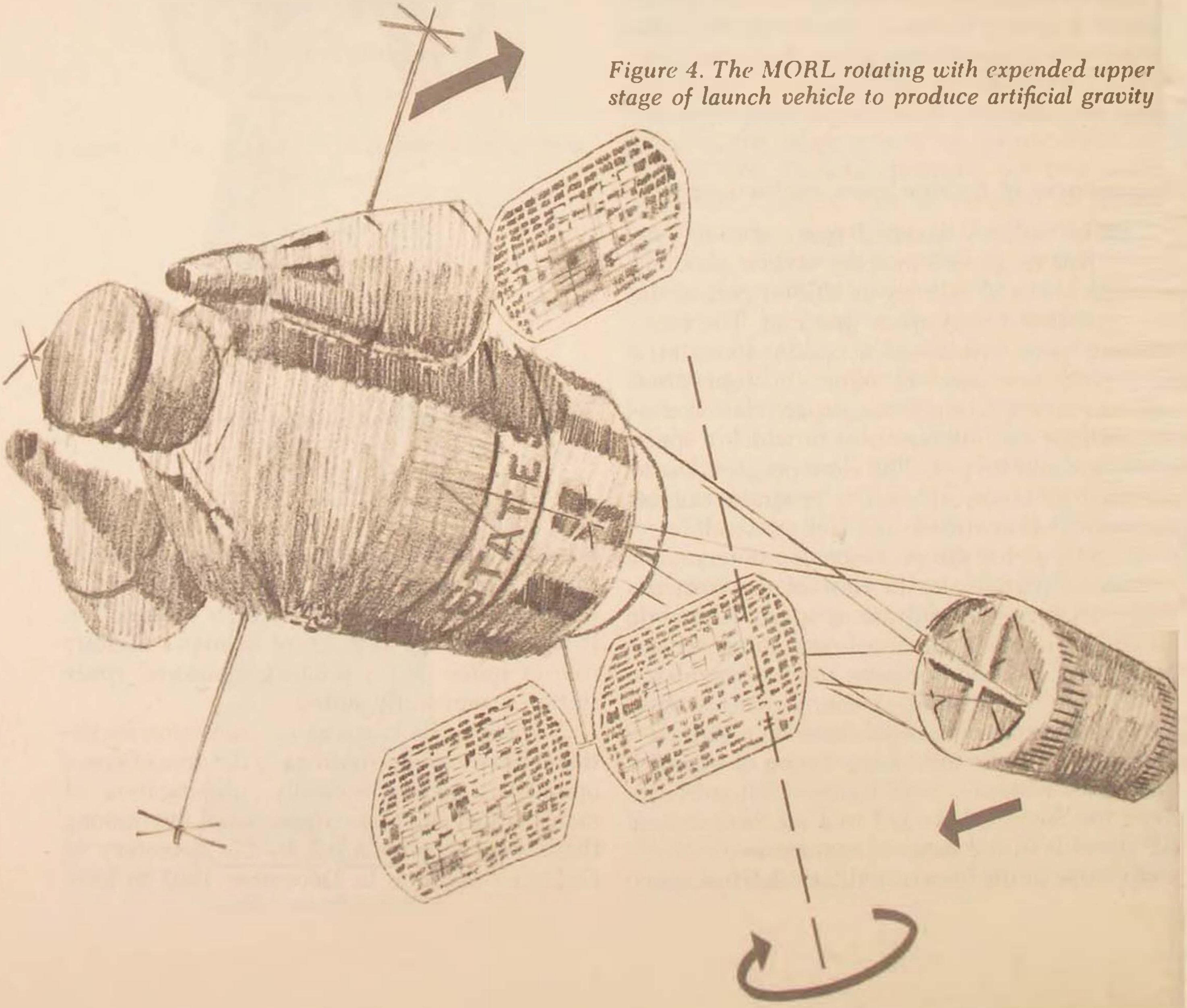


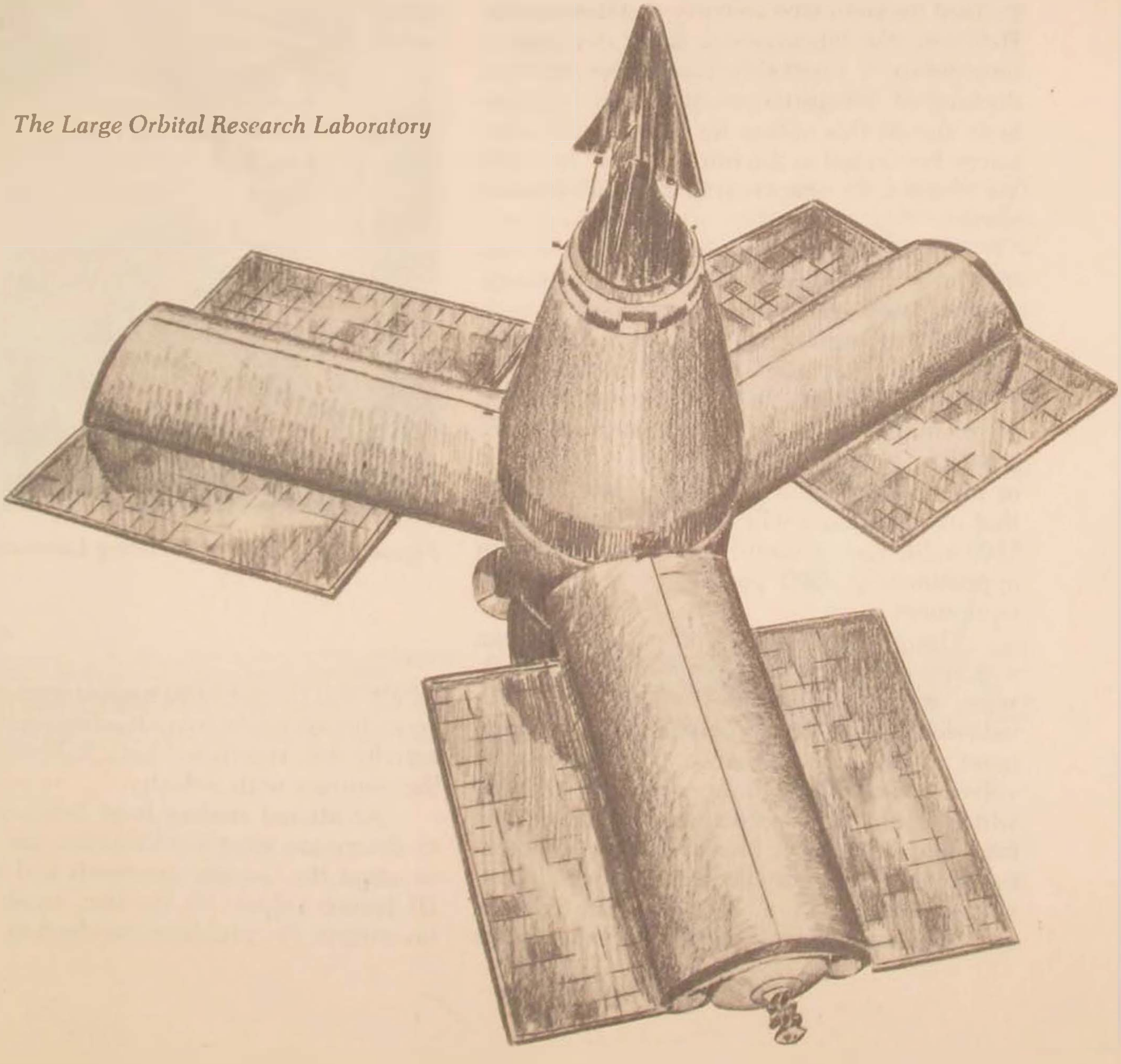
Figure 4. The MORL rotating with expended upper stage of launch vehicle to produce artificial gravity

results obtained must be analyzed to determine the quality of man's performance in the man/machine system. The overall performance of the man/machine combination can then also be compared against results which could be expected from use of highly automated equipment alone in performing the same tasks. The latter evaluation can serve as the basis for decisions as to whether future military space systems, if required, should be manned or unmanned, or conceivably a mixture of both.

The Manned Orbiting Laboratory Program will encompass and accomplish the first two tasks, i.e., it will provide the data and results prerequisite to performing the third task of analysis leading to formulation of valid conclusions concerning the requirement for man's presence in military space systems of the future. It should be apparent from the foregoing that the Manned Orbiting Laboratory is envisioned as a spaceborne test-bed; it will not provide a military operational capability, nor will it represent any prototype military operational system, although certainly its potential for growth into a military operational system will not be overlooked.

As described by the Secretary of Defense in his 10 December 1963 public release, the MOL system was to consist of the Titan IIIC booster, a cylindrical laboratory "approximately the size of a small house trailer," and a modified Gemini spacecraft, the Gemini B, as the

Figure 5. The Large Orbital Research Laboratory



personnel carrier for a crew of two. The entire vehicle could be launched into orbit, after which the laboratory and Gemini B would be detached from the Titan III booster. Once the orbit was established, the crew could transfer from the Gemini B to the laboratory. There, in a "shirt-sleeve" pressurized environment, the crew would remain for a 30-day period, during which the military and scientific experiments would be conducted. Upon mission completion, or in case of emergency, the crew would return to the Gemini B, separate from the laboratory, re-enter the earth's atmosphere, and be recovered in a preplanned ocean area using a recovery technique similar to that planned in the NASA Gemini program. The laboratory module could be abandoned in orbit, possibly to be programed for command re-entry and destruction. However, the laboratory is being designed to incorporate a capability for rendezvous and docking of a logistics resupply and ferry vehicle should this option for reuse of the laboratory be elected in the future. Initial launches are planned for execution in the 1967–68 time period.

A typical layout and operation of the MOL in orbit is illustrated in Figure 6. The arrangement shown is one of several concepts presently under consideration, providing for a laboratory area and a living or off-duty area, separated by a pressure wall which would permit the sealing off of one of the compartments should pressure loss occur due to meteorite penetration or other catastrophe. Studies to date indicate that the laboratory will provide approximately 1500 cubic feet of useful volume and allow for approximately 4500 pounds of experimental equipment.

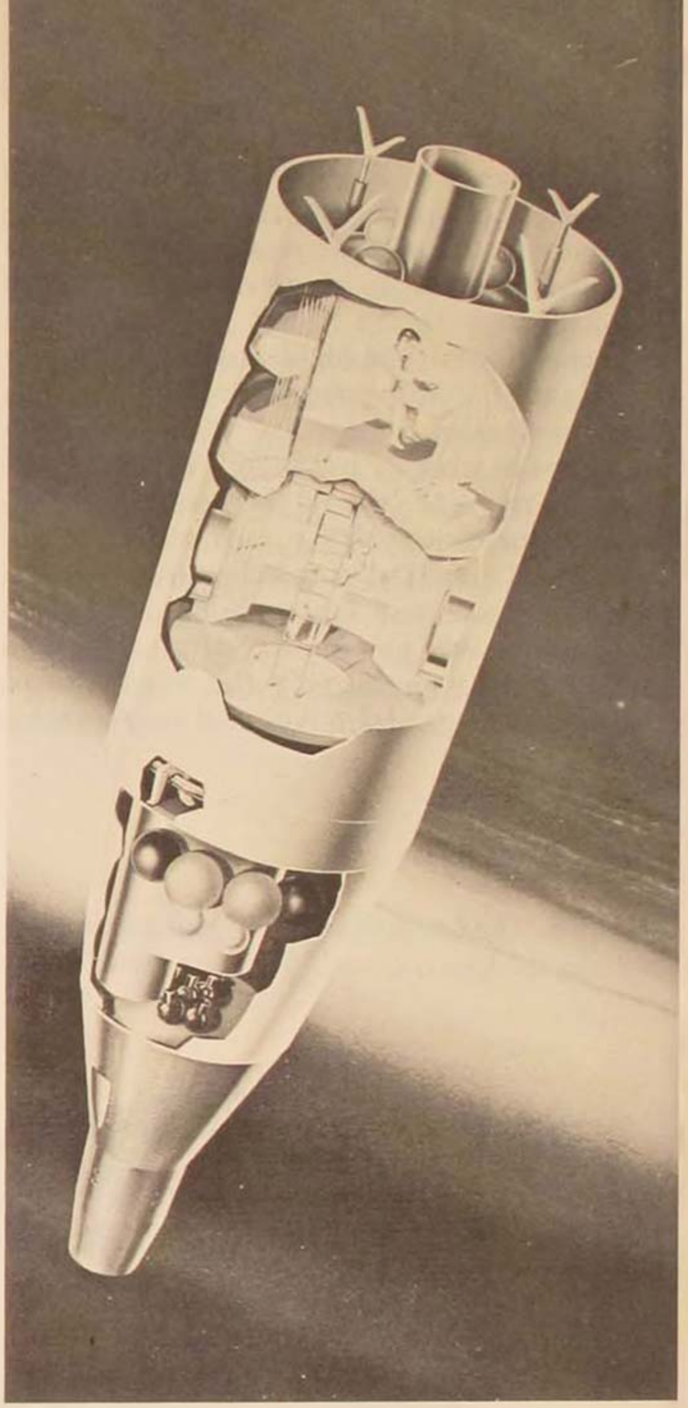


Figure 6. The Manned Orbiting Laboratory in orbit

The primary task, initiated last summer, was the definition of experiments, test hardware, and test procedures which would provide clear insight into and quantitative measurement of man's performance. This process involved consideration of those military missions which could most effectively be conducted from and in space, a breakout of these potential missions into specific functions which man might perform in a man/machine system, and the synthesis of particular experiments which will permit valid measurement of man's contribution in the man/machine system. This activity was conducted both in-house and under contract with industry.

Additional studies have been conducted to determine what modifications are required to adapt the Gemini spacecraft and the Titan III launch vehicle for the MOL mission and to investigate the problems involved in integrating these major subsystems into the overall MOL system. Other contracts were awarded to industry for the study of specific laboratory vehicle support subsystems such as environmental control, vehicle stabilization and control, electrical power generation, etc.

In the interest of overall program economy, all of the preprogram definition-phase activity has been based on making maximum use of existing hardware and facilities. Of particular importance in the effort is the formulation of plans to make optimum use of existing and planned NASA and DOD facilities for the MOL communications and tracking network and mission control center.

The scope of the program under which the above activities were conducted has recently been modified by the Secretary of Defense to incorporate additional objectives. As outlined by the Secretary on 23 January 1965, the MOL Program is to encompass development of technology to improve the capabilities for manned or unmanned operations of military significance. Also included is the development and demonstration of manned assembly and service in orbit of large structures with potential military applications, and possibly intermediate steps toward operational systems. Concurrent with the announcement was the release to industry of requests for proposals for design studies to assist in developing the cost and technical information required to proceed with full-scale development of the MOL system.

The Secretary's announcement and the renewed and increased planning and study effort presently under way point to a space station decision sometime during the current year.

Because of its scope and the resource expenditures involved, it is clear that any decision to embark upon a space station development program will be an important one. It is no less clear that the decision is a national one to be made at the highest levels of the Government. There is an evident duality in the requirement for a space station: by NASA in pursuing its mandate for peaceful exploration of space and by DOD in providing for national security in the near-earth space arena. Clearly there would be a beneficial fallout both to peaceful scientific progress in space and to military space capability from any space station program that might be initiated.

OUR PURPOSE has been to provide some insight concerning the role of the space station in the overall national space program and to describe space station planning efforts that are under way. This information should enable the reader to follow space station program developments, activities, and decisions as they unfold and to assess their significance meaningfully to the Nation and to the Air Force.

Hq Air Force Systems Command

Bibliography

1. Bleymaier, J. S., Brigadier General, USAF. "Mapping Out the MOL," *Air Force and Space Digest*, June 1964.
2. "Historical Origins of N.A.S.A.," NASA Historical Report No. 6 (revised). Office of Educational Programs and Services, NASA, Washington, D.C., 4 October 1962.
3. News Release No. 1556-63. Office of Public Affairs, Department of Defense, Washington, D. C., 10 December 1963.
4. News Release No. 42-65. Office of Public Affairs, Department of Defense, Washington, D. C., 23 January 1965.
5. Yarymovych, M. I. "Laboratory in Space." Address presented at First Space Congress, Canaveral Council of Technical Societies, Cocoa Beach, Florida, 20-22 April 1964.

LIFE SUPPORT IN SPACE OPERATIONS

Brigadier General John M. Talbot, USAF (MC)

THE CURRENT operational life-support systems and some developmental approaches to life support for manned space flight are the subject of this article. Depending mainly on length of mission and crew size, life-support systems fall into two basic classes: the all-expendable, nonregenerative systems and the partially regenerative systems. To date and through the NASA manned flight series, including the Manned Lunar Landing Program, the United States will rely on nonregenerative life support. However, for future missions exceeding 30 days the payload weight trade-offs make the principle of regenerative life support attractive. Thus in the Manned Orbiting Laboratory Program some of the life-support subsystems will have regenerative features.

Two very important problem areas that will have strong influence on space vehicle design for the future are weightlessness and penetrating space radiations. Reliable elucidation of the effects of prolonged weightlessness on man clearly requires extensive experience with the real thing. With regard to the effects of penetrating radiations on man, we are better informed. Nevertheless the qualitative differences between the effects of solar protons, for example, and the more familiar gamma radiation are only beginning to be defined. A great need also exists for better physical definition of the space radiation environment for those regions of space we wish to explore.

In a liberal sense, life support is a very broad area. It extends well beyond the restrictive concept of oxygen supply, carbon dioxide removal, and atmospheric pressure and temperature control. Its scope as broadly defined for this article is indicated by Table I.

environmental control systems

The Mercury spacecraft environmental control system is a good point of departure for this review. It was of the open or nonregenerative type. It provided a 100 per cent oxygen atmosphere at 5.1 pounds per square inch absolute (psia), good for approximately 32 hours. In the event of cabin decompression the system would maintain 3.5 psia in the pressure suit. There were actually two subsystems, one for the pressure suit and one for the cabin atmosphere. (See Figure 1.)

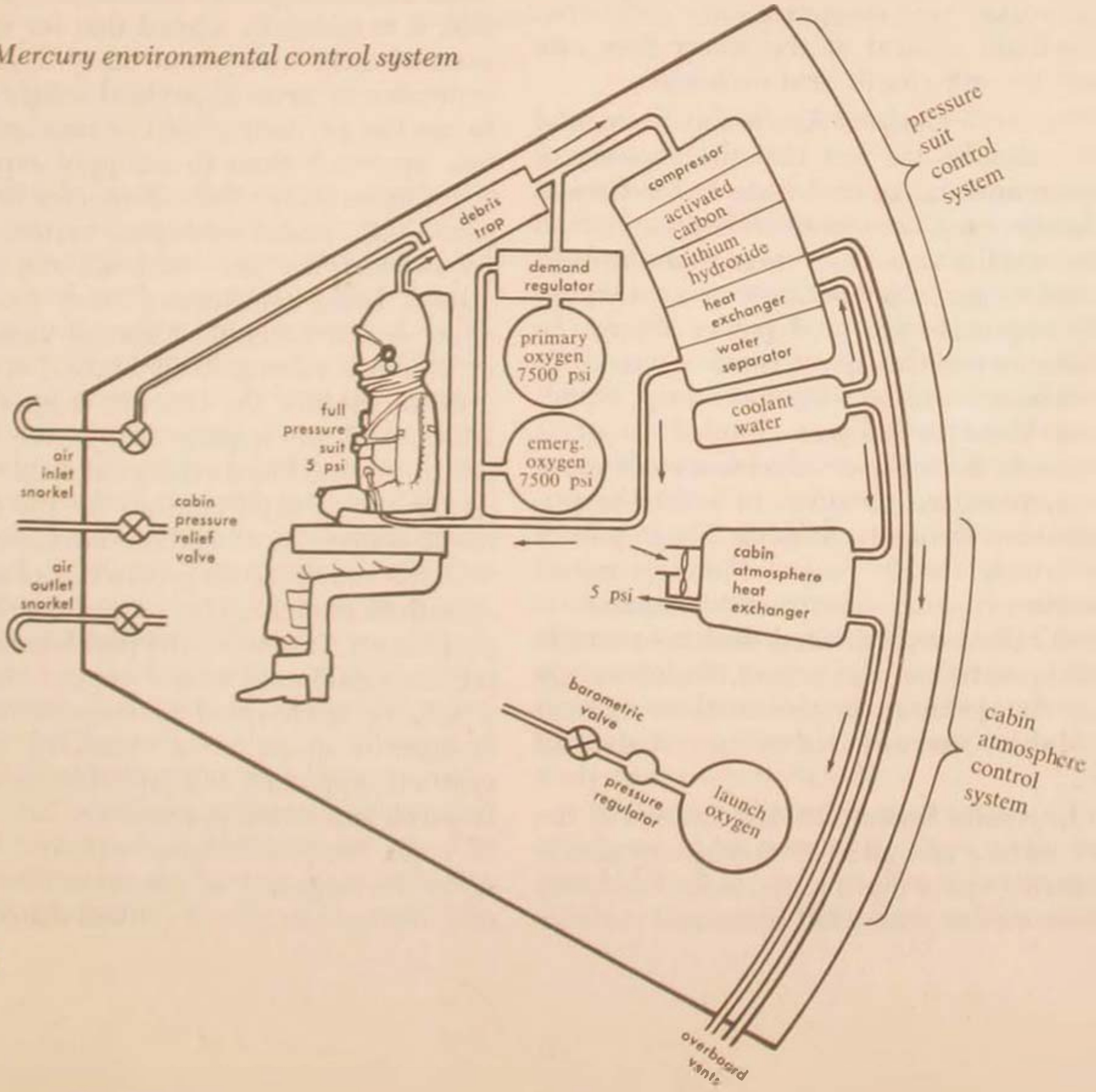
Pressure Suit Control System. Purified and cooled oxygen entered the pressure suit at the torso, ventilated the body and limbs of the astronaut, and flowed into the helmet. An exit connection at the helmet directed the gas flow through a solid-particle debris trap and into the compressor section, which maintained a flow rate of 10 cubic feet per minute (cfm) at 10 inches water pressure. The gas then passed through a chemical canister containing one pound of activated carbon for contaminant and odor control and two 2.6-pound beds of lithium hydroxide for removal of carbon dioxide. The purified gas then passed through the heat exchanger, emerging at about 45°F. The heat exchanger was of the water-evaporative type, rated at 1000 BTU/hour. Superheated water vapor from the heat exchanger was vented overboard. The last stage of the system was the water separator, which consisted of a vinyl sponge to absorb the condensed water droplets in the gas stream and an oxygen-driven piston to squeeze the sponge periodically.

Consumed oxygen was replaced from the primary supply through a demand regulator. As a safety feature an emergency oxygen supply would, in the event of a primary system failure, feed oxygen to the pressure suit and thence

Table I. Scope of Life Support

Environmental Control		*Biologistics*
natural space environment	*artificial environment*	
hard vacuum	gaseous systems	metabolic needs— nutrition water oxygen work-rest-sleep
penetrating radiation	temperature control	personal hygiene
visible, thermal, ultraviolet radiation	radiation shields	anti-isolation
temperature extremes	acceleration— launch, re-entry, maneuver, artificial gravity, vibration	preventive medicine
isolation	noise	medical aid
	waste management	

Figure 1. Mercury environmental control system



through an exhaust port in the pressure suit regulator to the cabin.

Cabin Atmosphere Control Circuit. The cabin atmosphere control circuit provided a circulating atmosphere of 100 per cent oxygen at 5 psia and 70°F ± 5°F. The source of oxygen was the primary supply used as well for the pressure suit circuit. The cabin circuit included its own heat exchanger, which delivered a stream of cooled oxygen to the cabin at the electronics bay. Following launch, a separate launch oxygen supply was used to purge the cabin and establish the cabin pressure at 5 psi.

In normal operation, since the cabin and the pressure suit circuit pressures were equal, the astronaut could open his helmet visor in flight. During re-entry, at the 20,000-foot level, barometrically controlled valves opened the snorkel inlet and outlet lines, which provided ambient air for breathing and ventilation for the remainder of the mission.

In Project Mercury, this all-expendable system worked well except for some difficulties with in-flight control of the water flow rate through the suit circuit heat exchanger.

Projects Gemini and Apollo Environmental Control. Besides the fact that the capacity of the environmental control systems for Gemini and Apollo must be considerably greater than in Mercury, the main differences from the Mercury system are in temperature control equipment, oxygen storage, and power supply. In both Gemini and the Apollo command module, primary heat exchange is by closed-loop, liquid-transport heat exchangers coupled to space radiators. In both these vehicles water-evaporative systems are provided to assist the primary systems during peak loads. The step from the relatively simple Project Mercury water-evaporative cooling system to this combination requires rather sophisticated design to ensure perfect operation of the system. Its advantage is that the primary system coolant is nonexpendable, thereby reducing the overall weight.

The Apollo Space Suit. Exploration of the moon's surface as well as extravehicular operations during space flight requires an insulating full pressure suit with a self-contained portable life-support system (PLSS). Four hours of complete environmental control for the temperature extremes and vacuum expected on the moon can be realized with a 50-pound system. This does not include the weight of the pressure suit and helmet assembly. Carried on the spaceman's back, the PLSS circulates cooled oxygen at 3.7 psi and at 17 cfm through the space suit. (Water cooling, now under intensive study and development, has proved more efficient and comfortable than oxygen cooling.) The present system reprocesses the used gas exhausted from the suit by removing carbon dioxide, other contaminants, heat, and excess moisture. Its design heat-exchange rating is 1570 BTU/hour, which includes 930 BTU of metabolic heat and 640 BTU/hour of system-generated and external heat loads. The PLSS may be recharged with expendables from supplies carried in the lunar excursion module.

Advanced Environmental Control Systems. Depending on available power and crew size, it is generally agreed that for space missions of more than about thirty days it is less expensive in terms of payload weight penalties to use the partially closed or regenerative system approach than to resupply expendables. The theoretically ideal solution for life support is the fully closed ecological system in which no expendables are used, all required substances being regenerated from the products of consumed materials. Although such a system in practical form is well beyond the present state of the art, the concept is good as a research and development objective. One approach to the closed ecological system is based on the use of algal cultures for the photosynthetic conversion of carbon dioxide to oxygen, with the simultaneous production of additional quantities of algae. This concept has had many proponents. However, the problems of designing an algal-based closed ecosystem, or even a reliable space-rated photosynthetic gas exchanger for spacecraft, are such that the photosynthetic approach will probably remain in the research and development stages for a number of years. Instead, reliance will first be placed upon development of physicochemical methods for regeneration of critical materials.

Table II. Closing the Loops in Life-Support Systems

subsystem	*Mercury, Gemini, Apollo*	*long-range spacecraft*
carbon dioxide removal	lithium hydroxide, non-regenerative system	regenerative molecular sieves backed by LiOH emergency equipment
oxygen reclamation from CO_2	none	Sabatier reaction or solid electrolyte cell + catalytic CO_2 converter
waste water reclamation	none	distillation subsystem
noxious contaminants control	activated carbon canister	catalytic burner (e.g., Hopcalite burner for CO)
spacecraft cabin temperature control	water-evaporative subsystem (Mercury) + closed-loop liquid transport with space radiators	closed-loop subsystem with space radiators and refrigeration

Meanwhile a number of environmental control system loops are being closed by important engineering developments. Perfection of the types of environmental control subsystems listed in the right-hand column of Table II will permit the design of practical semiclosed regenerative systems for truly long-range space missions lasting from months to a year or more.

An advanced environmental control system for five men on a six-week mission is shown in Figure 2. This is a two-gas atmosphere system instead of 100 per cent oxygen. It requires an independent power supply such as a solar cell system or auxiliary nuclear-electric system.

For carbon dioxide control, a most promising approach is the use of regenerative molecular sieves. Crystalline alumino silicates of alkali and alkaline earth metals known as synthetic zeolites have been prepared with myriad uniform channels in the crystal lattice. The pore diameters are such as to have selective adsorptive properties for the CO_2 molecule. An important characteristic of these materials is their ability to be desorbed by exposure to the vacuum of space, by heating the bed, or by other techniques such as purging with another gas of different molecular dimensions. Hence, if the environmental control system includes a device for reclamation of oxygen from CO_2, the molecular sieve could be one of the CO_2 collecting devices in the system. One type of regenerative CO_2 removal system is shown in Figure 3, taken from AFSC Manual 80-9 (20 February 1964).

Several other approaches are under investigation, including the reaction of CO_2 with silver oxide to form silver carbonate, followed by regeneration of the silver oxide and recovery of oxygen. Another is the ion exchange electrodialysis of carbon dioxide.

Oxygen reclamation may take one of several forms. Two procedures involve the reduction of CO_2 by hydrogen. The Sabatier reaction combines CO_2 with hydrogen to form methane

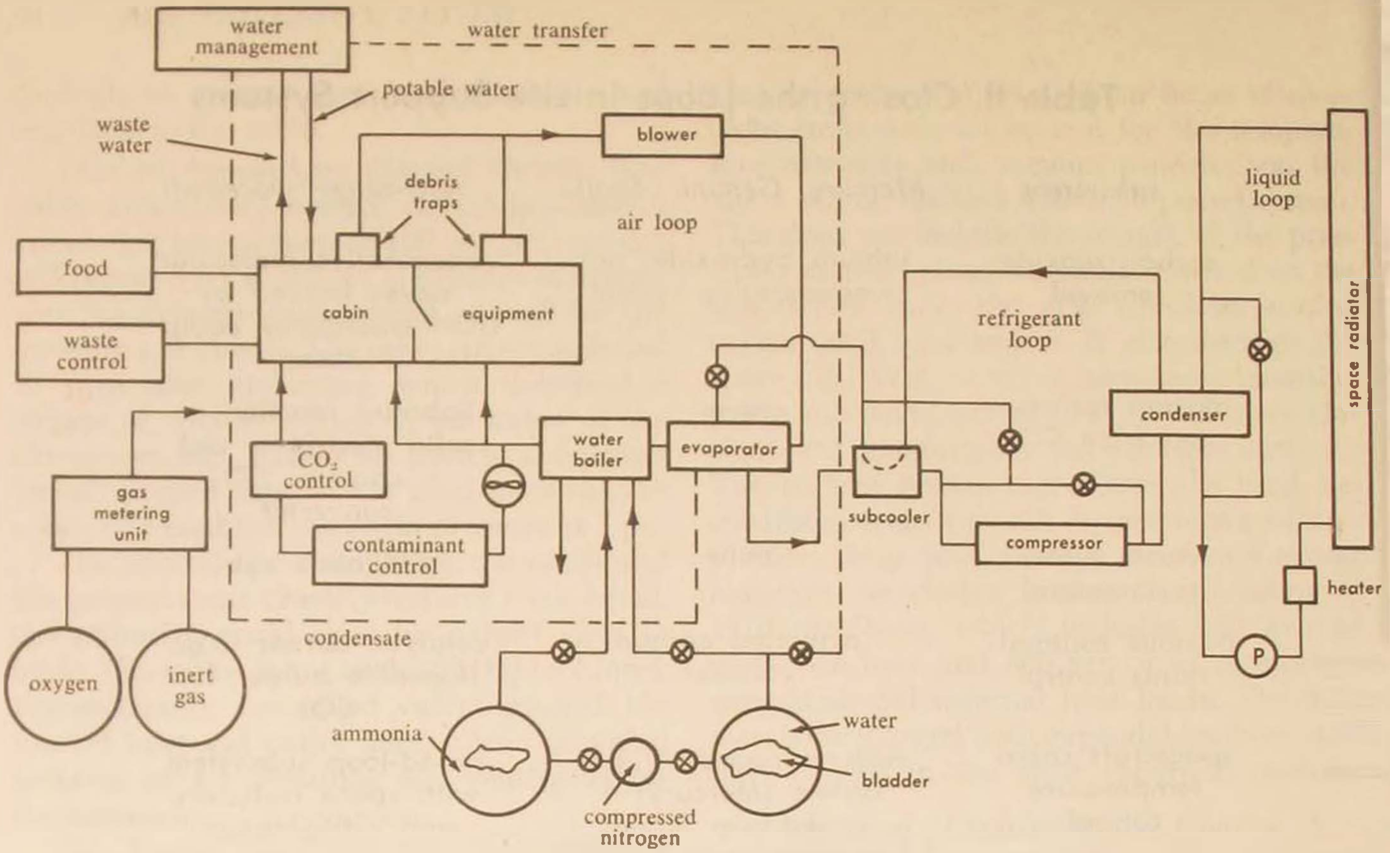
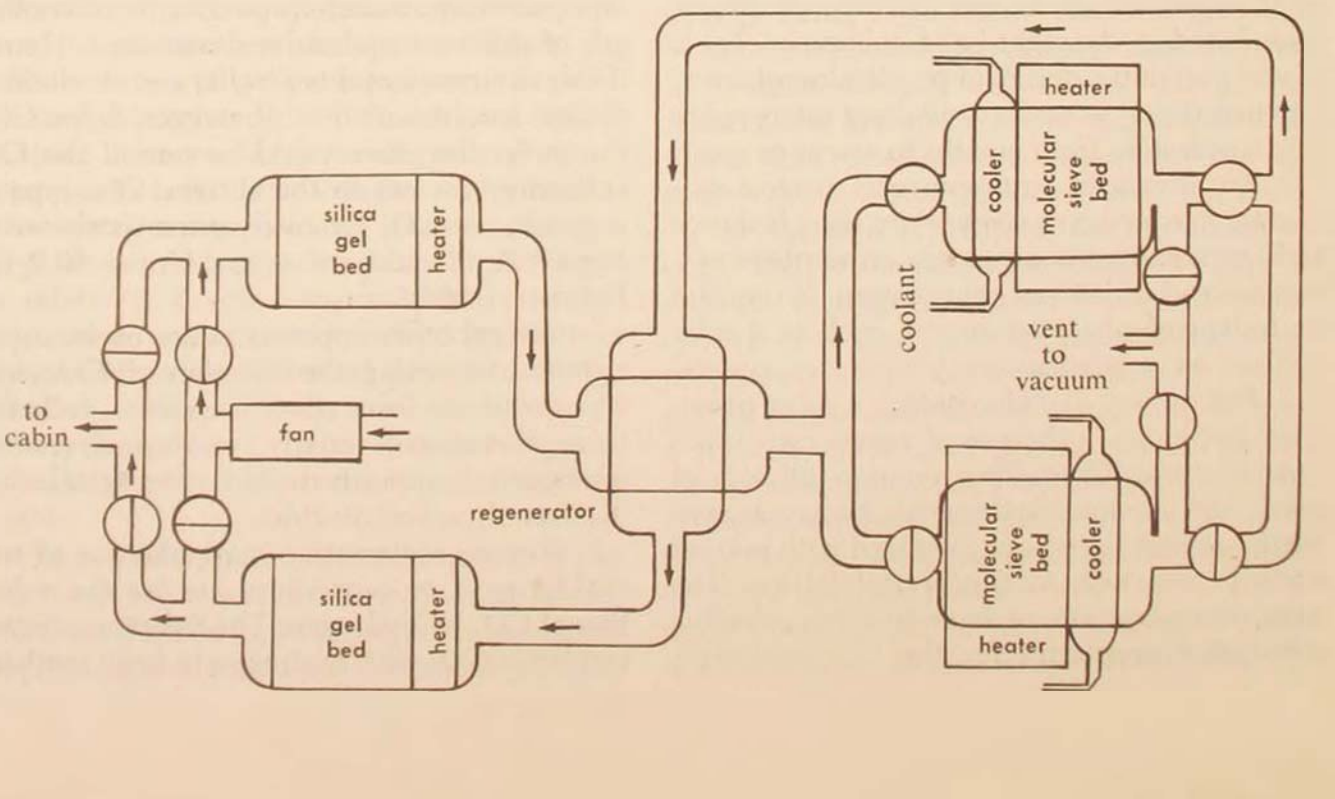


Figure 2. Active thermal and atmospheric control system, cabin and equipment in parallel

Figure 3. Regenerative carbon dioxide removal system



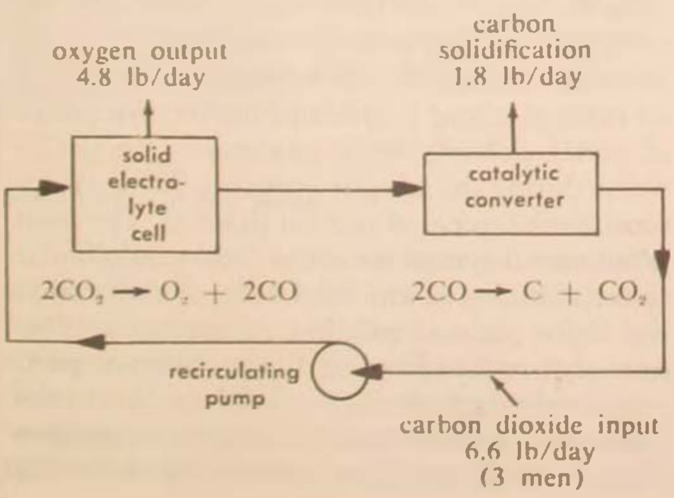
and water ($CO_2 + 4H_2 \rightarrow CH_4 + 2H_2O$). Electrolysis of the water yields oxygen and hydrogen. The methane is converted to carbon and hydrogen by pyrolysis. The hydrogen from the two processes can be recycled or could be diverted to fuel cells or other uses. The Sabatier reaction requires substantial electrical energy.

Another process which also is costly in electrical power is the hydrogen reduction of CO_2 directly to carbon dioxide and water followed by electrolysis of the water as in the Sabatier reaction.

Oxygen reclamation using the solid electrolyte cell offers the advantages of requiring no hydrogen and eliminating the liquid-gas interfaces of the processes described above. In the solid electrolyte approach, CO_2 passes through a cell in which yttrium and zirconium oxide strip one atom of oxygen from the CO_2. The resulting CO is then passed through a catalytic reactor in which two CO molecules convert to carbon and carbon dioxide. The CO_2 is then recycled. (See Figure 4.)

These oxygen-reclamation techniques are meant only as examples. As in most developmental efforts, the method which proves the most efficient, most reliable, least costly in power and weight, and most flexible in terms of system integration will be the top competitor for long-range environmental control systems.

Figure 4. Solid electrolyte oxygen reclamation system



Water reclamation has progressed to the point that practical recovery units have been designed. The waste water to be considered includes urine, water vapor from exhalation and perspiration, and wash water. Even though feasible, recovery of fecal water is not required for missions as long as two or three months. Tests have shown that change-of-state processes such as liquid → vapor → liquid yield the purest water from waste sources. The most promising approach appears to be compression distillation, which can function independently of vehicle orientation. Coupling with an activated charcoal filter increases the palatability of the recovered water. Another approach is the activated sludge process. A biological sludge system provided pure water for five men for thirty days in the recent NASA–Boeing advanced life-support ground-based run. The system is designed to operate for a one-year period.

Other facets of environmental control which have important biomedical considerations include space radiation shielding, meteoroid protection, launch and re-entry and landing accelerations, and the influence of weightlessness. Theoretical knowledge and engineering design data for spacecraft shielding are well advanced; however, the dilemma of shield weight versus payload capability remains serious in any manned space system based on near-future state of the art for propulsion systems. Three significant developments are needed for partial alleviation of this problem: first, a substantial increase in launch vehicle payload capabilities so as to afford greater shield weights; the acquisition of more reliable and extensive data on the space radiations to be encountered for all proposed space missions, including cislunar and interplanetary; and the development of reliable means of predicting solar flares and solar proton events sufficiently in advance to permit appropriate mission scheduling and/or evasive astronavigation.

Electromagnetic and/or electrostatic shielding against charged particles in space holds some developmental promise for the future.

Meteoroid Hazard. Estimates of probability of spacecraft penetration by a meteoroid vary widely,[1] but all authorities agree that on a statistical basis during long-range missions there exists a definite hazard of being struck by a meteoroid. For example, it has been estimated that a spacecraft of 1000 square meters surface with aluminum skin .070 inch thick will be penetrated once in 24 hours; with skin thickness of .1 inch, penetration would be once in 120 hours.[2] Even without penetration, a meteoroid striking a spacecraft may cause interior damage by collisions with spalled hull material. When penetration occurs, a life-support emergency may happen, not only from loss of cabin pressure but also from an oxidative explosion of finely disintegrated meteoroid and spalled material and vaporized wall material within the oxygen-bearing atmosphere. The oxidation may be associated with an intense flash of light and a transient high-temperature pulse. Thus five individual hazards may occur from a penetrating meteoroid: loss of cabin pressure, flash blindness, flash burns, mechanical injuries from scrap or spalled material, and spacecraft fires. Tests have shown that it is practical to shield spacecraft by means of extra outer layers of thin metal skin separated from the primary hull. In addition, it appears practical to include self-sealing material between the hull layers. These remarks concern small meteoroids, from a few microns to one or two millimeters in diameter. Present estimates indicate that there is no feasible way of designing to protect against the unlikely collision with a large meteoroid.

Acceleration Protection. Human tolerance for the accelerations of launch and re-entry of space vehicles and mission profiles through Apollo is well established from previous research, test, and manned space flight experience. Support of the body in a supine position during launch and re-entry provides the necessary antiblackout protection. Water-landing impacts during normal parachute descent of a spacecraft impose no undue stresses. However, impacts with hard earth in a Mercury or Apollo type spacecraft require that the astronaut support structures be designed for shock attenuation. Further test and space flight experience may show that additional protection must be provided for abnormal directions of impact forces, such as from the side or head to foot, should the parachute landing be complicated by abnormal swinging. The USAF has a vigorous program of research and test to define human limits for impact accelerations in which forces are applied in unusual directions.

Because a good argument may be made for artificial gravity in long-range spacecraft, a great deal of interest surrounds the question of man's adaptability to an artificially rotating environment. Design criteria for artificial gravity systems depend largely on man's physiological responses and behavioral capabilities while living in the artificial force field. If accumulating space flight experience proves that artificial gravity will be necessary for maintaining physical integrity of the crew during long missions (as well as for other practical reasons), the question of how much gravitational force is required becomes a critical item for engineering design.

Tests in ground-based simulators have shown that men tolerate exposure to moderate rates of constant rotation for periods of days without showing serious psychological or physiological decrement. For example, four men lived in the Slow Rotation Room (SRR) at the Naval School of Aviation Medicine for 14 days while the room turned at 3 revolutions per minute.[3] To simulate 1/6 of earth's gravity, a radius of 60 feet is needed at 3 rpm. The SRR has a diameter of 15 feet and is completely enclosed so that there is no outside visual reference. All subjects performed well in physical fitness tests, complex eye/hand coordination tests, mental tests, and special balance and psychomotor coordination tests. In earlier experiments with the SRR it had been shown that most test subjects required from several hours to a day or more to adapt to the rotating environment. At rates of about 5 rpm and higher, symptoms of canal sickness were moderately severe or worse during the adaptive process. These symptoms were typical of motion sickness. The most pronounced symptoms came from semicircular canal stimulation caused by head movements out of the plane of rotation. An interesting feature of these studies was the problem of postrotational symptoms during readaptation to the normal earth environment. Guedry summarizes the current concept of adaptation-deadaptation to a constantly rotating environment as follows:

> In the present experiments visual information about spatial relations within the room is essentially accurate, but locomotion within the room is accompanied by conflicting proprioceptive and vestibular information. As adaptation ensues, the intention involved in the movements permits learning of anticipated sensory conflicts from the proprioceptive and vestibular systems which apparently gradually results in a CNS [central nervous system] reorganization. Within a few days, walking and all movements are made without difficulty and without apparent sensory-motor disturbance. Upon cessation of rotation this new state of adaptation is now a source of difficulty. With movements of the head and body in the normal earth environment, the expected proprioceptive and vestibular information which was learned on the room is no longer elicited. As has been shown in previous experiments these movements now elicit reflex activity and sensory events directionally opposite to reactions which occurred soon after the beginning of rotation. With time these compensatory reactions appropriate to a rotating environment dissipate, . . ."

Despite the fact that such ground-based simulators as the SRR cannot truly duplicate the space flight situation, it is believed that the bizarre patterns of semicircular canal stimulation one experiences in the simulator are sufficiently like those postulated for a rotating space platform that good credence may be placed in simulator experimental results. It is clear that although artificial gravity in space vehicles may be feasible from an engineering standpoint, some problems of human adaptation and deadaptation must be considered not only in design of integrated artificial-gravity spacecraft systems but also in crew procedures during adaptation and deadaptation.

biologistics of life support

The provision of the spacecrew's needs in oxygen, water, food, and other sustaining necessities naturally goes beyond the scope of environmental control per se. Nevertheless there is a close link between the spaceman's metabolic output in heat, carbon dioxide, and other body wastes and the performance requirements of the environmental control system.

There has been insufficient experience to determine the precise nutritional needs of astronauts in long-term space flights. Some space medical authorities feel that the metabolic needs, in food, water, and oxygen, of the weightless man engaged in light physical activity will be significantly less than for the same man doing the same thing in a spacecraft simulator on earth. In spacecraft simulator studies, normal young men consumed an average of 1867 KCal per man per day in tests lasting from 14 to 30 days.[1] Minimum food intake in these studies was 1537 KCal/man/day; maximum 2160 KCal/man/day. Water was allowed without restriction. Since uncertainties exist, as mentioned above, it has been recommended that logistic design ensure a known safe allowance of food and water for maintenance of normal energy reserves and body weight, based on terrestrial experience. According to this guidance, the requirement for a young man weighing about 154 pounds is for 3000 KCal of food and 2.5 liters of water per day. If dehydrated foods are used, an additional liter of water per day for food rehydration is required. Proof of metabolic needs will accumulate starting with the longer missions of Projects Gemini and Apollo and may be well demonstrated in the Manned Orbiting Laboratory Program. The recommended ratios of protein (p), carbohydrate (CHO), and fat in the ration average about 14% protein, 54% carbohydrate, and 32% fat; however, variation in these ratios should be provided to improve food acceptability and perhaps to meet unexpected energy demands or unusual protein replacement requirements. Changes in the constituent ratios of foods bring about noteworthy differences in the weight of the food, in metabolic oxygen needs, and output of carbon dioxide and metabolic water. An example is shown in Table III.

Freeze-dehydrated foods show great promise for intermediate-range space missions. The foods are precooked, then frozen, dehydrated, and vacuum-packed in containers suitable for rehydration and use during weightlessness. For Project Gemini, the containers are clear, strong plastic bags with a feeding neck

Table III. Influence of Food Composition on Certain Biologistics Parameters (2800 KCal Ration)*

constituent ratios	*weight of ration*	*oxygen uptake*	*CO_2 output*	*metabolic water*
p 7%, fat 12%, CHO 81%	1.4 lbs	1.81 lbs	2.3 lbs	3.6 liters
p 16%, fat 35%, CHO 49%	1.2 lbs	1.83 lbs	2.16 lbs	3.3 liters

*Modified from Wu, General Dynamics Astronautics, 1963.

designed for direct transfer of the food to the consumer's mouth. Foods processed in this manner may be stored at room temperature without spoilage. Their weight and bulk are minimal. All that is needed for reconstitution is water. In tests of acceptability the freeze-dehydrated meals have competed well with meals prepared from fresh foods. Although some of the items such as gravy-vegetable mixtures have proved unpopular, the high acceptability of most of the menu items prepared in this way has established freeze-dehydration as the method of choice for the present.

Some of the average daily metabolic needs are listed below. Refined data from growing experience with manned space flights will be needed for precise biologistics planning for long-range missions.

Work-Rest Cycles. Not only is spacecrew efficiency dependent in part on the division of time between work and rest periods and between times of sleep and wakefulness, but also the spaceman's metabolic needs are tied to these patterns of work and rest. When one considers the numerous studies investigating the physiological, psychological, and industrial-economic aspects of human productivity, it may seem amazing that we have insufficient scientific information to prove that one pattern of work-rest-sleep is superior to another for such situations as space flight.

All investigations to date have confirmed that man's innate rhythms of wakefulness, work efficiency, desire for rest, relaxation, and sleep are strongly adapted to the 24-hour day-night cycle. When subjected to test conditions using other than a 24-hour "day," subjects show partial physiological adaptation as measured, for example, by body temperature patterns. Performance ability appears not to deteriorate as a result of changes in work-rest cycles provided that the work-rest and sleep-wakefulness ratios are held constant. For instance, studies by Adams and Chiles justify

Table IV. Average Metabolic Data for 25-Year-Old Man Weighing 154 Pounds Subjected to Normal Spacecrew Activity

oxygen uptake	1.9 lbs/day
carbon dioxide output	2.3 lbs/day
drinking water	5.5 lbs/day
food rehydrating water	2.2 lbs/day
food	3000 KCal/day
Water output (including water from metabolic oxidation)	
urine	3.52 lbs/day
respiration and perspiration	4.68 lbs/day
feces	0.3 lbs/day
Total heat output	11,100 BTU/day

the general conclusion that, in terms of performance efficiency in a simulated spacecrew situation, a continuous work-rest pattern as extreme as four hours on and two hours off is acceptable for at least two weeks and probably for thirty days.[5]

On the basis of current knowledge, several general conclusions seem warranted. Readaptation of men to a different sleep-wakefulness pattern and a different work-rest pattern than those to which they are accustomed is a slow process. It may require as long as two to three months and possibly will never be complete in the physiological sense. A second conclusion is that men vary markedly in their ability to adapt to new work-rest-sleep patterns. A third is that performance ability seems to be independent of work-rest-sleep patterns provided that, over periods of several days or weeks, the aggregate sleep approximates normal amounts. The fourth and most important conclusion at present is that the customary division of a man's day into approximately equal parts for work, rest, and sleep remains the best planning guide for long-range space missions in which payload capability affords sufficient crew to permit such scheduling. It is clear that additional research and experience with the real thing are needed before spacecrew duties can be programed most efficiently in harmony with innate metabolic cycles.

Personal Hygiene. One liter is an adequate although seemingly austere allowance of wash water per man/day. This relatively small amount of water used with sponge or disposable wash and rinse cloths permits a sponge bath, as well as hand washing when needed, on a daily schedule. More elaborate bathing provisions are possible provided that the extra payload capability is available. Even in weightlessness it is possible to bathe inside a body-covering plastic envelope designed to contain the water completely and channel it into a waste-water recovery system. Tooth brushing can be successfully done under weightless conditions after the astronaut masters a "closed system" technique; that is, brushing with lips closed around the brush handle and expelling the rinse water into a tubed and valved waste container. Shaving by mechanical means such as spring-driven or electric shavers is the preferred solution to the whiskers problem. A small vacuum device has been designed for use with the shaver to collect the shavings and thus prevent contamination of the cabin atmosphere with whisker clippings. In spacecraft with provisions for artificial gravity, washing, showering, tooth brushing, and other acts of personal hygiene may be performed in an essentially customary manner.

In spacecraft advanced enough to provide the so-called "shirt-sleeve environment," crews should change coveralls, underwear, and socks often enough for bodily comfort and acceptable aesthetic practice. On the basis of a change every three days, it is estimated that for a three-man crew on missions exceeding about 21 days the weight penalty for carrying extra changes of this basic clothing would exceed that of a lightweight, compact, closed-system washer-dryer for laundering.

Isolation. The disturbing effects of physical and psychological removal from the customary sensible environment have been described by many authors. For example, in an earlier issue of the *Air University Quarterly Review*, Dr. Hauty described the mental disturbances and depressions of Admiral Byrd and Dr. Bombard during their separate experiences of prolonged isolation.[6] Graybiel and Clark were the first to study methodically the experience of pilot detachment known as the "breakoff phenomenon."[7] This psychological experience of some pilots on high-altitude missions, characterized by strong feelings of detachment from the earth and its realities, exhilarates some, disturbs and even frightens others. More profoundly disruptive of mental efficiency and integrity, the experience of sensory deprivation incapacitates seriously in the test situation. Interestingly, the experimental removal of as many as possible of the customary sounds, smells, sights, and tactile and temperature sensations produces not a pleasant, durable somnolence as might be expected but at length a restlessness and ultimately a condition of frank hallucinosis. These and other possible effects of the isolation of space travel, such as fear of the astronomical distances involved, are the subject of much speculation.

We are encouraged by the lack of evidence of any such adverse effects on our astronauts in near-earth orbital flights. If anything, the U.S. astronauts have been too busy in meeting the procedural demands of their flights to pause for much speculation along these lines. Although they have obviously been weightless and confined, there is no evidence that they suffered from sensory deprivation or feelings of isolation. Indeed, Cooper reduced the number of in-flight status reports in order to gain extra time for other in-flight duties and rest. It would appear that isolation will not be a serious problem for multiman spacecrews equipped with reliable communications facilities and some additional aids to combat boredom and absence of the customary environment of civilized earth. Such aids might include recorded music, radio and television entertainment, books, games, etc.

Medical Support. We rely for the short-range space flights on careful preflight medical appraisal to ensure that the spacecrew members are fit at the time of launch. As mission duration lengthens, the risk of illness in flight naturally increases. Thus we must continue to strive for the capability for improved preventive medicine procedures for spacecrews. There is hope for at least partial control of common upper respiratory ailments by use of the new adenovirus vaccines and others yet to come. A breakthrough is needed in means of detecting infectious diseases during their prodromal phases as a part of premission medical evaluation. It is desirable to include a consideration of the incubation periods of certain common infectious diseases in planning premission crew control. Meticulous care in food service and medical surveillance over contacts between astronauts and their associates and families must be maintained during the several days before the mission, to eliminate as nearly as possible the transmission of infectious disease agents or toxic foods during these critical few days.

Medical selection of astronauts has advanced to a highly scientific and very thorough procedure. By this means the probability of an astronaut's having a latent medical defect of potential seriousness has been all but eliminated. This, then, is a powerful preventive medical capability in-being.

Until the day when spacecraft capacity and the nature of the space mission will justify having a physician on board, spacecrews may be given some training in the indications for and uses of a limited supply of drugs. At a minimum, spacecrews should be proficient in and pretested for an anti-motion-sickness drug, a stimulant such as dexedrine, an analgesic, and a broad-spectrum antibiotic. Coupled with a knowledge of these few essential drugs should be proficiency in advanced medical aid. The place of medical aid in total life-support is clear when one considers that space-mission accidents and unexpected emergencies can not be ruled out. We must plan for the medical aid equipment to be on board and for the training of the spacecrew in its use.

THIS REVIEW was prepared as an introduction to the subject of life support in manned space vehicles. Most of the important aspects of the subject have been treated in sufficient depth to give the reader an appreciation of the scope of life support for manned space flight and the current state of the art. It is hoped that the critical importance for mission completion of reliable life-support equipment will immediately be recognized as well as the fact that extensive research and development efforts will be necessary to evolve the kinds of high-quality, automatic life-support systems envisioned for future long-term space missions.

Office of the Surgeon General, USAF

Notes

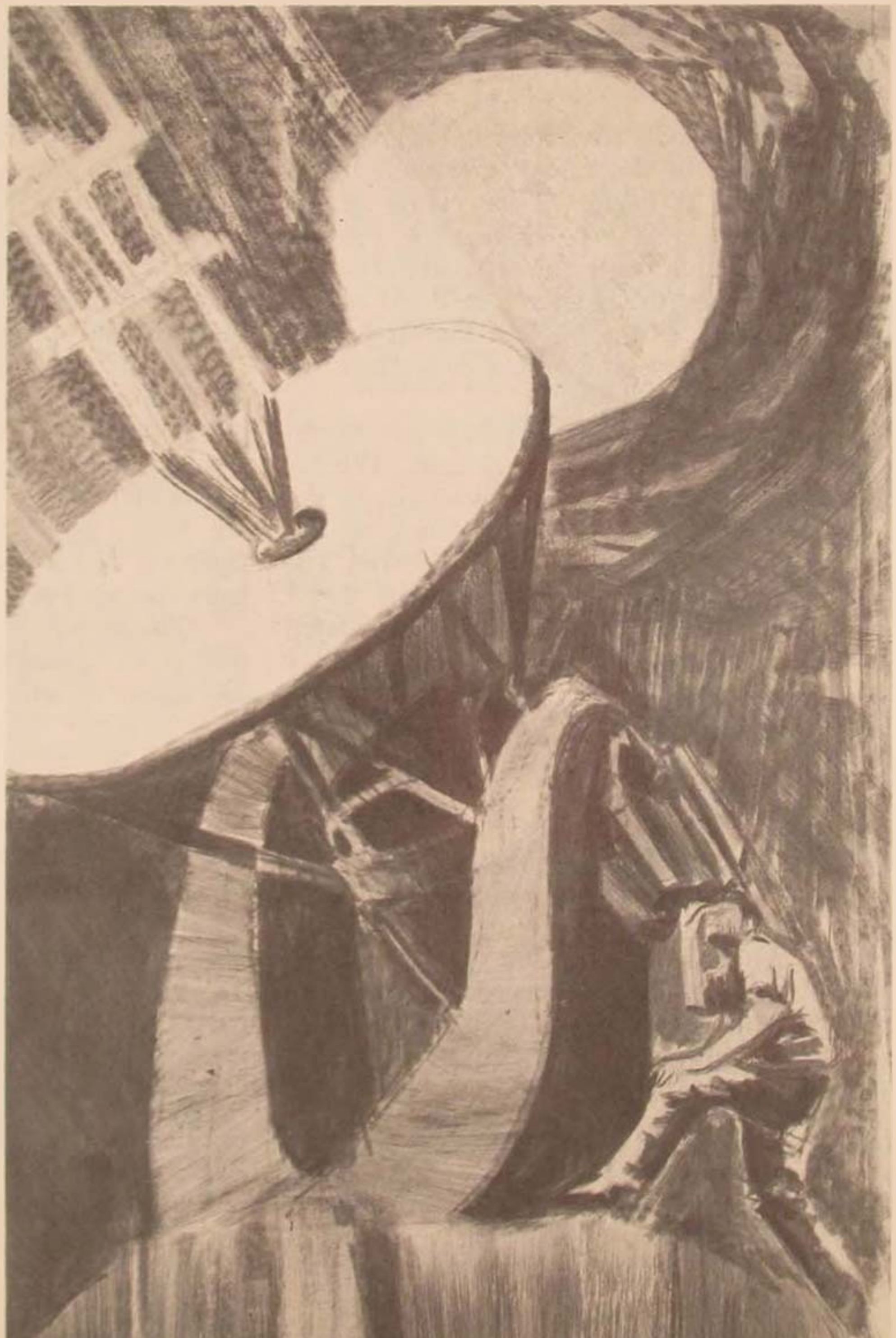
1. Francis B. Shaffer, "Probability of Meteoroidal Penetration," *AIAA Journal*, Vol. 2, No. 4, pp. 738-42.
2. Charles F. Gell, *et al.*, "Biological Effects of Simulated Micrometeoroid Penetration of a Sealed Chamber Containing Animal Specimens," *Aerospace Medicine*, Vol. 33, No. 2 (February 1962), 156-61.
3. Frederick E. Guedry, *et al.*, "Human Performance During Two Weeks in a Room Rotating at Three RPM," *Navy BuMed MR005.13–6001*, Subtask 1, Report No. 47.
4. B. E. Welch, *et al.*, "Sealed Cabin Experimentation," *ARS Journal*, 31 (November 1961), 1541-44.
5. O. S. Adams and W. D. Chiles, "Human Performance as a Function of the Work-Rest Cycle," *USAF WADD Tech Report*, 60 (March 1960), 248.
6. G. H. Hanty, "Human Performance in Space," *Air University Quarterly Review*, X, 2 (Summer 1958), 89-107.
7. B. Clark and A. Graybiel, "The Break-off Phenomenon: A Feeling of Separation from the Earth Experienced by Pilots at High Altitude," *Journal of Aviation Medicine*, 28 (1957), 121.

II

EXPLORING THE SPACE ENVIRONMENT

. . . but our minds have looked
Through the little mock-dome of
heaven the telescope-slotted
observatory eye-ball, there space
and multitude came in
And the earth is a particle of dust
by a sand-grain sun, lost in a
nameless cove of the shores of
a continent.

Robinson Jeffers, "Margrave"



THE SPACE ENVIRONMENT

Brigadier General Benjamin G. Holzman

THE SPACE environment begins at the center of the earth. It extends to infinity. It begins at the center of the earth because it is there in the slow, convective movement of molten material at the earth's core that the earth's magnetic field has its origin. It extends to the farthest known part of the universe, out to radio stars more than two billion light-years away.

Surely Air Force interest does not encompass this entire domain. Not directly. The Air Force's mission is to provide warning, to defend the country from aerial attack, and if necessary to carry out an aerial offensive. The aspects of space environment research that I will touch upon in this article are all related to this mission. The Air Force does not sponsor research on the space environment out of benevolence toward science. It sponsors research bearing on its mission. In the sense that the Air Force hopes to make application of the knowledge and techniques arising from its sponsored research, the work is applied research. But under a more acceptable definition, much of it is rather basic. At least, most people would consider that part of it which is the responsibility of the Air Force Cambridge Research Laboratories to fall under that category.

Only here and there can we at AFCRL clearly discern a relationship to more effective Air Force operations. For example, if one of our scientists designs an infrared sensor, places it as a piggyback package aboard a satellite launched from Vandenberg AFB, records its sensitivity to heat variations of the earth below, and publishes a report on this work entitled, "Improved Infrared Sensor," all would agree that this is of Air Force interest. But if another scientist places a spectrometer in the same satellite and ultimately publishes a paper, "Associative-Dissociative Mechanisms of Atmospheric Molecules," the relationship to Air Force interest would be far less apparent. The latter study, however, may ultimately lead to a satellite sensor—and thus may be of greater Air Force significance. Is there any way of knowing which is in fact the better program from the Air Force point of view?

The answer is a qualified no, but the question raises a tangential consideration, a brief discussion of which may serve to illuminate how decisions related to the support of space research are made. Not all research is good, per se. And certain kinds of good research are better from the Air Force point of view than other good research. How does the Air Force research manager go about selecting from among all the potentially promising research programs demanding support?

First of all, the research manager does not have to be a practicing scientist with an intimate and detailed knowledge of all the programs he is responsible for. It might actually be harmful if he were, since he might have his own faulty preconceptions on the scientific approach to a particular program. (An architect, for example, is not necessarily a good city planner.) But the research manager must know the conditions of research, the research environment, and the people populating that environment. He applies certain criteria, but he does so tacitly and to some extent intuitively. In choosing the work to be funded, he weighs the potential interest to the Air Force, scientific significance, chances of payoff in terms of some new technique, competence of the scientist conducting the work, resources needed, and so on.

The first of these criteria is that of potential interest to the Air Force. What then is the Air Force's interest in space? Operations in space may enhance Air Force capabilities in surveillance, detection, warning, tracking, and communications.

When I say "may enhance," I am injecting a qualification that should not be rapidly glossed over. The Air Force has been hard put to explain its role and mission in space. Generally we say we must explore the unknown, which is insufficient reason to those who control the Air Force budget. But in exploring the unknown we are extrapolating from past fruitful experience, which has taught us that we are seldom able to predict the technological outgrowth of research which in its incipient stages seems remote from Air Force operational missions.

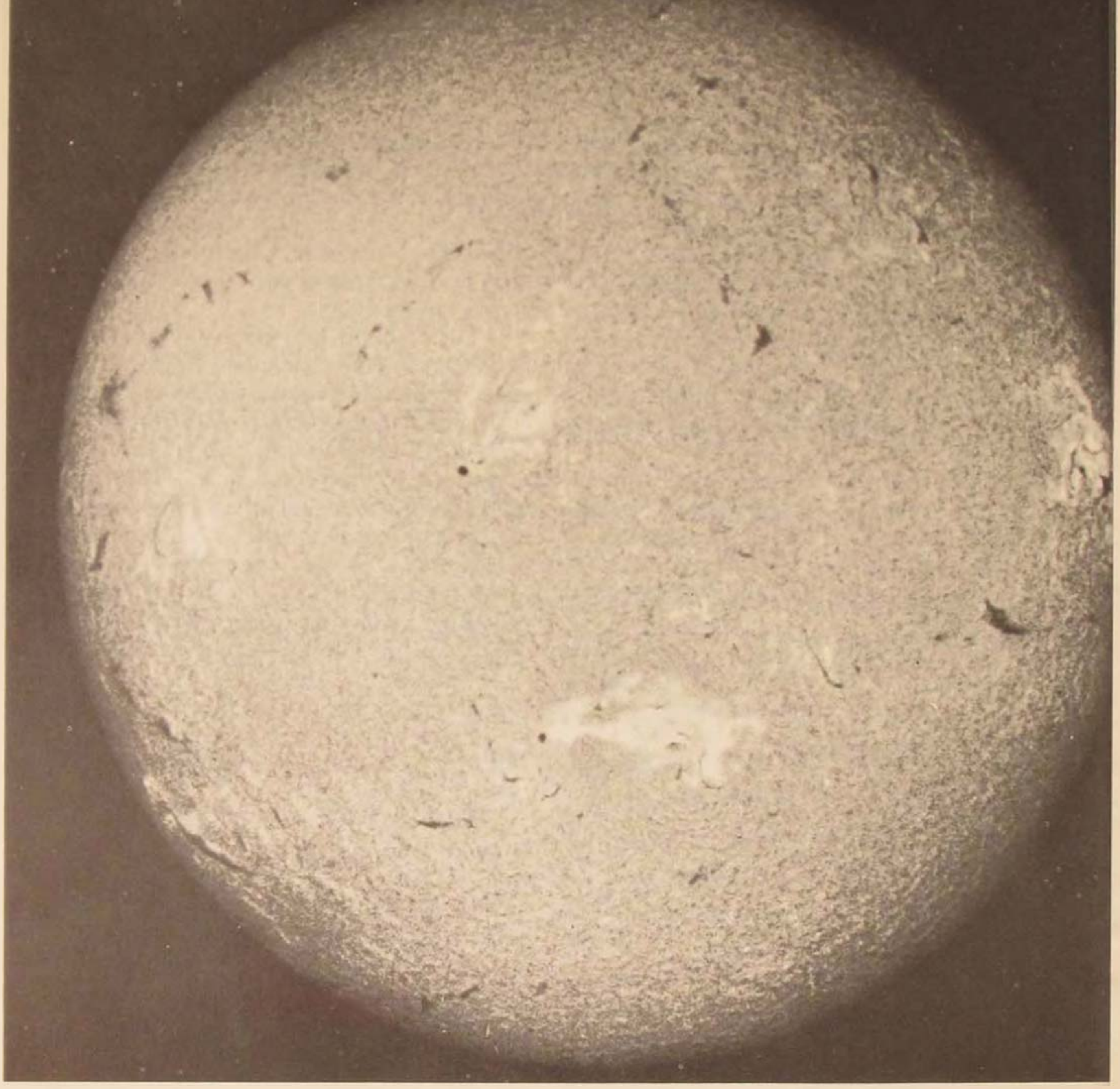
More to the point, it is logical that we can carry out various Air Force missions—particularly the most difficult mission, warhead destruction—if our operations are conducted in the environment in which the enemy missile will actually fly, namely, above the ocean of air surrounding the earth, rather than at the bottom of that ocean.

Most of the research supported by the Air Force at AFCRL relating to the space environment comes under the heading of improved sensor techniques. The sensing device may be an infrared detector, an ultraviolet sensor, a photocell, or an antenna for picking up radio signals. Atmospheric constituents, atmospheric refraction, galactic radio noise, heat radiation from cloud layers and the earth, and ionospheric absorption all may limit the effectiveness of these sensors. Thus the properties of the atmosphere—particularly the upper atmosphere—must be understood as well as techniques for building improved devices. They go hand in hand. For these reasons the research concerning sensors and the upper atmosphere is an essential part of Air Force research on the space environment.

Let's look at the space environment as it is more generally defined. Research on the space environment requires the talents of scientists in many specialties—astronomers, astrophysicists, geophysicists, chemists, spectroscopists, and mathematicians. What are the objects and phenomena that we are investigating through our research? I will list and briefly discuss them. The list is not very extensive. We are interested in the sun, the particles and radiation emanating from it and its magnetic fields; we are interested in the moon and its surface layer; we are interested in radio noise sources from deep space; in cosmic radiation; in meteors; in the environments of Mars, Venus, and Jupiter; in Van Allen radiation; and in free hydrogen in our galaxy.

All these are of scientific interest and either immediate or potential Air Force interest. We want to know what conditions we will find if in fact Air Force operations in space become feasible. We may detect some unsuspected phenomenon that we can exploit, or we may find some previously unknown hazard. An illustration of the latter can be taken from one Air Force research program.

The sun periodically emits high-energy protons which are a potential hazard to man in space and which can make delicate electronic equipment inoperative. Until several years ago we were unaware of this hazard. The emission of these protons seems to be correlated with certain types of sunspots of a certain age. It is of great scientific interest to understand just why certain sunspots emit dangerous protons and others do not. To be able to predict the occurrence of these proton showers would, of itself, be a scientific achievement because this implies an understanding of the mechanisms at play, and the understanding of natural phenomena is the goal of science. But from the standpoint of long-term manned operations in space, the ability to predict these



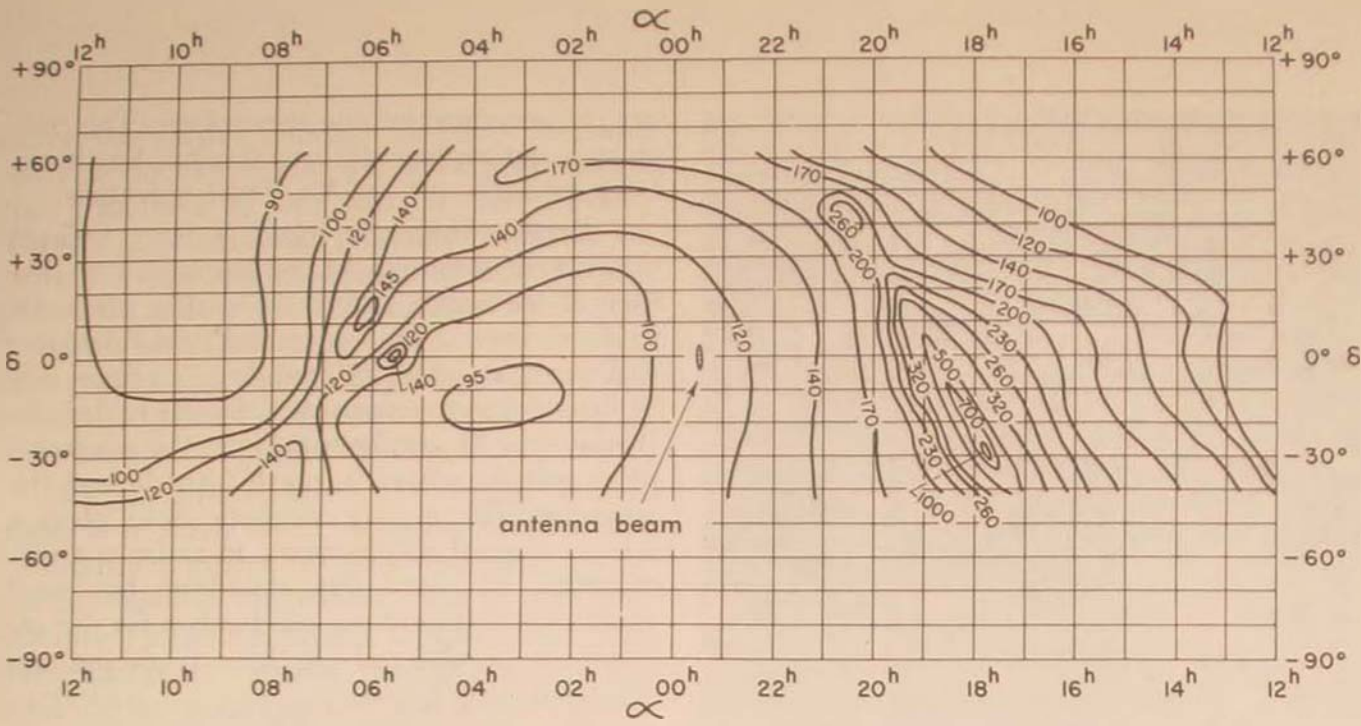
proton showers is also of great military significance.

Since it was initially the scientific interest which led to the military interest in this solar phenomenon, we can draw the inference that most space research funded by the Air Force is in some basic way related to more effective Air Force operations.

I would like to expand a bit on several areas of space research in which the Air Force is involved and hint at the reasons for our involvement in each case.

Radio Star Sources. We want to know all the discrete sources of radio energy in the heavens and to plot these sources. We use these sources as natural transmitters for the understanding and measuring of such things as ionospheric density and scintillation, which in turn are important to both ground-to-ground communications and deep-space communications. Through this study of radio stars we have developed novel methods of calibrating our large distant-early-warning antennas.

Galactic Noise. Galactic radio emissions are not confined to the point sources of radio stars but are found over broad regions of the heavens. We want to plot the intensities of this noise in various parts of the heavens and at



The solar photosphere was photographed in Hydrogen Alpha during a period of solar activity by astronomers at Sacramento Peak Observatory, New Mexico. The white areas are plages, and the darker spots are solar flares. . . . Plots such as this are made, at various frequencies, of the intensity of radio noise emanating from large general areas of space.

various frequencies. Some of the noise has its source in hydrogen emissions. From the detection of hydrogen emissions we learn of those parts of our galaxy where hydrogen gas is concentrated. But from the military viewpoint, we learn just where we are likely to encounter radio noise at certain frequencies of sufficient intensity to submerge the weak signals from deep-space probes. The knowledge permits us to establish, through choice of frequency, radio communication systems less subject to noise interference.

Solar Astronomy. I have already touched on one aspect of solar astronomy, namely, the prediction of solar proton showers. Our astronomers are also interested in the sources of the sun's continuing energy, the internal convective currents giving rise to sunspots, to solar prominences, solar flares, and spicules. They are interested in coronal temperatures and in the solar wind—that stream of protons and electrons which boil off the solar corona and are accelerated in a continuous wind out to the farthest limits of the solar system. The earth is embedded in this wind, which distorts the lines of magnetic force about the earth. This one aspect of the solar wind, its effects on the magnetic field, has implications to proposed very low frequency (VLF) communications schemes in which signals are coupled to the lines of magnetic force and channeled from one hemisphere to another. It has further implications in auroral mechanisms. The aurora must be fully understood in preparation for satellite observation systems in which the detectors are designed to operate in the infrared, visible, or ultraviolet portions of the electromagnetic spectrum. For such systems, auroral radiations represent interfering noise that reduces the sensitivity of the sensors.

Cosmic Rays. The large galactic and intergalactic magnetic fields are believed to provide the accelerating force of cosmic rays—those nuclei of hydrogen which are propelled to huge energies by a mechanism similar to that used to accelerate particles in our giant

The simplest and most economical way to measure high-energy particles is by photo emulsion techniques. When a high-energy particle strikes an atom in the emulsion materials, the number and density of "star" prongs produced denotes energy of the particle.

research particle accelerators on earth. Knowledge of levels of cosmic-ray activity must precede attempts to place man in space, on the moon, or on Mars for protracted periods. Except in the case of Mars, we now know that the level of cosmic-ray flux will not unduly inhibit these operations. But Mars is a special case. Although the earth's atmosphere and magnetic field shield us from the effects of primary cosmic rays, Mars, with an atmosphere only about one third that of the earth, is not well protected. As a matter of fact, the Martian atmosphere may even result in a greater danger from cosmic-ray activity than would be the case were there no atmosphere at all. Cosmic-ray secondaries, i.e., the scattering of high-energy particles resulting from the collision of cosmic rays with Martian atmospheric molecules, could result in scores of ionizing particles at the Martian surface for every cosmic-ray primary entering the atmosphere. This may or may not represent a real danger, but it has been pointed out by Air Force scientists.

Earth's Magnetic Field. I have defined the space environment as beginning at the core of the earth, where the earth's magnetic field originates. This magnetic field fluctuates and varies in intensity from hour to hour, day to day, season to season, and decade to decade. It changes its configuration as the magnetic poles wander about in north Canada and the antarctic. The earth's magnetic field plays a role in auroral mechanisms, in radio communications, in cosmic-ray shielding; but most important, it provides a mechanism for the entrapment of charged particles from the sun and from deep space.

Van Allen Belts. Although cosmic rays have much greater energies than the particles making up the Van Allen belts, the greater density of ionizing particles in the Van Allen belts makes them a much greater hazard to man and his electronic equipment than cosmic rays. Since the Argus series of atomic explosions in the South Atlantic in 1958 we have known that we can greatly increase the level of ionizing radiation within the Van Allen belts by detonating an atomic bomb in or near the belts. If these belts can be enhanced, there may also be techniques for depleting the belts of high-energy particles. Much of our research with respect to the Van Allen belts at the present time involves measurement of the densities and the energies of particles through different cross sections of the belt.

Meteors. The earliest orbiting satellites carried detectors for recording the numbers of meteors and micrometeorites in space. From data gathered by these satellites, it is estimated that a man exploring the lunar surface might be struck by one micrometeorite each second. Even though these micrometeorites are traveling at velocities as great as 100,000 feet per second, they provide no real hazard because of their small size. About 10^{12} (1,000,000,000,000) micrometeorites penetrate the earth's atmosphere each day. Meteors with diameters 8/10 of a centimeter or greater are, of course, less frequent. Some 10,000 of these enter the earth's atmosphere each day, and each produces a visible trail greatly exceeding the brightness of any star in the heavens. Through a continuing research program, we hope to refine our knowledge in order to predict with great statistical precision the extent to which man, his space vehicle, or his lunar base will be exposed to danger by meteors and micrometeorites. Our present estimates may be in error by as much as two orders of magnitude.

Upper Atmosphere. The upper atmosphere is in many ways the most complex and multifaceted aspect of space research, giving rise to scores of individual areas of research. Obtaining profiles of winds, temperatures, densities, and pressures from ground level out beyond 700 km at various latitudes and at various seasons is only one aspect of this research, although it is an aspect which must always figure in the calculations of those designing our rockets, missiles, and satellites, and in planning the trajectories of these vehicles. To avoid satellite tracking errors and guidance errors during launch, we must continue to conduct research on such prosaic matters as refractive index, which forces us into intensive studies of the humidity and layering structure of the lower atmosphere. The sensors that we would place in our satellites are affected by many atmospheric features, some of which I have already noted. They include natural airglow of the atmosphere, the aurora, ionospheric structure and irregularities, noctilucent clouds, and absorption and re-emission of energy for atmospheric molecules. Since the late 1940's, AFCRL alone has launched well over 300 research rockets to investigate these aspects of the atmosphere. To understand the mechanisms, Air Force scientists have simulated in the laboratory a variety of dynamic processes in the upper atmosphere.

In 1961 Air Force scientists discovered that a band of micrometeorite dust permanently surrounds the earth at 80–100-mile altitude. The discovery was made by use of this specially designed rocket, which collected micrometeorites and returned them to earth.

THE WORK THAT I have covered thus far shows some direct relationship to the space environment. It would be well to touch upon research that has an essential, although indirect, bearing on space. At Cambridge Research Laboratories, side by side with the large environmental research program, there is a substantial program in electronics. All these electronics projects relate in some way to space operations. At the same time, none of the programs was undertaken for the basic purpose of improving space capabilities. Almost all have as their primary goal the enhancing of more or less conventional Air Force missions.

We have, for example, a large effort in speech research. This would seem to be well removed from space. This program was undertaken to compress the amount of bandwidth needed to transmit human speech. By way of showing some relationships, an ordinary telephone line has a bandwidth of about 3000 cycles, and a good home hi-fi system covers a bandwidth of at least 15,000 cycles. Our goal is to transmit natural-sounding voice messages over a bandwidth of only 50 cycles. What will this achievement buy us, insofar as the space program is concerned? Because power needed to transmit a wide-bandwidth signal is proportionally much larger than is needed to transmit a narrow-bandwidth signal, it will result in

great savings in power requirements (or transmissions over much greater distances, assuming equal power), and power requirements are always a basic consideration in space vehicle design. For similar reasons, we are attempting to compress the bandwidth needed to transmit video pictures.

Studies of radiation damage to electronic materials and components, studies of methods for converting energy from one form to another, studies in microminiaturization, in error-correcting coding, and in antenna techniques —all have a bearing, usually a quite obvious bearing, on space operations. I cite these instances in passing, to emphasize that it is difficult to find research of any kind, whether carried out in Government laboratories, at universities, or in industry, that does not relate in some fundamental way to the space effort.

The space program has, of course, given a special impetus to environmental research. For one thing, the Government has heavily sponsored research in this field, and this alone has led to rapid advances in the understanding of the physics of the universe. Related to this support, and most important, are satellites and space probes that have permitted us to place our sensors above the atmosphere and into the very environment that we are investigating. New discoveries in the space environment area are being made more rapidly than in any other single field of science, with the notable exception of the advances in the field of microbiology-genetics.

In spite of our emergence above the frequently opaque ocean of air made possible by our space vehicles, ground-based facilities still provide us with much of our essential sensory data on space. By ground-based facilities I am referring primarily to telescopes, radio and optical. Data acquired by satellite and data acquired from our ground facilities are complementary.

Let us now take a closer look at the facili-

At its Sagamore Hill Radio Observatory the Air Force has one 84-foot radio telescope (foreground) and a second new one of 150-foot size. The two instruments can be used together as an interferometer.

ties and equipments required for space environment research. Air Force facilities are fairly typical of the national pattern. In addition to huge telescopes, we use instrumented balloons, rockets, and satellites. On a par with these as tools for understanding our environment are laboratory simulation facilities.

telescopes

The Air Force has one of the Nation's most important centers for astronomy at the Air Force Cambridge Research Laboratories. To conduct research in astronomy, you must have large observatories. I will cover some of our existing and planned observatories. First, we operate a large radio observatory, Sagamore Hill Radio Observatory, at Hamilton, Massachusetts. For several years the primary instrument at this observatory was an 84-foot telescope. Two years ago we erected a new telescope at Hamilton, one measuring 150 feet. These two telescopes can be used either separately or together as an interferometer. Research at this facility involves lunar reflection communications, refractive index studies, measurement of scintillation of radio stars, measurements of hydrogen in space, and the study of atmospheric densities.

Cambridge Research Laboratories had a primary role in the design and construction of the world's largest radio telescope, the 1000-foot radio telescope at Arecibo, Puerto Rico. With this dish we can look deeper into space than we have been able to do before. The dish is carved out of a natural depression in the ground. This telescope, built with ARPA funds, is being operated by Cornell University. Air Force radio astronomers will make extensive use of the telescope in their programs.

The Arecibo telescope was placed in operation late in 1963, but already a new and more powerful generation of radio telescopes is on the horizon. The multiplate antenna concept for the new telescope originated in AFCRL's antenna research program, and hopefully it will be built in the Southwest within the next few years.

Antenna sensitivity—or resolution—depends on the size of the collecting area. The Arecibo antenna has a collecting area of some 18.5 acres. This is just about the practical limit for large dish-type antennas, but it is not the practical limit for other types of antennas with large collecting surfaces. The multiplate antenna grew out of the need for greater and greater sensitivities of the kind that can only be achieved by larger collecting areas. This sensitive antenna can be used as a radio telescope to obtain man's deepest view of space. It can also be used as a sensitive radar to receive signals from deep space probes at distances far beyond the range of any other antenna.

Operating in the radar mode with a 2.5-mw transmitter, the antenna in the proposed plan can detect targets of one square meter at 80,000 miles. (An 84-foot-diameter dish with a similar transmitter would provide detection ranges of 1 m^2 at 4000 miles.) This extreme sensitivity is perhaps better appreciated by a comparison of detection at a range of 15,000 miles. At this distance the minimum cross section seen by the smaller antenna system is 200 square meters; the multiplate antenna can see one-thousandth of a square meter. This factor of over 10^5 in performance makes this antenna not merely another subsystem improvement but a radical new tool in the Air Force inventory.

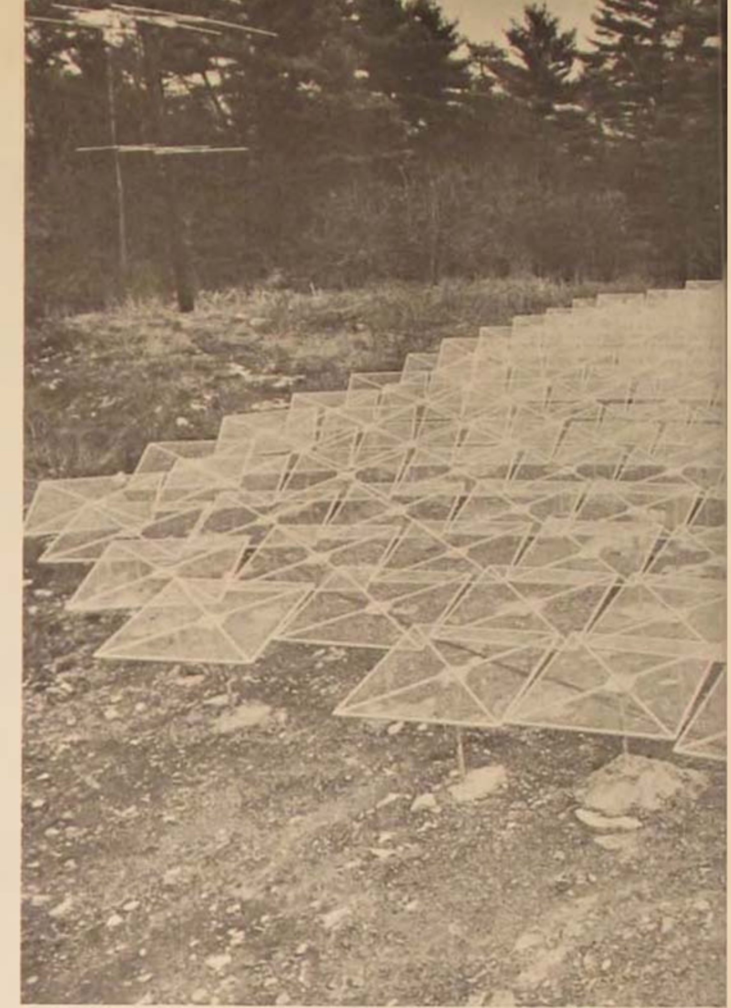
For the past two years we have evaluated a model segment of this antenna, a scaled-down segment measuring 70 by 120 feet. The capabilities of the antenna have been thoroughly tested and documented. As long as we move deeper and deeper into space, and as long as the Air Force has the need to locate and track all man-made objects in space, the requirements for such an antenna are incontestable. In research and development, where great uncertainties are always at hand, we can be certain that antennas of this type will be built within the coming decade—and will perhaps become the standard antenna configuration for deep-space communications and detection.

The important feature of the multiplate concept is that the antenna can be built to any arbitrary size, with construction costs increasing only lineally. With fully steerable dishes, costs increase exponentially with size. The multiplate antenna consists of thousands of flat plates, each of which measures 20 by 20 feet. The antenna will have a maximum aperture of 2500 feet or about a half mile.

Our historic window to the universe is the optical telescope. Most of the optical telescopes in observatories all over the world are stellar telescopes. But if research is concentrated on solar astronomy, it is desirable to construct specialized facilities for solar research, as in the case of the Air Force observatory at Sacramento Peak, New Mexico. Similarly, if primary interest is in observing the planets, laboratory facilities will be specialized for this purpose. The Air Force is interested in all three of these specialized observatories—for stellar observations, for solar observations, and for planetary observations.

Astronomers are generally agreed that the planning and construction of new astronomical observatories have not kept pace with space-spawned needs. Certain kinds of observations can only be made from the ground by use of large telescopes. Where there is a choice of observations from the ground or from instrumented rockets and satellites, ground-based observatories offer an incomparable economic advantage.

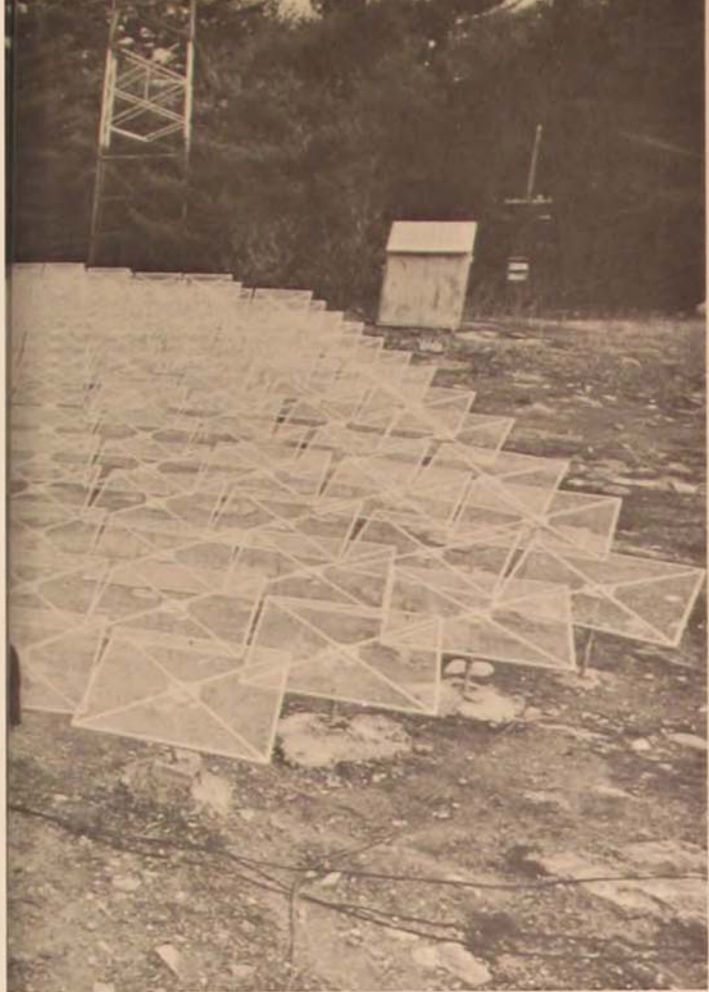
A new stellar observatory, planned over the past several years to be located in Chile, is now under construction. The new observatory was jointly funded by the Air Force and the National Science Foundation. It is located northeast of Santiago at one of the best locations in the world from the standpoint of seeing. "Seeing," I might note, refers to a variety of conditions which influence resolution. Lack of cloud cover and haze are obviously desirable. But most critical are the vertical and horizontal movements of air immediately surrounding the telescope. This air turbulence gives rise to scintillation and jitter, which in effect make sharp focusing impossible. The Chilean telescope with a 60-inch lens will be one of the finest in the Southern Hemisphere.



This experimental model of the multiplate antenna was located at Concord, Massachusetts. Initial the-

At Sacramento Peak Observatory in New Mexico, the Air Force operates one of the most complete solar observatories in the world. Many studies are being carried out at this observatory that have a bearing on future space operations. One such study of immediate and critical importance involves ionizing proton showers emitted from the sun, which I have previously noted.

Scientists at Sacramento Peak Observatory are studying methods for predicting the safe periods when there is an absence of the showers. During the present quiet sun period,



oretical and experimental work on the antenna was done at AFCRL. It was dismantled in December 1964.

the observatory has been making seven-day predictions with about 97 per cent accuracy and can predict the absence of proton showers for periods up to a month with about 75 per cent accuracy. The true test of the validity of this study, however, must await the 1968–1970 period when the sunspot activity again approaches maximum.

rockets, satellites, and balloons

AFCRL uses rockets and satellites in its experimental research programs to a greater extent than any other research activity in the free world with the exception of NASA. During 1965 about 40 major rocket firings are scheduled, at least two AFCRL satellites will be launched, and scores of individual experiments will be carried aboard Air Force and NASA satellites. Experiments are planned for Air Force satellites such as the Gemini vehicle, the Manned Orbiting Laboratory, and the SATAR (satellite for aerospace research) series.

Rocket and satellite instrumentation places three demands on the experimenter. First, he has to decide what aspect of the environment might bring fruitful results. Next, he must design sensors—often highly specialized and ingeniously contrived—that will obtain desired data. Last, and the most tedious and time-consuming aspect of research, he must analyze and reduce the data. If he is a good analyst, he will be particularly attuned to any unsuspected anomaly in the data. In these anomalies are often found discoveries of significance.

A curious fact that I have often observed is that the experimenter is generally slow to recognize these anomalies and their significance. He will question the performance of his instrumentation, will assign more prosaic causes for the anomaly, and will spend long months reviewing the data. A case in point was the discovery three years ago of X-ray sources from deep within our galaxy during an AFCRL rocket experiment designed to measure soft X-ray emissions from the moon. This discovery suddenly opened a whole new field of astronomy, but we did not rest easy with this discovery until we had sent a second rocket aloft to make additional measurements. In the last year the Navy Research Laboratories have inaugurated a major program to catalogue these X-ray stars, and one of the Nation's leading astronomers has called the discovery the most important in astronomy in recent years.

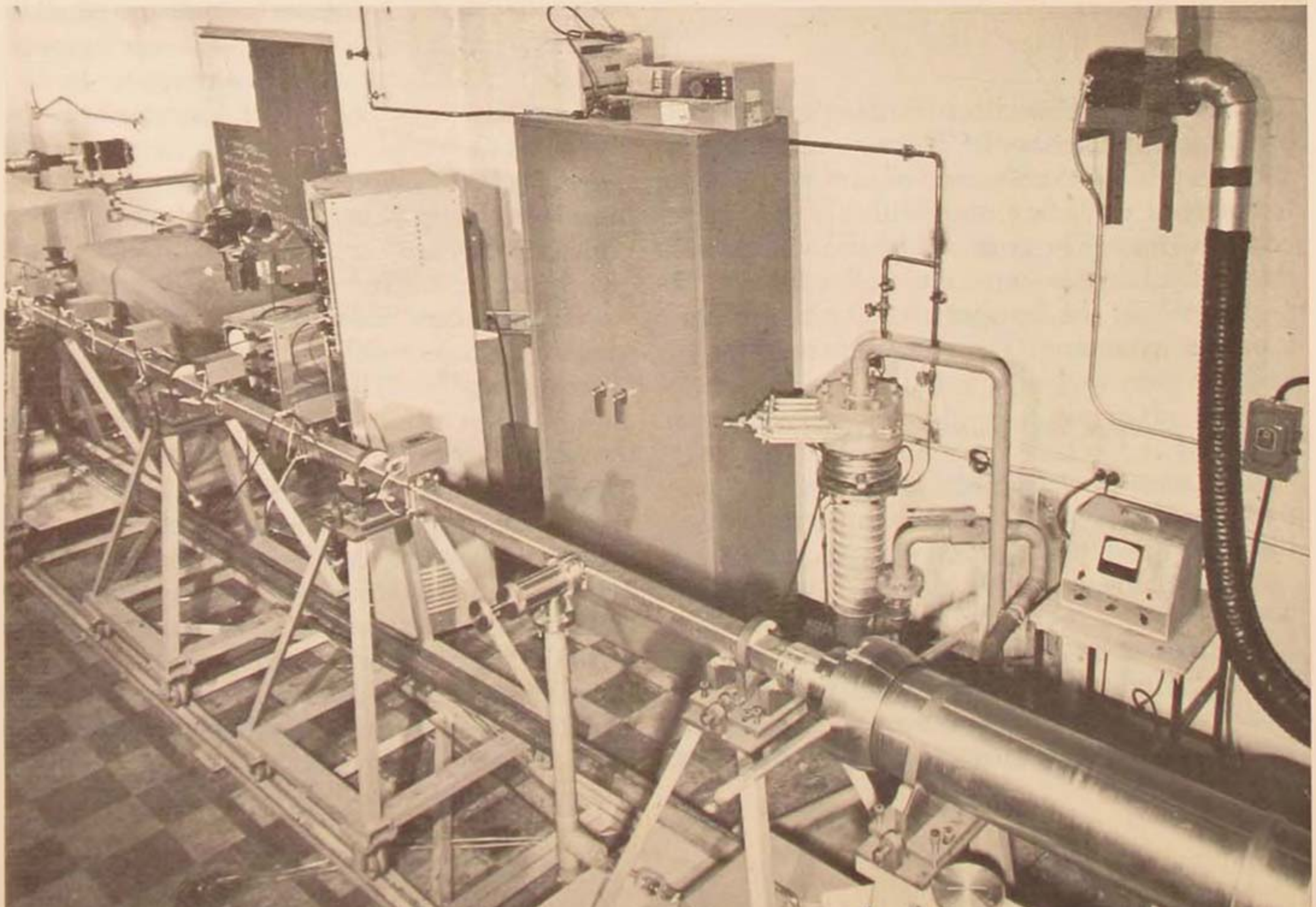
The long history of balloon development would seem to make balloon research and space a strange technological mixture. But at altitudes of 100,000 feet or so, balloons rise above all but a fraction of the atmosphere and thus have a clear view of the heavens. AFCRL uses balloons widely as astronomy platforms. We have transported large telescopes into the near-space regions to view the sun, moon, and planets. Balloons have the advantages of economy, of being able to transport huge payloads to the fringes of space, and ease of recovery. For certain kinds of research, we consider them to be a research tool of importance equaling that of rockets and satellites.

astrophysics and laboratory simulation

Astrophysics is defined as the application of the laws of physics to problems of astronomy. For this work one does not necessarily need telescopes, rockets, and satellites or laboratory equipment. It is possible to sit at a desk with a pad of paper and, using only the astronomical observations of others and the laws of physics, derive the structure of the universe and all its parts, if one is a good theorist. The precision of this theoretical structure, however, will rest on the precision of observations and on the ingenuity of laboratory experiments. Therefore we conduct space research in the laboratory using equipment to simulate astronomical phenomena. Without going into detail, I will briefly note some of these experiments.

Lunar Environment Chambers. Scientific opinions differ as to the composition of the lunar surface layer. From work carried out at AFCRL in three lunar environment chambers, in which we attempt to duplicate the temperatures, the high vacuums, and the radiation levels found at the lunar surface, we believe the moon to be covered with an extremely fine-grain powder. But we have shown, using our environment chambers, that because of the absence of a lunar atmosphere this powder is bonded into a hard, firm layer. We were extremely gratified to find our model of the lunar surface substantiated by the recent flights of the Ranger series. Work with the lunar chambers, however, has uncovered an unsuspected problem likely to be faced by lunar explorers. Because of the adhesion of lunar dust to all surfaces with which it comes in contact, there is the prospect that observing ports, solar cell arrays, and other surfaces may become coated with dust. Because of its strong adhesion, there is no simple mechanism for removing it.

Simulation of the Solar Corona. It is possible to simulate to some extent the solar corona, the solar wind, and the general solar magnetic field by means of rotating conducting fluids in a magnetic field. Fluids of differing



conductivity are used, a fluid of a given viscosity and conducting coefficient being in an inner container and a second fluid with differing properties in a surrounding outer container. These fluids simulate, respectively, the visible surface of the sun and the solar corona. By rotating these containers relative to one another in the presence of a relatively high gauss field, we can derive information on electrical currents, thus enhancing understanding of solar mechanisms.

Shock Tubes and Spectroscopy. One aspect of research in astrophysics centers around high-precision shock tubes. The Air Force has two such shock tubes at AFCRL. The first of these is related to spectroscopic studies. Spectroscopy has long been the most important single tool of research not only for astronomy but for research in physics in its broadest sense. Our knowledge of the stars, their dynamic mechanisms, their composition, has its basis in the analysis of spectral lines. One of the shock tubes is designed for measuring the absolute spectral line intensities of the elements forming the sun and stars. We make certain measurements in the laboratory using this shock tube, and the data we obtain in our controlled laboratory observations lead to a greatly improved accuracy in the analysis of the chemical composition of space bodies.

The second shock tube, measuring 50 feet long, is used to study plasma turbulences. Plasmas fill the universe. We find them in the solar corona, in solar winds, and in the vast reaches of interstellar space. They are in continuous turbulence. We can understand the nature of these turbulences in shock tubes by using ionized gas samples whose exact temperature, density, and homogeneity can be determined and by controlling these gases by means of surrounding magnetic fields.

Within Air Force laboratories and in laboratories all over the country are scores of laboratory equipments designed to help us understand more about our space environment. They have a place in space environment research no less important than satellites and telescopes.

A 23-foot-long optical shock tube is used in the investigation of high-temperature chemical reactions occurring in the space environment.

Our knowledge of the upper atmosphere and composition of the sun and stars derives for the most part from analysis of spectral lines.

The Air Force has pioneered in balloon technology, which furnishes versatile and economical test-beds for carrying instrument packages above all but a fraction of the earth's atmosphere. Launching giant balloons is an exacting operation.

ALL OF THE research relating to the space environment that I have covered and all future research in this field sponsored by the Air Force must be responsive to the needs of the Air Force, both immediate and anticipated. "Anticipated" is a key word with implications that are at the heart of Air Force sponsorship of space science. More properly, the words should carry the qualification, "anticipated on the basis of scientific inquiry." The laws governing physical behavior impose formidable constraints on Air Force ambitions for its future operational systems—but these same laws also set limits to the ambitions of systems designers of other countries as well. Through a strong position at the forefront of science, we maintain a vigilance on the potential enhanced military technologies of others. Enhanced military technologies—both our own and others—are most likely to have their origins in the collection, analysis, and interpretation of data from environmental sensors.

Air Force Cambridge Research Laboratories

PROJECT SUPER

Lieutenant Colonel John D. Peters and
Captain George Mushalko

THE AIR FORCE and the National Aeronautics and Space Administration have embarked on a unique effort to combine USAF and NASA research and development capabilities to meet the Nation's space goals. Project SUPER (Support Program for Extraterrestrial Research) is the culmination of an idea developed by several people within the Air Force Systems Command whereby that command's laboratories already engaged in space-oriented research would support the national effort by conducting and sharing all costs of the research in areas of common interest to both the Air Force and NASA. The plan proposes to accomplish several objectives. First, it seeks to promote economies in space research. The Air Force has the facilities for accomplishing certain types of research such as materials development, hypervelocity impact studies, studies in atmospheric physics, and other space-oriented studies. With a slight reorientation and with a minimum of extra effort, certain objectives desired by NASA can be accomplished with a minor additional manpower requirement and a minimum amount of additional funds. On the other hand, if NASA were to pursue the same research separately, the time, manpower, and funds required would be appreciably more than that required in the extended effort of a project already under way in Air Force facilities. The second objective is to utilize to the maximum extent possible the scientific resources available in the United States for the solving of problems facing the Nation in its space programs. Many of the technical personnel assigned to Air Force Systems Command laboratories, divisions, and centers are already experienced and have done extensive research in the areas of interest. Under SUPER their skills and knowledge are well used. The third objective of the proposal is to permit the Air Force to keep abreast of the research being contemplated by NASA and to advance the technical competence of the Air Force personnel by their doing research in areas where the Air Force does not have a mission but where it has a vast capability for the performance of the required research. In addition, Project SUPER offers the Air Force an opportunity to train junior officers in the management of research projects and programs.

Project SUPER, as presented to NASA, was primarily oriented toward studies and basic research. These types of projects are needed to meet the national space objectives, and the work would be accomplished in areas of interest expressed by NASA.

At the beginning of Project SUPER, NASA outlined several areas in which it felt that the program could be responsive. These included thermal model studies aimed at the definition of similitude laws that could be used in developing theory for the heat balance on full-sized vehicles through testing of models in small environmental chambers; hypervelocity impact studies where the target materials and the projectile materials, mass, and shape could be rigidly controlled in order that the hypervelocity impact phenomena could be well defined; adherence of dust particles to surfaces in a vacuum environment; investigation of self-sealants for space vehicles; study of measure of gas density by radiation scattering; and thermal testing techniques. These investigations are discussed more fully below.

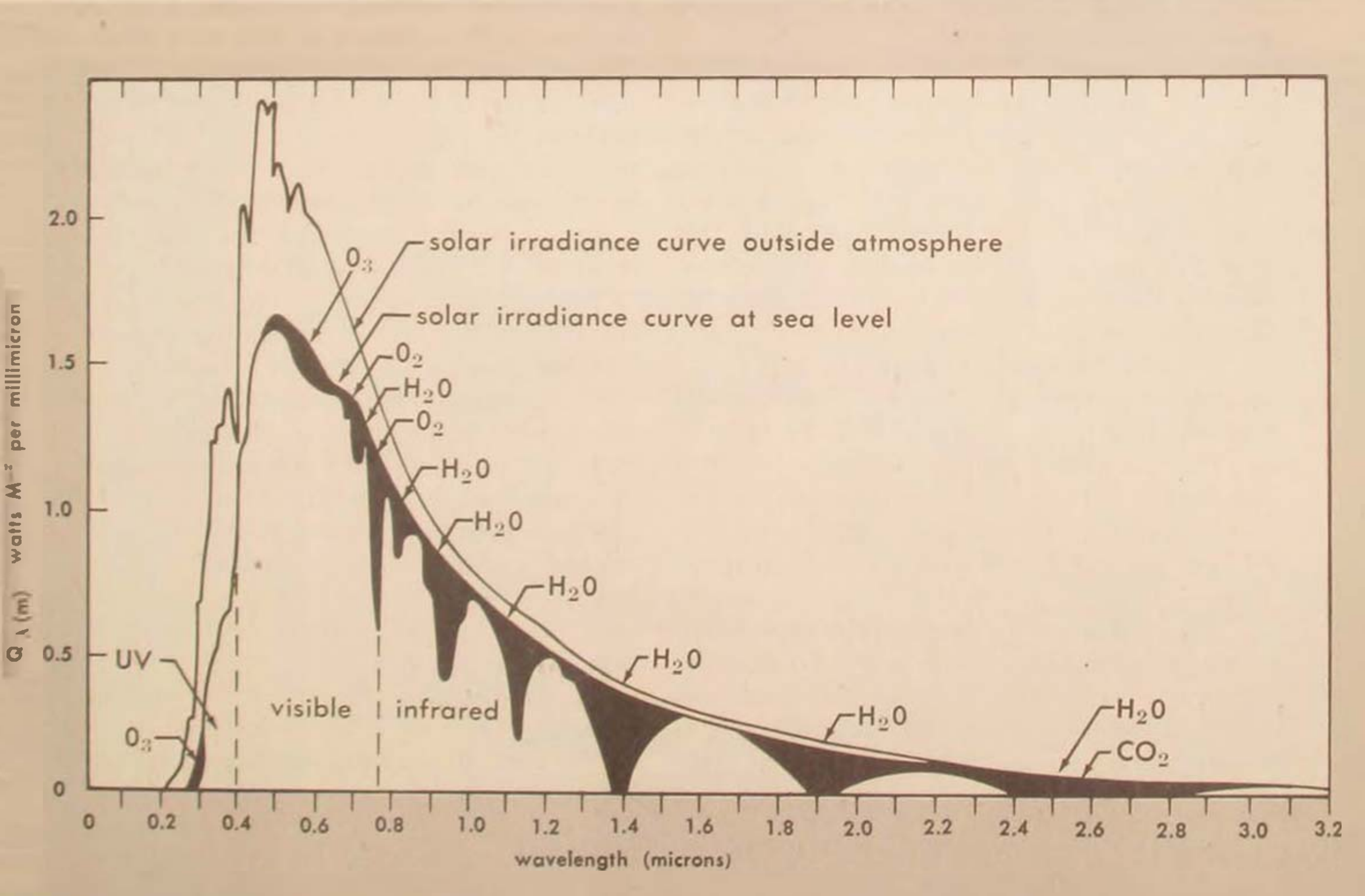
solar/thermal testing and simulation

Outside the earth's atmosphere the spectral energy distribution of the sun is different from that on the earth at sea level. As seen from Figure 1, almost all the ultraviolet and a large part of the infrared spectrum are filtered out by the earth's atmosphere. Due to fluctuations in the composition of the atmosphere, the amount of solar radiation scattered or absorbed by the various atmospheric constituents can vary over quite wide ranges, as indicated by the diagonal lines in Figure 1.

The vehicle in earth orbit receives radiation from the sun (solar radiation), solar radiation reflected from the earth's atmosphere (albedo radiation), and radiation transmitted to the vehicle due to the earth's temperature (planet radiation). It reradiates an equivalent amount of thermal energy to cold dark space.

Figure 2 shows the spectral energy distribution of a carbon arc solar simulator and a tungsten filament lamp superimposed on the spectral energy distribution of the sun in space. There is a growing opinion now among space engineers that the necessity for simulating the true spectral energy distribution on some environmental tests is not as critical as previously thought. Some of these thermal balance tests, it is felt, can be accomplished with heat flux equipment. Of course on solar cell tests where the voltage and current output are strongly dependent upon both the spectral characteristics and the uniformity of the incident radia-

Figure 1. Spectral energy curves related to the sun. (Source: Handbook of Geophysics, *revised edition, Air Force Cambridge Research Center, 1961.)*



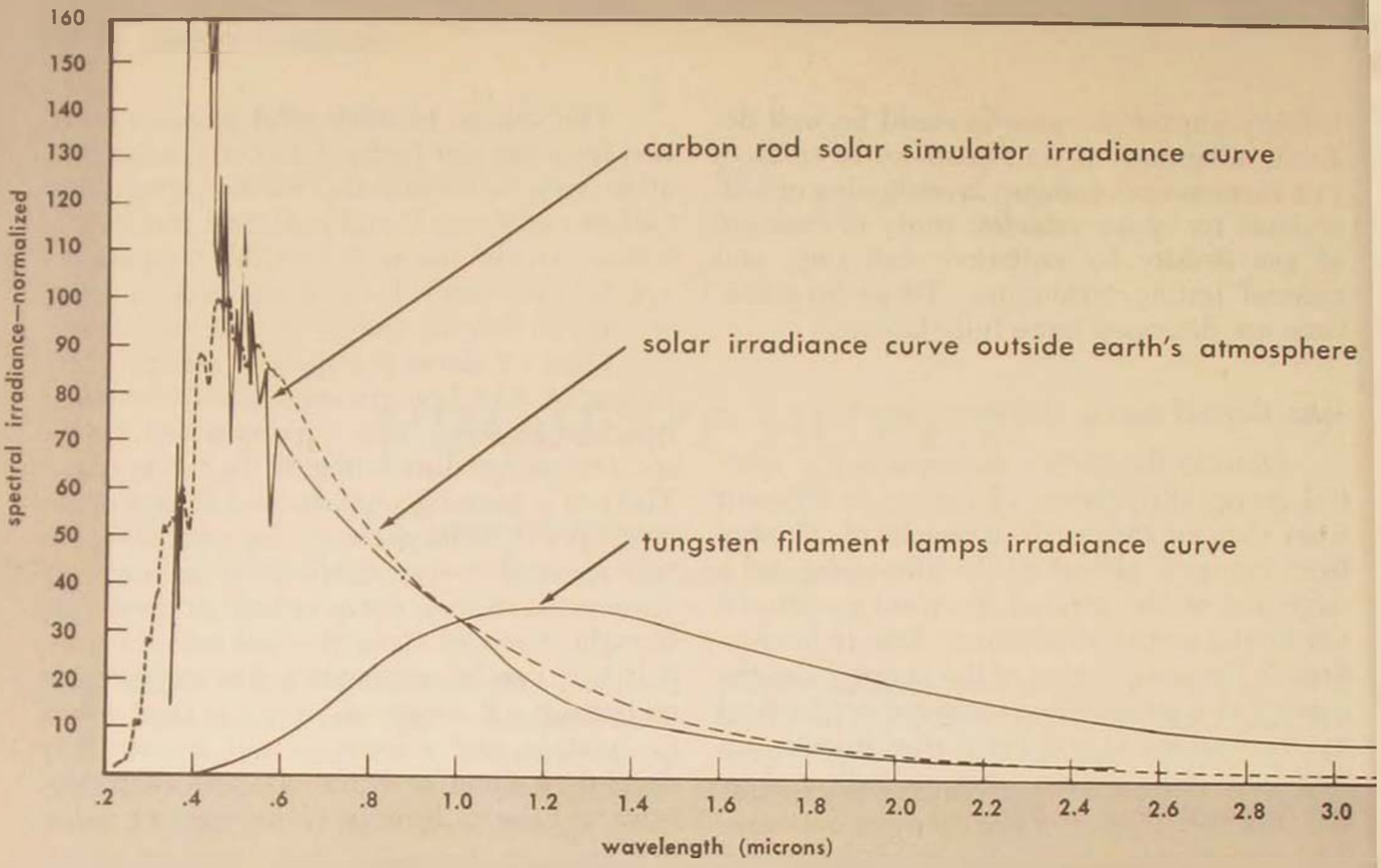


Figure 2. Spectral energy curves

tion, an environmental test with a true spectral match would be required.

The purpose of one project under SUPER, Thermal Testing Techniques, is to provide techniques for properly utilizing solar equipment and to develop techniques for supplying the desired thermal conditions from less expensive heat flux equipment. Work with this heat flux equipment will lead to a technique of using tungsten filament lamps to supply the desired amounts of thermal energy.

The 5V vacuum chamber (5′ x 5′ x 13′) located at Arnold Engineering Development Center (AEDC) was used in this study (Figure 3). The chamber is pumped down to the 10^{-7} mm Hg range (earth orbit pressure) to prevent heat transfer by convection. (Remember that all heat transfer to and from a vehicle in space is by radiation.)

Figure 4 is a photograph of the experimental setup in the 5V space chamber. To describe the experiment briefly, the solar simulator provides one solar constant (1400 watts/m^{-2}) at earth orbit. The heat absorbed by the flat plate from the solar simulator operating at one solar constant is then measured. The next step is to obtain this same thermal energy on the flat plate by use of the tungsten filament lamp. To perform the experiment, the thermal properties of the plate and its coating must be known. In one case an aluminum plate is coated with a special black paint that has a total absorptivity of .97, that is, the plate absorbs 97 per cent of the radiant energy incident on it and reflects 3 per cent. The temperatures of the wall and the plate are measured with thermocouples. The total energy radiated to the liquid-nitrogen-cooled wall of the chamber from the plate is then computed. To minimize liquid-nitrogen consumption, the heat load to the chamber wall must be reduced. To do this a calorimetric technique is used whereby cooling water is circulated through tubes on the back of the plate to remove heat by conduction. The amount of heat removed by this process is then easily computed from the relationship $Q = \dot{m}C_p(T_{water\text{-}out} - T_{water\text{-}in})$ where $\dot{m}$, the mass flow rate of the water, is known;

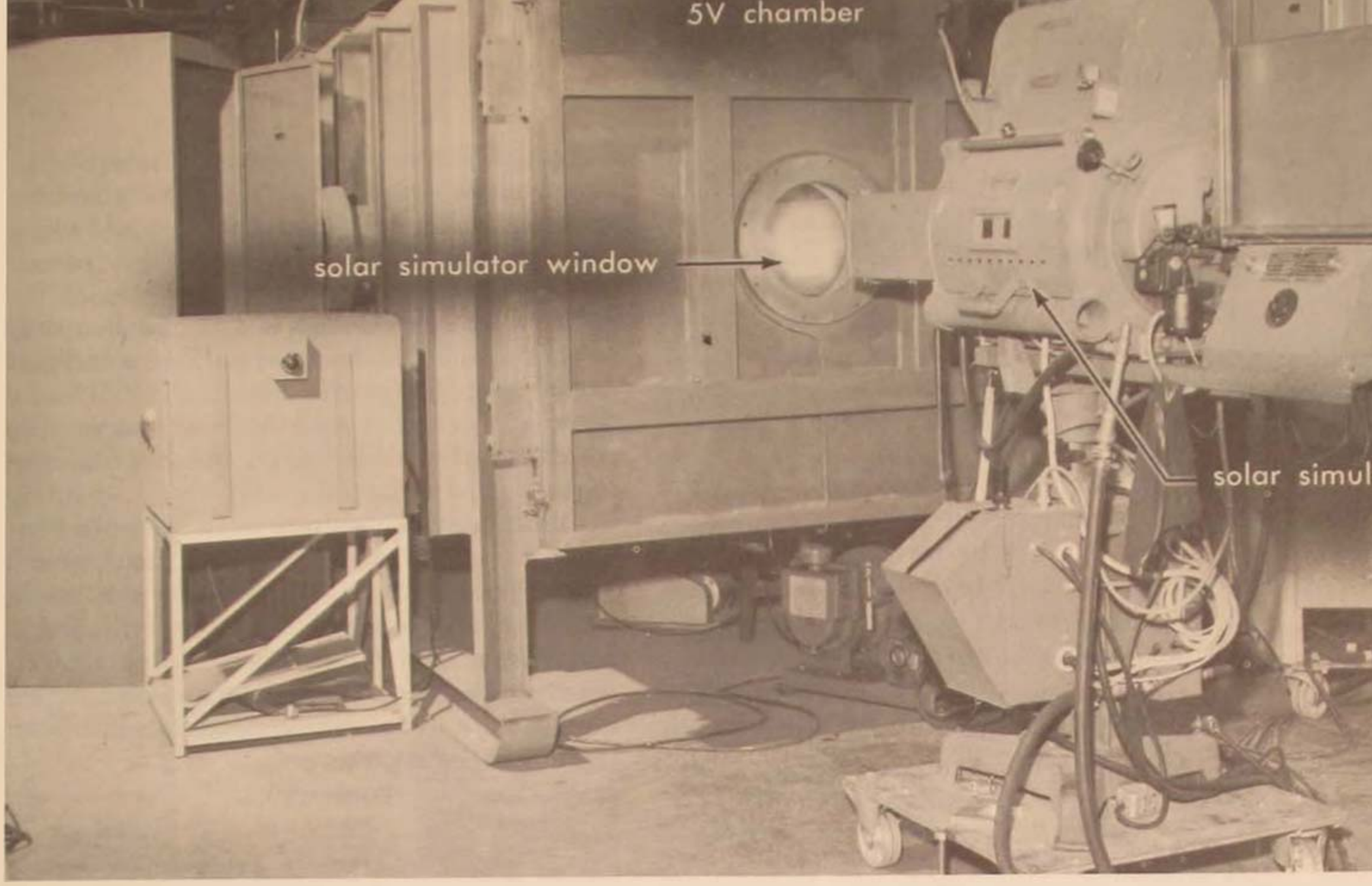
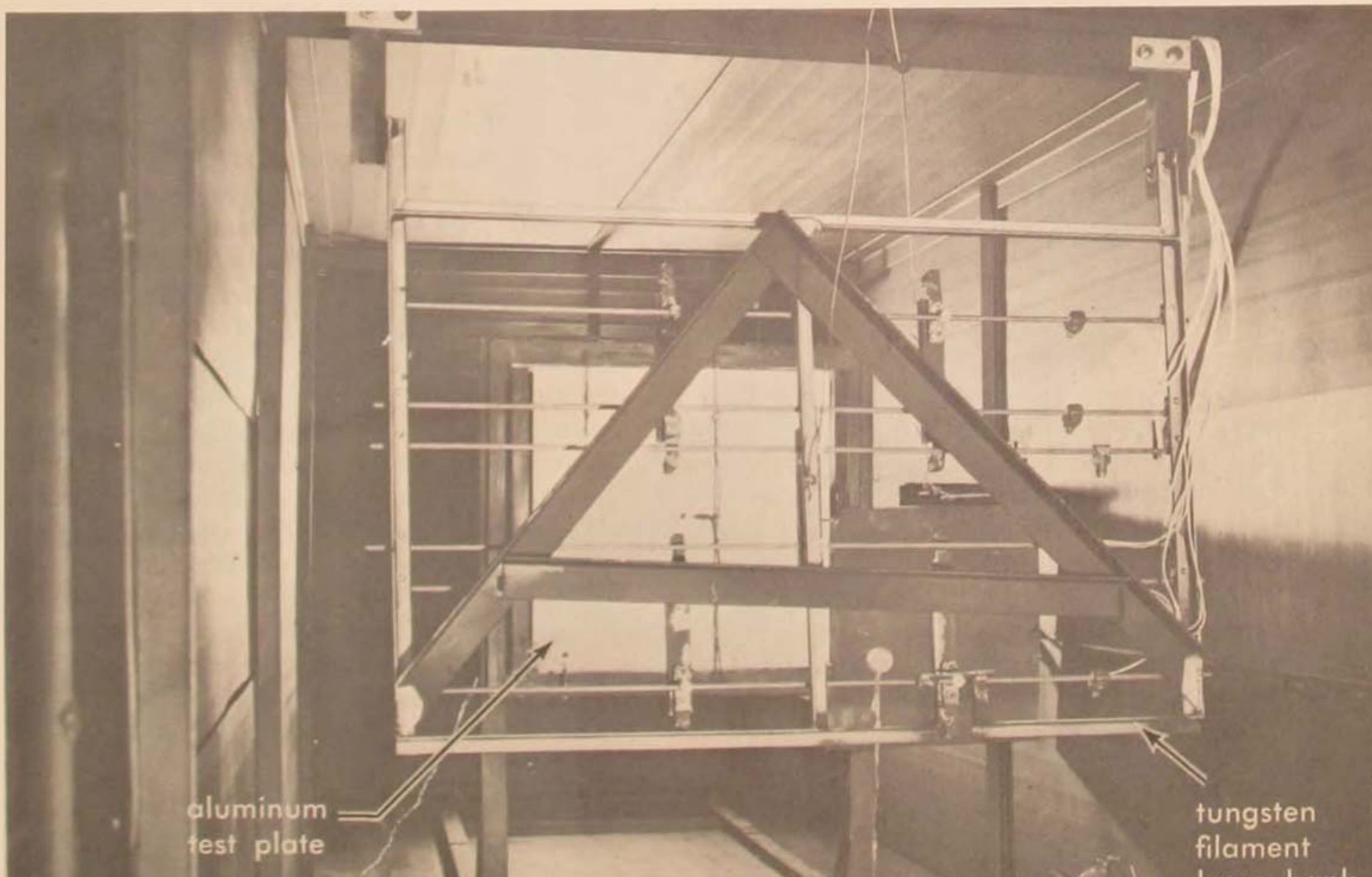


Figure 3. 5V vacuum chamber

Figure 4. Tungsten filament lamps irradiating test sample



C_p (specific heat at constant pressure) for the water is known; and the temperature of the incoming and outgoing water is known. Several radiometers located on the front of the plate measure irradiance or heat input to the plate. As previously mentioned, the absorptivity of the plate with the black paint is .97. The desired conditions are attained when the thermal energy absorbed by the plate is equivalent to 97 per cent of the thermal energy radiated to the plate by the solar simulator operating at one solar constant.

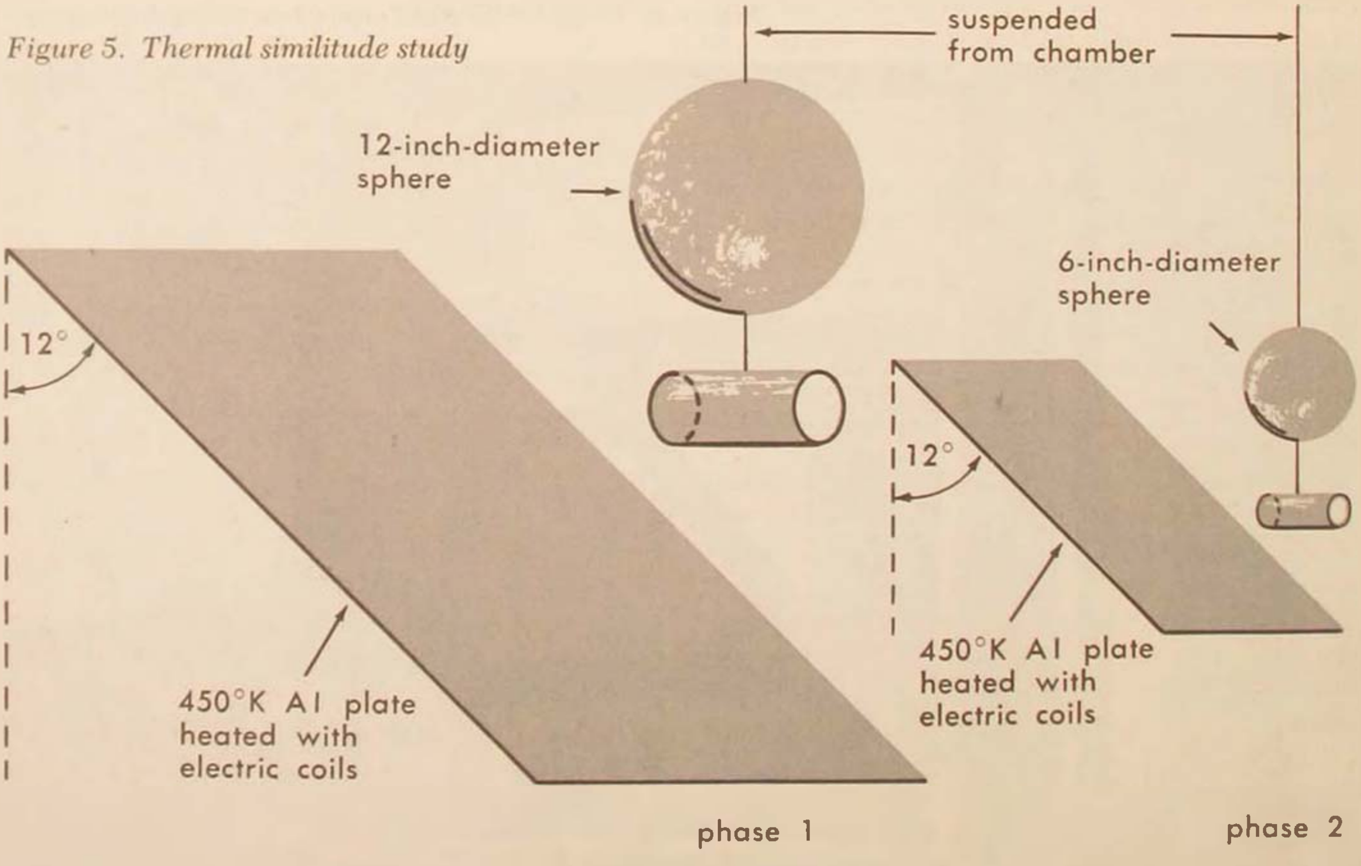
Another project under SUPER in the thermal testing area is Thermal Similitude Studies, wherein the objective is to scale the temperature measurements obtained from a thermal scale model to the actual vehicle. For instance, even without a thermal test of the full-scale space vehicle it will be possible through the use of models to determine the heat flux through the surface of a vehicle and thus the heat load to which an instrumentation box or fuel tank in a space vehicle will be exposed. This experimental program will be run in the 5V chamber at a pressure of 10^{-7} mm Hg. (Figure 5 shows the experimental setup.) The temperature distribution on the 12″-diameter sphere and 4″ x 8″ cylinder is measured when the plate is at 450°K with a certain input power. Through scaling laws the thicknesses of the sphere and cylinder are increased when they are reduced to one-half size for the second part of the test. The power input to the half-size plate is then varied until the same temperature distribution is achieved on the half-size test models.

Still another program in the thermal testing area is Thermal Radiation Measurement Techniques, wherein the objective is to study total radiation intensity detectors for in-chamber adaptability. Part of this study is to investigate thin-film thermopile detector construction as a means of reducing the detector size, thereby reducing the area of the test article blocked out by the detector.

impact data

Another area of joint interest to the Air Force and NASA is the problem of meteorite

Figure 5. Thermal similitude study



impact and possible perforation of critical subsystems and components on space vehicles. The S2 Impact Range at AEDC will be used to furnish experimental data to NASA.

The objective of the study is to provide cratering data on materials useful for space applications. This program will be combined with other cratering studies sponsored by Marshall Space Flight Center, where the overall aim is to develop a theoretical model with experimental justification for impact and cratering processes. Such a model will be useful for extrapolating cratering data to higher velocities on a sound physical basis. A matrix of 80 shots at velocities ranging from 16,000 to 30,000 ft/sec will be conducted using various projectiles and targets. Crater depth, diameter, and volume will be measured, and targets will be weighed before and after each shot to determine mass loss to ejecta. Tests will be run using 1″ and 2″-thick target plates and 1/16″ and 1/8″-diameter spherical projectiles. The accompanying photograph (Figure 6) shows a 1″-thick 304 stainless steel target plate after impact of a 1/8″-diameter 304 stainless steel spherical projectile fired at a velocity of 25,400 ft/sec. Depth of crater was measured to be 0.308″, width of crater was 0.5402″, and volume was 0.0592 cubic inch.

Figure 6. Target plate after impact

low density measurement by electron scattering

The purpose of this investigation is to define the range over which gas density may be measured by electron scattering. Methods will be evaluated for using an electron beam to measure density in a hypersonic gas flow such as in facilities simulating re-entry conditions. The advantage of this method is that no material probe which may disturb the electrical or flow properties of the gas or which may melt when inserted in the region of interest need be inserted into the flow. Basically the density is measured by collecting electrons scattered from the beam by collisions with gas molecules. Charged particles are scattered by the electric field of the atom with which they come into contact. Electrons have the largest charge to mass ratio (e/m) of any known particle and are therefore more easily deflected by electric fields than are other particles, and for a given particle velocity they would be scattered through larger angles than others would. Electrons are thus used in this project because they are the most sensitive probes that can be utilized to measure gas properties.

In the apparatus used in this experiment, an electron beam is directed through the test volume, and then the scattered electrons are detected by a counter collimated so that only electrons scattered in the test volume of the beam can reach the counter directly. (See Figure 7.) After calibration of the system, the comparison of the known density with the indicated density will indicate the accuracy of the measurement and the feasibility of the method.

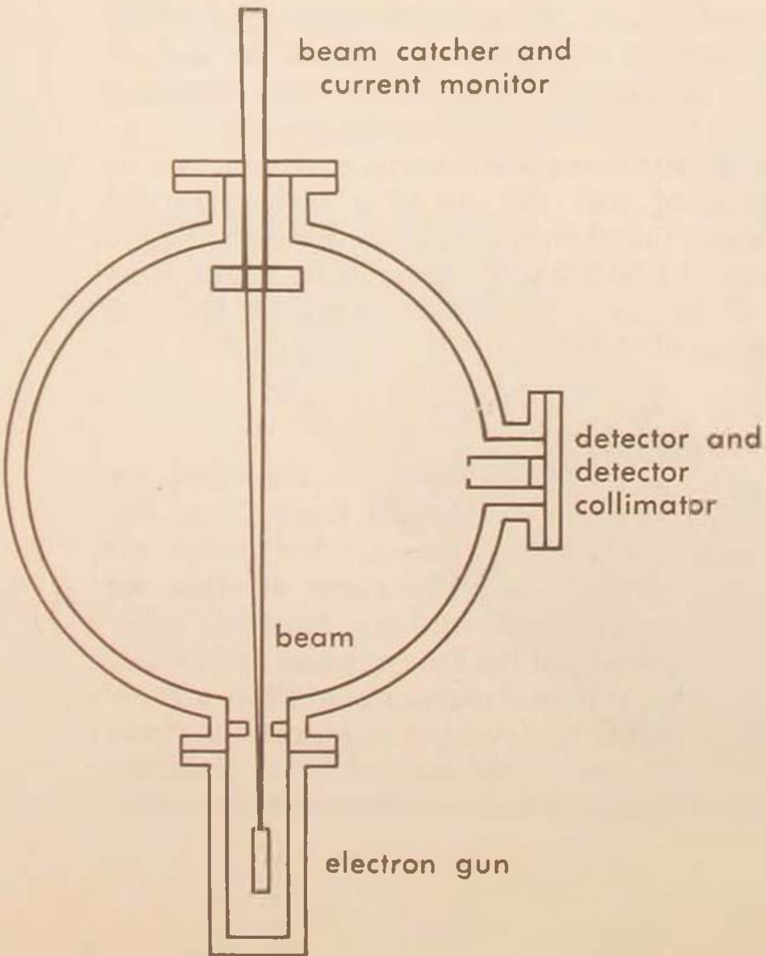
It is believed that the two major problems with this method of measuring gas density, i.e., beam scattering and electron collecting, will be resolved with the use of a higher-powered beam. Initial work on the project entailed the use of a 50,000-volt beam. Under SUPER, NASA will supply a 350,000-volt beam for the next phase of the study.

THE five projects described are being conducted at Arnold Engineering Development Center. However, other projects are being carried out under SUPER at other Air Force centers. At the Air Force Materials Laboratory (AFML) of the Research and Technology Division, Wright-Patterson AFB, Ohio, a study is under way to develop a self-sealant material that will seal a void created by a moderate-velocity particle both at ambient and cryogenic temperatures. The primary objective of this research effort, then, is to develop one or more substituted cyano-sorbic-acid compounds that are useful as self-sealants. AFML will synthesize, determine mechanism of polymerization, and test the cyano-sorbic-acid systems. All polymeric materials proposed under this project will be characterized by AFML as to elemental chemical analysis, determination of molecular weight, melting point, boiling point, viscosity, etc. One side of the sample will be subjected to vacuum and the other to atmospheric conditions during the course of testing.

Materials are now under study which are being used to modify the thermal radiation properties of the "skin" of space vehicles. Such a study, Thermal Control Surfaces for the Extraterrestrial Environment, is being carried out under SUPER at the Air Force Materials Laboratory. Part of the study is to make use of organic dyes to increase absorptance in a metallic pigmented system. Another part of the study deals with obtaining surfaces with very high reflectance throughout the electromagnetic spectrum.

Exposed cryogenic storage tanks, various types of radiators for heat rejection, communication antennas, optical surfaces, and overall structures for use on the lunar surface will have surfaces coated with specially developed materials. Any adherence of lunar dust to the surfaces would greatly deteriorate the passive thermal control characteristics upon which the design of these systems was initially based. At the Air Force Cambridge Research Laboratories, Laurence G. Hanscom Field, Massachusetts, a study is progressing under SUPER to determine the likelihood of dust adhesion to metallic surfaces and to propose possible countermeasures. Studies of the phenomenon of adhesion will be undertaken, since little is known of the nature of bonds that hold dust particles together. The role of each type of bond will be determined in order to develop effective countermeasures.

Figure 7. Schematic of electron beam apparatus



ALL THESE programs were initiated in FY 1964 and are being presently pursued. Additional and follow-on projects were added to the SUPER program by Marshall Space Flight Center and various Air Force centers and laboratories in FY 1965.

The resources required to accomplish this kind of work for NASA are technical competence, facilities, experience in the technical areas of concern, and management capability. All these resources are available within the Air Force and can be used in accomplishing the NASA tasks under Project SUPER. In addition, the Air Force has an established educational program which trains young officers for assignments in the various research facilities operated by the Air Force. These officers are technically competent but need management training and experience. Under the concept of Project SUPER both NASA and the Air Force derive benefit from this working relationship. In the first place, NASA pays only for materials and such equipment or modification of existing equipment as are required to perform tasks not otherwise included in the Air Force research program. NASA also pays for any supplemental contractual service required relative to the program and for travel expenses incurred by Air Force personnel during the actual accomplishment of the work. On the other hand, NASA does not pay for the in-house scientific and management manpower, nor does NASA pay facility utilization costs on available research facilities.

The Air Force in its R&D laboratories and centers has current work efforts as well as current and potential capabilities for expanded effort in areas of interest to NASA in the general field of extraterrestrial research. The Air Force has been engaged in basic as well as applied research and development since its establishment in 1947. With the advent of the missile and space era in 1954, much of this in-house capability became space oriented. A vast complex of research and experimental facilities and companion management teams was developed by the Air Force to meet the requirements of this new era. Many of our present space programs were conceived and proved feasible by these Air Force teams.

With this accumulated experience the Air Force, through SUPER, can assist NASA by participation in advanced studies, concepts, and testing programs concerned with extraterrestrial research. The Air Force effort, by clear identification with NASA interests and by expansion within existing manpower capabilities, will serve to fulfill some of NASA's requirements at a lower cost to NASA than by other means. The result will be a net savings to the Nation. This experience, coupled with the Air Force's space facilities training program, will enable the Air Force to maintain a high degree of proficiency in space technology. This proficiency in space research will be ready for application should a military mission in space be identified at a later date.

The Air Force also is in a position to give the NASA research program an adequate priority rating to ensure results in a timely manner. These priorities are derived from the complementary programs in-being in the Air Force research facilities. There are other intangible advantages to NASA through Project SUPER channels. First, there need not be any contractual arrangement. Since both are Government agencies, it is only necessary to have an administrative agreement whereby NASA furnishes the Air Force funds for the material, travel, special equipment, etc., and the Air Force agrees to furnish labor and facilities. Second, all NASA research projects have the advantage of related studies that are already completed or that are under investigation by Air Force personnel. Both of these advantages provide for a "quicker start," since the research which the Air Force accepts blends into existing programs already under way, so that there is no waiting period until existing work stops and NASA research is actively started.

THE FOREGOING is intended to present a comprehensive picture of the concept of Project SUPER. The project is now one year old. The administrative procedures have been completely worked out, and there are eight active tasks under way at Arnold Engineering Development Center, Air Force Cambridge Research Laboratories, and the Research and Technology Division laboratories at Wright-Patterson AFB. At the present time only one of the NASA centers, the Marshall Space Flight Center at Huntsville, Alabama, is actively involved in the program. It is anticipated that the program can be further expanded to other NASA centers with equal success.

Project SUPER can never become a vast program because of the nature of the ground rules under which it was established. The size of the program is determined by the ability of the Air Force to accomplish the NASA research without any increase in manpower or funding at the center and laboratory level. This means simply that the work accepted from NASA must blend with the work already under investigation at the laboratories and centers in such a way that only a minimum amount of additional effort and funds is required to obtain the answers to the NASA questions. The important part of the Project SUPER concept is that it is a national effort to provide maximum utilization of both material and manpower resources in support of the national space goals.

Arnold Engineering Development Center, AFSC

LUNAR CHARTING

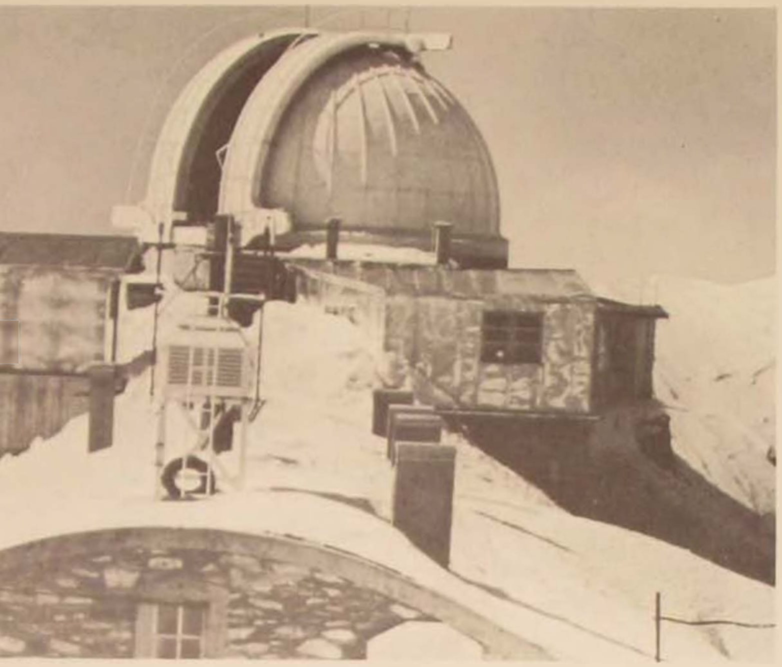
Colonel John G. Eriksen

MAPPING the moon is not the easiest thing on earth! Challenges facing the lunar cartographer must be met head-on to keep pace with the advances in space technology. These challenges take the form of questions like: What is the resolution of lunar photography? How much detail can we really see on the moon, even with the largest of telescopes? How can we reconcile the many variables in determining lunar elevations, since there is no "sea level" on the moon? What is the best scale for the lunar charts that one day will be used by our astronauts?

These and other equally baffling questions are being pursued—and many answered—by Air Force cartographers at the Aeronautical Chart and Information Center (ACIC), where, back in 1960, an ambitious program to chart the moon was begun.

Some of the factors in lunar topographic mapping present fundamental problems that limit the accuracy of lunar charting. One of these is the frustrating fact that all photographs of the moon taken from the earth have imperfections that affect resolution. The best lunar photographic resolution ever achieved is about four tenths of a second of arc, which is about a half mile on the surface of the moon, whereas visual observations with large telescopes under the best viewing conditions can detect a multitude of minute details never captured on a photograph. Thus visual observation permits the annotation of the best photographic images with small and subtle details.

Lunar features can be best mapped near the terminator, where a low sun casts long shadows, as seen in the twenty-one-day-old moon at the left. In the other, shown just after full moon, note the crater rays, which are quite prominent under full illumination. The dark areas are the maria or "seas" of the moon. Photographs were taken at Pic du Midi Observatory, France.



Pic du Midi Observatory, nearly two miles high in the Pyrenees Mountains of France, is the site of an Air Force program for photographing the moon.

Another problem—and probably the chief cause of trouble—is the earth's atmosphere. It absorbs part of the light passing through it, and its turbulence produces small random variations in refraction of the image of detail points on the moon.

Still another troublesome fact is the moon's distance from the earth, an average of 239,000 miles. At this distance an object one mile across measures only three-thousandths of an inch in a typical telescopic photograph.

And then there is the disturbing phenomenon of the moon's rotating on its axis in the same period it takes to go around the earth. It keeps nearly the same face turned toward us at all times, thereby limiting the variation in perspective or libration° that can occur. For this reason conventional stereoscopic mapping techniques cannot be applied extensively to lunar charting, and new techniques have had to be developed to provide needed information for ACIC lunar charts.

Despite these obstacles, the Aeronautical Chart and Information Center is producing a series of charts, mosaics, and atlases of the moon. Collectively they constitute the best graphic representation of the moon available. These graphics and the methods used in producing them are the subject of this article.

°**libration.** Usually the real or apparent oscillation of the moon that allows us to see some of the hidden side.

the LAC series

The predominant ACIC lunar chart effort is concentrated on the production of a 1:1,000,000-scale series of Lunar Astronautical Charts. The LAC series, as it is called, is coordinated for the entire surface of the moon in much the same way as the 1:1,000,000-scale World Aeronautical Charts (WAC) are coordinated for the earth. The LAC is also similar to the WAC in format, except that only 144 LAC's are needed to cover the moon whereas 1851 WAC's cover the earth.

Since only a maximum of 59 per cent of the lunar surface (under all conditions of libration) can be seen from the earth, about 80 charts or portions thereof can be produced. As of January 1965, 20 LAC's, covering the central portions of the lunar disk, have been published.

The LAC is designed primarily as a topographic map to show the surface of the moon in the greatest possible detail. The 1:1,000,000 scale was chosen for its compatibility with the maximum resolution that can be realized from earth-based observations.

The projections on which the LAC's are drawn are generally standard. The Mercator projection (used from 16° N to 16° S) embraces two bands of charts which join together perfectly to form a continuous strip around the lunar equator. The Lambert Conformal projection is used from 16° N and S to 80° N and S, while the polar areas are drawn on the stereographic projection.

Using an idealized light source simulating the evening illumination, scientific illustrators artistically render surface features of the moon to give a three-dimensional effect. Very low or high features are drawn as though no shadows were cast to hide their details. Differences in steepness of features are portrayed by varying tone densities: the lighter the tone, the more shallow the feature.

Another important surface characteristic is the ray systems, distinctive features evident in full-moon photography. Though it is unnatural to see both the relief and rays on a single lunar photograph, both features are shown on the same chart simply by printing them in two different colors. Olive green for relief and blue gray for the full-moon ray system have been selected for the LAC because of their aesthetic values. (The colors on the moon—if indeed there are any—have not been positively determined.)

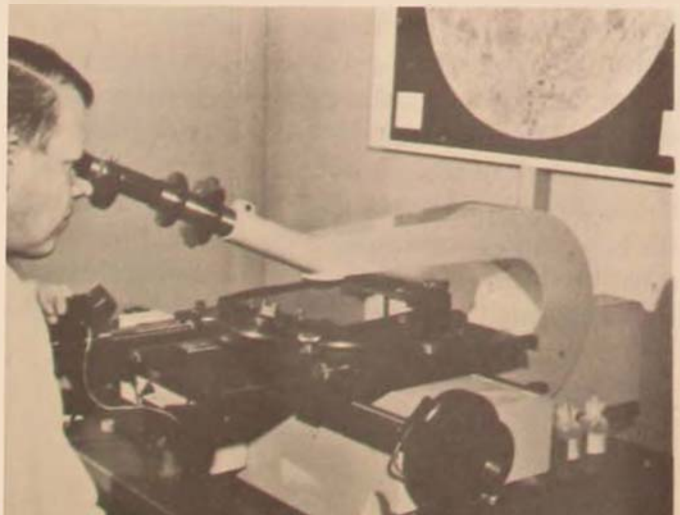
Contours, printed in brown to complement the olive green relief, are related to an assumed spherical figure of the moon whose radius has been established at 1738 kilometers. The contour interval of 300 meters was selected for its convenience in converting to the American usage of the 1000-foot interval. (So that most of the contours would have positive values, ACIC established a zero point or "moon datum" at 2.6 kilometers below the 1738-kilometer radius.)

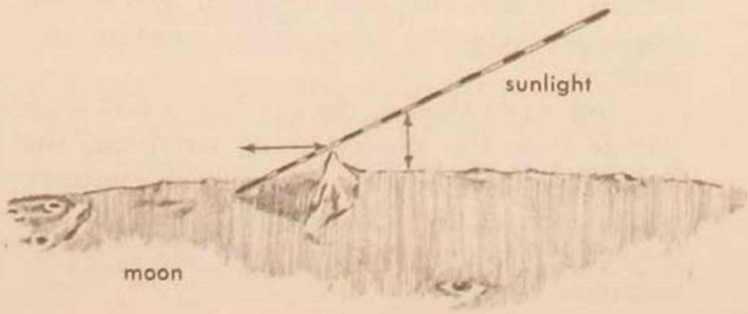
visual telescopic observation

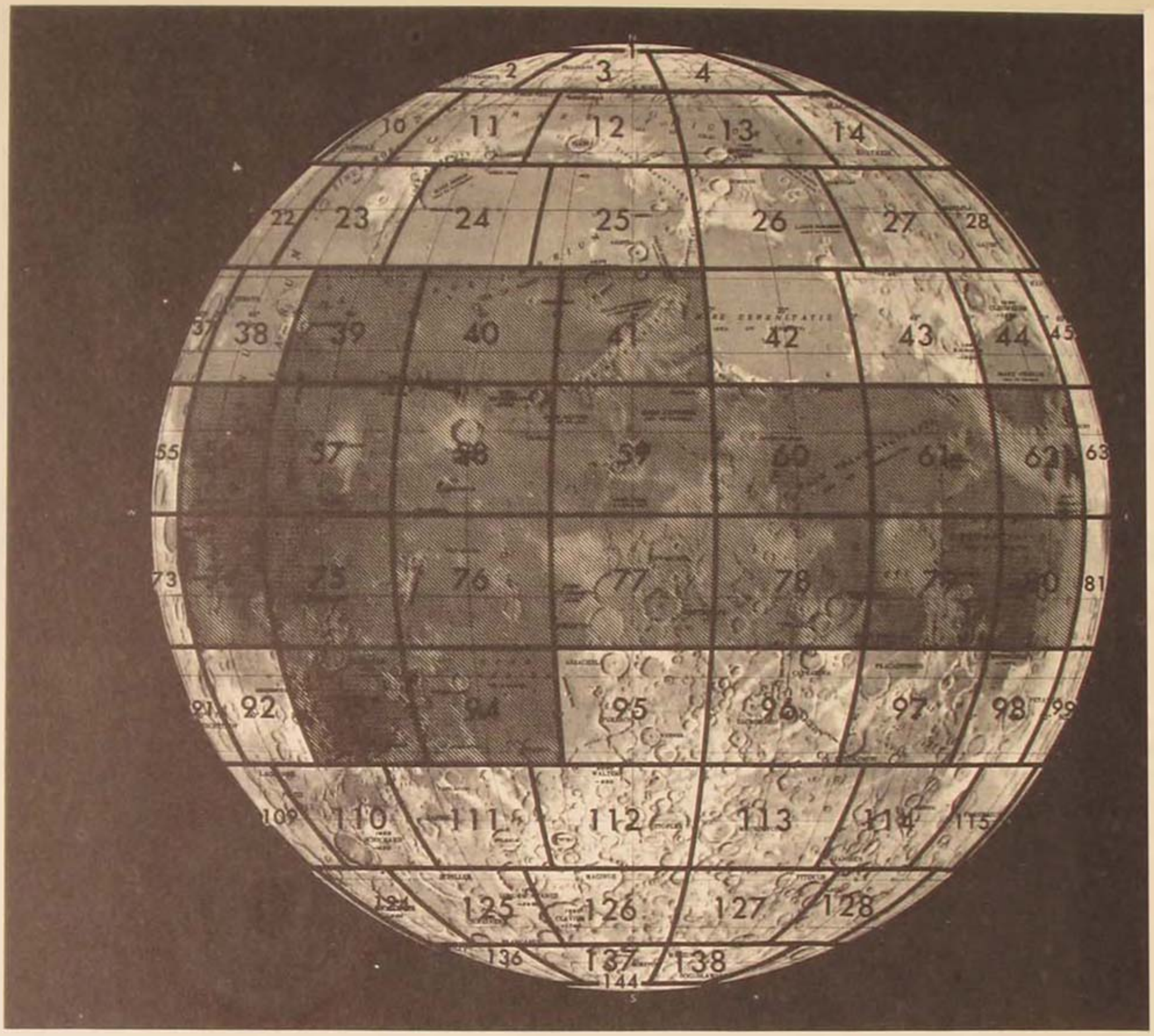
Early in 1960, ACIC recognized that a good

Determining heights of lunar mountains by measuring the length of the shadow cast by the peak dates back to the 18th century. Refined techniques developed at Manchester University are used by USAF cartographers to determine relative heights on lunar charts. . . . ACIC uses linear comparators (below) to measure the length of lunar shadows in determining the elevation of mountains and the depth of craters.

Piton







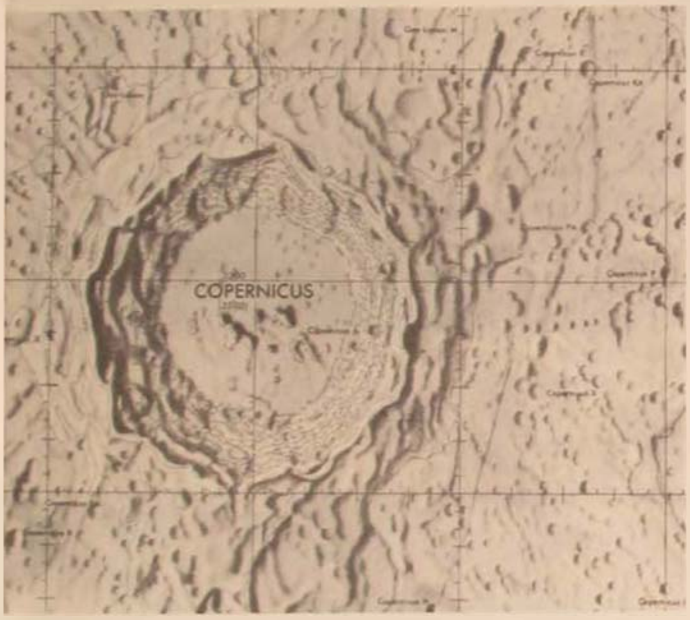
telescope located where the best viewing conditions would prevail was most vital to the success of a modern lunar mapping program. Consequently, in September 1961, a permanent ACIC observation unit was established at Lowell Observatory in Flagstaff, Arizona. There observers at 20-inch and 24-inch refracting telescopes keep the moon under constant surveillance.

But, even though the eye can detect finer details than the photographic emulsion, the turbulence of the earth's atmosphere produces image motion and defocusing similar to the wavering image movement seen above a hot radiator.

The period of stability or good "seeing" may be very short, requiring instantaneous response and recognition of visible details. Since the human eye has the characteristic of rejecting poor images and retaining good ones (even of short duration), the observer builds a composite picture of the details of a feature over a period of time. He then sketches the feature or annotates photographs at the telescope with the particular feature he has observed.

Under troublesome viewing conditions, this is a slow process because the observer must concentrate on a single feature at a time and must wait for periods of stability. As conditions improve, the observer may be steadily occupied

An index to the Aeronautical Chart and Information Center 1:1,000,000 scale Lunar Aeronautical Charts series (left). By November 1964, nineteen of the charts had been published. . . . A section of ACIC lunar chart LAC-58 covers the Copernicus crater. Elevations are shown by 300-meter contours.



in recording detail. Then, too, there are the rare moments of stability when a rush of fine detail reaches the eye at one time, making it virtually impossible for the observer to capture it all. At times like this, the camera is a most valuable tool.

The telescopes at Lowell Observatory are equipped for both visual and photographic observations so that one may be supplemented by the other. The supplementary photography is obtained with a 70-mm motion-picture camera. The eyepiece and camera are arranged as one unit, with reflex prisms diverting the optical beam to the eyepiece. Because of the ever present small residual motion, the camera cannot capture the details with all the sharpness and clarity that the observer can, even during the longest periods of stability or steadiness. However, even a slight image of the small features is a great help to the observer: having seen them clearly with the telescope, he can readily interpret them on the photograph.

Several observers are involved in describing the detail within a chart area over a six-month period of observations. Most observations are made under oblique illumination so that vertical dimensions are emphasized by the shadows cast by the feature.

Normally, observations are made along and within 30° of the terminator.* In the illuminated portion, the shadows are optimum for detail interpretation. Along the terminator, very low and gentle features (such as maria** ridges and valleys) will show up prominently. Craters, small prominences, and rills can easily be interpreted at 5 to 15 degrees from the terminator. Very large or steep craters or mountains can best be interpreted when the sun angle is between 15 and 30 degrees. However, higher illuminations may be necessary to see some crater floors and the finer details of crater rays.

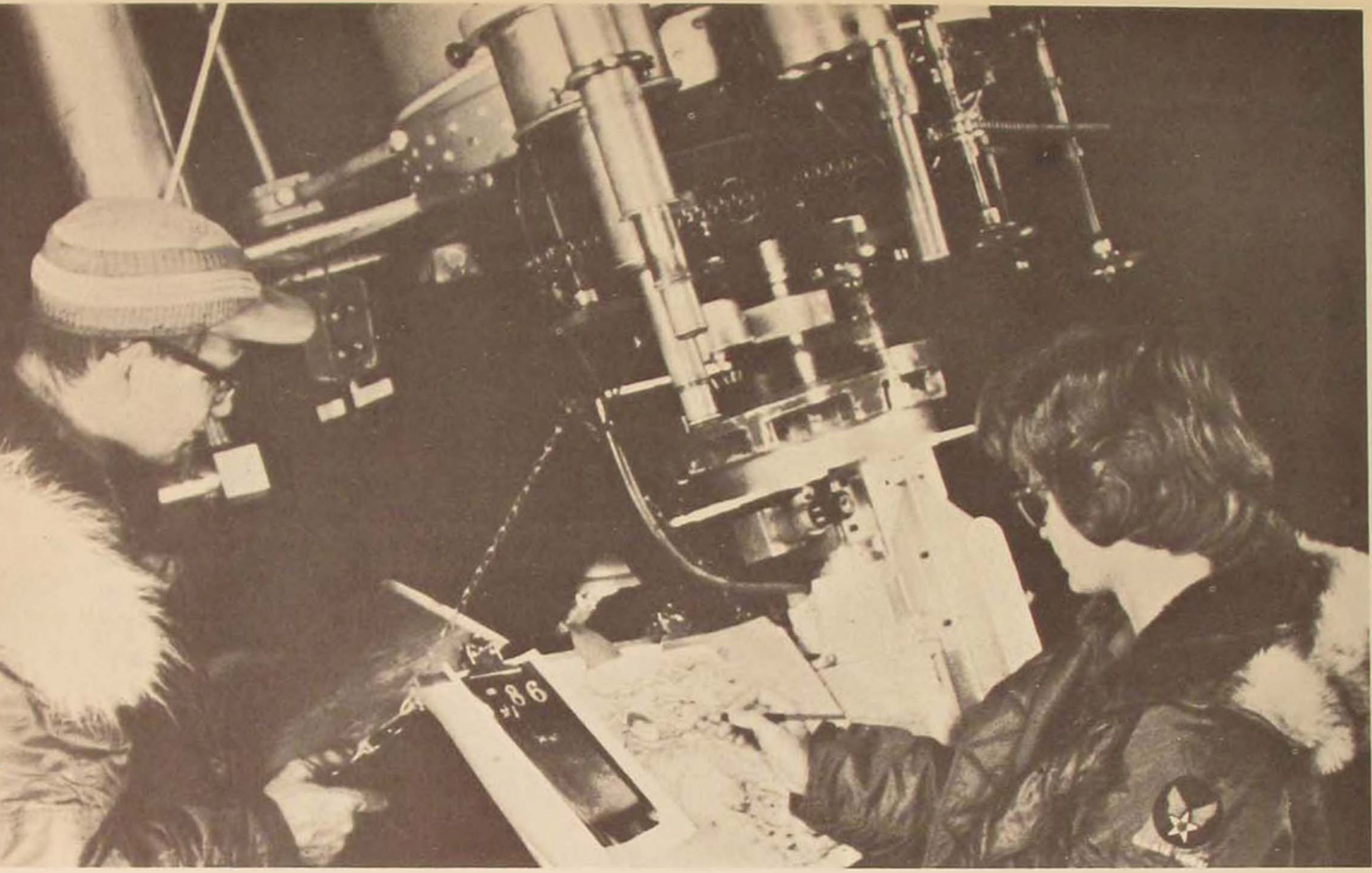
determining lunar elevations

Pic du Midi Observatory, located in the Pyrenees Mountains of southern France and noted for its excellent "seeing" conditions, is probably the highest permanently manned observatory in the world. Sitting atop a 9400-foot peak, this observatory is engaged in photographing the moon for the Aeronautical Chart and Information Center, taking low sun-angle photographs (timed to the nearest second) needed in the shadow-measuring technique for determining elevations shown on Lunar Astronautical Charts.

The shadow-measuring technique involves a simple trigonometric solution relating several angles which have been previously recorded

*terminator. The great circle on the moon that is the boundary between day and night. We speak of the sunrise and the sunset terminators.

**mare (*plural* maria). Any of the large, dark areas on the moon. Because the earliest observers believed such an area to be a body of water, they named it "mare" which is Latin for "sea."



At Lowell Observatory, Flagstaff, Arizona, (above) ACIC selenographers use large telescopes to identify minute lunar features. . . . Photogrammetric technicians skillfully match many lunar photographs to construct a composite photographic picture of the moon.

or are being measured. With the help of ephemeris tables,* and knowing the time at which a photograph was taken, one can compute the relative positions of the observer, the sun, and the object on the moon from their respective latitudes and longitudes. Variations in perspective due to libration must also be accounted for, and because of the elliptical nature of the moon's orbit the scale of the photograph can change by as much as 14 per cent. After these factors have been determined, the measured length of the shadow in the photograph can be converted to the physical length of the shadow on the lunar surface. Again using the time of exposure, one can establish the angle of the sun above the horizon at the particular point on the moon. From these combined factors, the relative height of the object can be found. (Reducing this mass of data to a usable answer was rather time consuming by desk calculator. Today's electronic computers, however, make it a routine task.)

Obviously the reliability of heights determined by the shadow-measuring technique depends to a large degree upon how accurately the lunar shadow can be measured. Two instruments are being used at ACIC to accomplish these measurements: the linear comparator and the microdensitometer.

The linear comparator uses a precision lead-screw, calibrated to an accuracy of 1/25,000 of an inch, to move the photograph beneath a viewing spot over which cross-hairs are projected. The number of turns of the screw determines the measurement of the distance across the shadow.

The microdensitometer uses a photoelectric cell to measure the amount of light going through a negative image. As the photograph is moved through the light path, density traces are automatically recorded on calibrated sheets, from which the shadow lengths can be measured.

The linear comparator has the advantage of speed over the microdensitometer and is therefore used to measure the numerous smaller craters of the moon. The microdensitometer is better suited for measuring shadows of large craters that have terraced walls and irregular floors.

When the computations have been completed, it can be stated that a given peak is a certain calculated distance higher than the point on the lunar surface occupied by its shadow-tip. For any given LAC area, several thousand of these measurements may be needed to correlate with other computed elevations on the lunar surface in order to establish the network of 300-meter contours.

lunar control

The shadow-measuring technique provides relative heights only, i.e., the depth of a crater floor as related to its rim and the height of its rim as related to the surrounding area. There is no natural datum on the moon such as "mean sea level" on earth. To obtain such a datum, the shape of the moon must be determined—a problem which has defied selenographers* for many years.

Since a photograph is a two-dimensional recording medium and the moon is a nearly spherical object, one of the major efforts of the lunar cartographer is directed toward "surveying" or measuring the round moon on the flat surface of a photograph. Remember, too, that taking positions of lunar features at a distance of almost a quarter of a million miles adds considerably to the problem. This problem has been attacked by many astronomers and mathematicians of the past, and the attack is still going on.

The efforts of several of these eminent scientists are worth noting. In 1901 Professor Julius Franz published a summary of positions of 150 moon craters. This work was based on nine fundamental positions derived from heliometric measures.

In 1958 G. Schrutka-Rechtenstamm reviewed the original measurements and derived a new set of positions for each crater and a new determination of the height above or below a mean sphere of the 1738-kilometer radius of the

***ephemeris tables.** The American Ephemeris and Nautical Almanac contains the relative positions of the earth, moon, and sun for any day of a given period.

***selenographer.** One who studies the surface of the moon. The lunar equivalent of a terrestrial geographer.

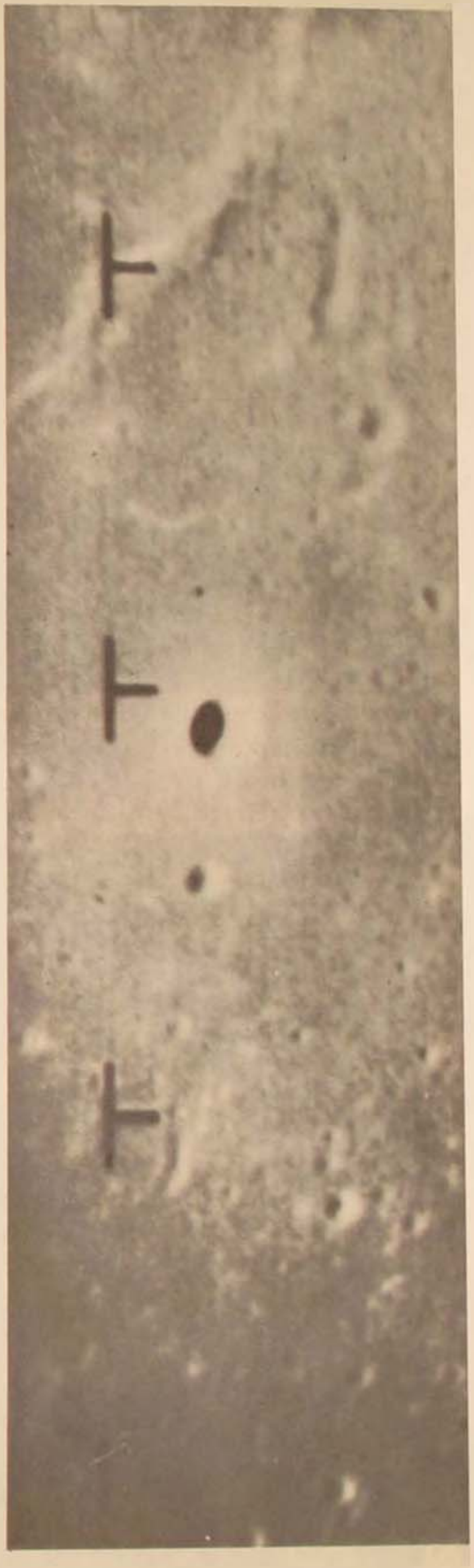
moon. Schrutka gave his determination of the probable error of a single height as ±1.23 km from results ranging from ±0.44 km to ±4.7 km. In 1963 Dr. Ralph B. Baldwin extended the Franz-Schrutka positions to 696 selenocentric control* points from measurements made on five Lick Observatory plates.

In June 1961 Dr. Gerard P. Kuiper and his associates, in collaboration with ACIC, produced an orthographic grid of the moon based on approximately 5000 positions superimposed on high-resolution lunar photographs. This predominantly selenographic control** consisted of evaluated positions by S. A. Saunder and J. Franz, including unpublished positions by D. W. G. Arthur.

The positioning of innumerable lunar formations is still dependent on old existing selenographic control. The inherent error in the old plates and antiquated measuring equipment cannot be assessed satisfactorily. However, some estimation can be made of the error resulting from disregarding elevations of the positions above or below the mean lunar radius. For example, the horizontal position of a point whose elevation is two kilometers above the mean surface would be displaced 700 meters at 20° arc from the center of the disk, 1700 meters at 40° arc, and 3500 meters at 60° arc. Obviously, excessively large errors exist in the old system.

Therefore, a new program to develop selenocentric control points is in progress at the Aeronautical Chart and Information Center. By use of a method of multiple intersections, it employs 31 selected Franz-Schrutka positions that have been accepted as fundamentals for a first approximation and 165 new positions evenly distributed over the lunar disk. Successive iterations are designed to establish the three-coordinate derivations of the new positions and thereby improve the positions of the fundamentals.

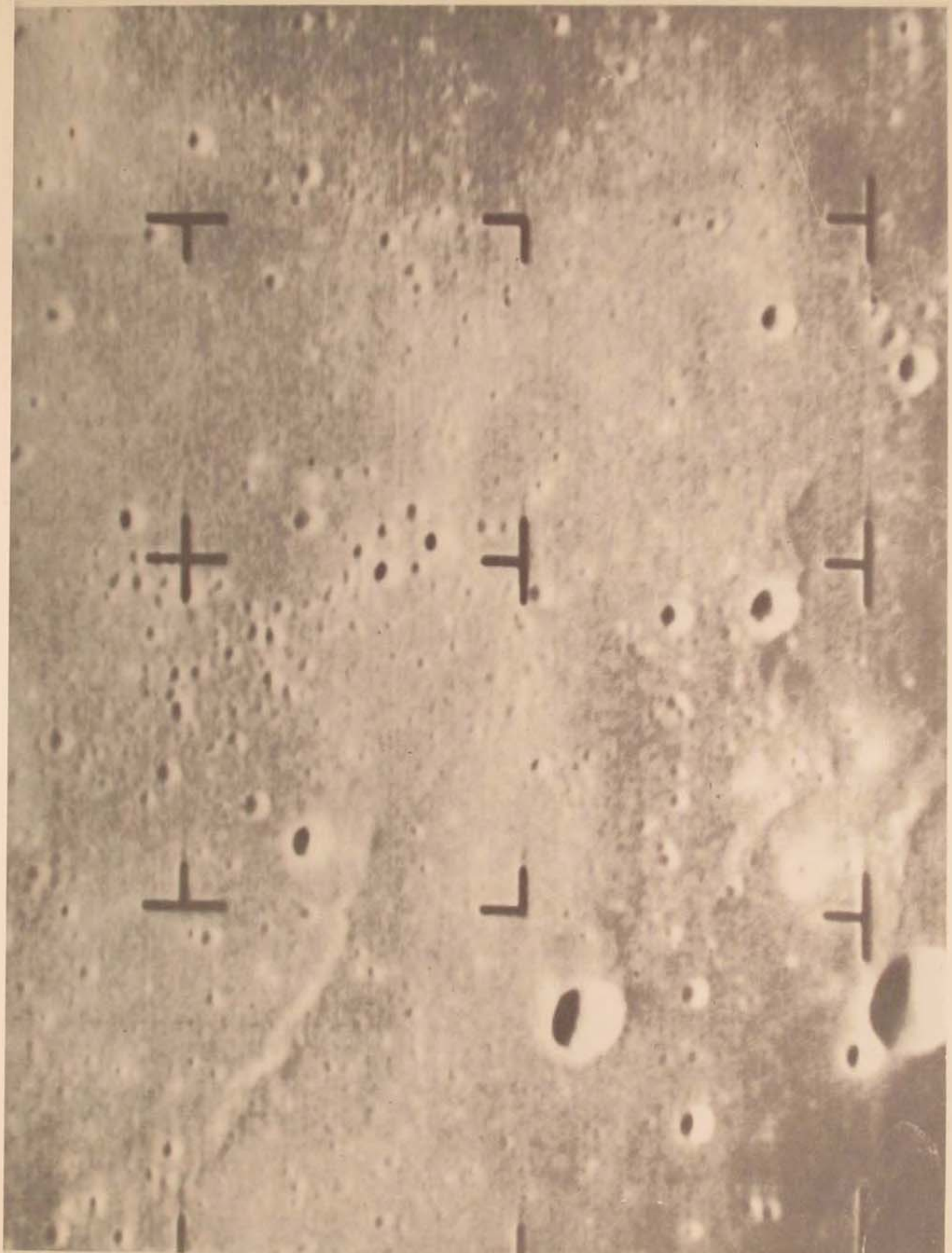
Four sets of five sequential photographs



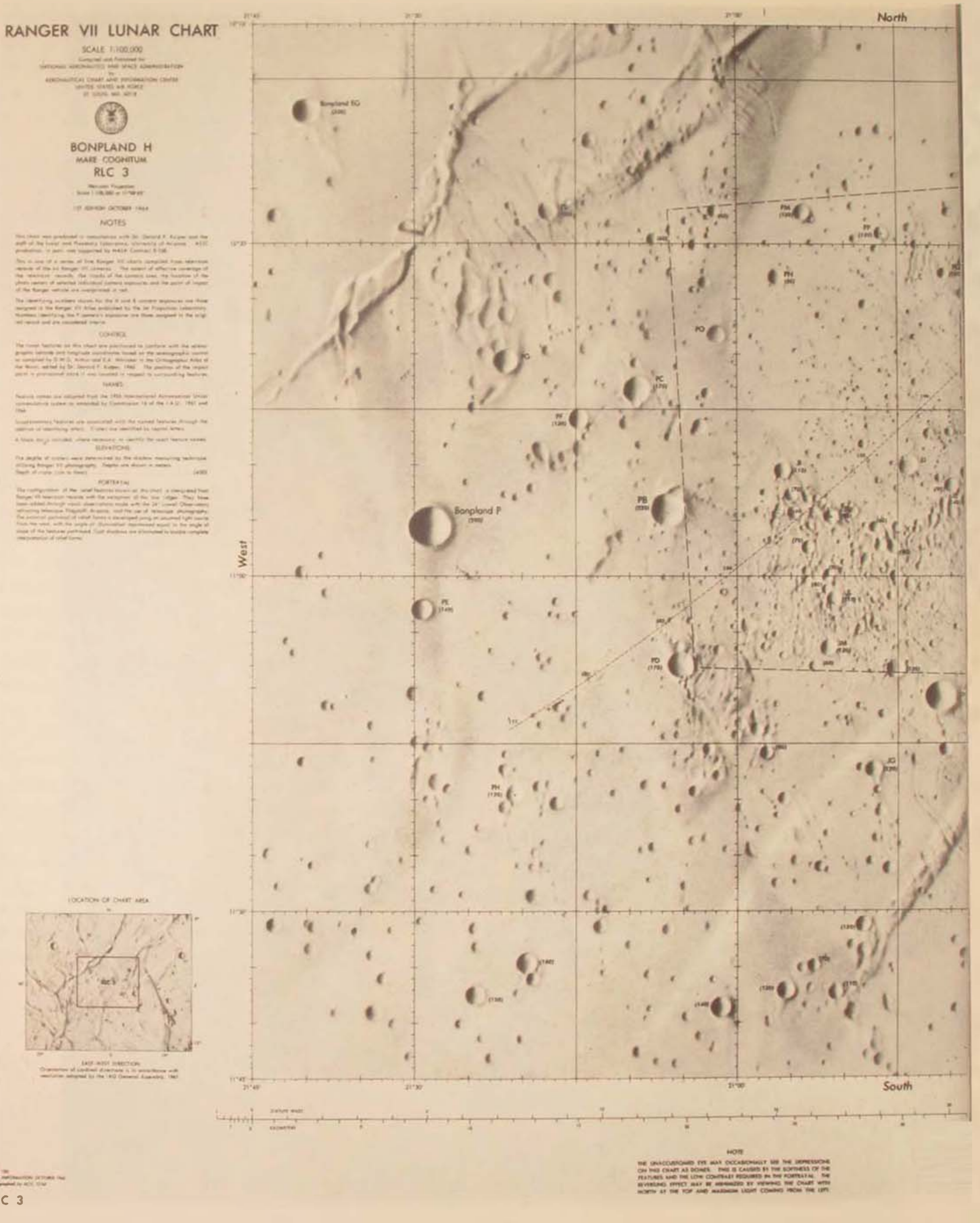
*selenocentric control. Three-coordinate derivations of lunar surface positions referenced to the center of the moon. Coordinates are expressed in latitude, longitude, and elevation above or below an assumed reference surface.

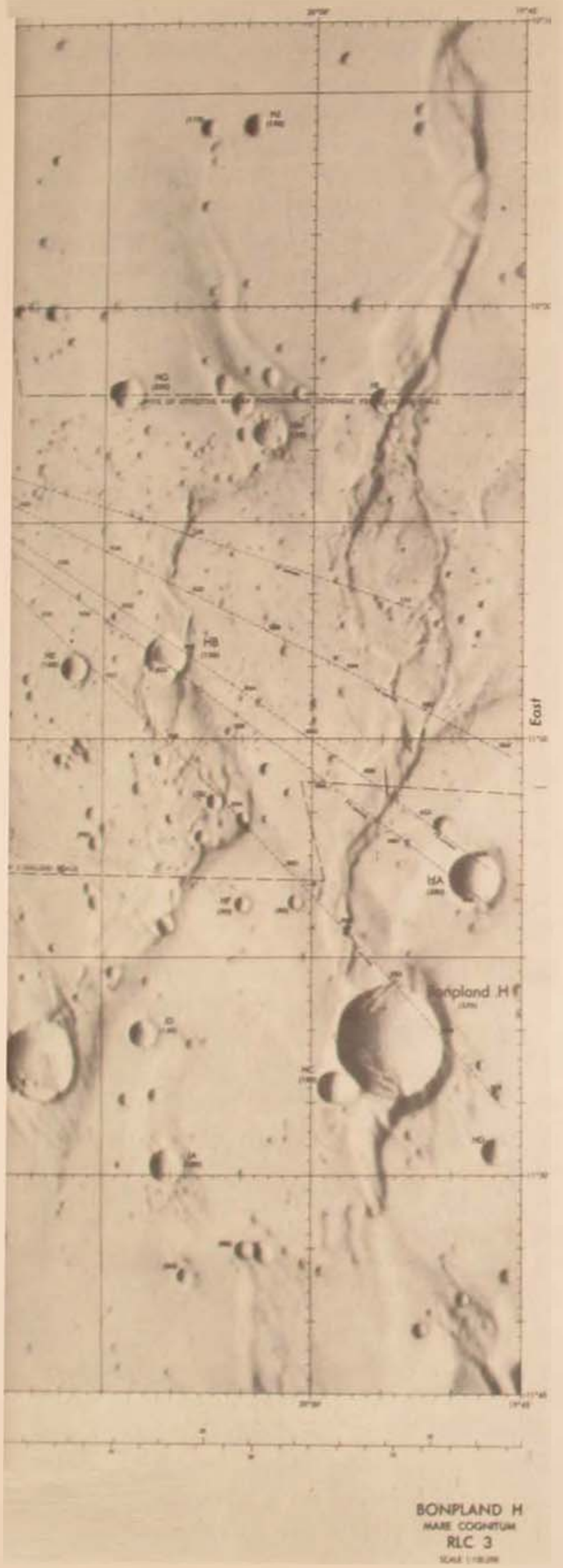
**selenographic control. Two-coordinate derivations of latitude and longitude based on the assumption that all lunar surface positions lie on a smooth surface.

A photograph taken by the Ranger VII "A" camera from an altitude of 101 kilometers (about 63 miles)



One of the five Ranger Lunar Charts prepared from Ranger VII material. It includes an overprint of lunar nomenclature and a latitude-longitude grid.





(80 photographs total) representing libration in each quadrant of the moon are selected for measurement. Two operators, working independently, measure the 196 points (31 plus 165) on each photograph. (The dual effort minimizes the possibility of accidental errors in measuring and atmospheric distortions of the lunar photographic image.)

Measurements in each librated plane are reduced separately to a single set of coordinates by a series of least-square linear transformations. The accepted coordinates of the 31 fundamental points are transformed into their perspective positions on the librated planes. Then the measured coordinates of the fundamentals are transformed into perspective coordinates. The parameters of this transformation are applied to the 165 points measured in each librated plane to establish their perspective positions. All perspective coordinates thus established are transformed finally into selenocentric coordinates.

When the four references have been reduced to this phase, each point will have four positions on different librated planes and will be traversed by four perspective rays which will intersect at that point's surface position. In practice, the perspective rays of any point will not intersect because of errors which cannot be eliminated entirely. Therefore, the final coordinate position of each point is determined from the maximum convergence of these rays through the method of least squares.

As of January 1965, 75 per cent of the references have been reduced. A comparison of the differences between coordinates established by ACIC's method and the coordinates in Schrutka's reduction of the 31 common positions revealed the following ± average differences:

	Schrutka	*ACIC*
Latitude ($\Delta\beta$)	451 meters	244 meters
Longitude ($\Delta\lambda$)	777	225
Height (Δh)	1828	468

The relative accuracies obtained by the ACIC reduction method indicate a marked improvement over the Franz-Schrutka positions. Also the ACIC method yields a network of coordinates related both to one another and to the moon's geometric center. As a result of these encouraging developments, we are hopeful that our final results will give a standard circular error of ±250 meters in horizontal position and ±500 meters vertically.

USAF lunar atlases

The first significant cartographic contribution by the Air Force to the national space effort was the *USAF Lunar Atlas* issued in 1960. It represents a comprehensive selection of the finest lunar photography ever assembled. Since then two supplements have been published: Supplement No. 1, the *Orthographic Lunar Atlas*, and Supplement No. 2, the *Rectified Lunar Atlas*. The three atlases were sponsored by the Air Force Cambridge Research Laboratories and produced by Dr. Gerard P. Kuiper of the University of Arizona, with technical assistance from the Aeronautical Chart and Information Center.

Selecting 280 photographs out of hundreds analyzed from plate collections at Mt. Wilson, Lick, McDonald, Yerkes, and Pic du Midi Observatories for the *USAF Lunar Atlas*, Dr. Kuiper divided the visible lunar disk into 44 fields, each covered by a minimum of four photographs taken under four different illuminations. The illuminations usually include one morning and one evening view under moderately high sun, a full-moon view, and a supplementary view under low grazing illumination. The different angles of illumination are necessary because some lunar features show up more clearly under one illumination than under others. Depending on the angle of illumination, some features, in fact, disappear completely.

The photographs contained in the *USAF Lunar Atlas* have been enlarged to a lunar diameter of 100 inches, which is about 20 miles to the inch near the center of the disk.

The *Orthographic Lunar Atlas* contains photographs selected from the basic atlas. The photographs display the selenographic grid, plotted at intervals of one-hundredth of the lunar radius. The grid is based on approximately 5000 measured points, 3500 of which are from Franz and Saunder and 1500 from Arthur. Superimposed on the same sheets are selenographic latitude and longitude, which represent the basic horizontal control for all ACIC lunar charts.

The *Rectified Lunar Atlas*, as the title implies, contains rectified photographs. Rectified photography of the moon removes the foreshortening which naturally increases toward the limb. Craters near the limb which are elliptical in shape on normal lunar photography become nearly circular on the rectified photograph.

The photographs are obtained by projecting normal photography onto a hemispherical easel and then photographing the new image with a camera aimed toward the center of the hemisphere. This technique results in a vertical view of that part of the moon being photographed.

While the projected image on the hemispherical easel can never show more detail than is recorded on the original photograph, rectified photography provides the selenographer with a new perspective of the moon. This type of photography is an aid to the lunar cartographer in his studies of the lunar ray systems as well as the concentric and radial structures associated with maria and large craters.

Ranger VII photography

On 31 July 1964, NASA's Ranger program opened the way for a new era in lunar cartography. Over 4000 television records were obtained, which was the first significant amount of lunar image data collected from outside the earth's atmosphere. Through the experience gained in other areas of selenography, ACIC was able to provide valuable assistance in the interpretation of this new coverage.

The Ranger VII mission was a model in technological perfection. The vehicle was launched into an orbit about the earth, then boosted into an earth-moon transfer trajectory. A mid-course correction was applied in order to produce impact at the desired location, and no further maneuvers were required. After approaching the moon on a hyperbolic trajectory, the vehicle struck the surface at approximately 20.6° west longitude and 10.7° south latitude.

Ranger VII contained six television cameras having several combinations of focal length, angular coverage, and cycling frequency. The cameras were turned on about 17 minutes before impact and continued to operate until the vehicle struck the moon. The last frame, taken 0.19 seconds before impact, was still being transmitted when the crash occurred. During the time of camera operation, 4316 frames were scanned and transmitted to earth. The signals were received at NASA tracking stations and were regenerated into television images for photographic recording on film.

Through Dr. Gerard Kuiper, the principal investigator for the Ranger program, ACIC agreed to assist in the interpretation and portrayal of the data contained in these television records. The major part of this response by Air Force selenographers was in the identification and location of features and in providing topographic interpretations and cartographic drawings.

After the photographic negatives were received at ACIC, a series of enlarged prints was prepared, showing the recorded image detail at a scale and contrast which were optimized for visual interpretation. Upon examination of these prints, it was decided that the best medium for portrayal of the new information would be a special series of charts. These charts, designated as the Ranger Lunar Chart (RLC) series 1 through 5, were planned to cover the new lunar detail through representation at several steps in scale. The scales selected were RLC-1 at 1:1,000,000; RLC-2 at 1:500,000; RLC-3 at 1:100,000; RLC-4 at 1:10,000; and RLC-5 at 1:1000. Thus it is possible not only to correlate the coverage from one of these charts to the others but also to relate the newly acquired Ranger material to the LAC charts generated from earth-based observations.

Since the Ranger VII documents were a new type of data input for selenography, the preparation of the RLC's involved several innovations in the lunar cartographic field. This was particularly true in regard to the scale and location of lunar features resolved in the latter portion of the mission—features which had never before been observed. The problem was simplified by two characteristics of the Ranger VII mission: first, the geometric distortions of the cameras were calibrated prior to the flight; and second, the coverage of the "A" camera was nested, that is, each frame fell within the limits of the frame before. The geometric corrections could be applied to remove scale and position errors which were introduced by the optical systems. The nesting characteristics were then utilized, since the geometry of each frame could serve as a position and scale reference for the next.

The first stage in the RLC production was the preparation of a set of controlled photo mosaics at the desired chart scales. For the first base, it was only necessary to use the relief drawing from the previously compiled LAC 76, which covered approximately the same area. For each successive step, the previous scale RLC base was enlarged by an appropriate factor, and rectified prints of the Ranger images were joined in a mosaic to fit the projected pattern of lunar features. Each frame used was selected on the basis of its area of coverage and the new detail it contained.

After the mosaic bases were completed, work was begun on the preparation of the cartographic drawings. ACIC selenographers who were trained to recognize and identify lunar images at the telescope started making annotations to the photo mosaics. Each frame was meticulously studied in order to bring the maximum amount of information into the cartographic drawings. Experienced scientific illustrators then combined these annotations into shaded relief drawings at each chart scale. Special drawings were prepared for the last two frames, which showed the lunar surface under maximum magnification and resolution.

While the drawings were being prepared, other studies were being conducted to extract additional information from the television records. These studies included measurements of the sizes, depths, and slopes of craters and the distribution of features that would be obstacles to a landing. When the RLC charts were printed, they not only contained the selenographic drawings but also depicted the limits of coverage of the various Ranger cameras, the ground track of the incoming vehicle, and approximate point of impact on the moon.

It would be shortsighted to consider that the LAC charts, or even the RLC series, represent the ultimate in lunar charting. More sophisticated missions, particularly photographic lunar orbiters, will provide additional information about the lunar surface. The interpretation and reduction of these photographs will be necessary before a site can be selected for the eventual landing of men on the moon. It is significant, however, that ACIC selenographers have placed the Air Force in a position of leadership in the field of lunar cartography—a position which carries with it both a sense of pride in achievement and a responsibility toward man's further progress in aerospace navigation and exploration.

Aeronautical Chart and Information Center

THEORY OF SPACE OPERATIONS

He lies upon his bed
Exerting on Arcturus and the moon
Forces proportional inversely to
The squares of their remoteness and conceives
The universe.
Atomic.
He can count
Ocean in atoms and weigh out the air
In multiples of one and subdivide
Light to its numbers.

Archibald MacLeish, "Einstein"

SPACE ORIENTATION

Some Problems of Satellites in Earth Orbit

Major William C. Ross

To introduce some theoretical considerations and terminology of space operation and thus give an insight into the orbital problems of a spacecraft in the near-earth environment, this article will point out some elementary principles from physics and mathematics governing satellite behavior. First, the basic relation between acceleration and force will be reviewed, then the principles of inverse square attraction due to gravity, the conservation of energy, and the conservation of momentum. Rocket propulsion parameters and applications of rocket power will be covered next. Orbital motion about a central body, including changes in the plane of the orbit, will then be presented. Finally, problems in maneuvering an orbital spacecraft over selected points on a rotating earth will be discussed.

From basic physics come the definitions and principles of acceleration, velocity, and distance:

Velocity is the rate of distance covered per unit of time. Average velocity may be obtained by dividing the total distance covered by the time required to cover this distance. Instantaneous velocity at a point in time may be closely approximated by dividing the distance covered during a short interval of time (containing the required point in time) by the short interval of time.

Acceleration is the rate of velocity change per unit of time. Average acceleration may be obtained by dividing the difference in the velocities at the beginning and end of a time period by the length of the time period. Instantaneous acceleration at a point in time may be closely approximated by dividing the velocity change during a short interval of time (containing the required point in time) by the short interval of time.

The force required to accelerate a mass is directly proportional to the product of the mass and the desired acceleration, or, expressed as an equation:

$$F = Kma$$

where F is the required force for acceleration
m – mass of the object to be accelerated
a – acceleration produced by force on mass
K – a constant dependent on units of F, m, and a.

Since one pound force will produce an acceleration of about 32.2 ft/sec^2 on one pound mass, in the English system if F is given in pounds force, m in pounds mass, and a in feet per second, then

$$K = \frac{1}{32.2}\left(\frac{\text{pounds force} \times (\text{seconds})^2}{\text{pounds mass} \times \text{feet}}\right)$$

Often m will be expressed in slugs (a slug is 32.2 pounds) so that a K value of one (using slugs mass instead of pounds mass) may be used to make the mathematics easier. Thus in a closed system, a constant force of one pound applied to a one-pound mass will change the velocity of the mass 32.2 ft/sec in one second. Of course this change will take place in the direction in which the force is applied.

In orbital flights near the earth but outside the atmosphere, the two main forces that are applied to a spacecraft are rocket thrust and the gravitational attraction of the earth. Obviously other forces exist, but their influence may be considered as negligible for short periods of time. Some examples of these negligible forces are the gravitational attraction of heavenly bodies other than earth, the drag caused by collision with the tiny gaseous particles in outer space, and the force caused by the light from the sun impinging on the surface of the spacecraft. These negligible forces for long time periods will eventually cause an orbit to decay, but it may take years. The rocket engine thrust acts for a limited time only while the gravitational attraction of the earth acts throughout the time of orbital flight. The force of gravitational attraction between two masses may be expressed by the equation:

$$F = \frac{Gm_1m_2}{d^2}$$

where F – the force
G – universal gravitational constant
m_1 – mass of first body
m_2 – mass of second body
d – distance between the two.

If the distance between the two bodies is very large compared to the dimensions of the masses, then it makes little difference which point on either mass is used as a reference in measuring the distance. However, since the earth has quite large dimensions as compared to the distance to an orbital spacecraft, we use the center of the earth as the best reference point for earth's mass. Because the earth is not a perfect sphere made up of concentric shells of uniform density, the gravity force does not always pull the satellite precisely toward the earth's center. For short time periods, errors resulting from use of the earth's center as the reference point are negligible; but for longer periods more exact methods of computing the earth's gravity force must be used for accurate orbital path prediction. One such method would be to divide the earth's mass into a number of small masses, then compute the cumulative effect of the gravity force of all the small masses on the orbiting satellite. As the volume of these masses would be small, the selection of a reference point within each small mass would present no problem, and a more accurate gravity force computation could be achieved than by considering all the earth's mass as concentrated at the earth's center.

The law of conservation of energy, which simply states that energy can neither be created nor destroyed, is also helpful in solving orbital mechanics problems. Thus, for a closed system the total energy at the beginning of a time period is the same throughout the time period. This law is used to equate the sum of the kinetic energy (energy due to velocity) and potential energy (energy due to position) to a constant. We may assume that all other forms of energy (such as chemical, nuclear, or thermal) do not change enough to be noticeable during free orbital flight above the earth's atmosphere. Under the conservation of energy law, it is apparent that the sum of kinetic energy and potential energy equals the total energy minus all other forms of energy in the system. As both the total energy and all other forms of energy in our closed earth-orbital-satellite system are constant, the following useful relation is apparent:

kinetic energy + potential energy = total energy − all other forms of energy = constant.

$$\text{Kinetic energy} = \tfrac{1}{2} K m v^2$$

where m – mass
v – velocity
K – constant of proportionality, dependent on system used.

$$\text{Potential energy} = \frac{-K\mu m}{r}$$

where m – mass
μ – $g_R R^2$ where g_R is force of gravitational attraction on a unit mass at the distance R from center of central body (earth in this case)
r – distance of satellite from center of central body
K – constant of proportionality, dependent on system used.

We can now combine terms and divide out the mass and proportionality constant to obtain the specific mechanical energy (energy per unit mass) E:

$$E = \tfrac{1}{2}v^2 - \frac{\mu}{r}$$

E will be constant for a closed system.

The law of conservation of momentum is also very useful in operating orbital spacecraft systems. The essence of this law is that for a closed system momentum is conserved. The momentum of an object can be defined as the product of the mass of the object and the velocity of the object. In the case of an explosion of an object at rest, many particles go in various directions at various velocities. If the size and velocities of all these particles could be measured soon after the explosion (so that the effects of any gravitational accelerations would be negligible), the momentum of each particle could be computed. By adopting a three-dimensional coordinate system which would assign signs (such as + for up and − for down), we would find that the momentum of all the particles comprising the original object would total zero for any given direction, since the original object was at rest.

Let us consider how the law of conservation of momentum is used to develop a rocket equation. As the rocket motor ejects a particle in one direction, the spacecraft must pick up an increment of velocity in the opposite direction. Although the particles ejected are small, they are ejected in such quantity and at such a high velocity that large velocity changes of the spacecraft can be effected. Rapid discharge of the propellant mass is quite important when large forces (such as gravity) are exerted on a spacecraft. Picture a spacecraft on a launch pad. The spacecraft's rocket motor would not cause lift-off if it ejected particles so slowly that the thrust produced were less than the earth's gravitational force. Instead the spacecraft would just vibrate on the pad until all its fuel was expended. On the other hand, let us make the theoretical assumption (not a practical one) that the rocket motor could expel all the propellant in one instant. Then we could use the equation derived from conservation of momentum:

> velocity increase of spacecraft × spacecraft mass = exit velocity of propellant × propellant mass.

Of course this is not a practical formula, as the propellant must be ejected at a finite rate, with gravity and drag (atmospheric resistance) acting throughout, and with the first propellant particles not being as efficient in imparting velocity to the spacecraft as the last propellant particles because the first particles are imparting velocity to the spacecraft *and* the remaining propellant.*

With a given rocket motor containing a certain amount of a specific propellant, a spacecraft has the ability to change its velocity by a certain increment. Let us consider methods of applying rocket thrust in order to make maximum use of this available velocity increment (ΔV).

Case 1. Spacecraft is traveling at velocity V_o; a maximum increase in the magnitude of the velocity is desired.

Solution. Eject propellant in direction opposite to V_o; then ΔV will be added directly to V_o. (See Figure 1 for vector diagram.)

*Major Roper's article in this issue, "Rocket Propulsion for Space—Fundamental Considerations," gives a more quantitative treatment of rocket propulsion principles.

Figure 1

It is also quite apparent that a maximum decrease in velocity occurs when the rocket motor ejects propellant in the direction of the existing velocity. Now let us consider some more advanced problems in the use of rocket propulsion. (See Figure 2.)

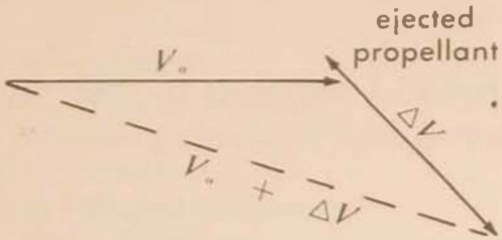


Figure 2

By ejecting propellant from the rocket motor in a certain direction, an increment of velocity (ΔV) in the opposite direction is added to the initial velocity. In Figure 2 a sample V_o and ΔV are added as in vector addition. The lengths of line V_o and line ΔV are proportional to the numerical values of V_o and ΔV. The angle between V_o and ΔV is that which would occur under real conditions. The new velocity ($V_o + \Delta V$) is proportional in numerical value to the length of the line from the base of V_o to the tip of ΔV. Similarly the angle between the old velocity (V_o) and the new velocity ($V_o + \Delta V$) could be measured by protractor on Figure 2.

As there is no restriction on the direction of application of ΔV, a number of possible magnitudes and directions are possible with a given initial velocity (V_o) and a given available magnitude of change in velocity (ΔV). (See Figure 3.)

By connecting the base of the line representing V_o to any point on the circle (center at tip of V_o and radius equal to ΔV), quite a few possible new velocities ($V_o + \Delta V$) can be seen.

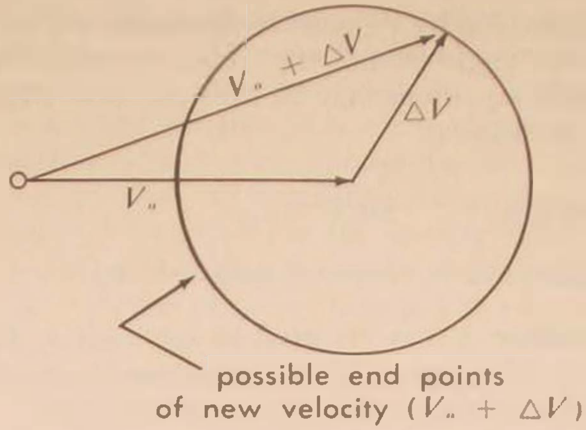


Figure 3

Case 2. Which of the possible new velocities in Figure 3 would result in the maximum direction change between the old velocity (V_o) and the new velocity ($V_o + \Delta V$)?

Solution. Construct a tangent to the circle in Figure 3 from the base of V_o. Two such tangents are possible, one on either side. (See Figure 4.) The length of the tangent is proportional to the magnitude of the new velocity, and the angle θ between the tangent and V_o is the maximum angle change possible. Either tangent will give the same result, as the figure is symmetrical. Those readers familiar with the Pythagorean theorem (sum of squares of legs of right triangle equals square of hypotenuse) can quickly deduce that $(V_o + \Delta V)^2 = V_o^2 - \Delta V^2$ and that the angle θ between the old velocity (V_o) and the new velocity ($V_o + \Delta V$) has a sine of $\frac{\Delta V}{V_o}$. By using the numerical answer of

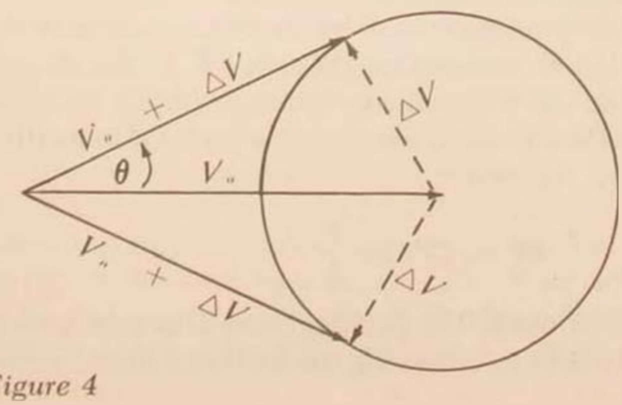


Figure 4

ΔV divided by V_o, one can determine the value of the angle from a trigonometric table. Two useful equations may be deduced from Figure 4, as follows:

Equation 1 $\quad \sin\theta = \dfrac{\Delta V}{V_o}$

Equation 2 $\quad$ New velocity $= V_o \cos\theta$

Equation 1 may be used to solve for θ; then using this value of θ in equation 2, one can find the magnitude of the new velocity.

In some cases it might be desirable to cause a change as large as possible in the direction of the original velocity (V_o) and also to have the final velocity ($V_o + \Delta V$) the same magnitude as the original velocity. (See Figure 5.) Figure 5 may be constructed graphically from Figure 3 by striking a circular arc with the center of arc at base of V_o and radius equal to V_o. Draw a line from the base of the original velocity (V_o) to the point of intersection of the arc with the circle (either of the two points of intersection); this line will represent the new velocity. By trigonometry, a useful relation can be deduced between the original velocity magnitude (V_o), the magnitude of the velocity change (ΔV), and the angle (θ) between the new and old velocities. This relation is:

$$\Delta V = 2V_o \sin\frac{\theta}{2}$$

Case 3. The problem may also arise where, in addition to having the final and initial velocities equal in magnitude, we would like to obtain as large an angle change as possible while maintaining the same velocity magnitude (V_o) all during the change in direction. (See Figure 6.)

Note that ΔV is applied in a continuously changing direction (i.e., always perpendicular to the instantaneous direction of flight). In all the other cases, ΔV was applied in a constant direction. With this method of change in velocity, the following relation exists between the velocity increment available for change (ΔV), the angle of change (θ), and the initial velocity (V_o):

$$\theta = \frac{\Delta V}{V_o}$$

The symbol θ represents an angle expressed in radians (a form of angle measure expressed as the ratio of the length of subtended circular arc to the length of the radius).

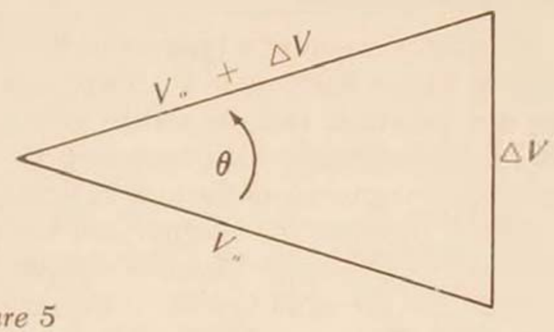


Figure 5

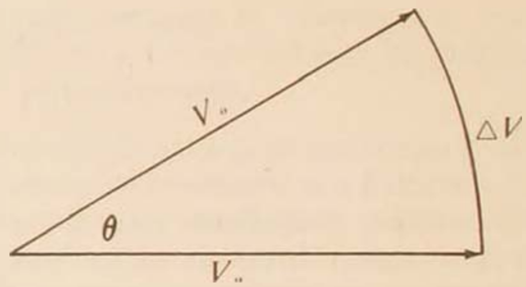


Figure 6

The geometric path of any of the methods shown in Figures 4, 5, and 6 could be followed to change the direction of motion of a spacecraft. But what effect does a change of direction have on the orbit of a near-earth spacecraft? To appreciate the answer to this question, one must have an understanding of how a spacecraft in orbit moves around the earth.

The spacecraft in orbit moves around the earth in an elliptic path (a circle is a type of ellipse). It is possible to cause a spacecraft to follow a parabolic or hyperbolic path near the earth by increasing the velocity of the spacecraft by a sufficient amount. Note that all of the types of paths mentioned are plane curves. Let us consider why these paths are all plane curves. Begin by considering the following equations as given:

Equation 1 $F = ma$

Equation 2 $\Delta V = a\Delta t$

Equation 3 $\Delta s = V\Delta t$

Equation 4 $F = \dfrac{Gm_1m_2}{d^2}$

where F – force

m – mass (subscripts may be used to denote specific masses)

a – acceleration

V – velocity

ΔV – change in velocity

Δt – change in time (an increment of time)

Δs – change in distance

G – universal gravitational constant

d – distance between m_1 and m_2.

The first and fourth equations are facts observed in nature. The second and third equations are obtained from definitions of velocity and acceleration in terms of distance (s) and increments of time (Δt).

Now let us relate these equations to the motion of a satellite in orbit near to the surface of the earth (but out of the earth's atmosphere). We might compute the gravitational forces on the satellite due to the gravity of the earth, sun, moon, and other heavenly bodies (using formula 4). Since the force of gravity of the earth on the satellite is very large compared to the other gravitational forces, our primary problem is that of the satellite in motion about one point, the center of the earth. In fact, we can learn most about satellite motion by studying and computing solutions to an isolated earth-satellite system and then considering minor perturbations to this motion that might be caused by gravitational forces of the sun, moon, and other heavenly bodies, by the fact that the earth is not a true sphere, and by any drag on the satellite due to collisions with gaseous particles in space.

Now let us consider this isolated earth-satellite system. In equation 4, F represents the force of the satellite on the earth and the earth on the satellite. They are identical in numerical value but act in opposite directions. However, when we substitute these values in equation 1 to solve for the accelerations of the earth and satellite, it is quite apparent that the acceleration of the earth is practically zero as compared to acceleration of the satellite, since the mass of the earth is much greater than the mass of the satellite. Therefore, let us just ignore any motion of the earth due to gravitational pull of the satellite and let the center of the earth remain stationary. Let us now take a small increment of time, say one second, and proceed to compute the motion of the satellite starting at the initial position P_0. (See Figure 7.) In fact, with computers avail-

Note: Not to scale; P_1 would appear much closer to P_0 in a scaled drawing.

Figure 7

able today that compute in microseconds (millionths of a second), we could easily take smaller time increments.

Figure 7 is a diagram showing distance traveled by the satellite during one second, assuming that the velocity of motion during the one-second interval will not vary appreciably from V_0. Now let us compute a new velocity value for P_1; this is necessary because the force of the earth's gravity has been accelerating the satellite for one second.

The magnitude of the acceleration may be computed from equations 4 and 1. Equation 2 may then be used to compute ΔV. Now add to V_0, the original velocity at P_0, the velocity change (ΔV) caused by the earth's gravity.

This is done either graphically or algebraically. (See Figure 8.) ΔV is directed toward the earth's center, and V_0 is in the original direction of satellite motion at point P_0. The addi-

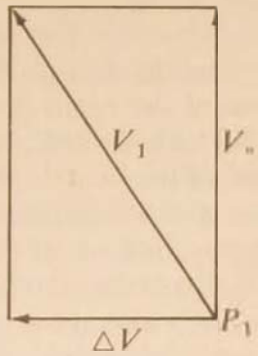


Figure 8

tion is accomplished by forming a parallelogram with ΔV and V_0 as sides. The diagonal of the parallelogram drawn from P_1 is the graphic addition of ΔV and V_0. This process of addition is called vector addition. We can now repeat this process for as many seconds as desired, plotting the points P_0, P_1, P_2, etc. as we compute. We could take smaller increments of time to get a more accurate plot or take an average of the velocities at P_0 and P_1 over the interval from P_0 to P_1 if we desired more accuracy. Care must be taken to ensure that the velocity variation between any two consecutive points (as P_1 and P_2) is very small in both magnitude and direction. Should the variation exceed some small velocity variation that has been established as critical, then it will be necessary to use smaller time intervals until the variation is less than the established critical variation. After computing the satellite's position for many seconds following a given V_0 and P_0, we could learn how the satellite would move. Actually, for an orbiting satellite, we are only interested in those cases where the satellite's position would remain above the earth's atmosphere and where the satellite would remain in orbit (elliptic path). By experimenting, we could find a range of values for V_0 that would keep the satellite in orbit starting at P_0 (P_0 is above the atmosphere). For larger values of V_0 the satellite would get too far away from earth to remain in orbit; for smaller values of V_0 the satellite would enter the earth's atmosphere and might even intersect the earth's surface. We would also find that the paths of the satellite in orbit would be elliptic.

Ellipses may be classified as to their eccentricity. Eccentricity is a measure of an ellipse's deviation from a circle; an ellipse which is a circle has an eccentricity of zero. Other satellites in orbit follow elliptic paths around a central body. An ellipse is a plane curve, not a three-dimensional curve. A satellite in orbit remains in an elliptic orbit under assumptions made, as there are no forces acting to pull the satellite out of the orbital plane. Very good mathematical equations exist for elliptic orbits of a satellite around a stationary central body.

In summary, we should note that the formulas for isolated earth satellites in orbit are very useful in computing ΔV requirements to place satellites in the required orbits.

If the space vehicle has enough power to give us the required ΔV, we can now refine our calculations for satellite motion by bringing in the gravitational forces of other heavenly bodies, the drag forces, and variations of earth's force due to its deviation from a perfect sphere. We can rest assured that these forces will be of a small enough magnitude so that the original computed velocity will maintain the satellite near the desired orbit for some time. In fact, if necessary we can compute the precise motion of the satellite with a computer, using all available accelerations as we did with the stationary central force. If deviations from the desired orbital path exist, they will be small, and only small corrections will be necessary.

Now let us relate the orbital motion to locations on the surface of the earth. In space operations, a coordinate system consisting of altitude, latitude, and longitude is used for giving locations of spacecraft above the earth.

In Figure 9, which depicts a satellite in orbit around a rotating earth, the track (trace on ground formed by points directly underneath satellite) would appear as the dotted line if the earth were not rotating. The plane

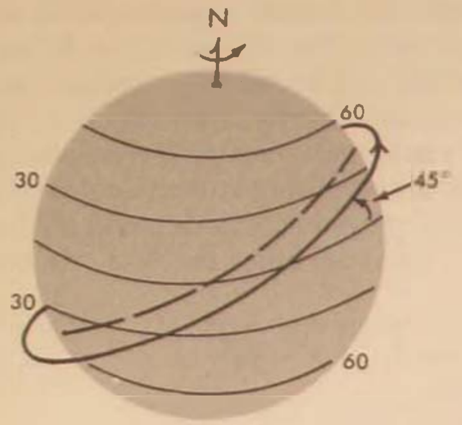


Figure 9

of the orbit in this figure makes an angle of 45° with the equatorial plane; hence we say that the orbit has an inclination of 45°.

With an inclination of 45°, the satellite will not pass over any point on the surface of the earth with a latitude greater than 45° (either north or south latitude). Suppose we wanted to change the inclination from 45° to 90° so that the satellite would pass over the poles. If the spacecraft is turned with rocket power at a point over the equator, the spacecraft would only have to turn through 45°. But if the satellite attempted to turn into polar orbit at 45° latitude (north or south), a full 90° turn would be required because the satellite flight path is parallel to the plane of the equator when it reaches 45° latitude. (See Figure 10.)

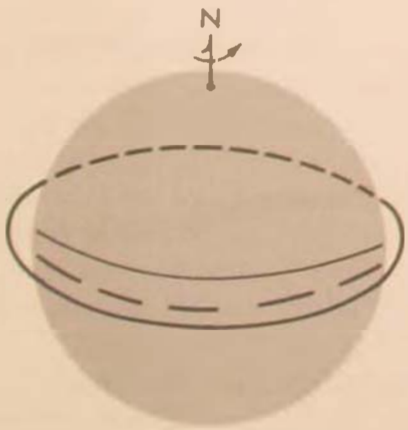


Figure 10

Thus the reader can see that it is important to consider where a turn should be made to get maximum benefit from propulsion. This example was presented to demonstrate that the change of orbital inclination must occur at the equator if we wish to conserve propellant power while changing. (In the example presented, a 45° plane change may be considered too great, and a new spacecraft might better be launched into polar orbit should one be needed there.)

Now let us consider the problem of launching a satellite into a 300-nautical-mile circular orbit so that it will fly over a certain desired point on the first flight. (See Figure 11.) By using spherical trigonometry, the speed of the satellite in orbit, and the earth's rotational velocity, we can compute a good solution. For a more precise computation, the oblateness of the earth and other factors must be considered.

Anyone familiar with spherical trigonometry and the isolated-spherical-earth-and-satellite formula could make this computation. However, it is not even necessary to be familiar with these subjects in order to obtain practical though less accurate results. The equipment required consists of a large globe and a flexible string graduated in minutes after launch for a circular orbit. The first mark on the string represents three minutes while all others are for one-minute graduations. It is necessary that the first graduation represent a longer time period because the spacecraft requires time to gain altitude and velocity for the desired circular orbit. All other graduations are equal in length and represent the track of the spacecraft on earth for one minute in orbit.

Each parallel on the earth is divided into segments representing rotational time. As the earth rotates on its axis approximately every 24 hours, a point on the surface of the earth rotates through one-fourth spherical degree of longitude every minute. Compare this to the track of a satellite in a 90-minute circular orbit, one minute's flight time on the string being equal to 4 spherical degrees on the globe. (One spherical degree is 1/360 of a circle with the radius of the globe.)

To check the launch angle required in order to fly over a point (F in Figure 11), one

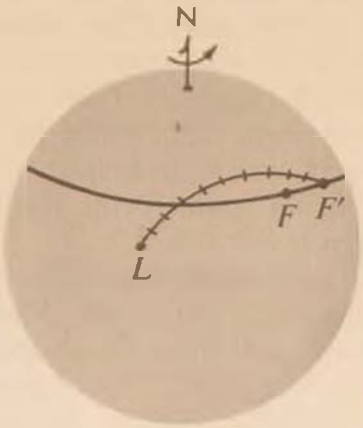


Figure 11

need only stretch the string tight with one end at the launch point (L) and the other on the same parallel as the flyover point (F). Then move the string (or shorten or lengthen) so that the time on the string matches the time required by the earth to rotate from F to F' (indicated on the parallel). Figure 12 shows a better view of the adjustment. After the string

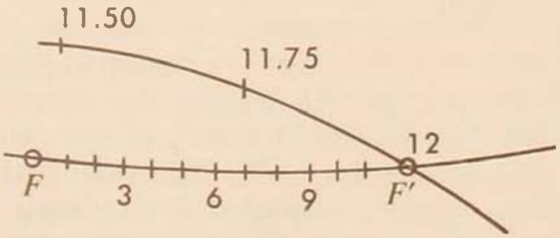


Figure 12

is adjusted, the launch angle required may be measured at L. (See Figure 13.) It is the angle between the string and the meridian connecting L to the North Pole. Such a graphic method could well serve as a check to analytic methods of computing launch angles. More sophisticated graphic methods could be designed should a more accurate check be desired.

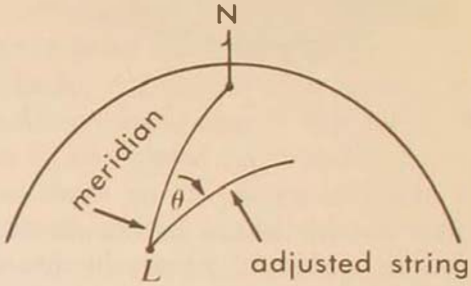


Figure 13

A QUALITATIVE presentation of some orbital astronautics information has been made. Starting with basic physics laws, some relations in orbital mechanics have been developed and later related to a problem involving the rotating earth. A graphic computation to solve this problem was given. Our intention has been to help the reader understand other articles in this issue. Any officer wishing to gain more knowledge about astronautics could profit by attending the three-week Aerospace Operations Course at Air University.

Hq Air Force Systems Command

REFLECTIONS ON LAUNCH WINDOWS

Colonel Francis X. Kane and
Major William C. Ross

ONE of the basic concepts of space operations is that of the "launch window" or the amount of time available on a given day during which a spacecraft must be launched in order to accomplish a given mission. In the Mercury flights, the time available for launch varied from 5 to 7 hours. For some Tiros launches, this "window" is open for only 45 minutes.

Some of the factors which determine the time dimension for launch operations are orbital plane inclination, altitude, eccentricity, launch site location, and in-space maneuver capability, whether by transfer or dogleg. The purpose of this article is to show the interaction of these factors by discussing some of the techniques of space rendezvous. To illustrate the major problems involved in determining the "launch window," a hypothetical situation is posed. Assume the following situation: You are planning to resupply a space station which has been in orbit for some weeks. The personnel on board must be rotated, consumable supplies must be replenished, and equipment must be exchanged for new and different experiments.

How do you determine when to launch your resupply vehicle? Two of the factors to consider are launch site location and launch capability. Let us assume that you have two launch pads at one launch site; you have four resupply vehicles, but it takes ten hours to change resupply vehicles on the pad. To increase reliability, you plan to prepare both pads for launch and countdown simultaneously, but you will launch only one resupply vehicle. Also, let us assume that the launch site is located at latitude 28.5° north.

Other factors to be considered are inclination and orbit characteristics. (See Figure 1.) The launch site rotates with the earth and passes through the plane of the orbiting laboratory twice each day (to be more precise, twice every 23 hours 56 minutes 4 seconds). The satellite in turn passes over the 28.5° north parallel twice every 92 minutes 32 seconds. It is in a circular orbit at 250 statute miles, at an inclination of 50°.

We shall consider four possible launches.

Alternative I—Shortest Time to Rendezvous. One way for the resupply craft to rendezvous quickly with the space laboratory would be to launch from the site *L* (Figure 2) so that the supply spacecraft would arrive at point *P* at the same time as the orbital laboratory. This time can be predicted if the ephemeris of the lab is known. The supply craft would then execute a turn at *P* to match the velocity and direction of the orbiting laboratory.

Unfortunately, this alternative requires the

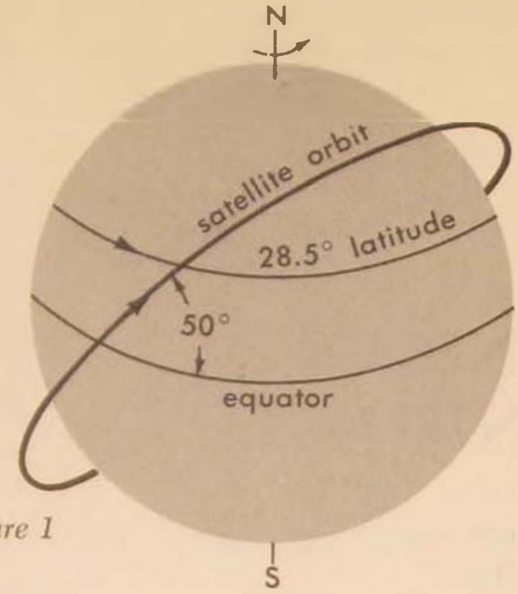


Figure 1

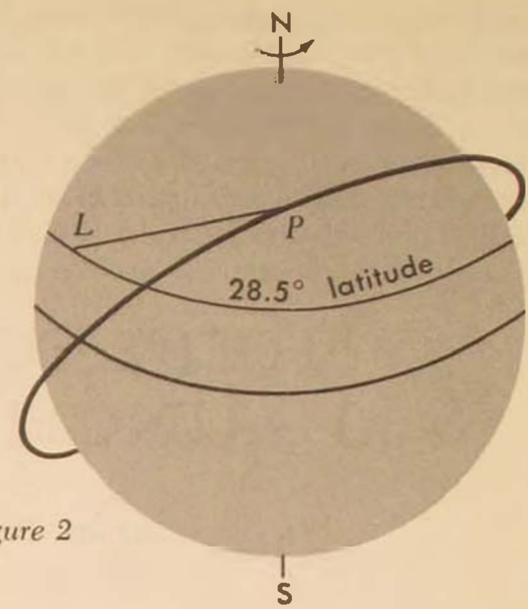


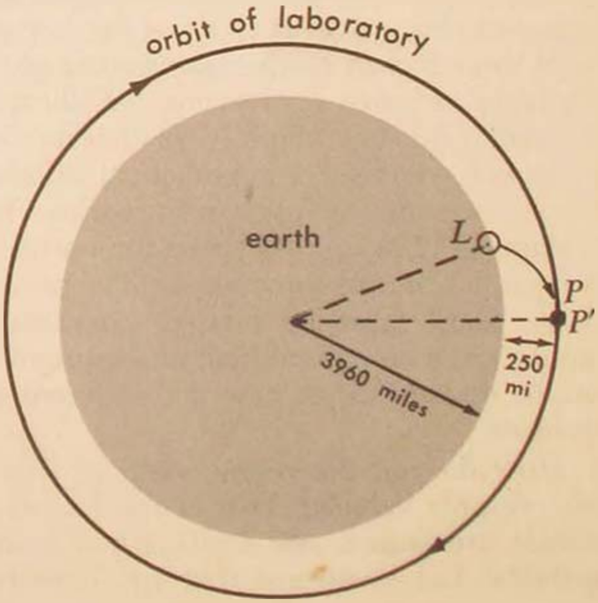
Figure 2

most thrust for in-space maneuvering, and for most cases it will not be available on board the supply craft. Thus the thrust required to change the velocity and direction of the resupply spacecraft at P will be the limiting factor in launch window planning (and in most other maneuvers as well). For example, a spacecraft traveling in 250-statute-mile circular orbit would expend an amount of propellant equal to almost half its weight in turning 20°, assuming that the spacecraft has a powerful propellant with a specific impulse (I_{sp}) of 400 seconds.*

Alternative II—Minimum Thrust. The ideal situation for launch with minimum thrust will occur when the resupply spacecraft is launched in the plane of the orbiting laboratory and is met by it just as the spacecraft is injected into 250-mile circular orbit. This procedure calls for very exact timing. Let us investigate the problems entailed in launching the supply craft directly into the plane of the orbiting laboratory.

Figure 3 depicts a view of the plane of the orbital laboratory with a cross section of the earth at the time when the launch site turns into the plane of the orbital laboratory. Only one of the many possible paths for the resupply craft is shown. The craft moves almost vertically during the first portion of its flight in order to get away from the high drag of the atmosphere. Then it turns so that at propellant-burnout time it is on an elliptic path that is tangent to the circular path of the orbiting laboratory. At this time thrust is applied to give the spacecraft the added velocity nec-

Figure 3



*For simplicity in computations, velocity changes will be considered as delivered in impulses (ΔV). In a real situation, propulsion will be applied continuously over a definite period of time in order to change the velocity vector, and more propellant will be required.

essary to place it in a circular orbit. (The final circular velocity is 4.768 miles per second.) Ground tracking units near the launch site can closely control this maneuver, but their location relative to this in-space maneuver becomes a limiting factor.

Theoretically, the in-plane launch method is the most efficient. However, to use this solution we must be sure that we can launch exactly on time. There is always a possibility that the supply vehicle might not be ready for launch when the window is open. Then we would be in trouble, for it will be many days before the relative positions of orbiting laboratory and launch site will again be ideal for launch. To ensure against failure in case of launch aborts, let us look at some more-flexible alternatives.

Alternative III—Catch-Up Maneuver. We can launch the resupply spacecraft into the plane of the orbital laboratory even though the two spacecraft may not be in position to rendezvous immediately. The smaller supply craft can be maneuvered so as to achieve rendezvous with the other craft at a later time. In planning maneuvers of this type, we must be careful to prevent the supply spacecraft from re-entering the earth's atmosphere, to minimize the amount of propellant required. and to minimize the time to rendezvous. Three factors in this maneuver are inclination, velocity, and relative positions in space.

We can operate as follows. Launch the resupply spacecraft directly into an elliptic orbit the apogee of which is near the launch site and at the same altitude as the orbit of the space station. The day and hour for the launch should be selected so that the orbiting laboratory will be only a short distance ahead of the resupply spacecraft at apogee (as in P' in Figure 3). For the resupply craft, select a less-than-orbital velocity so that the two spacecrafts will meet at the apogee of the resupply spacecraft. Then a velocity increment must be added to the resupply spacecraft so that it will be in the circular orbit of the orbiting laboratory.

Suppose that on the day selected the orbital laboratory will be 12° ahead of the resupply spacecraft when it arrives at its apogee of 250 statute miles. In Figure 3 note the angle at the center of the earth formed by lines to L and P'. It is not necessary that the two spacecraft meet after only one period; two or more trips around the earth before rendezvous could be considered. To determine what the factors are for this operation, we must make two calculations. First we must compute the velocity that the booster must give the supply craft so that it will arrive at apogee when the orbiting laboratory is there (Problem 1). Second, we must compute the velocity which the on-board maneuvering engine must add to the supply spacecraft to circularize its orbit when it meets the orbiting laboratory later (Problem 2).

The following formulas may be used to solve the two problems.

Formula A $\quad V^2 = \mu \left(\frac{2}{r} - \frac{1}{a} \right)$

This holds for all elliptic orbits.

V – Velocity of orbiting craft.

μ – A constant which is a function of the mass of the central body. For earth it is 9.58×10^4 (miles)^3/(sec)^2 or 14×10^{15} ft^3/sec^2.

r – Distance from orbiting craft to the center of mass of the earth. Use 3960 statute miles as the radius of the earth.

a – Semimajor axis of the ellipse.

Formula B $\quad a = 3155\, T^{2/3}$ or $T = \left(\frac{a}{3155} \right)^{3/2}$

a is given in statute miles.

T is the period (in hours) for one rotation around the earth.

Formula *B* is a good approximation for orbiting bodies with the earth as central body. It does ignore effects of the oblateness of the earth, any drag, and gravitational attraction of other bodies, as the sun and moon.

Let us now use Formulas A and B in solving the two problems involved in this alternative.

Problem 1. Using the smallest perigee allowable for the resupply spacecraft—100 statute miles—compute the period of the resupply

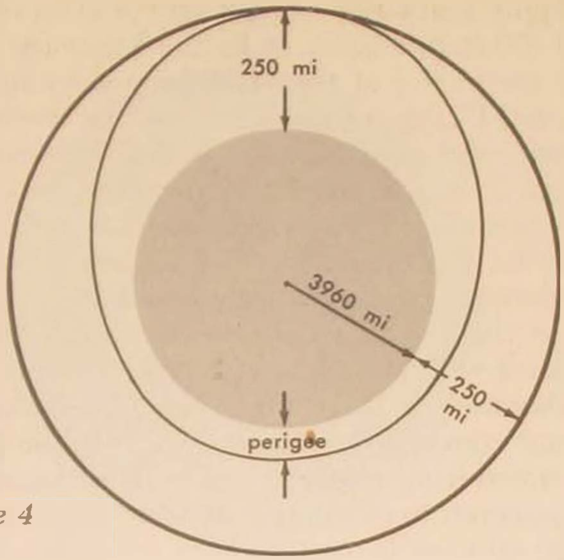


Figure 4

spacecraft by solving for a, the semimajor axis. (See Figure 4.) Then compute for T in Formula B.

$$r \text{ max (at apogee)} = 3960 + 250 = 4210$$
$$r \text{ min (at perigee)} = 3960 + 100 = 4060$$
$$\text{Major axis} = 8270$$
$$\div 2 = 4135 \text{ statute miles} = \text{semimajor axis}$$

This is a for our orbit.

Using the formula $T = \left(\frac{a}{3155}\right)^{3/2}$

$$T = \left(\frac{4135}{3155}\right)^{3/2} = 90.06 \text{ minutes,}$$

the time to travel one revolution or 360°.

Thus the resupply spacecraft will return to apogee in 90.06 minutes (travel 360°). In this time the orbiting laboratory (period 92.53 minutes) will cover:

$$\frac{90.06}{92.53}(360°) = 350.4°.$$

Adding: + 12.0°, initial lead
= 362.4° traveled by lab while resupply craft travels one revolution.

Thus after one revolution the laboratory would still be 2.4° ahead. The resupply spacecraft cannot catch the orbital laboratory in one revolution. They must meet at the apogee, for only there do the two orbits intersect. To solve this, let us put the supply craft into an elliptic orbit so that it will get 6° closer to the orbiting laboratory on each revolution. Then the two spacecraft will meet after two revolutions.

Computation for 6° closure per revolution:

$$\frac{X}{92.53}(360°) = 6°$$ X = minutes difference in periods of the two spacecraft.

$X = 1.542$ minutes difference in period

92.53 minutes
−1.54
90.99 minutes, period of desired ellipse of the supply spacecraft.

Substitute this period in the semimajor axis formula to find the semimajor axis: $a = 3155\,T^{2/3}$.

Now to find the required velocity for the supply craft at apogee, substitute the semimajor axis (a) and apogee distance to earth center (r) in Formula A:

$$V^2 = \mu\left(\frac{2}{r} - \frac{1}{a}\right)$$

V – unknown
$\mu = 95{,}800$ miles³/sec²
r – distance from earth center at apogee = 4210 miles
a – semimajor axis = 4160 miles

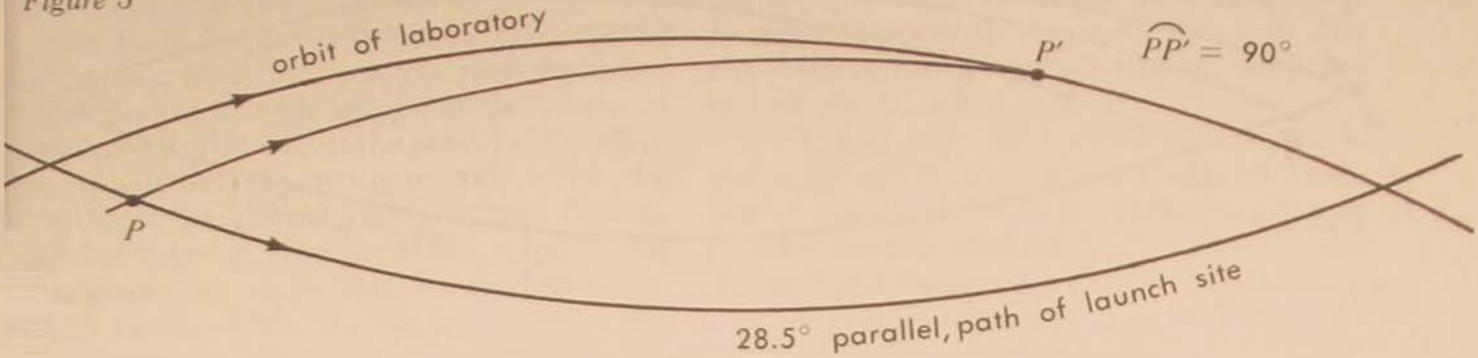
$$\left[a = 3155\,T^{2/3} = 3155\left(\frac{90.99}{60}\right)^{2/3}\right]$$

$$V^2 = 95{,}800\left(\frac{2}{4210} - \frac{1}{4160}\right) \text{miles}^2/\text{sec}^2$$

$2/4210 = .0004750$ miles
$1/4160 = .0002405$ miles
$V^2 = 22.470$ miles²/sec²
$V = 4.74$ miles/sec, required injection velocity for resupply craft.

Note how critical the velocity control must be for this maneuver, as the circular velocity is 4.768 miles per second. You can see that an error of ½ of one per cent in the final resupply spacecraft apogee velocity would cause failure to rendezvous. Errors of less than one hundredth of one per cent are needed.

Figure 5



Problem 2. Find velocity required to inject resupply spacecraft into circular orbit when it meets the orbiting laboratory at apogee.

Solution: 4.768 miles/sec, circular velocity required
−4.74 miles/sec, apogee velocity
= .028 miles/sec to be added.

By giving the resupply craft a velocity at apogee (250 statute miles) of 4.74 miles per second (when orbiting laboratory is in circular orbit in same plane at apogee altitude and 12° ahead), the two spacecraft will meet after two orbits. Then by adding .028 miles per second to the resupply craft, the two spacecraft will be in the same circular orbit and a docking could take place.

This alternative may not be satisfactory because every trip around the earth makes our control problem more difficult. With each return to apogee, the supply craft is over locations which are progressively more westerly of the launch site. After the first revolution and return to the 28.5° parallel, the supply spacecraft will be over the Big Bend National Park in Texas; after the second trip around the earth, it will be over the Pacific Ocean southwest of Los Angeles. The supply craft is returning to the same place, but the earth is rotating underneath. In order to check closely the velocity increments required to circularize the elliptic orbit and to verify correct time for application, extensive tracking facilities west of the launch site will be required. The next alternative considers a launch method which could concentrate the location of the tracking facilities.

Alternative IV—Rendezvous Turn. Another method would be to launch when the launch site is near the plane of the orbital laboratory and to execute a turn into the orbit of the orbital laboratory. (See Figure 5.) The procedure could be standardized by always executing this turn over a point on earth that is 90 spherical degrees (5400 nautical miles) downrange from the launch site, for here the angle of turn is the least. (See Appendix for solid geometry proof that this will be the least turn angle. The circle "*O*" in the theorem may be considered the track of the orbiting laboratory on a nonrotating earth, and the point *P* in the theorem considered as the launch site.)

Look at Figure 6. As the launch site rotates with the earth from *A* to *B* to *C* (fixed points on a nonrotating globe), points *A*′, *B*′, and *C*′ are generated by the flight path of the laboratory. These points are only a few degrees from each other, so the corresponding uprange position for the orbital laboratory to effect rendezvous has moved very little. The difference in longitude of *A* and *C* on the nonrotating globe is 24°. As the earth rotates about 15° an hour, it will require $\frac{24°}{15°} \times 60$, or 96 minutes

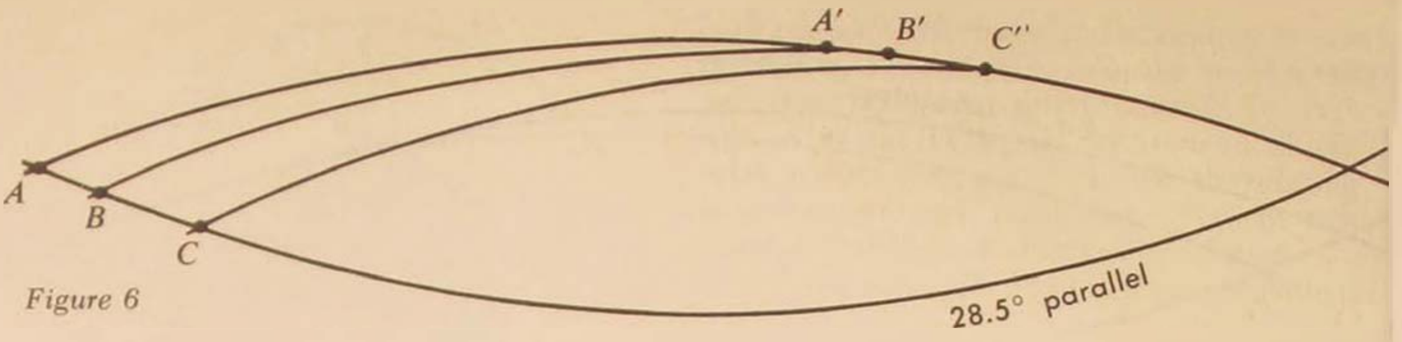


Figure 6

for the launch site to move from point A to point C. During this time, the orbiting laboratory must be at the corresponding uprange position at least once, for it goes through all points in its orbit every 92.53 minutes. Using these data we can determine when the laboratory will be at the correct uprange location for rendezvous to be made. This gives us the time of launch.

Knowing this time, we would launch, reach 250 miles altitude, turn when 5400 miles downrange, and rendezvous.

Although each of the four alternatives is a logical rendezvous method, it may not be feasible to accomplish a rendezvous every time under present performance restrictions by any of these alternatives. Some additional considerations in using Alternatives III and IV are as follows.

With Alternative III, at a given time the orbiting laboratory may be so far ahead of the resupply spacecraft at launch time that rendezvous could be accomplished faster by waiting and launching the next time the launch site was in the plane of the orbiting laboratory.

With Alternative IV, the turn angle, although a minimum, may be so large that again it would be advisable to wait until the next time the launch site passed near the plane of the orbiting laboratory in order to obtain a smaller turn angle.

In both of these cases, more available thrust through more powerful propellants, lighter structures, or other methods could make launches possible every time. But with more available thrust, we may decide to wait for a more advantageous launch time and take up more payload.

Of course, if we had available more launch sites with resupply spacecrafts, we would increase our probability of having a short time to rendezvous. By choosing launch sites which are properly separated, we could increase our chances of being able to launch from at least one on short notice.

Someday we may have to plan to rendezvous when the launch site is never in the plane of the orbiting laboratory. To illustrate this case, let us suppose the laboratory is in orbit over the equator and again our launch site for the resupply spacecraft is on the 28.5° north parallel. The circle of the 28.5° north parallel lies in a plane parallel to the equatorial plane. Thus the launch site could never pass into the plane of the orbiting laboratory. A rendezvous of the type described in Alternative IV (90° uprange from crosspoint) would be the type requiring least thrust, but still this type would require a 28.5° angle of turn, an angle quite prohibitive for a resupply spacecraft under present standards. A new launch site closer to the equator would be a must in order to accomplish rendezvous with adequate payload under present propulsion restrictions.

We have shown four alternatives for rendezvous with a spacecraft in circular orbit:

(1) Launch so as to intercept the orbiting laboratory, then turn the resupply spacecraft into the plane of the orbiting laboratory so as to match velocities for rendezvous.

(2) Launch into the plane of the orbiting laboratory so that interception takes place when the resupply spacecraft first reaches apo-

gee; then circularize orbit of the resupply spacecraft at apogee.

(3) Launch into the plane of the orbiting laboratory, place the resupply spacecraft into an elliptic orbit with apogee at the height of the circular orbit, adjust the period of the elliptic orbit so that the two spacecraft will arrive together at the apogee at a later time, then on arriving together circularize the elliptic orbit of resupply spacecraft to obtain rendezvous.

(4) Launch when the site is near the plane of the orbiting laboratory, adjust time of launch so that interception will take place over a point 5400 nautical miles (90° of great circle arc) from the launch site, add thrust at the interception point to turn and circularize the orbit of the resupply spacecraft.

The first alternative has a launch window that is open all of the time, but most of the time thrust requirements are too high. The second alternative has the lowest thrust requirements, but the launch window is open for only a very short time. The third alternative allows a launch twice a day, but the time required to achieve rendezvous might be too lengthy. The fourth alternative allows a launch twice a day with a short fixed time required to achieve rendezvous, but with a higher thrust requirement than that of the third alternative. However, the fourth alternative has a smaller requirement for downrange tracking and control of the launched vehicle.

The first rendezvous efforts will be simple and allow ample time for checking orbital data before any rendezvous is attempted. A proposed NASA Mission Plan for the First Gemini Agena Rendezvous Flight makes ample provision for ground computation as well as advice to the Gemini astronauts on all maneuvers. Three separate and distinct maneuvers, ending at the third apogee of the Agena spacecraft, will be made respectively to change plane into that of the Gemini vehicle, to correct the Agena perigee, and to circularize the Agena orbit. The final rendezvous maneuver will take place over 150° of arc with corrections at 90° and 30° before projected rendezvous. Later efforts should progress in complexity so as to have greater precision.

It will be interesting to follow the progress of other space operations to see which alternatives are used in getting through the launch window.

Hq Air Force Systems Command

Bibliography

1. Leighton, H. L. C. *Solid Geometry and Spherical Trigonometry*. New York: D. Van Nostrand Company, Inc., 1943.
2. Proposed Mission Plan for the First Agena Rendezvous Flight, Plan No. 2, 24 March 1964, by NASA Manned Spacecraft Center Mission Planning and Analysis Division, Rendezvous Analysis Branch, Houston, Texas.

Appendix

Theorem

Given: Great circle O on sphere S, point P on sphere S which does not lie on O.

Hypothesis—Of the family of great circles containing P, those two which intersect O 90° from P will have the least angle of intersection with O.

Proof

Construction:

Through P and the poles of circle O, pass a plane. This plane will be perpendicular to O as any plane containing both poles of a great circle will be perpendicular to plane of the great circle; this plane will intersect S in a great circle. Label the shorter arc from P to O PP'. Draw an arc of

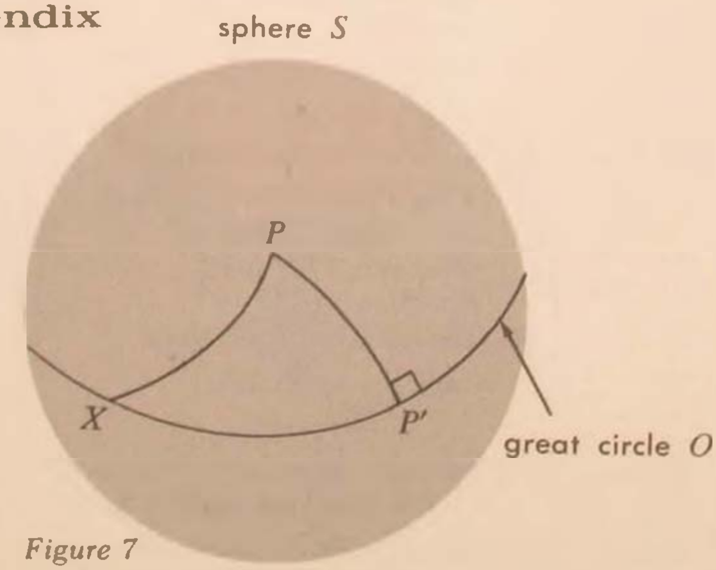


Figure 7

another great circle from P to X, where X is any point on circle O. (See Figure 7.)

Analysis: Find the length of PX when $\angle PXP'$ is a minimum.

Proof

Steps	Reasons
1. Either $\widehat{PP'} = 90°$ or $\widehat{PP'} \neq 90°$	1. Enumeration of possibilities
2. When $\widehat{PP'} = 90°$, P is at the pole of O.	2. A pole of a great circle lies 90 spherical degrees from the given great circle along any great circle arc which is $\perp$ to the given great circle. (Spherical Trigonometry)
3. Thus when $\widehat{PP'} = 90°$ $\angle PXP'$ is always 90°.	3. Any great circle through the pole of a given circle intersects the given great circle at right angles. (Spherical Trigonometry)
4. Henceforth consider case where $\widehat{PP'}$ is less than 90°	4. Enumeration of remaining possibility and PP' was labeled so as to be the shorter arc from P to great circle O.
5. $\angle PP'X = 90°$	5. Construction
6. $\dfrac{\sin \widehat{PP'}}{\sin \angle PXP'} = \dfrac{\sin \widehat{PX}}{\sin \angle PP'X}$	6. Law of sines for spherical triangles.
7. $\dfrac{\sin \widehat{PP'}}{\sin \angle PXP'} = \sin \widehat{PX}$	7. Step 5 and sin 90° = 1.
8. $\widehat{PP'} =$ Constant	8. Given
9. $\sin \angle PXP' = \dfrac{\text{Constant}}{\sin \widehat{PX}}$	9. Substitution
10. $\sin \angle PXP'$ is a minimum when $\sin \widehat{PX}$ is a maximum.	10. Algebra
11. $\sin \widehat{PX}$ is a maximum when $\widehat{PX}$ is 90°.	11. Trigonometry; sine is maximum at 90°.
12. $\sin \angle PXP'$ is a minimum when $\widehat{PX} = 90°$.	12. Steps 10 and 11 and substitution.
13. $\angle PXP'$ is a minimum when $\sin \angle PXP'$ is a minimum.	13. For positive acute angles, the angle with smaller sine is the smaller angle.
14. Thus $\angle PXP'$ is a minimum when $\widehat{PX}$ equals 90°.	14. Steps 12 and 13 and substitution.

Similarly, it can also be shown that $\widehat{P'X} = 90°$ when $\angle PXP'$ is a minimum. There will be two positions for X so that $\widehat{PX}$ is 90°, as one may measure either clockwise or counterclockwise from P'. (However, for our case we will eliminate one of these, as we will take the $\widehat{PX}$ with the velocity component in the direction of flight of the orbiting laboratory.)

SYNERGETIC ORBITAL PLANE CHANGE

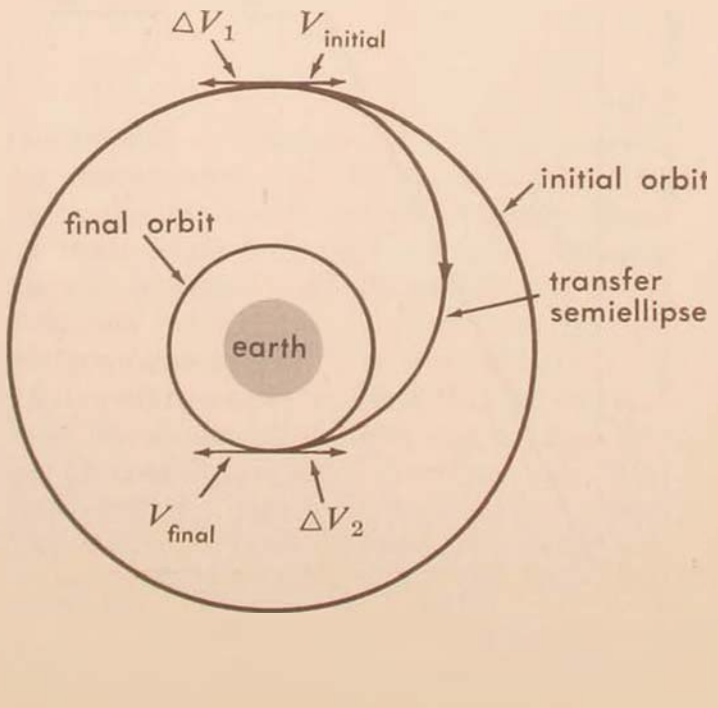
A Key to In-Space Maneuverability?

MAJOR JACK W. HUNTER

ALTHOUGH maneuvering in space will probably never be similar to "dog-fighting" in the atmosphere, the capability of changing altitude and orbit plane will be an inherent performance characteristic of military spacecraft. Our discussion will look beyond present space operations into a planning frame of reference for future spacecraft.

The ability of an orbiting spacecraft to change its orbital plane (flight path direction) or altitude is limited by the amount of energy retained by the spacecraft after it has attained an initial orbital condition. Normally, altitude changes will be made by a minimum-energy maneuver called the Hohmann transfer. (See Figure 1.) In this maneuver the spacecraft simply follows a semielliptical path to either a higher or lower orbital altitude, accomplishing the transfer by the application of two velocity impulses. As seen from the curve in Figure 2, the incremental velocity required to ascend (or descend) through altitudes from 100 to 600 nautical miles is on the order of hundreds of feet per second ΔV.* On the other hand orbital plane changes, normally accomplished by the application of a single velocity impulse perpendicular to the flight path, require a much greater amount of energy, or ΔV. For example,

Figure 1. Hohmann transfer (descent)



*An earlier article by Major Hunter and Colonel Francis X. Kane, entitled "Are You Ready for Space? or Lost in the Land of ΔV?" appeared in the *Air University Review*, XV, 1 (November-December 1963), 52-59.

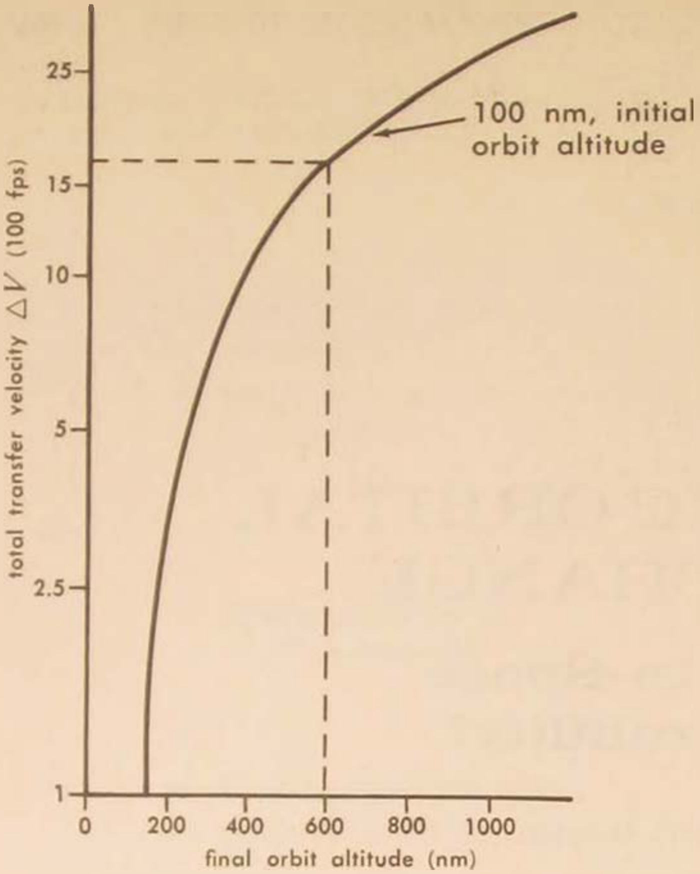
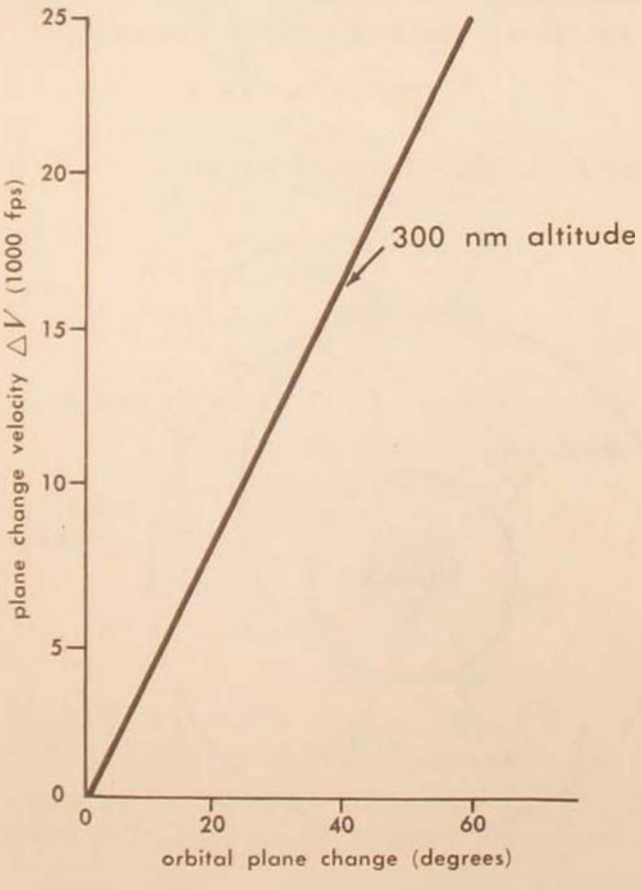


Figure 2. Hohmann transfer velocity requirements

Figure 3. Plane change velocity requirements



if a plane change of 60 degrees were required at a particular altitude, it can be seen from the following basic formula that as much energy is needed to accomplish the plane change as is required to maintain a circular orbit at that altitude:

$$\Delta V = 2V_s \sin \frac{\Delta i}{2}$$

where V_s is the satellite velocity and Δi is the plane change. A plane change of more than several degrees at a typical near-earth orbital altitude of 300 nm requires thousands of feet per second of energy or ΔV (Figure 3).

If representative values for specific impulse and weight ratio are substituted in the ideal velocity equation,

$$\Delta V = I_{sp}\ g \ln \frac{W_{\text{initial}}}{W_{\text{final}}}$$

where I_{sp} is specific impulse of the propellant, g is earth gravity, and ln is natural logarithm, it will be found that the maximum ΔV available is on the order of 5000 feet per second. Again referring to Figure 3, we find that a plane change of only about 10 degrees can be made with this amount of ΔV.

A 10-degree plane change capability would be more than adequate for orbit corrections and terminal rendezvous maneuvers that could be required for resupply of a space station, for example. Also, in-space maintenance techniques could be employed by spacecraft at the same altitude and within several degrees of the malfunctioning satellite.

But suppose we project our thinking into the middle of the next decade—or into the mid-80's—or simply into the future. Space-based maintenance shuttles, rescue spacecraft, and even perhaps some type of operational spacecraft—all may need substantially more in-space maneuvering capability than technology can provide in some particular time period.

How can in-space maneuvering capability be increased? There are two obvious ways. Reference to the ideal energy equation shows, first, that higher values of specific impulse could be obtained through advances in high-energy fuel technology, and, second, that higher weight ratios could result from advances in maneuvering-engine propulsion technology. At the present time it appears that the first of these technological advances, a several-fold increase in specific impulse, will be necessary to provide adequate in-space maneuvering capability for future space operations. However, it is not at all clear just when this breakthrough might occur, and in fact there is some question that it will.

Regardless of whether future manned spacecraft have 5000 fps, 10,000 fps, or even greater ΔV, there is a third potential method of obtaining increased maneuvering capability. This method, called the synergetic maneuver, employs both propulsion and aerodynamic forces to change a spacecraft's orbital plane.

The initial study which investigated the theoretical feasibility of synergetic plane changing was conducted by the RAND Corporation during 1962.* Two basic considerations were involved in the analysis: (1) Changes in orbital altitude, particularly below 600 nm, require relatively small amounts of energy; and (2) Manned spacecraft employed in future military space operations will be designed for lifting re-entry.

"Synergetic" is defined as working together or cooperating, and that is exactly what happens between the two forces, propulsion and aerodynamic lift, during the synergetic plane change. The maneuver consists basically of four phases: (1) deorbit and descent to the upper atmosphere, (2) pullout and constant-altitude gliding turn, (3) acceleration to orbital speed and ascent, and (4) injection into the new orbit at the original altitude. Figure 4 illustrates the complete synergetic maneuver. Note that three impulsive velocity increments (ΔV) are required: ΔV_1 = velocity increment for deorbit; ΔV_2 = velocity increment for acceleration and initiation of ascent; and ΔV_3 = velocity increment for injection. Also note that the turn maneuver, which results in the actual orbital plane change, requires no propulsion but is accomplished by aerodynamic lift.

The Synergetic Plane Change for Orbiting Spacecraft, by F. S. Nyland, RAND Memorandum RM-3231-PR, August 1962.

Now let's briefly examine each of the synergetic maneuver phases, recalling that the synergetic plane change will be applicable only to lifting spacecraft whose hypersonic lift-to-drag ratio (L/D) is somewhat greater than one, and to near-earth space operations (up to about 600 nm altitude).

Deorbit and Descent. The velocity impulse that initiates deorbit actually causes a decrease in the velocity of the spacecraft. This impulse is generally called retrofire or retroimpulse and is provided by some type of maneuvering engine attached to the spacecraft.

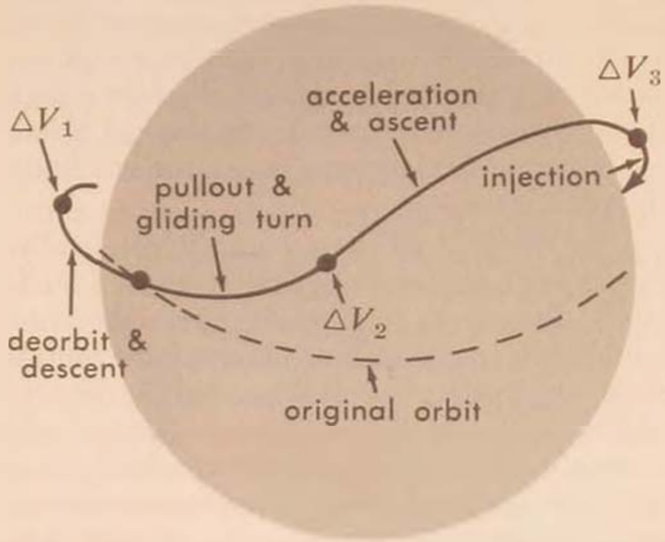


Figure 4. Synergetic plane change

The amount of retroimpulse (ΔV) determines the characteristics of the descent trajectory which the spacecraft follows. Actually, there are many such ballistic trajectories, but the one which would normally be selected is the Hohmann semiellipse. Not only is this the minimum-energy path but it would result in small re-entry angles, which in turn would minimize the deceleration force and heating rate encountered during atmospheric re-entry. (Remember that a spacecraft executing a synergetic maneuver must descend far enough into the atmosphere to make an aerodynamic turn, i.e., to an altitude of approximately 35 nautical miles.) Assuming that the descent path is a Hohmann transfer to an altitude of 200,000 feet, we find that the conditions at the termination of descent are zero flight path angle and a velocity greater than that of a circular orbit at 200,000 feet. Normally, if a spacecraft were simply changing orbit altitude, for example from 600 nm to 300 nm, a second velocity increment or retroimpulse would be required to circularize in the lower orbit. (See Figure 1.) This occurs because the point at which the transfer ellipse is tangent to the new circular orbit is the perigee of the transfer ellipse. If we recall the basic laws of orbital mechanics, we note that the velocity at this elliptical perigee is greater than the corresponding circular velocity. However, since the spacecraft will encounter a velocity loss during the next phase of the synergetic maneuver, the descent phase is considered to terminate under the condition of excess velocity.

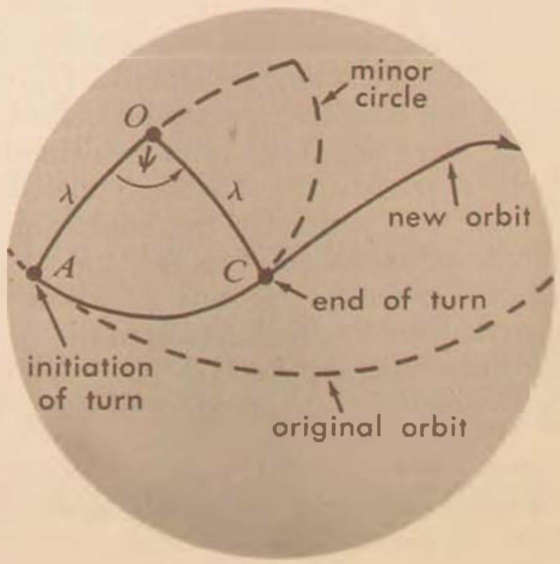
Pullout and Gliding Turn. Theoretically, no pullout maneuver is required when a Hohmann descent is employed. This maneuver would be initiated only when accurate control of the flight path angle at the end of the descent trajectory cannot be maintained. In this case a certain amount of velocity would be lost due to atmospheric drag as the spacecraft approaches 200,000 feet, but this velocity loss would be minimized by increasing the lift-to-drag ratio and/or decreasing the re-entry angle. Because of the many variables involved at the end of descent, the determination of optimum operating techniques cannot be made entirely on a theoretical basis but would undoubtedly require actual experience.

When the spacecraft has attained a zero path angle at the desired altitude of 200,000 feet—and, as we stated earlier, at a velocity in excess of circular orbit velocity—a so-called glide is begun, and a constant-altitude aerodynamic turn is initiated. The types of turning trajectories by which orbital plane changes can be obtained fall primarily into two categories: one method would be to hold a constant bank angle, and the other method would be to vary the bank angle of the spacecraft so that its trajectory is a circle. The latter trajectory is more readily analyzed, and the geometry of the trajectory, known as a minor circle turn, is shown in Figure 5. Note that a turn from equatorial orbit to a new orbit has been selected for easier visualization. Although this selection does not affect an analysis of the synergetic plane change, since such a maneuver can be employed at any orbital inclination, one point should be emphasized. If a change in orbital plane is to be equal to a change in orbit inclination,* the line of intersection of the two planes must be in the equatorial plane. Stated in more practical terms, this trigonometric fact simply means that a plane change made at any latitude other than the equator will not result in the same amount of changed orbit inclination. Furthermore, it follows that a minimum-energy orbit inclination change is always made when the spacecraft is in or crossing the equatorial plane.

Referring now to Figure 5, we see that the minor circle turn is defined by the parameter λ, which is simply its radius measured as an arc length about the center of the earth. The minor circle defined by $\lambda = 45°$, for example, would pass through the North Pole if it were tangent to the equator. The position of the spacecraft during a turn along the minor circle

*Orbit inclination, or, more loosely, orbital plane, is defined as the angle of the plane with respect to the equatorial plane.

Figure 5. Minor circle turn



AC is defined by ψ, the angle between the spacecraft position and the initiation point of the turn measured about the center of the minor circle, O.

By the use of spherical and plane trigonometry, an equation can be derived from the geometry shown in Figure 5 which relates the new orbital plane to the old for different values of the angle ψ, or position coordinate. Also, since this position coordinate is a function of the range along the minor circle, which in turn is related to the velocity loss encountered by the spacecraft as it decelerates in the gliding turn, logarithmic expressions can be established which permit the calculation of this velocity loss. The equations also show that again, as we noted in the case of a possible pullout manuever at the end of descent, the velocity loss during the aerodynamic plane change turn is decreased as the lift-to-drag ratio is increased. This fact, of course, is the basis for the variation in propulsion energy required for a particular plane change as the maximum hypersonic lift-to-drag ratio of the spacecraft varies. This variation will be illustrated later.

Acceleration and Ascent. When the gliding turn is completed, the spacecraft must accelerate to the orbital velocity corresponding to an altitude of 200,000 feet before a Hohmann transfer maneuver back to the original altitude can be initiated. Theoretically, a velocity increment for acceleration and a second velocity increment for initiation of ascent can be calculated from the basic equations of orbital mechanics. Operationally, however, a single total impulse would be applied by the maneuvering engine.

Two further points should be noted. First, the spacecraft does not have to return to its exact original orbit altitude. The use of identical initial and final altitudes simply eliminates one calculation because of the symmetry of the Hohmann transfer: ΔV for ascent $= \Delta V$ for descent. Since the synergetic maneuver can be used effectively from altitudes up to only 500 or 600 nautical miles, and since most operations of any duration would employ orbits of at least 150 to 250 nm, a maximum variation in final altitude of several hundred nautical miles would still be insignificant. Secondly, the velocity increment required to accelerate—that is, to offset the velocity loss during the aerodynamic turn—is approximately 5 to 10 times that required to initiate the Hohmann ascent maneuver.

Injection. The final phase of the synergetic maneuver is injection of the spacecraft into the new orbit. As we noted above, for return to the original altitude using the Hohmann transfer, injection requires the same velocity increment as that used for deorbit. The only difference, of course, is that the spacecraft velocity is decreased at deorbit and increased at injection.

The description of the synergetic plane change presented here has been greatly simplified. Those interested in a detailed analysis should obtain the RAND report referred to.

As stated earlier, the synergetic maneuver is a *potential* method of augmenting a spacecraft's plane-changing capability. The theoretical advantage of employing the synergetic plane change is indicated graphically in Figure 6. The total velocity requirements are plotted for the two methods of plane changing—the synergetic maneuver and propulsion only—for spacecraft with varying hypersonic lift-to-drag ratios. Values for the synergetic maneuver were calculated by use of the analysis described above, and the pure propulsion velocity requirement was computed from the basic equation shown earlier, $\Delta V = 2V_c \sin \frac{\Delta i}{2}$. For spacecraft with lift-to-drag ratios of about one or less, there is no increase in plane change capability (with a given amount of propulsion) through use of the synergetic maneuver. For lift-to-drag ratios beginning at about 1.5, the reduction in ΔV required for a particular plane change becomes significant; and for the higher values of L/D, the saving in energy over the straight propulsion method is appreciable. It is apparent, however,—and this point should be emphasized—that the synergetic plane change can be economically employed only for plane changes greater than about five degrees, regardless of the lift-to-drag ratio of the spacecraft.

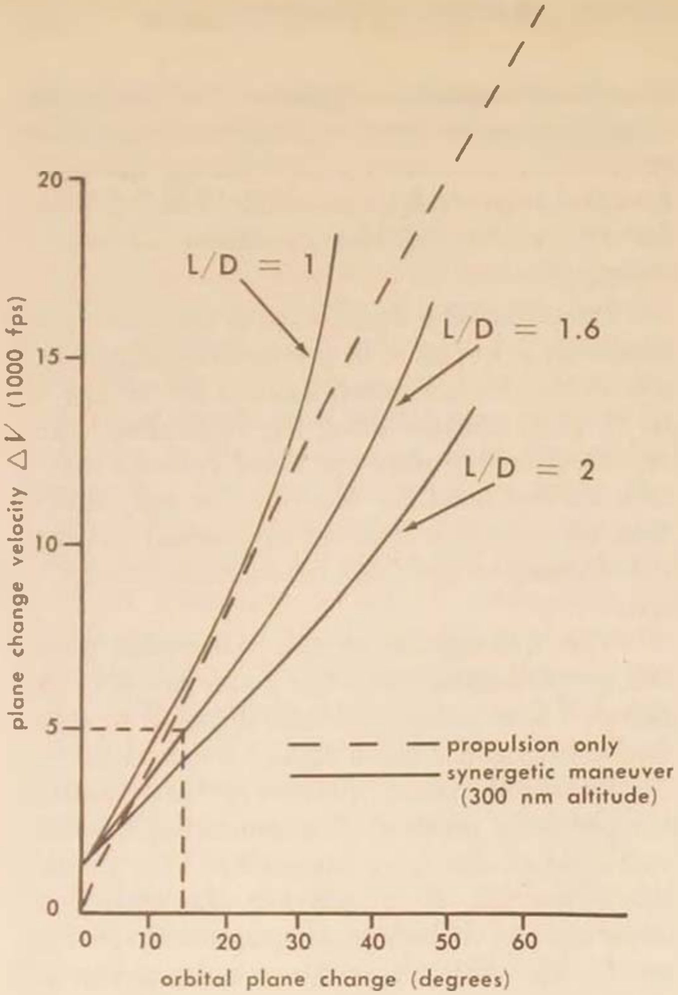


Figure 6. Comparison of plane change velocity requirements

So much for the theoretical analysis of synergetic plane changing. As far as future space operations are concerned, how practical could it be?

First, of course, there is the question of spacecraft lift-to-drag ratio. The semiballistic spacecraft being developed by NASA, Gemini and Apollo, will not have a high enough L/D to employ the synergetic maneuver. In fact, there is no spacecraft under consideration at the present time which could effectively use the maneuver. Keep in mind, though, that our discussion has been focused toward future planning and has assumed possible operational requirements and technological capabilities of the mid-1970 time period.

Second, from the standpoint of practical application, the synergetic maneuver will require a very efficient spacecraft cooling system and highly accurate guidance and control equipment. Re-entry, just into the upper atmosphere, will require a significant advance in cooling technology in order to avoid any structural degradation. "Flying" the spacecraft in a variable-bank turn along a minor circle—or holding a constant-bank turn—will be a very exacting maneuver, even for the most experienced astronaut using the best instruments available.

Finally, the most critical aspect of the operational potential of the synergetic plane change will be the actual situation. To illustrate this point, let us consider a hypothetical situation that could occur during the next decade.

The United States is maintaining a space station for the purpose of conducting scientific and military experiments. The station is in a 30° circular orbit at an altitude of 300 nm. Six men are on board, one of whom is an expert electronics maintenance technician. There are three two-man ferry spacecraft attached to the station (in case of an emergency, the entire six-man station crew might have to be evacuated to earth). Two of the spacecraft are the advanced, lifting-body type which were developed during the early 1970's. The third is one of the few remaining Gemini vehicles. Each of the ferry vehicles has a maneuvering engine rated at 5000 fps ΔV, and the Gemini spacecraft and one of the lifting-body vehicles have a full load of propellant.

After some ten years of employing inclined synchronous and random type communication satellites, the United States has developed a 24-hour synchronous system to be launched into an equatorial orbit. Because of booster and launch azimuth limitations, the communication satellite will first be launched into a parking orbit at 300 nm and 15° inclination. After a final checkout by telemetry, it will be injected into its operating orbit at 19,300 nm and 0° inclination.

The situation is this. During checkout it is determined that the power supply of the communication satellite is not functioning properly. Should the satellite be abandoned, or should an attempt be made to repair it? Quite a few in-space maintenance experiments have been conducted from the space station during the past year, and there is a good chance that the communication satellite can be repaired if it can be reached. An earth-launched maintenance mission would not be economical, but if one of the ferry spacecraft could be used to repair the satellite on the way back to earth on a scheduled crew-rotation flight, millions of dollars would be saved.

A quick check of the situation reveals that 6250 fps ΔV is required for a ferry spacecraft in the 30° space station orbit to change to a 15° orbit, and only 5000 fps ΔV is available. But let's reconsider: Could one of the spacecraft reach the malfunctioning satellite by employing the synergetic plane change?

The Gemini spacecraft has an L/D of about .30, so it is obviously incapable of such a maneuver. However, the lifting-body ferry vehicle has a hypersonic lift-to-drag ratio of 1.6, and according to the theoretical curve shown in Figure 6, such a spacecraft is capable of a 15° plane change by a synergetic maneuver. Furthermore, with its lateral maneuvering range of 1000 nm during landing re-entry, the lifting-body spacecraft would have no difficulty returning to its normal recovery site.

Apparently, then, the two-man lifting-body spacecraft could augment its ΔV capability an equivalent of 1250 fps by deorbiting, making an aerodynamic turn at the equator to a 15° orbit, and returning to 300 nm altitude. Close-in rendezvous with the communication satellite could be accomplished by the ferry astronaut, and the electronics technician might correct the malfunction by using extravehicular repair techniques.

The situation postulated here is obviously preconceived. However, there is a point which is more significant than illustrating how the synergetic plane change could be employed to an economic advantage. Regardless of future spacecraft design and regardless of their in-space maneuvering capability, situations could occur during military space operations which would necessitate additional plane change capability. The laws of orbital mechanics are irrevocable. If a plane change of x degrees were required to accomplish an urgent mission and the spacecraft deployed did not have enough ΔV to change planes by x degrees, there would be absolutely no way in which it could perform the mission—except, perhaps, by a synergetic plane change.

Hq Air Force Systems Command

Bibliography

1. Nyland, F. S. *The Synergetic Plane Change for Orbiting Spacecraft.* Santa Monica: RAND Corporation, August 1962.
2. Warfare Systems School, Air University. *Space Handbook,* July 1964. (Instructional text only.)
3. White, J. F. *Flight Performance Handbook for Powered Flight Operations.* New York: John Wiley & Sons, 1962.
4. Wolverton, R. W. *Flight Performance Handbook for Orbital Operations.* New York: John Wiley & Sons, 1963.

ROCKET PROPULSION FOR SPACE

Fundamental Considerations

Major Kenneth H. Roper

Any consideration of general military operations in space must give ample thought to space propulsion. This is true simply because we are propulsion-limited with respect to any capability for *general* operations in space. Operations which we now perform in space are peculiarly adapted to minimal propulsion requirements. In a sense, the space propulsion limitation is to space flight as the so-called sonic and thermal barriers are to higher-velocity atmospheric flight.

The following discussion attempts to do three things: to present in a summary but comprehensive fashion the fundamental concepts and parameters upon which rocket propulsion depends; to show how these parameters impose current limitations; and to indicate in a general way the avenues of improvement that are open to the future.

Propulsion is a particular space flight problem for a number of reasons. First, the velocities involved in space flight are much greater than those in atmospheric flight, with which we are all more familiar. Also in space flight we have no thrust multiplier such as we have in atmospheric flight, wherein the wing produces a lift-over-drag ratio typically in the neighborhood of 15. So that we can disabuse ourselves of some possible misconceptions, it is worthwhile to spend just a moment on these two considerations.

For greater velocity, the first consideration, we require a larger acceleration, or an acceleration over a longer period of time, to go from rest to an initial condition of stable flight. But this is not the whole story. Because velocity is a vector quantity, we also require a larger incremental velocity whenever we want to change our *direction* of travel, as is illustrated in Figure 1. With a very small velocity, represented by a short vector, V_1, a small change in the direction of our velocity requires a small incremental change, ΔV. On the other hand, if the initial velocity is much greater, V_1', and it is desired to achieve the same change in direction—again without any change in the magnitude of the velocity—a proportionately greater incremental velocity, or $\Delta V'$, is required.

The question of change in direction leads to the second consideration. During horizontal flight by an aircraft in the atmosphere, a change in direction is produced by banking the aircraft to one side or the other so that a significant portion of the lift force acts horizontally in the direction in which it is desired to turn, as illustrated in Figure 2. For instance, in a 45° bank, a component equal to about 0.7 of the lift force is in the horizontal direction toward the center of curvature of the flight path. With a lift-over-drag ratio of 15, a force is thus produced normal to the direction of

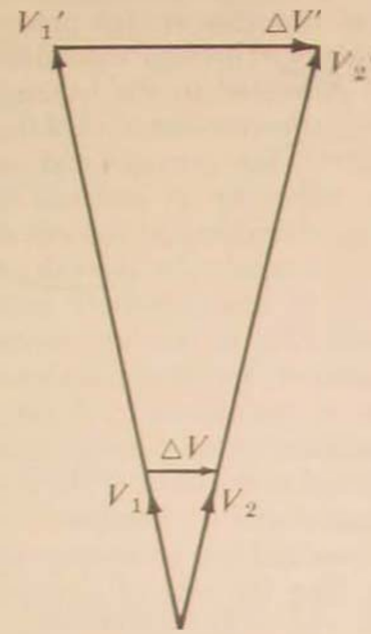


Figure 1. Vector change of direction

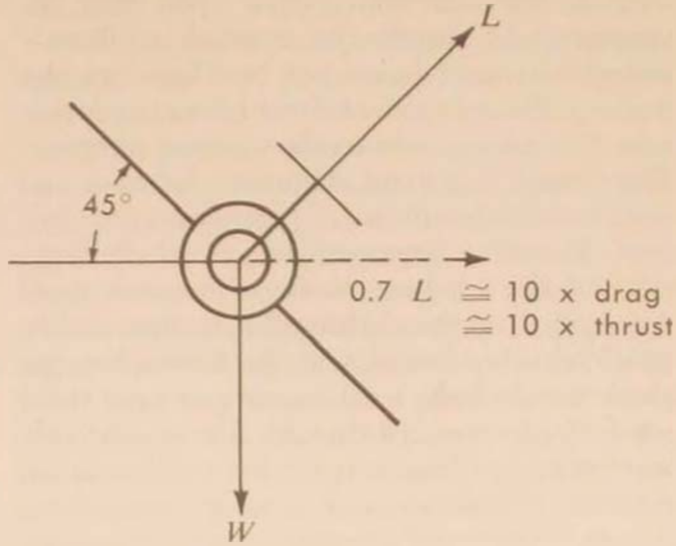


Figure 2. Forces on an aircraft in a horizontal turn

flight that is about 10 times greater than the drag on the aircraft, the drag being equal to the thrust for constant-speed flight. In space there is no such effective force multiplier for changing direction. Consequently the thrust to provide *any* incremental change in velocity—including change in direction—must be provided directly by the propulsion system.

There are additional considerations that complicate propulsion for space flight. One is the fact that the oxidizer must be carried along as part of any chemical propellant system. Even if nuclear or electrical rocket propulsion is employed, the working medium to be accelerated rearward to produce the forward thrust must be carried along in the space vehicle and accelerated from rest, through the velocity history of the flight, to the time when it is discharged to produce thrust. In contrast, the atmosphere is continuously available in atmospheric flight as the working medium to be accelerated rearward to produce the desired forward thrust.

From these considerations it is abundantly clear that propulsion for space flight is different from that for atmospheric flight. Accordingly, to understand space flight we must first understand the fundamentals governing space propulsion, i.e., rocket propulsion. Most of the concepts and principles presented in the following discussion are applicable to nuclear as well as chemical rockets, and many are likewise applicable to electrical rocket propulsion.

In the study of rocket propulsion, four fundamental natural phenomena are pertinent: (1) conservation of momentum, as stated in Newton's laws of motion, (2) conservation of mass, (3) conservation of energy, and (4) the behavior of gases as represented by the perfect gas equation, which interrelates pressure volume and temperature for any gas or mixture of gases not near conditions of liquefaction.

Rocket propulsion, like any familiar form of propulsion through a fluid medium, obtains its forward thrust from the dynamic reaction to the rearward acceleration of matter (Newton's third law of motion). Included in the "familiar" forms of propulsion are the canoe paddle, the ship's screw, the airplane propeller, and all forms of jet propulsion.

While rocket propulsion is identical to the other forms in this very fundamental respect, it differs from them in two important respects. First, the rocket (chemical, nuclear, or electrical) carries its working medium, i.e., the matter to be accelerated rearward, within itself, whereas the other forms draw upon their environment to supply the working medium—water or air, as the case may be. Therefore, the rocket is the only one of these forms of propulsion that can operate in the vacuum of space. The second important difference between the rocket and other forms of propulsion is as follows. The other forms utilize a relatively large mass of the working medium per unit time, accelerated rearward through a comparatively small velocity change, whereas the rocket employs a relatively small mass per unit time, accelerated rearward through a comparatively large velocity change. Indeed it will be shown that the effectiveness of a rocket propulsion system is improved proportionately with an increase in the velocity change through which the working medium is accelerated rearward.

To understand the basic operation of the rocket as a source of thrust, one must understand the nature of what is appropriately labeled the fundamental thrust equation. This equation is derived by application of Newton's second and third laws of motion.

Newton's second law of motion states that an unbalanced force acting on a mass produces a rate of change (with respect to time) of the momentum that is proportional to the magnitude of the unbalanced force and is in the same direction. Momentum is the product of mass, m, times velocity, V. Therefore, using appropriate units for force, mass, velocity, and time, one can write:

$$F = \frac{d}{dt}(mV)$$

where $\frac{d}{dt}$ denotes rate of change with respect to time. Newton's third law of motion states that for any dynamic action there is an equal and opposite reaction. Thus if a person pushes on a mass with a force of ten pounds, the mass pushes back on him with an opposite force of ten pounds.

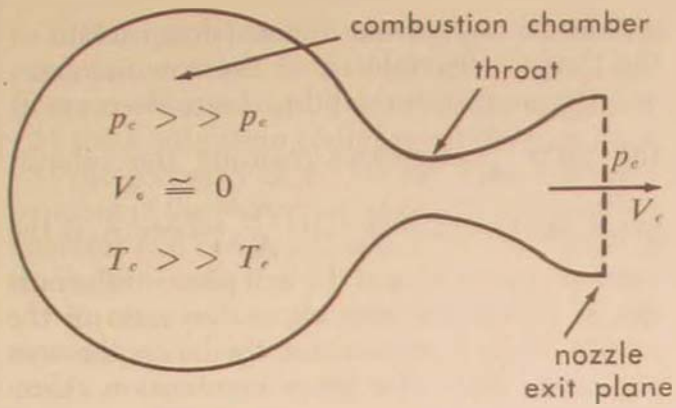
These laws of motion can be applied to the analysis of thrust from a rocket engine. (See Figure 3.) Typically, propellant enters the combustion chamber at high pressure and negligible velocity. Through combustion, the propellant is converted to the exhaust gases, which depart the downstream end of the rocket nozzle at relatively low pressure and very high velocity. The difference in pressure between the combustion chamber and the exit plane of the nozzle is what accelerates the exhaust gases from a velocity of approximately zero in the combustion chamber to the high velocity, V_e, at the exit plane of the nozzle. In accordance with the law of conservation of matter, the mass of the exhaust gases passing through the nozzle is identical to the mass of the propellant entering the combustion chamber.

Newton's second law of motion can be applied by equating the rate of change of momentum within the rocket combustion chamber and nozzle with the forces producing that change in momentum. We denote the mass flow per unit time through the nozzle as $\dot{m}$. From conservation of matter, it follows that this is the flow rate of propellant into the combustion chamber and also the flow rate of exhaust gases past *any* cross section of the nozzle. We then recall from the discussion above that this amount of mass each second is accelerated within the rocket from an initial velocity approximately zero to a final exit velocity V_e. Thus the rate of change of momentum within the rocket is

$$\dot{m}(V_e - 0) = \dot{m}V_e$$

We now must equate this rate of change of momentum to the sum of all the forces acting on the gases to produce this momentum change.

Referring again to Figure 3, the forces acting on the gases within the rocket engine to produce the above change in momentum are of two types. First, we have the sum of all the force exerted by the walls of the combustion chamber and nozzle on the gases. This we denote as F', taken positive *rearward*, i.e., in the direction of V_e. Second, we have the force exerted at the exit plane of the nozzle by the downstream gases. This is the force deriving from the static pressure, p_e, of the exhaust gases



p_c - pressure in chamber | V_e - velocity at exit

p_e - pressure at exit | T_c - temperature in chamber

V_c - velocity in chamber | T_e - temperature at exit

Figure 3. Schematic diagram of rocket engine

at the exit plane of the nozzle. By "static pressure" we mean the pressure that would be observed by a pressure sensor riding along with the exhaust gases, at the same velocity, so as to experience no dynamic pressure or ram effect. The magnitude of this pressure force, then, is p_eA_e, where A_e is the cross-sectional area of the exit plane of the nozzle; and the direction of this force exerted *by* the gases downstream of the nozzle *on* the gases within the rocket is opposite to V_e.

The sum of the forces can then be written as $F' - p_eA_e$, where the positive direction is rearward, i.e., in the direction of V_e, or of increasing momentum of the exhaust gases. Equating the sum of the forces to the change of momentum we then have

$$F' - p_eA_e = \dot{m}V_e$$

Recalling Newton's third law of motion, we recognize that the total force, F', of the walls of the combustion chamber and nozzle *on the exhaust gases* is exactly equal and opposite to the force of the exhaust gases *on the rocket*, which we shall denote F, taken positive *forward*. Then we can write

$$F = F' = \dot{m}V_e + p_eA_e$$

If the rocket is operating in a vacuum, no additional pressure or viscous forces act on the rocket, and the fundamental thrust equation is

$$F = \dot{m}V_e + p_eA_e$$

If the rocket is operating in an environment of ambient pressure p_a and we ignore any drag owing to the velocity of the rocket, then that pressure acts over the entire exterior of the rocket *except* the area A_e (upon which we noted previously that the static pressure, p_e, of the exhaust gases acts). Thus the ambient pressure, p_a, produces a net force on the rocket corresponding to the absence of p_a acting over area A_e. This net environmental force, then, is p_aA_e acting in the direction of V_e, or opposite to the thrust F. Accordingly, the fundamental thrust equation for a rocket engine operating in an environment of ambient pressure p_a is

$$F = \dot{m}V_e + p_eA_e - p_aA_e$$

or

$$F = \dot{m}V_e + (p_e - p_a)A_e$$

We shall want to discuss the terms and parameters on the right-hand side of the equation in some detail because they reveal a number of significant facts about rocket performance. Before doing that, however, we should consider briefly just how the high exhaust velocity V_e is obtained.

In a chemical rocket, the combustion process heats the resultant exhaust gases to a very high temperature. In a nuclear rocket, heat transfer from the nuclear core heats the working medium, which thus becomes the exhaust gas, to a very high temperature. These high-temperature gases are also under high pressure, determined by (1) their temperature, (2) the mass flow rate of the propellant, or working medium, and (3) the cross-sectional area of the *throat* of the nozzle, the throat being that region of the nozzle where the cross-sectional area is a minimum. These "combustion chamber" conditions of temperature, T_c, and pressure, p_c, are also referred to as stagnation temperature and stagnation pressure respectively, because they are associated with a condition of nearly zero, or negligible, gas flow velocity.

In accordance with the kinetic theory for gases, the high combustion chamber temperature, T_c, is characterized by a high kinetic energy, per unit mass, of random molecular motion of the molecules of the gases. That is, the molecules of the hot gases are in high-velocity, random molecular motion. Because of the high pressure, p_c, existing in the combustion chamber and the low pressure, p_a, existing outside the nozzle, the molecules of the hot gases, or working medium, are accelerated rearward. This acceleration produces the rearward flow through the nozzle, with progressively increasing velocity relative to the nozzle. As the gas expands and moves down the axis of the nozzle with this progressively increasing velocity, its pressure, density, and temperature decrease. This change is associated with the transformation of kinetic energy of *random* molecular motion of the hot gases in the combustion chamber to kinetic energy of *ordered* molecular motion, rearward along the axis of the nozzle. Thus, at the point of minimum pressure and temperature within the nozzle—at the exit plane—the flow has a high velocity, V_e.

One additional characteristic of the flow through the nozzle is noteworthy. It is associated with the throat of the nozzle. At that point the velocity of the flow is always equal to the local speed of sound; that is, the mach number is always 1. This velocity of flow is related to the velocity of random molecular motion associated with the stagnation temperature, which is approximately equal to the combustion chamber temperature, as mentioned earlier. For any rocket with a convergent-divergent nozzle, the flow will always be subsonic and the temperature high on the *upstream* side of the throat, and the flow will always be supersonic and the temperature comparatively low on the *downstream* side of the throat.

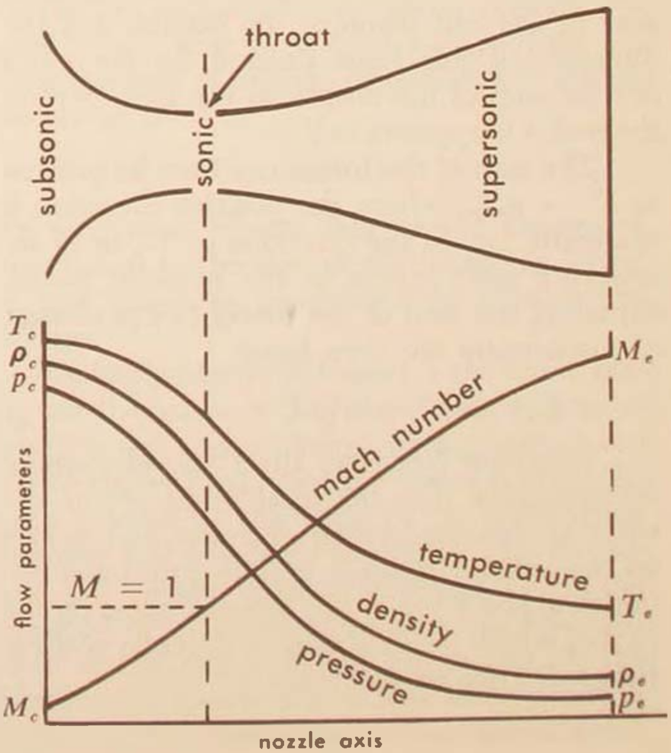
With certain assumptions that are good approximations of the real case, it can be shown analytically that the mach number at *any* station along the flow axis of a nozzle is a function of (1) the particular gas, or mixture of gases, and (2) the dimensionless ratio $\frac{A}{A_t}$, where A is the cross-sectional area of the nozzle at the station in question and A_t is the cross-sectional area at the throat. This analysis shows that, within the supersonic region (downstream of the throat), the velocity of the flow increases and the temperature and pressure decrease as the ratio $\frac{A}{A_t}$ increases, causing the exhaust gases to expand. The ratio $\frac{A_e}{A_t}$, where A_e is the cross-sectional area at the exit plane of the nozzle, is called the area expansion ratio of the nozzle. Thus it is seen that the larger the area expansion ratio (for given combustion chamber conditions), the greater will be the exit velocity, V_e. These phenomena are graphically summarized in Figure 4.

We are now in a position to consider in some detail the terms and parameters of the fundamental thrust equation:

$$F = \dot{m}V_e + (p_e - p_a)A_e$$

It is clear that this equation represents the thrust, F, as the sum of two terms. Because of their inherent makeup, the first term is called the *momentum thrust term*, and the second is called the *pressure thrust term*.

Figure 4. Variation of flow parameters along nozzle axis



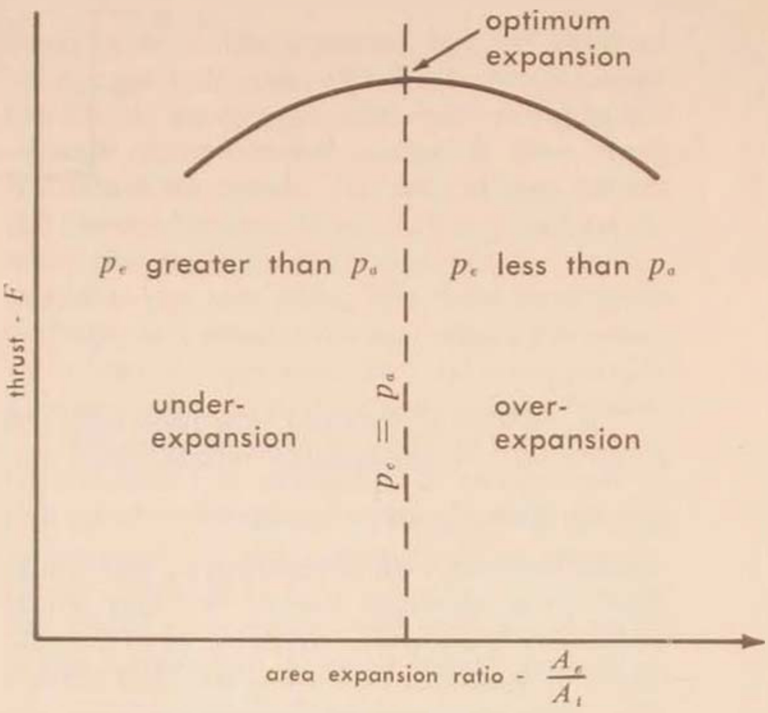
The momentum thrust term is by far the predominant one. It is dependent upon (1) the particular exhaust gas, or mixture of gases, (2) the combustion chamber pressure, p_c, and (3) the geometry of the nozzle. Thus it is independent of the environment in which the rocket operates and is a function only of the design of the rocket engine.

In contrast to the predominance of the momentum thrust term, the pressure thrust term can represent either a positive or a negative increment of thrust, depending upon the relative values of p_e and p_a. It therefore *is dependent* upon the rocket environment. If the nozzle area expansion ratio, $\frac{A_e}{A_t}$, is sufficiently large to reduce p_e below the ambient pressure, then the pressure thrust term will be negative. This circumstance, wherein the pressure thrust term is negative, is referred to as *overexpansion*. If the nozzle area expansion ratio is too small to reduce p_e to a value as low as the ambient pressure, then the pressure thrust term will be positive. In this case, the existing condition is referred to as *underexpansion*.

It is obvious, then, that a condition can exist, for any given ambient pressure, wherein the geometry of the nozzle dictates that the exhaust gas exit pressure be exactly equal to the ambient pressure. This is known as the condition of *optimum expansion*. With given combustion chamber conditions and mass flow rate, it represents a condition of maximum thrust for any fixed environment, i.e., fixed ambient pressure. An understanding of this fact is seen through reference to the fundamental thrust equation. Where a rocket is operating in a condition of underexpansion, with respect to a given p_a, additional thrust can always be obtained by increasing the area expansion ratio, $\frac{A_e}{A_t}$, to the point where $p_e = p_a$. This change will decrease the pressure thrust term from a positive value to zero. At the same time, however, the resultant increase in V_e will *always* increase the momentum thrust term by an increment greater than the decrease in the pressure thrust term; thus the total thrust will increase. Beyond this point additional increase in the expansion ratio will always produce a negative pressure thrust term greater in magnitude than the incremental increase in the momentum thrust term.

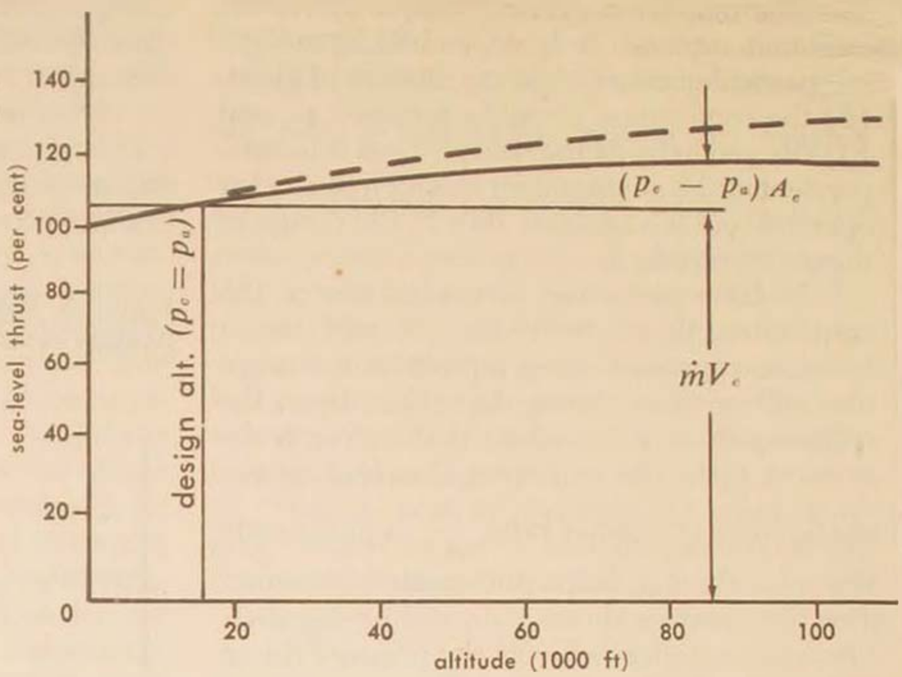
Thus maximum total thrust for a given p_a is obtained when $p_e = p_a$, the condition of optimum expansion. This condition is portrayed in Figure 5.

Figure 5. Variation of thrust with area expansion ratio for a fixed ambient pressure



The above discussion relates to a condition of fixed ambient pressure, p_a. It is now appropriate to consider the case of varying ambient pressure. Here the thrust will always increase as p_a decreases, even for a fixed expansion area ratio. This phenomenon also can be seen clearly by reference to the fundamental thrust equation. Therein, the momentum thrust term will remain constant as p_a decreases, because it is independent of the environment; but the pressure thrust term will increase with decreasing p_a all the way to the limit, a perfect vacuum, where $p_a = 0$. Thus a rocket of fixed design will invariably produce maximum thrust when operating in a vacuum. The above phenomenon is depicted in Figure 6.

Figure 6. Typical thrust variation with ambient pressure (i.e., altitude)



If, instead of the fixed expansion ratio just considered, an expandable nozzle were employed, then the nozzle expansion ratio, $\frac{A_e}{A_t}$, would increase with decreasing p_a and would allow p_e to decrease toward p_a. This would continuously allow operation closer to the condition of optimum expansion and thus allow a somewhat greater growth in thrust with decrease in p_a. This additional consideration is shown graphically by the somewhat more rapid growth in thrust with altitude, depicted by the broken line in Figure 6.

THUS FAR this discussion has treated the rocket engine only with respect to the thrust it can produce. Also important is the rate at which it consumes propellant.

A measure of performance of the rocket engine, then, is the ratio of the thrust produced to the propellant flow rate. In standard practice, thrust is measured in pounds; the propellant flow rate is measured in pounds per second, measured at standard sea-level conditions of gravity, where a force of one pound acting on a mass of one pound produces an acceleration $g_o = 32.2$ feet/sec/sec. In order to write $F = ma$, as we do in deriving the fundamental thrust equation, we must use the unit of mass known as the slug, which is g_o times the pound mass. The above defined ratio thus becomes

$$\frac{F}{\dot{m}g_o}\left(\frac{\text{pounds of thrust}}{\text{pounds/second of propellant mass}}\right)$$

If the numerator and denominator in the parentheses are multiplied by *seconds*, we then have

$$\left(\frac{\text{pounds-seconds of impulse}}{\text{pounds mass of propellant}}\right) = I_{sp}$$

This quantity we call *specific impulse* (I_{sp}) because it is seen to be the impulse delivered per pound mass of propellant consumed. I_{sp} is the primary measure of the performance of a rocket engine. It is a function of the design of the engine as well as the propellants used. It is also a function of the ambient pressure, as will be shown below. Hence for purposes of comparison it must be specified under standard conditions, such as sea level or vacuum.

Referring to the fundamental thrust equation, we can write

$$I_{sp} = \frac{F}{\dot{m}g_o} = \frac{\dot{m}V_e + (p_e - p_a)A_e}{\dot{m}g_o}$$

Recalling that the momentum thrust term is large in relation to the pressure thrust term (which is zero for optimum expansion), we may logically define a quantity, *effective exhaust velocity*, c, such that

$$\dot{m}c = \dot{m}V_e + (p_e - p_a)A_e$$

The specific impulse can thus be expressed

$$I_{sp} = \frac{\dot{m}c}{\dot{m}g_o} = \frac{c}{g_o}$$

Since g_o is a constant, the specific impulse is directly proportional to the effective exhaust velocity. This relation holds true for all forms of rocket propulsion—chemical, nuclear, or electrical. Thus it is clear that the higher the effective exhaust velocity the greater the specific impulse obtained.

For optimum expansion, $V_e = c$, and the expression for specific impulse becomes

$$I_{sp} = \frac{V_e}{g_o}$$

The question thus becomes: What determines the exit velocity, V_e? For chemical and nuclear rockets utilizing a gaseous working medium, an expression for V_e can be obtained directly from the equation for conservation of energy in a one-dimensional, *steady, adiabatic* flow. A *steady* flow is one in which all the properties of the flow are constant, or invariant, with respect to time. This is true for the flow through an ordinary rocket nozzle except for that extremely small fraction of the operating time when starting or shutdown transients are present. An *adiabatic* flow is one in which no thermal energy, or heat, is transferred to or from the fluid across the boundaries of the flow. This condition holds to a good degree of approximation for the flow through a rocket nozzle because of the very short time it takes any element of the exhaust gases to traverse the length of the nozzle. Although the walls of the nozzle do tend to become heated, the fraction of the total energy of the flow that goes into heating the nozzle walls is very small.

Thus the energy equation (for a unit mass) for the flow through a rocket nozzle can be written

$$\tfrac{1}{2}V^2 + C_pT = C_pT_o$$

where V = axial flow velocity at any point (feet per second)

C_p = specific heat at constant pressure (ft-lbs per slug per °F absolute)

T = temperature of fluid at same point (°F absolute)

T_o = temperature of fluid at stagnation point ($V = 0$) (°F absolute).

Each term in the equation has the units of energy per unit mass. The first term represents the kinetic energy per unit mass owing to the velocity of the ordered motion, or flow, along the axis of the nozzle. The second term carries the thermodynamic label *enthalpy* and represents the energy, other than kinetic energy, available per unit mass. The third term is the enthalpy at a point in the flow where the velocity is zero; it represents the total energy available per unit mass of the steady, adiabatic flow.

Because the flow velocity in the combustion chamber is negligible in comparison to that at the exit plane of the nozzle, we can assume that the combustion chamber temperature, T_c, is equal to T_o. We can then express the exit plane temperature and velocity in terms of the combustion chamber temperature as

$$\tfrac{1}{2}V_e^2 + C_pT_e = C_pT_c$$

Through algebraic manipulation we obtain

$$V_e^2 = 2C_p(T_c - T_e)$$

$$V_e^2 = 2C_pT_c\left(1 - \frac{T_e}{T_c}\right)$$

The term in the parentheses is the thermodynamic efficiency. It can be seen that decreasing T_e (by increasing the expansion area ratio of the nozzle) increases the theoretical thermodynamic efficiency.

To understand better the factors affecting V_e, we use a relationship from thermodynamics:

$$C_p = \frac{\gamma R}{(\gamma - 1)M'}$$

where $\gamma = \frac{C_p}{C_v}$, the ratio of specific heat at constant pressure to specific heat at constant volume (dimensionless)

R = universal gas constant—the same for all gases and mixtures of gases (49,700 foot-pounds per slug mole per °F)

M' = average molecular weight of the exhaust gases (dimensionless).

According to simple kinetic theory for gases, the ratio of specific heats, γ, for any given gas is a constant. For a real gas or mixture of real gases, γ is found to be a weak function of temperature. It is therefore customary to treat it as a constant, using an average value for the temperature range under consideration. Between different gases, γ theoretically varies in accordance with molecular complexity as shown in Table I. For chemical rockets, a γ of about 1.25 is typical.

Table I. Theoretical Value of Ratio of Specific Heats as a Function of Molecular Complexity

Molecular Complexity	Theoretical γ
Monatomic	1.67
Diatomic	1.40
Complex	$1.0 < \gamma < 1.4$

Substituting the above expression for C_p in the equation for V_e^2, we have

$$V_e^2 = \frac{2\gamma R}{(\gamma - 1)M'} T_c \left(1 - \frac{T_e}{T_c}\right)$$

It is customary to express the temperature ratio $\frac{T_e}{T_c}$ in terms of the pressure ratio $\frac{p_e}{p_c}$. If we assume that there are no losses due to friction (fluid viscosity) in the flow through the nozzle, then the flow is said to be *isentropic*, and the following relationship from thermodynamics applies:

$$\frac{T_e}{T_c} = \left(\frac{p_e}{p_c}\right)^{\frac{\gamma - 1}{\gamma}}$$

For a well-designed nozzle, the assumption of negligible losses from friction is a good one, and very little error is introduced. We thus have

$$V_e^2 = \frac{2\gamma R}{\gamma - 1} \frac{T_c}{M'} \left[1 - \left(\frac{p_e}{p_c}\right)^{\frac{\gamma - 1}{\gamma}}\right]$$

From the above definitions and derivations we now can write

$$V_e = \sqrt{\frac{2\gamma R}{\gamma - 1} \frac{T_c}{M'} \left[1 - \left(\frac{p_e}{p_c}\right)^{\frac{\gamma - 1}{\gamma}}\right]}$$

and

$$I_{sp} = \frac{1}{g_0}\sqrt{\frac{2\gamma R}{\gamma - 1} \frac{T_c}{M'} \left[1 - \left(\frac{p_e}{p_c}\right)^{\frac{\gamma - 1}{\gamma}}\right]}$$

For chemical rockets, γ, T_c, and M' are determined by the propellant selected. The ratio $\frac{p_e}{p_c}$ is determined by the design of the rocket nozzle.

If other parameters are held constant, it is clear that I_{sp} is proportional to the $\sqrt{\frac{T_c}{M'}}$. Thus from the point of view of specific impulse, it is desirable to achieve the highest possible combustion temperature and the lowest possible molecular weight for the exhaust gases.

The role of molecular weight in determining specific impulse is a principal reason why the nuclear rocket is attractive. For chemical rockets, the lowest practicable exhaust gas molecular weight is somewhat lower than 18 (the molecular weight of H_2O). In the nuclear rocket, wherein the working medium is heated by a nuclear reactor by means of a heat exchanger, we can select as the working fluid hydrogen (H_2), which has a molecular weight of 2. This ninefold reduction in molecular weight of the gas expanded through the nozzle would, other things being equal, give a threefold increase in specific impulse. However, because of temperature limitations in the reactor and heat exchanger, the hydrogen cannot be heated as hot as the combustion temperature for hydrogen and oxygen; hence the full benefits of the reduced molecular weight of the exhaust gas cannot presently be realized.

From the expression for I_{sp}, it is also apparent that the pressure ratio, $\frac{p_e}{p_c}$, has an effect. Other things being equal, the smaller the ratio $\frac{p_e}{p_c}$ (corresponding to a larger expansion area ratio for the nozzle) the greater the specific impulse. The expansion area ratio to be employed is limited by considerations of nozzle weight and, at lower altitudes, the phenomenon of overexpansion, discussed earlier.

The ratio of specific heats, γ, also has an effect, albeit small, on the I_{sp} developed. For typically large values of the expansion area ratio $\frac{A_e}{A_t}$, a decrease in the value of γ will produce a modest increase in the value of the specific impulse.

With the above discussion of the fundamental thrust equation and specific impulse, we now have a basic understanding of the rocket as a source of thrust. In this respect we saw that specific impulse is the primary measure of rocket engine performance. Now we must consider the rocket in the light of its ultimate mission, namely, to impart a change in velocity, ΔV, to some finite mass, its useful payload. In this process we shall evolve another performance parameter important when considering the rocket as a source of ΔV.

To determine the ΔV obtainable, we apply Newton's second law of motion in the form $F = ma$ to the total mass, M, of the vehicle. This mass includes the propellant and inert mass of the rocket, the payload, the guidance system, and all additional structure. Ignoring gravity and drag, the net force on the vehicle is the rocket thrust, F. Recalling that acceleration is the rate of change of velocity with respect to time, we can write

$$F = Ma = M\frac{d}{dt}(V)$$

From the definition of effective exhaust velocity we have that $F = \dot{m}c$. Thus we can write

$$\dot{m}c = M\frac{dV}{dt}$$

But the propellant mass flow rate, $\dot{m}$, is also the rate of *decrease*, with respect to time, of the total vehicle mass, M. Writing this in mathematical form,

$$\dot{m} = -\frac{d}{dt}(M)$$

Substituting for $\dot{m}$ in the previous equation,

$$-c\frac{dM}{dt} = M\frac{dV}{dt}$$

This equation is readily converted to a simple differential equation with the variables M and V separated. Integrating between the limits of initial vehicle mass M_1 and velocity V_1 and final vehicle mass M_2 and velocity V_2, we get

$$\Delta V = V_2 - V_1 = c \ln \frac{M_1}{M_2}$$

Recalling that $c = I_{sp}g_o$, we can write this equation as

$$\Delta V = I_{sp}g_o \ln \frac{M_1}{M_2}$$

Since mass is proportional to weight for a given gravitational field, we can also express the mass ratio as $\frac{W_1}{W_2}$ and write

$$\Delta V = I_{sp}g_o \ln \frac{W_1}{W_2}$$

In this form, however, W_1 and W_2 must be measured under conditions of equal gravitational attraction so that the weight ratio is identical to the mass ratio.

Since g_o in this expression is a constant, ΔV is directly proportional to (1) the specific impulse and (2) the natural logarithm of the mass ratio. As the logarithm of a number varies much less rapidly than the number itself, ΔV is more sensitive to I_{sp} than to $\frac{W_1}{W_2}$. However, it is apparent that the mass ratio, $\frac{W_1}{W_2}$, is another important performance parameter for a rocket vehicle.

The final weight, W_2, is the initial weight, W_1, minus the weight of the propellant consumed. Therefore W_2 includes the inert portion of the rocket engine, any unburned propellant, the payload, the guidance system, and any additional structure. If the inert portion of the engine has significant weight in relation to the other constituents of W_2, then this inert engine weight is a constraining factor on the mass ratio achievable.

Thus, between two rocket engines of equal specific impulse, the one giving higher performance would be the one that had the higher propellant mass fraction, or lower inert mass fraction. Conversely, two rocket propulsion systems can be compared on a basis of specific impulse only if similar propellant (or working medium) mass fractions are obtainable. For example, two liquid-propellant, chemical rocket engines can generally be compared on the basis of specific impulse. However, chemical rockets, nuclear rockets, and rockets using electrical propulsion cannot, in general, be compared on a basis of specific impulse alone.

Table II shows the relationship between mass ratio and ΔV for an assumed specific impulse of 400 seconds (460 sec is approximately the theoretical maximum I_{sp} for a liquid-hydrogen/liquid-oxygen engine operating in a vacuum, and 390 sec is the equivalent value for operation at sea level). The "Comment" column indicates roughly the practicability of achieving the corresponding mass ratio in a single stage. Note that for a mass ratio of 7 or 8, which is about as good as is practicable, a ΔV of only 26,000 feet per second is achievable with an I_{sp} of 400 seconds. Since the velocity for a vehicle in low-altitude, circular, earth orbit is also approximately 26,000 feet per second and velocity is a vector quantity, this ΔV limits the maneuver of a single stage (at a constant altitude of orbit) to a total of less than 60° of plane change. For a vehicle on the ground, this ΔV is not even sufficient to get into low-altitude orbit because the losses due to drag and gravity (in getting up to orbital altitude) constitute a propulsion requirement for an additional ΔV equivalent of 3000 or 4000 feet per second over and above the 26,000 feet per second required for orbit. For a total ΔV of 30,000 feet per second, a mass ratio greater than 10 is required, assuming the specific impulse of 400 seconds.

Table II. Variation of ΔV with Mass Ratio

$\frac{W_1}{W_2}$	Comment	$\ln \frac{W_1}{W_2}$	ΔV (ft/sec)
2.7	Easy	1	12,880
7.3	Difficult	2	25,760
20.0	Not practicable	3	38,640

Assume $I_{sp} = 400$ seconds
$c = I_{sp}g_0 = 12{,}880$ feet per second

On the basis of the expression for ΔV, we have two alternatives for increasing our capability beyond that described for a single-stage chemical rocket. We have to increase either the specific impulse or the mass ratio.

To get a manifold increase in specific impulse, we have to abandon chemical rockets and resort to nuclear rockets or electrical rocket propulsion in some form. Hence the development efforts in these areas today.

Because of payload and structural considerations, it is impossible to get a comparable improvement in mass ratio for a single stage. So we resort to multiple staging; i.e., the propulsion system is broken up into two or more increments which operate in series. When the propellant for the first increment of propulsion is consumed, the fixed, or inert, propulsion mass for this increment and the associated structural and flight-control mass are discarded. Thus, this mass no longer needs to be accelerated as the succeeding increments of propulsion come into operation.

The discarding of the used increment of propulsion is known as staging. With respect to the mass ratio for a given stage, it should be noted that the W_2 for that stage includes the weight of all succeeding stages as well as the payload. Thus to achieve a significant mass ratio (and hence ΔV) for a given stage, its propellant weight must be greater—usually several times greater—than the total weight of all succeeding stages plus payload. Hence, as the number of stages increases to obtain greater ΔV, the total weight of the vehicle increases very rapidly.

The benefits of staging are shown quantitatively by reference to the equation for ΔV. Assuming the same I_{sp} for each stage and simply adding the ΔV for each of n stages, the expression for the total ΔV becomes

$$\Delta V_T = I_{sp} g_o \ln\left(\left.\frac{W_1}{W_2}\right|_1 \times \left.\frac{W_1}{W_2}\right|_2 \times \ldots \times \left.\frac{W_1}{W_2}\right|_n \right)$$

Thus the effective mass ratio for the total vehicle of n stages is

$$\left.\frac{W_1}{W_2}\right|_T = \left.\frac{W_1}{W_2}\right|_1 \times \left.\frac{W_1}{W_2}\right|_2 \times \ldots \times \left.\frac{W_1}{W_2}\right|_n$$

As a specific example, a three-stage missile with each stage having a mass ratio of 4 would have a total, or overall, mass ratio of $4 \times 4 \times 4 = 64$.

In view of this relationship, we see that there is no structural or design state-of-the-art limitation on the magnitude of mass ratio achievable. However, increasing the number of stages rapidly increases the complexity of the vehicle. This complexity then becomes a limiting factor because of cost and reliability considerations.

The preceding discussion has been a quick review of the governing considerations with respect to rocket propulsion, the only means of propulsion in space. The limitations imposed by today's available propulsion and the possibilities for improvement have been outlined. In conclusion, it seems appropriate to quote from an address by the Honorable Brockway McMillan, Assistant Secretary of the Air Force (R&D), before the American Rocket Society on 18 July 1962:

> I realize that there are laws of physics which control the degree to which the flexibility I have just described can be achieved, and fix the minimum price thereof. At the moment, we also have many other engineering problems to think about, but in the long run the usefulness of space vehicles to the military, if they are useful at all, will be limited by the efficiency and capability of their propulsion systems. High energy fuels, storable in orbit, techniques of refueling, restartable and throttleable engines that realize the maximum specific impulse from their fuels, nuclear engines, nuclear impulse engines, electric engines, and radiation engines, must all be considered for their practicability and applicability to maneuvering vehicles.

Hq Air Force Systems Command

IV

SPACE APPLICATIONS

O vast Rondure, swimming in space,
Cover'd all over with visible power
and beauty,
Alternate light and day and the
teeming spiritual darkness,
Unspeakable high processions of
sun and moon and countless
stars above . . .

Walt Whitman, "Passage to India"

ROCKET ENGINES AND PROPULSION

MAJOR JOHN H. WATTS

PROPULSION is one of the keys to man's future in space. As his ambition for greater achievement in the vastness of space expands, propulsion will more than ever be the limiting barrier to unhampered movement in space and will continue to be the paramount technological problem. It is to be expected that as the other technological problem areas such as life support, shielding, communications, navigation, and guidance are solved for near-earth and lunar missions an adequate technological base for conducting other more distant space voyages will exist. In a sense, these are one-time problems. For example, life-support requirements at 300 million miles (or 300 billion) are not likely to differ greatly from those at 300 miles. The difference in propulsion requirements, however, is staggering. The success of present as well as future space missions will depend on our continuing scientific and economic ability to provide adequate propulsive power.

When compared with conventionally powered, in-atmosphere aircraft, which carry only fuel and obtain oxidizer from the atmosphere, the space system undergoes a severe penalty in payload capability as a result of the requirement to carry both fuel and oxidizer.

A simple example will illustrate the degradation in payload and performance that a rocket-propelled vehicle undergoes because it must carry not only its fuel but also the oxidizer. If one takes some liberty with technical possibilities, he can visualize for a moment a T-33 that has been modified so that its tanks will carry liquid oxygen (lox) and JP-4 and that its engines will burn this combination. In certain rocket boosters liquid oxygen and RP-1 (JP-4) are burned in the approximate ratio of 2.4 to 1 by weight. The normal T-33 fuel load is 5280 pounds. Divided into the proper ratio, the T-33 propellant load of oxidizer and fuel would be as follows:

1550 lbs JP-4
3730 lbs lox
5280 lbs propellant

Assuming normal consumption of JP-4 for the T-33, the fuel required for start, taxi, take-off, and straight-out climb to 30,000 feet is 1170 pounds. Approximately 2800 pounds of lox would be required during this period for combustion. A level-flight return to the field would consume the remaining 380 pounds of fuel and the last 930 pounds of oxygen. In terms of our present-day requirements, a mission of this kind would have little practical value.

As mission requirements place greater and greater burdens on propulsion systems and as the ability of these systems to meet requirements is stretched to the utmost, the need grows for a greater understanding of the fundamentals of space propulsion systems by planners and decision-makers. Although the design and development of a high-performance rocket engine constitute a complex endeavor, the basic principles and processes involved are easily understood.

propulsion system classification

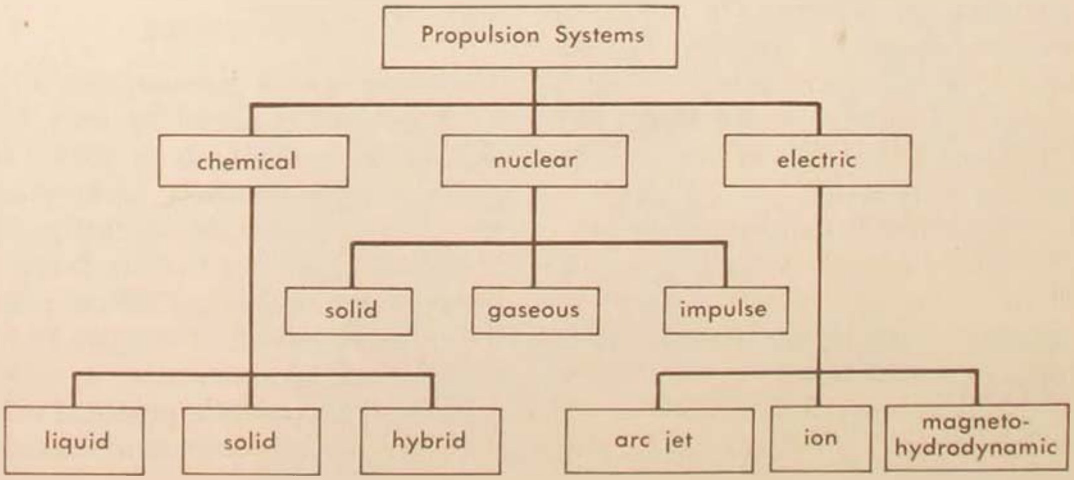
Space propulsion systems are often thought of as consisting of the engine alone. The propulsion system, however, is made up of the engine and accessories and, of great significance, the propellant and propellant tanks. The impact of the propulsion system on the design of the overall space system becomes evident when one realizes that its combined weight may easily exceed 97 per cent of the total space vehicle weight at launch. Most of this poundage, as the vehicle rests on the launch pad, is the weight of the propellants. In the Mercury-Atlas combination, for example, the weight of fuel and oxidizer is approximately 94 per cent of the total weight of the vehicle at launch. The weight of the structure, engines, other equipment, and payload accounts for the remaining 6 per cent. The useful payload, the capsule itself, represents approximately 1 per cent of the total. This latter percentage is called the payload fraction.

Rocket propulsion systems may be classified in several ways, depending on the distinctions to be made. One method of classification which relates propulsion systems to each other is shown in Figure 1. Of these classifications only the chemical propulsion system is in operational use today. Chemical propulsion systems have the lowest theoretical performance potential. The nuclear and electric systems as well as the hybrid chemical are considered advanced systems and will require considerable research and development to prove feasibility and provide operational models. Nuclear systems generally rank next higher in theoretical performance, with certain of the electric systems promising the highest performance of all. It is from these higher performance concepts that we receive encouragement that propulsion requirements for future deep space missions can be met.

propellant characteristics and performance

Chemical systems, the main topic of this article, can be further classified by physical properties of the propellants used in the system and by the chemical composition of these forms (Figure 2).

Figure 1. Classification of propulsion systems



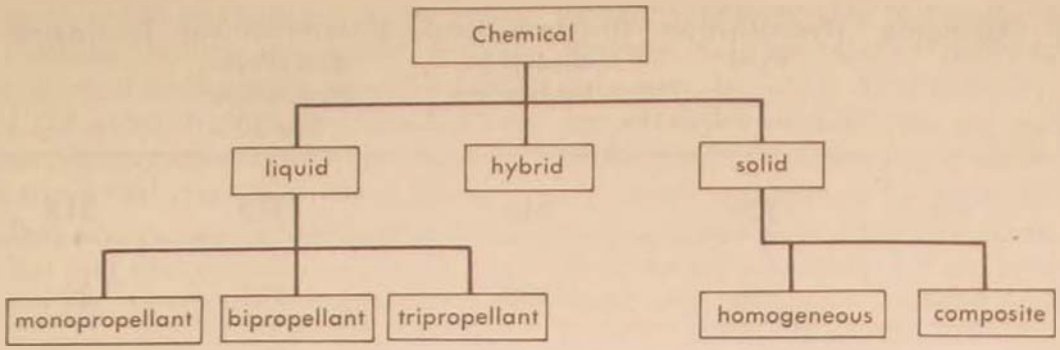


Figure 2. Classification of chemical propulsion systems

Liquid Propellants. Liquid propulsion systems use only fluids for the propellant. These fluids may be either monopropellants, bipropellants, or tripropellants. They may be mixtures or chemical compounds, cryogenic or storable. All chemical propulsion systems must contain both a fuel and an oxidizer, which, under the proper conditions, react chemically and release the desired energy.

In the monopropellant fluids, the fuel and the oxidizer are contained in a single substance. That is to say the fuel and the oxidizer are mixed together and are carried in a single tank. There is no requirement to separate the two as is necessary with other more chemically active propellants. The single substance may be either a mixture of the fuel and oxidizer, as with hydrogen peroxide and alcohol, or it may be a chemical compound containing the fuel and oxidizer in a single molecule. Nitrocellulose is an example of the latter. The monopropellant combination is stable at ordinary temperatures and pressures, but when heated or brought into the presence of a catalyst, the fuel and oxidizer react and provide the energy release. Combining the fuel and oxidizer into a single ingredient results in a simpler system in comparison with bipropellant systems, which must keep the propellants separated until they are mixed in the combustion chamber. With monopropellant systems a single tank is required, only one pump is necessary, and the plumbing is greatly simplified. Injector design is not as critical, since there are no problems of mixing fuel and oxidizer in the thrust chamber, the proper proportions having been previously combined in the fuel tank. Because of these simplifications in design, monopropellant systems are generally more reliable than bipropellant systems; on the other hand, they are generally lower in performance.

Most of our present-day systems use bipropellants. The greater energy release thus obtained more than offsets the greater handling, storage, and reliability problems inherent in the use of bipropellants.

Bipropellants use two chemical substances which for several reasons must be kept separate until combustion is desired. Some propellant combinations, such as nitric acid and analine, ignite spontaneously upon contact with each other. Spontaneously ignitable propellants are commonly called hypergolic propellants. Although it is not a great technical problem to provide an ignition system for nonhypergolic fuels, spontaneous ignition does simplify the system. Other, more important propellant characteristics, however, make this a secondary consideration in propellant selection.

In addition to the basic fuel and oxidizer in the bipropellant systems, chemical additives are sometimes used to improve storage and handling characteristics. Certain catalysts may also be added to speed up the chemical reaction and to release more energy. Bipropellants are kept separate until they are injected into the thrust chamber in the proper proportions. Figure 3 lists the specific impulses of several common bipropellant fuel/oxidizer combinations in use today.

Oxidizer \ Fuel	Ammonia	Hydrocarbon Fuel	Unsymmetrical-dimethyl-hydrazine	Unsymmetrical-dimethyl-hydrazine 50/50	Hydrazine	Hydrogen
liquid oxygen	294	300	310	312	313	391
chlorine trifluoride	275	258	280	287	294	318
hydrogen peroxide	262	273	278	279	282	314
red fuming nitric acid	260	268	276	278	283	326
nitrogen tetroxide	269	276	285	288	292	341
fluorine	357	326	343	353	363	410

Figure 3. Theoretical specific impulses of certain liquid bipropellants with chamber pressure 1000 psia, atmospheric pressure 14.7 psia, and conditions of shifting equilibrium

All oxidizers shown will react with all the fuels listed. This is the characteristic of the oxidizer family that makes them good oxidizers and also contributes to the fact that, in general, they tend to be very corrosive and toxic. Oxygen of course is an exception to this, but in its liquid form it is difficult to store and handle. Certain of the fuels and oxidizers are liquid only at extremely low temperatures. Liquid oxygen boils at $-297°$ F, fluorine at $-306°$F, and hydrogen at $-423°$ F. Propellants having very low boiling points are called cryogenic propellants and require special storage and handling techniques.

Tripropellants contain a third substance in addition to the fuel and oxidizer. The third substance may react with oxidizer or fuel, or may just act as a working fluid by providing lighter-weight exhaust products for added thrust.

Specific Impulse. The specific impulse (I_{sp}) is a general measure of the energy content or efficiency of a propellant combination. The specific impulse indicates the thrust in pounds per pound of fuel consumed per second (Figure 3). The unit of specific impulse is the second, derived as follows:

$$I_{sp} = \frac{\text{thrust (lbs)}}{\text{flow rate of propellants (lbs/sec)}} =$$

$$\frac{\text{lbs} \times \text{sec}}{\text{lbs}} = \text{sec}$$

Specific impulse may be referred to either as delivered or theoretical. Theoretical specific impulse is the maximum specific impulse theoretically possible with a given propellant combination under given conditions. Delivered specific impulse is that actually achieved from the propellant combination when used in a given system. Differing characteristics of engine systems will result in different delivered specific impulses even though the same propellants may be used.

In comparing values for the specific impulses of different propellants, it is necessary to ensure that the specific impulses were computed or were measured under the same conditions of chamber pressure and atmospheric pressure. It is also necessary to ascertain whether the specific impulses were derived by assuming a condition of shifting or frozen equilibrium. Under conditions of shifting equilibrium, the chemical composition of the exhaust gases is considered to be changing as the gases proceed rearward through the nozzle. An assumed condition of unchanging chemical composition of the exhaust gases is called frozen equilibrium. Shifting equilibrium more nearly represents the actual processes taking place.

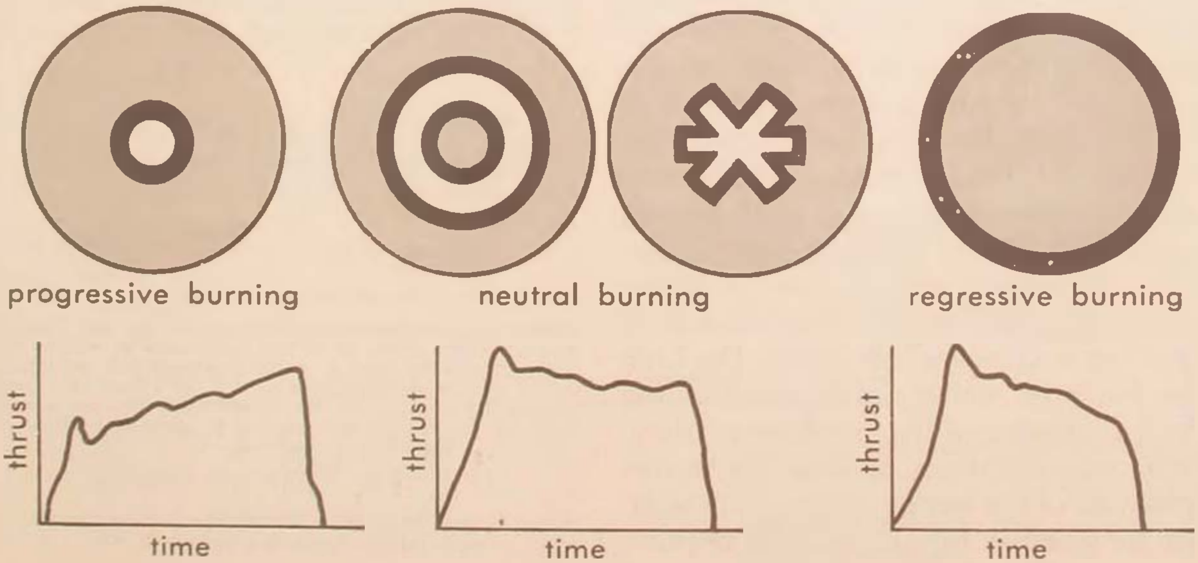
Solid Propellants. Solid-propellant systems have the fuel and the oxidizer combined in a single solid mass. As with liquid monopropellants, this combination of solid fuels and oxidizers can be accomplished either as a mixture or as a chemical compound. A composite or heterogeneous solid propellant is a mixture of the fuel and oxidizer. The oxidizer is dispersed uniformly throughout the mixture and usually represents from 70 to 80 per cent of the mass. The fuel, often called the binder, holds the propellant grain together. A homogeneous solid propellant contains the fuel and the oxidizer in a single chemical compound. Occasionally, in order to improve handling or performance characteristics, two separate compounds, each with its own oxidizer and fuel, are mixed together. These are called double-based propellants.

A solid propellant will not react below a certain temperature and pressure, which are usually well above those of the normal environment. This characteristic makes the solid propellant relatively safe to handle. Additives to improve handling characteristics and performance may be used. Aluminum and beryllium are examples of additives for increasing the performance of a given propellant.

Solid propellants produce thrust in the same manner as liquid propellants, that is, the thrust obtained is proportional to the mass, velocity, and pressure of exhaust products at the nozzle exit. In liquid-propellant engines the mass of propellants consumed per unit of time can be held constant either by controlling the propellant pumps or with propellant valves. With solid-propellant engines, special configurations of the propellant grain are used to control the burning area and thus the mass of propellant consumed and exhausted through the nozzle. Certain surfaces of the propellant grain may also be coated with an inhibitor to restrict burning to only the area desired. Figure 4 illustrates three typical solid-propellant grain configurations and their effects on combustion and thrust.

The mass rate of a solid propellant consumed is determined by the amount of area and the rate of burning of the propellant under combustion. The burning rate is the velocity with which the flame front passes through the propellant in a direction normal to the burn-

Figure 4. Typical solid-propellant grain configurations. Black areas of the grain configurations indicate the burning surfaces.



ing surface. With progressive burning, the area of propellant under combustion increases with time. This increasing propellant consumption rate provides a continually increasing mass of exhaust products and a corresponding increase in thrust. In neutral burning, by special design of the core, the area under combustion is held relatively constant, and the thrust from this effect is constant. With regressive burning, combustion area continually decreases, with a resulting decrease in thrust.

rocket engines and propulsion systems

Certain relatively small changes in the design parameters of rocket-propelled vehicles may have a large effect on the final performance of the vehicle. The thrust-to-weight ratio of the engine is an example of this. In general the thrust-to-weight ratio of an engine is an indicator of just how good the engine is and is somewhat comparable to the horsepower-per-pound ratio used in describing conventional reciprocating aircraft engines. Increasing the thrust-to-weight ratio of an engine system having a given thrust may be accomplished by decreasing the weight of that system. Advantage can be taken of the savings in weight in several ways. Additional payload can be placed into orbit or additional propellants may be carried to give increased performance.

Another indicator of engine performance effectiveness is the specific impulse delivered by the engine of the propulsion system. The specific impulse may be rather loosely compared to the conventional engine performance measurement of miles per gallon. In the rocket engine as with the conventional engine, it is not just the propellant or the engine that counts; rather it is the combination of the two. Two different engines using the same propellant might well deliver differing specific impulses. The propulsion system with the higher specific impulse would be the more efficient of the two in terms of propellant usage. The high I_{sp} alone, however, would not be the deciding factor in the selection of the propulsion system, since other considerations, such as the thrust-to-weight ratio of the engine or the total thrust, may not be suitable for the mission requirements. Certain advanced propulsion-system concepts which promise very high theoretical specific impulses* will not be suitable for application where high thrusts are required.

The thrust-to-weight ratio of the entire vehicle is another important parameter to be considered in selecting the propulsion system. It is immediately obvious that the thrust-to-weight ratio of the entire vehicle at launch must be greater than unity in order for the vehicle to leave the launch pad. That is, the thrust of the engines in pounds must be greater than the total weight of the system. Simply stated, the acceleration of the system at any instant is dependent on the amount by which the thrust of the booster exceeds the sum of the weight of the vehicle and the aerodynamic drag.**

Since the major part of the powered flight of the vehicle occurs in the atmosphere, it is subjected to velocity losses from aerodynamic drag. These losses are proportional to the density of the atmosphere and the square of the velocity. From the standpoint of minimizing drag, it is fortunate that the velocities in the more dense atmosphere near the surface of the earth are usually relatively low. As the velocities increase at the higher altitudes, the effects of drag become less because of the decreasing densities. Drag losses are reduced by streamlining and by reduction of cross-sectional area. By selection of a combination of high bulk density and high I_{sp} propellants, cross-sectional area and thus drag can be minimized. The reduction in propellant volume by use of a high-bulk-density propellant also results in smaller

*Electric propulsion systems promise specific impulses for the future ranging from 2500 seconds with ion engines to 30,000 seconds for MHD (magnetohydrodynamic) engines, yet the thrust may be only in fractions of pounds.

**From the basic relationship, $F = ma = \frac{W}{g} \times a$, the acceleration of a mass at any instant is $a = \frac{F \times g}{W}$ where a is the acceleration in ft/sec², F is the sum of the instantaneous forces acting on the mass, W is the instantaneous weight of the vehicle in pounds, and g is the gravitational acceleration. The forces (F) acting on the mass are the thrust (T), gravity and atmospheric drag (D). By substitution, the acceleration for a vertically launched mass may be developed as follows:

$$a = \left[\frac{T - (W + D)}{W}\right] g \quad \text{or} \quad a = \left(\frac{T - D}{W} - 1\right)$$

When drag is zero, as at lift-off, the instantaneous acceleration in g's is equal to the thrust-to-weight ratio minus one.

I_{sp} (sec) / F (lbs)	250		300		350	
	Acceleration (g)	Final Velocity ft/sec	Acceleration (g)	Final Velocity ft/sec	Acceleration (g)	Final Velocity ft/sec
100,000	.10	11,210	.10	14,500	.10	17,000
125,000	.39	11,210	.39	14,500	.39	17,000
150,000	.67	11,210	.67	14,500	.67	17,000

Initial weight of vehicle (w_1) = 90,000 lbs

Weight of vehicle at burnout (w_2) = 20,000 lbs

$$\text{Mass ratio} = \frac{w_1}{w_2} = 4.5$$

Figure 5. Lift-off acceleration and final velocities (ideal)

structure and a corresponding reduction in weight.

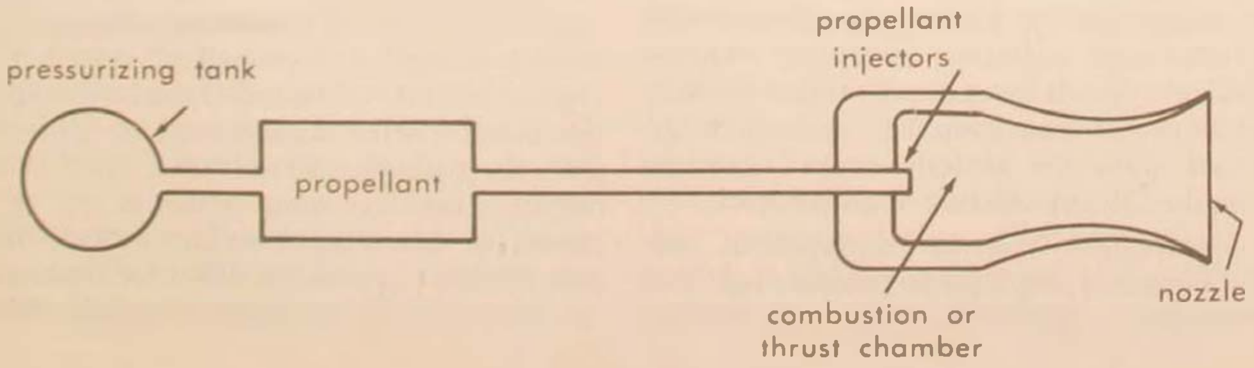
Figure 5 shows the effect of varying the I_{sp} and thrust on lift-off acceleration and ideal final velocity.* The example used is that of a hypothetical single-stage rocket-propelled vehicle. It is interesting to note that acceleration of the vehicle is a function of the thrust-to-weight ratio of the system and is independent of the specific impulse. Conversely the final velocity achieved is independent of the thrust (assuming no losses from drag or gravity) but rather is dependent on the value of the specific impulse.

*The ideal final velocity is the velocity that would be achieved if there were no losses from the effects of gravity and atmospheric drag. Actual velocity losses from these effects in a typical system may range from 3000 to 5000 feet per second.

liquid-rocket engine operation

Monopropellant Systems. Figure 6 is a schematic representation of a pressure-fed monopropellant rocket engine. Simplified for clarity, it shows the main propellant tank with the premixed solution of fuel and oxidizer and a second, usually very small, pressurizing tank in which a high gas pressure is accumulated to operate the system. Pressure from the pressurizing tank forces the propellant through the system for injection into the combustion or thrust chamber. Since the fuel and oxidizer

Figure 6. Pressure-fed monopropellant system



have been previously combined in the proper proportions in the main propellant tank, mixing at the injector is not required. The propellant is ignited in the combustion chamber, and the exhaust products are forced rearward out the nozzle, providing the desired thrust.

Mass flow rate of propellant and thus the level of thrust delivered by the engine may be controlled by a valve located between the injector and the propellant tank. One disadvantage, which is applicable to all pressure-fed systems, is the requirement to provide relatively heavy propellant tanks. These heavy tanks are needed to withstand the high pressures necessary to force large volumes of propellant through the system. This extra weight results in reduced payload. On the other hand, pressure-fed monopropellant systems have the advantage of relative simplicity. The requirement to mix the fuel and oxidizer in a precise ratio at the injector is eliminated. The absence of turbopumps for the fuel supply gives added simplicity and increased reliability. Monopropellant systems are generally thrust-limited, but because of their simplicity and reliability they are used for attitude control and for similar low-thrust mission requirements.

Bipropellant Systems. Bipropellant systems (Figure 7) may also be pressure fed. The advantage of simplicity is also applicable, although the weight penalty of sturdier tank design is still applicable. Bipropellant systems, both pump and pressure fed, introduce a complication that is not a factor in the monopropellant rocket engine. To make sure that the combustion chamber will be provided with large quantities of fuel and oxidizer (one or both of which may be cryogenic and corrosive) in differing amounts, in a very precise ratio, with complete mixing, requires complex analyses and design. Variation in the mixture ratio or incomplete mixing causes changes in combustion temperatures and pressures which in turn may result in changes in thrust and engine efficiency. The rate of propellant flow is controlled by adjusting the pressure in the tanks or by manipulation of flow valves in the propellant feed lines.

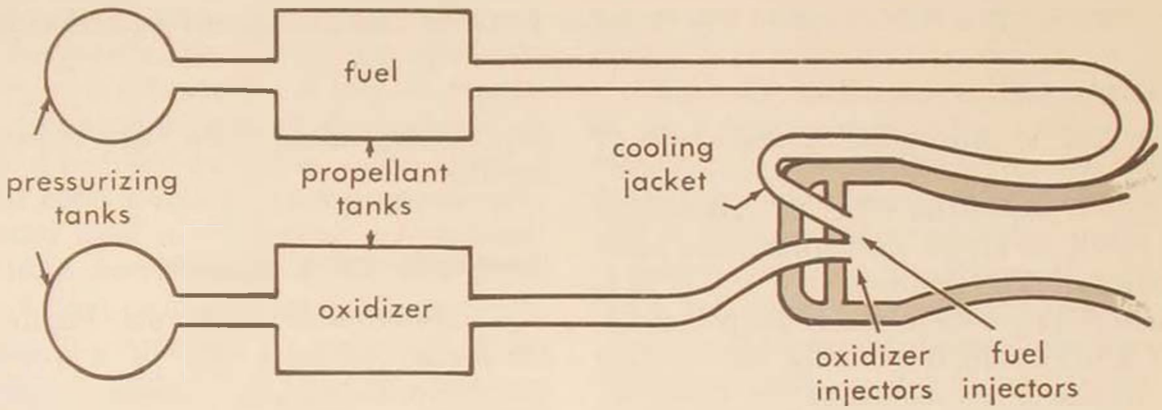


Figure 7. Pressure-fed bipropellant system

Pump-Fed Systems. Most high-performance space propulsion systems today are pump-fed bipropellant systems. Figure 8 illustrates a greatly simplified functional schematic of a typical pump-fed bipropellant system. This engine system is often called the bootstrap system because of the manner in which the turbine from the turbopump feeds itself from its own output. There are many schemes and refinements for this type of engine system, but in general they may be simplified for explanation as shown. The gas generator uses fuel and oxidizer from the main tanks to provide the gases to operate the turbine. The mixture ratio required is usually different from that of the main rocket engine. As the pumps begin to supply propellant to the engine, an increased supply is also furnished the gas generator for the turbine. When an adequate supply of propellant is introduced into the thrust chamber, the propellant is ignited. In high-thrust systems the large mass rate of flow of propellants requires high pump impeller velocities. Cavitation, with a loss of propellant flow, may result. simple instance of the type of engineering challenges that have been encountered in designing turbopump systems is the stress analyses of the shaft required for such a pump. The gas turbine attached to one end of a relatively short shaft may be operating at a temperature of many thousands of degrees Fahrenheit while at the other end the impeller may be immersed in propellants with temperatures less than −400°F. In some rocket engine system designs, cost of the propellant pumping system has been the largest single development item.

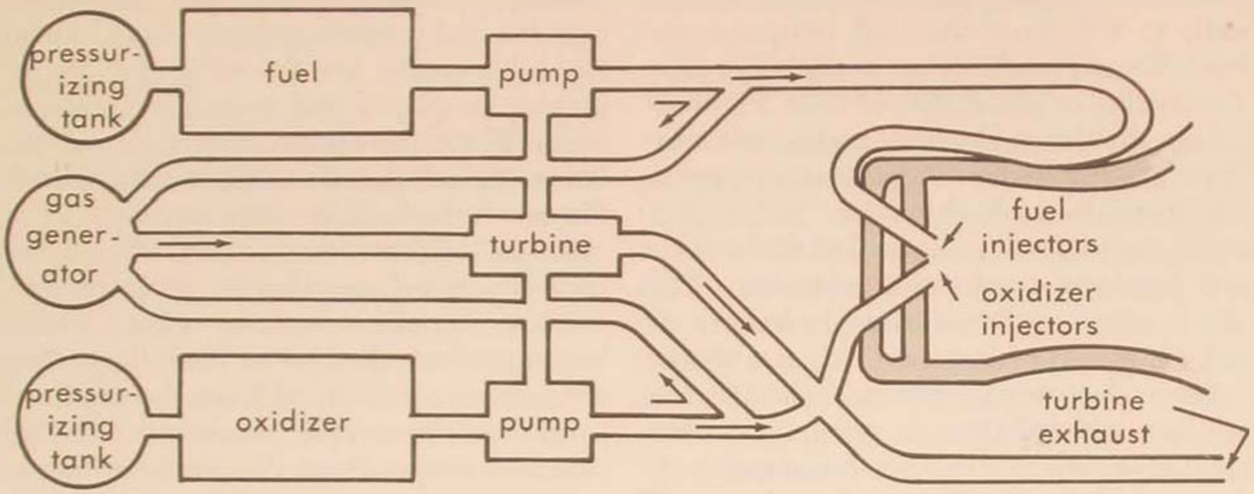


Figure 8. Schematic of pump-fed liquid bipropellant (bootstrap) system

To counteract this undesirable effect, positive pressure is usually maintained in the main tanks. One method of providing this pressure is by the use of pressurizing tanks, as shown in the illustrations. Pressures necessary to prevent cavitation and the increased structural weight required for the propellant tanks are not as high as those for a purely pressure-fed system.

Pump-fed systems are able to deliver propellant to the thrust chamber at greater flow rates than comparably sized pressure-fed systems. This greater mass flow rate produces greater thrust. On the other hand, pump-fed systems are generally not as reliable as pressure-fed systems. The requirement to pump large quantities of propellants with varying densities at precise rates is difficult to meet. A

Velocity Adjustment. In order to achieve a precise orbit, the space vehicle must be injected into its orbit path with a very precise final velocity. The guidance system furnishes a very accurate signal to close the propellant valves and shut down the propulsion system; however, because of residual propellant in the lines, burnout is not instantaneous and some "tailing off" is experienced. It is not possible to determine the amount of tailing off that will occur in each case, and thus the exact final velocity is difficult to predict. To correct for this error in velocity, engine burnout may be adjusted to ensure that the velocity at burnout is slightly less than that desired. The final velocity increment is then added by small vernier engines. Although "tailing off" and small inaccuracies also occur in vernier engine shutdown, the thrust of the vernier engine itself is small, so these variations and the error in the velocity increment are correspondingly small and are held to an acceptable level.

engine cooling

Because of the quantity of heat generated in the thrust chamber and exhausted from the nozzle during operation of the propulsion system, great emphasis is placed on the materials from which the engine components are constructed. The efficiency and output of rocket engines may be limited by the inability of certain materials to withstand the high temperatures involved. Several methods are available to provide for cooling or dissipation of heat. Figure 8 shows how the thrust chamber and nozzle may be cooled regeneratively. Propellant is pumped directly from the tank through a jacket that surrounds the surfaces to be cooled and is then injected into the combustion chamber. The propellant, often cryogenic, absorbs heat from the high-temperature surfaces, and, as a bonus, heat energy that would otherwise have been wasted through radiation is returned to the combustion chamber. The heated surfaces may also be cooled by introducing liquid around the perimeter of the combustion chamber. This liquid, which may be either the fuel or oxidizer, forms a thin film on the walls and provides enough heat absorption to give the cooling required. Transpiration cooling is accomplished by constructing the surfaces to be cooled from a porous material. A fluid, again usually the fuel or oxidizer, is forced through the walls into the combustion chamber. The walls of the chamber "sweat," so to speak, and provide the cooling. For test-stand operations, water may be used for cooling in place of the propellant, in a system similar to the regenerative cooling system. Rocket engines may also be cooled by radiating the heat into space. This method is best used for upper-stage and space applications.

solid-rocket engine operation

With respect to simplicity, solid-propellant rocket engine systems have a clear advantage over liquid-propellant systems. The absence of plumbing, pumps, pressurization, cryogenics, and propellant injection and mixing in the solid system are the major factors in this greater simplicity and generally greater reliability. The basic components of a solid-propellant engine system are shown schematically in Figure 9. Note that since combustion takes place in the port area, a thrust or combustion chamber is not specifically designed into the system. Propellant burning takes place when hot gases from the igniter blow down the port area and are exhausted from the nozzle. Solid propellants burn under conditions of high pressure and temperature; therefore, as long as the required temperature and pressure are maintained, burning is sustained and thrust is produced. When the vehicle reaches the velocity desired for that stage, thrust is terminated by "blowing off" thrust-termination ports, and thus pressures are reduced below that required for combustion. With the port opening oriented forward, rearward thrust from escaping gases assists in separation of the stage. Without liquids available for regenerative cooling, solid propellants must rely on radiation, thermal ca-

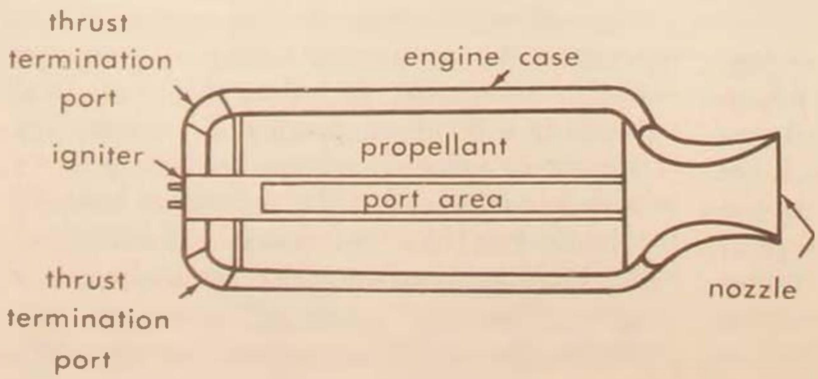


Figure 9. Typical solid-rocket engine

pacitance, or insulation for dissipation of heat.

Whereas liquid-propellant propulsion systems can be designed for repeated starting and stopping by controlling the flow of propellants, solid-propellant systems are a one-time operation. Proposed hybrid propulsion systems, utilizing both liquid and solid propellants, would combine some of the advantages of both liquid and solid propulsion systems. In the hybrid system, one of the propellants is liquid and the other is solid. As an example, the solid propellant may be the fuel, and the liquid propellant the oxidizer. Control of the engine is achieved by metering the supply of the liquid oxidizer.

thrust vector control

For many obvious reasons directional control of the space vehicle during the powered portion of flight is required. The vehicle must be placed on the proper path to achieve the desired trajectory or orbit. Perturbations in the atmosphere or slight deficiencies in the propulsion system output will necessitate corrections. In some earlier space systems and in some current high-acceleration rocket-boosted missiles, control of the vehicle during the powered portion of the flight is achieved aerodynamically by the use of controllable fins. As long as the vehicle remains in the sensible atmosphere, this method is acceptable. For longer-duration powered flight with slow velocity buildup and correspondingly low aerodynamic forces and for missions that extend into the vacuum of space, thrust vector control is more desirable than aerodynamic control. Control is obtained by deflecting the thrust vector of the main rocket engine from the longitudinal axis of the vehicle. The small transverse component of thrust produces the desired change in missile attitude.

Several methods of providing this thrust vector control are in use or are under development. The method most frequently used for thrust vector control of liquid-propellant engines is flexible mounting of the engine. By gimbaling the entire engine, the thrust is deflected in the direction that the missile is to go. Since the entire stage in a solid-propellant engine is an integral unit and there is no engine as such, gimbaling of the stage is impractical. Thrust vector in solid-propellant engines is controlled at the nozzle. The most common method is swiveling the nozzle. Controllable tabs placed in the nozzle exhaust have also been used. Injection of liquid or gases into the nozzle produces disturbance in the exhaust and provides some control.

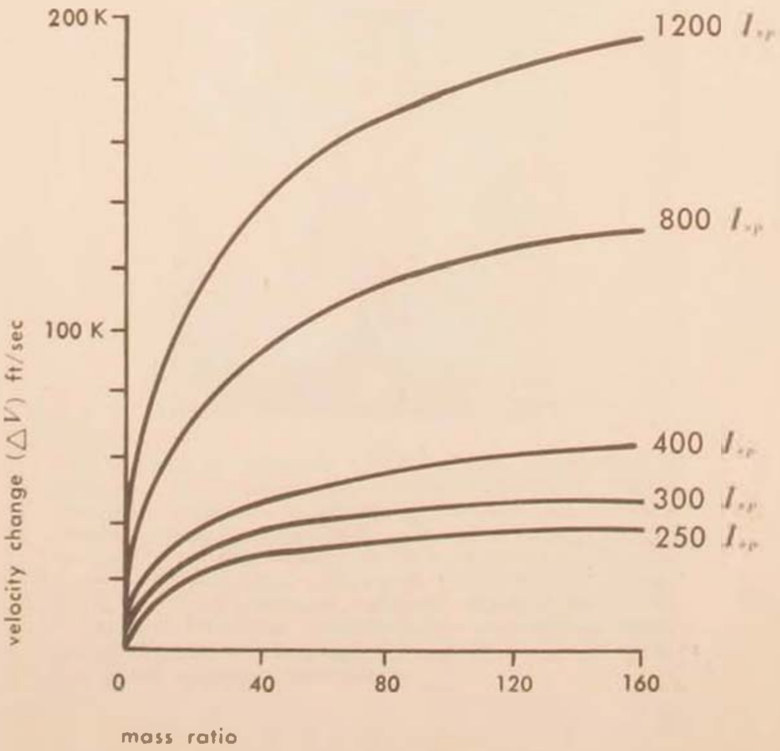
From the standpoint of engine efficiency, the gimbaled engine or movable nozzle is the most desirable method. Those methods of thrust vector control that produce disturbance in the flow of gases from the nozzle cause some loss in efficiency of the engine.

future propulsion requirements

New concepts for high-performance propulsion systems hold great promise for the future. The development of technology and materials for future missions is absolutely necessary as we near the theoretical upper limit of our chemical systems. By continual exploitation of every avenue open to improvement of present chemical systems, we are slowly but surely edging our way to that theoretical limit.

Figure 10 helps to illustrate one reason for fostering interest in the development of pro-

Figure 10. Velocity change versus mass ratio



pulsion systems with high specific impulses. Using velocity change as a measurement of mission performance, one can make a few comparisons. Note from Figure 10 that with a mass ratio in the region of 40 to 60 and with a specific impulse of from 250 to 400 seconds, relatively little ΔV is gained by increasing the mass ratio, which is almost at its practical upper limit already. On the other hand, note the increased ΔV available with mass ratios in the same region but with specific impulses of 800 to 1200 seconds and beyond.* Future missions involving large orbit changes and distant space voyages will require velocity changes available only through high I_{sp} propulsion systems. These advanced systems will operate under the same basic laws of physics as contemporary systems, but they will require different concepts, energy sources, and types of equipment.

Hq Air Force Systems Command

*Other considerations, such as thrust delivered, supporting equipment required, and type of mission involved, can affect the conclusions reached in the use of the graph in Figure 10; however, the basic data are still valid.

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Roll Call

UMPIRING AND EVALUATING JOINT EXERCISES...LESSONS OF LEBANON...THE JAPAN AIR SELF DEFENSE FORCE

JULY-AUGUST 1965