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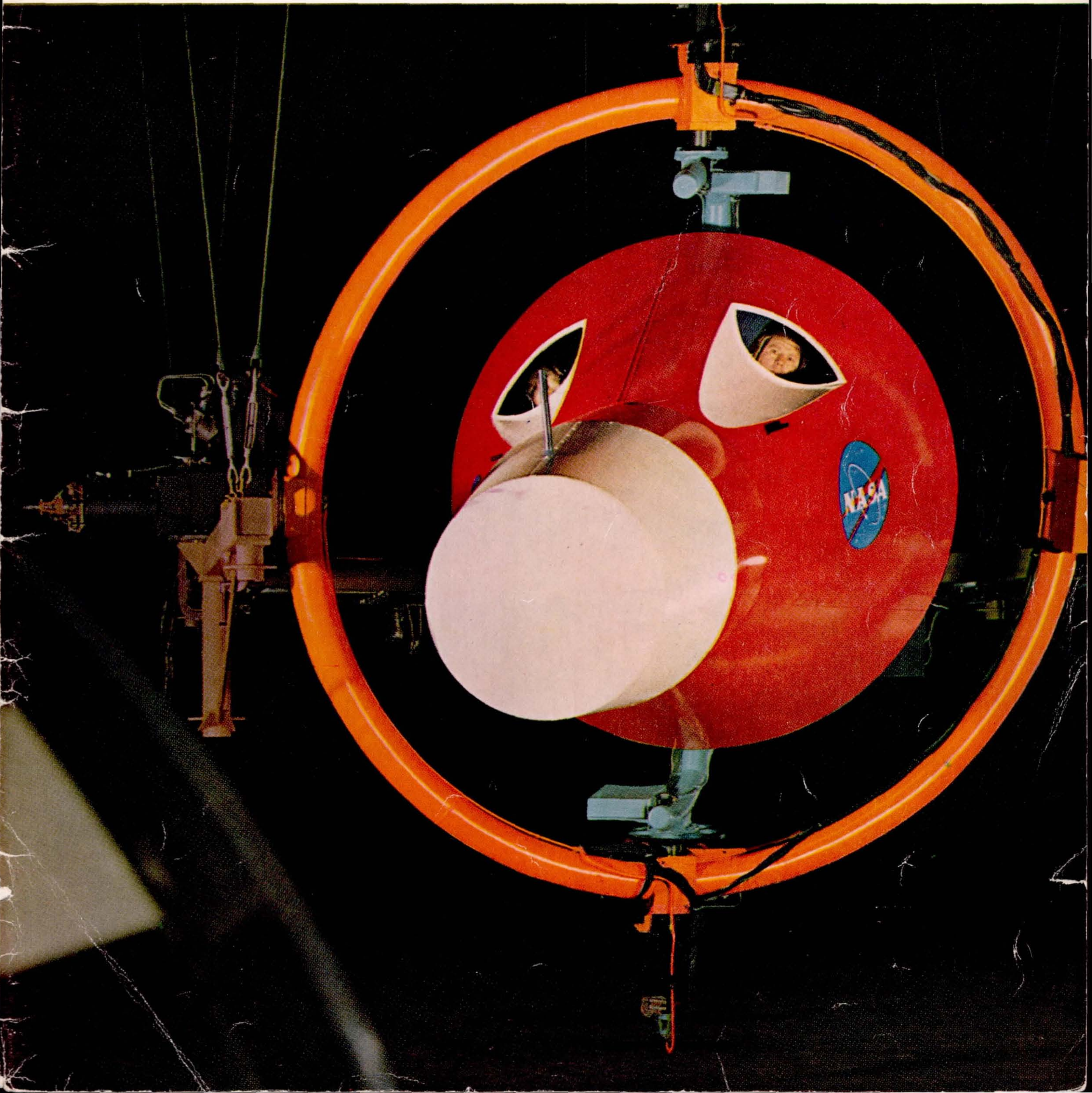


Special section on
MANNED ORBITAL OPERATIONS

Space progress, 1962

BY ROBERT R. GILRUTH

It is an honor to provide the editorial comments for this first issue of *Astronautics and aerospace engineering*. I believe that the merger of ARS and IAS exemplifies a trend of progress that is very prominent in our fast-moving space age. Working together to achieve common aims was one premise on which our nation began, and still operates. The progress of our space program during the past four years has been strongly influenced by the exceptional cooperation of the Federal civilian-military-industry team.

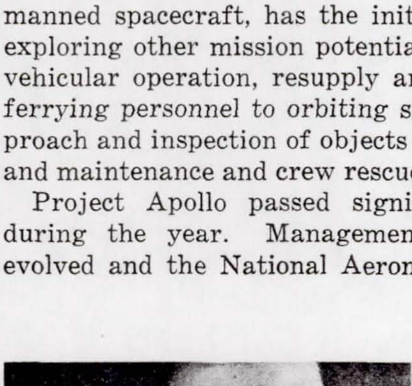
During the past year, we have seen an excellent example of this progress, the three successful manned orbital flights that met the original objectives of Project Mercury. We will soon take a more ambitious step into space, a one-day manned mission with a modified Mercury spacecraft. This mission will provide an opportunity to examine further man's physiological responses to weightlessness and will provide experience concerning the adequacy and reliability of our basic Mercury systems during more prolonged periods in the space environment.

During the past year, moreover, Project Gemini became an approved program. This project is the necessary step that bridges the "relatively simple" Mercury missions and the highly complex Apollo lunar program. Project Gemini will afford an opportunity to explore the problems associated with more prolonged weightlessness and to gain the necessary experience in new techniques such as rendezvous and docking. Without the benefits gained from this program, the technological jump would be difficult, if not impossible. In addition, Gemini, being a second-generation manned spacecraft, has the initial capability for exploring other mission potentials, such as extravehicular operation, resupply and crew transfer, ferrying personnel to orbiting space stations, approach and inspection of objects orbiting in space, and maintenance and crew rescue.

Project Apollo passed significant milestones during the year. Management concepts were evolved and the National Aeronautics and Space Administration built an organization to implement the many facets of the program. Its Office of Manned Space Flight in Washington, D.C., is responsible for planning and controlling the total program and for integrating its various phases. The Marshall Space Flight Center has responsibility for the launch vehicle and for managing both the Michoud assembly plant and the Mississippi Test Facility. The Launch Operations Center at Cape Canaveral is responsible for providing the facilities at the Cape and for the launch of the vehicle. The Manned Spacecraft Center provides the manned spacecraft, and the mission-planning and flight-control aspects of the program. The past year saw prime contracts awarded for the various elements of the spacecraft. Decisions have been made and are being implemented with respect to use of the three-stage Saturn C-5 as the launch vehicle, the use and expansion of the Cape Canaveral area as the launch site, and the use of lunar-orbit rendezvous as the mode of operation for lunar landings. Major emphasis is presently being placed on design, development, and construction of these vehicles and facilities.

With regard to future progress, NASA utilizes all of its Centers directly or indirectly in working toward long-range goals of spaceflight. The various Centers must make contributions in the areas of research and studies on propulsion systems, orbital rendezvous, lunar landing, flight control, navigation and guidance, entry and environmental physics, and space biology. These efforts, however, constitute only part of what will be needed to achieve the goals of the manned spaceflight program.

Other organizations must produce flight hardware, test facilities, and general support. The Department of Defense, the Weather Bureau, the Atomic Energy Commission, and other federal agencies have directly supported the program, for example, as in Project Mercury. There the Air Force provided the launch vehicles and acted as NASA representative to the launch-vehicle contractors. This same procedure still exists for



ROBERT R. GILRUTH directs the NASA Manned Spacecraft Center at Houston, Tex., and previously headed the NASA Space Task Group for Project Mercury. His long and distinguished career with NACA and NASA began in 1937 at Langley Memorial Aeronautical Laboratory. During the past year, he has received the ARS Robert H. Goddard Award, the IAS Hill Transportation Award, and an honorary D.Sc. degree from Indiana Technical College. Dr. Gilruth has served as Guest Editor for this Manned Orbital Operations Section.

Project Gemini, a program that will use three different propulsion vehicles. DOD provided medical monitoring support during Mercury flights and assisted in analyzing the data obtained, provided facilities for use in the selection and training of the astronauts, and provided the recovery forces. There are still other contributions too numerous to mention; but, without this cooperation, Mercury could not have been implemented, and the other manned space programs would not have advanced as far as they have.

Industry provides another area of direct support, and the amount of this support is as large and varied as the number of contractors contributing. Besides the prime contracts and subcontracts for furnishing the space-vehicle hardware and supporting equipment, there are many minor contracts with universities and small companies for services, associated equipment, studies, and testing.

It takes a powerful common effort, welding these national resources, to make a program like Mercury a success. The programs of the future will be more ambitious. I am speaking not only of the numbers of people or cost, although these will grow greatly for a few years before the trend begins to stabilize. I am more concerned with the general attitude toward manned spaceflight.

There will be massive research, engineering, and development efforts necessary for advanced programs, such as the design and development of vehicles capable of interplanetary travel; for knowledge of the universe can be obtained only by using space vehicles of considerably increased capability. The design and development of advanced systems will aid in achieving this increased capability, but probably an equally important and influencing factor will be the talent to employ intelligent and decisive methods of keeping pace with the momentum of our technology. We must have aggressive planning and thinking to take full advantage of the invention, research, and development that will effect our over-all plan. Industry and government agencies must take the initiative in their respective fields to push the state of the art to the limits.

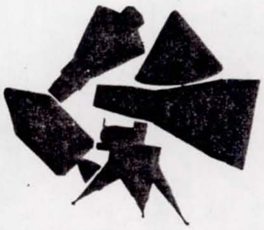
Although we can see over the horizon toward subsequent programs (and we must possess the capability to look far ahead), we must approach programs through logical steps. Each step has its own milestones that must be traversed to reveal clearly how we can reach our goals in space.

Then, too, we must be constantly aware of reliability and quality-control requirements in the advancement of space technology. The first consideration is simplicity of design and the budgeting of element reliabilities from an over-all numerical value. The responsibility must rest on the spacecraft designer, who budgets to the subsystems the requirements on the degree of redundancy and other measures for improving reliability. Second, the confidence of randomly selecting samples for qualification purposes cannot be justified unless all supposedly identical parts from the assembly are truly identical in all essential features. To achieve a degree of control, all components requiring certification through qualification should be made up from sets of parts whose members have been produced consecutively on the same assembly line without an intervening change in design, process, or materials. Third, a strict control on the identification and use of parts is necessary to insure that all suspect parts can be readily located, should a need arise to remove and replace any that reveal a deficiency.

In the area of inspection, flight-safety considerations and the limited number of articles involved in our programs make it reasonable to require 100% inspection of all items. This selection process should help insure that defective and marginal items are found and rejected. When an equipment malfunction does occur, failure analysis and decisions for corrective action must take place immediately. This can often be done best at the scene of the failure, where the availability of the part, the test apparatus, and the people involved in the test, provide the best opportunity for accurate determination of the pertinent facts.

We have just mentioned people; and, in the final analysis, we will be concerned fundamentally with people—the many thousands of people who will be involved in the national space program. Each and every one of them must have that pride of workmanship which will assure quality. Achievement of true reliability will demand the individual who will never overlook or ignore, but rather will recognize the slightest sign of trouble, and who will freely give the last bit of extra effort that so often spells the difference between success and failure.

Our progress in space will be the result of a large team rising to meet the most challenging assignment ever given to the American scientific, engineering, and industrial community. Every individual involved in this vital national program must give his best in a united effort to achieve our space goals. ••



From Mercury to Mars

BY MAXIME A. FAGET AND PAUL E. PURSER

The cumulative technology of Mercury, Gemini, Apollo, and space-station operations will establish a sound base for manned interplanetary flight

The NASA manned spaceflight program has the basic aim of exploring as much of the universe as practical, using man as a sensor, computer, and decision-maker to improve exploration. In existence since the establishment of NASA in 1958, this program now includes as approved developments the Mercury one-man spacecraft, the Gemini two-man rendezvous spacecraft, and the Apollo lunar-landing mission.

Under study in the various NASA centers are more advanced missions, such as the orbiting space station, a lunar base, and interplanetary manned missions. Although not yet approved programs, these advanced studies undertake to provide the basic technology for extending our efforts when advisable.

How can these various projects provide orderly progress in the goal of solar-system exploration? This is the question we would like to discuss, first, in terms of general comments on the development process.

The over-all planning of a total manned spaceflight program should be based on a logical sequence of steps. The planning of each individual project within such a total program should be similarly based. Individual projects should normally be established only to cover the greatest reasonable advance in capabilities that seems feasible within the state of the art at any given time. Future steps within the total program should be planned to take advantage of foreseen progress in the state of the art and individual projects should be so planned as to allow the insertion or use of unforeseen real advances or breakthroughs. Yet the goals of each phase of the program should be rather firmly established beforehand and care must be exercised to avoid delays resulting from continual changes brought about by the insertion of apparent or less-consequential advances. The proverbial wisdom of the ages would be required to completely avoid this paradoxical situation of planning for advances but yet not letting changes introduced by the advances result in undue delays.

The successful demonstration of man's capabilities in space and the advances made in spacecraft and launch-vehicle technology open a broad vista of possible manned spaceflight programs. Yet the national economy and technical resources cannot conceivably support all the possible programs. For this reason, the attack on the space frontier must be pointed and deep, rather than broad. Each succeeding program must be planned not as an end in itself, but as both a useful mission and as a stepping-stone in technology leading to the next program.

Many of us would agree scheduling such far-reaching programs needs improvement over past practice. It has been traditional, in this country at least, for planners to overestimate progress during the immediate future and to underestimate progress for the more distant future. The graph appearing on page 27 illustrates this point. The estimate made at $T_{(0)}$ will likely be optimistic for short times, because of a tendency to set tight schedules, based on the assumption that every system will work as planned. Setting tight schedules keeps each element of the system moving ahead at its maximum pace; and, for those systems which do not exhibit developmental troubles, allows the introduction of advances in the state of the art.

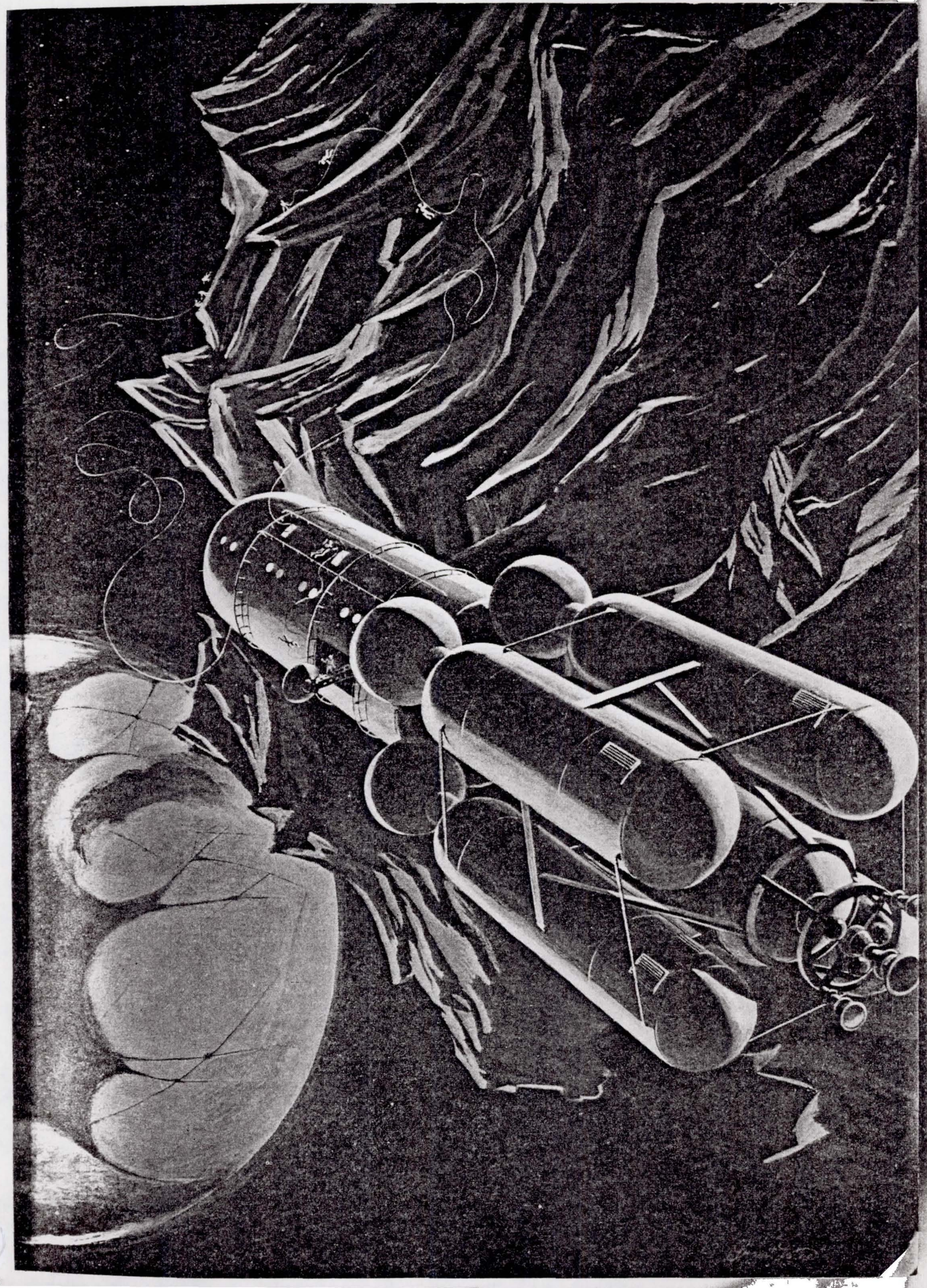
Some systems are prone to developmental troubles. The tight schedule is not met and progress in the near future is less than estimated. The increased progress in the far future results both from the advances introduced as just stated and from the normally cautious approach of planning on step-by-step progress and not purposely counting on breakthroughs occurring.

To get the best progress, we think planners of future programs and projects should:

1. Plan pointed, specific, and orderly programs that provide useful short-term knowledge and lead logically into the next longer term step.

2. Insure that estimates of near-future progress are not lowered.

RENDEZVOUS WITH PHOBOS, illustrated by MSC artist Grant Lathe—a goal for manned spacecraft development.



3. Be prepared to make less progress than estimated in the early phases of a program.

4. Be prepared to encounter otherwise unforeseen advances which will speed progress in later stages of a program.

5. Plan the approach to new programs so that, although the goals are firm and not subject to continual change, the plans are flexible enough to be modified to accept breakthroughs and advances in the state of the art.

The present manned spaceflight program—Mercury, Gemini, and Apollo—each prosecuted to a successful termination, will develop a fairly strong technical base for planning future projects.

The initial experience of manned spaceflight has been obtained in Mercury. This experience is not only applicable to flight and ground-operations crews but is also important in all phases of design engineering and management. Since Mercury is a simple spacecraft system, this experience will be greatly broadened in Gemini and then in Apollo.

Gemini will provide the first attempts at maneuvering in space in which the magnitude and direction of the velocity changes made will be computed during the flight in response to the situation created during the mission. Similarly, the capability will be developed to land at a predetermined point by guiding the spacecraft in re-entry and descent attitudes. Gemini will also allow longer flights and more complex experiments.

Apollo will give the first deep-space navigation experience. In many respects, Apollo will also have the first real mission. Its crew will be transported to the moon, and experiment, explore, and gather samples there. In Mercury and Gemini, except for incidental experiments, the mission objective is to learn about spaceflight. The Apollo crew is expected to guide the craft down to the surface of a hostile world only vaguely understood in detail and extremely difficult to reach. This requirement is challenging our technology and is also stimulating the growth in this technology needed for our future projects. When Project Apollo achieves its initial goal, our technology will have attained greatly increased capability in launch vehicles, high-energy propulsion systems, deep-throttling rocket engines, guidance and navigation equipment of high accuracy and reliability, and great increases in propulsion system reliability, streamlined launch procedures, a greatly improved and expanded deep-space network, and many other such attainments.

Future projects in the manned spaceflight program must be considered in view of our present efforts. Developments in the progress of both spacecraft and launch vehicles will represent significant increases in capability. As we have said, these improvements will be obtained only through large investments in money and manpower, and for this reason future projects must both complement these efforts and represent in themselves significant improvements in desirable capabilities.

Presently, three advanced projects are receiving serious consideration—the orbital space station, the lunar base, and the interplanetary spaceship. Both the orbital space station and the lunar base are achievable within the capabilities of the advanced Saturn launch vehicle, and in this sense will profit from developments of the present program. The lunar base is not quite as clearly defined as the orbital station (see page 52). A better assessment of this project will result when more is learned about the character of the moon from the Ranger and Surveyor projects as well as from Apollo.

Research has been in progress on multimanned orbiting space stations during the last two years both within NASA and by contractors.

The space station will allow scientific studies of meteoroids (density, velocity, size, and direction of flight); studies of the complete spectrum of space radiation; and astronomical observation at visible, ultraviolet, infrared, and radio frequencies; detailed investigations of weather patterns on earth and of the solar- and earth-radiated heat balance as affected by cloud cover and as it affects cloud cover; and detailed studies of the earth's geography, defining much better the relative location of many geographic features on the earth's surface.

The space station or the lunar base will put investigations of materials, structural systems, electrical power systems, communications, etc., in the real space environment, and so will eliminate the need for simulating or partially simulating these environments on the earth's surface. They will allow engineers to study systems in space over long periods of time under more adequately controlled conditions of both observation and measurement. For instance, a large multimanned orbiting space station will allow long-term research into maintenance-free communications satellites.

The large size and weight and extremely long-duration missions of the manned space station, moreover, will permit investigation in the real space environment of many specific systems—environmental control, electric power, propulsion, communication, navigation and guidance, etc.—required for manned planetary missions.

This cumulative technology should indeed establish a sound basis for manned interplanetary flight and all this foresees.

The first interplanetary spaceship project will probably be designed for an initial exploration of Mars; and the next generation of space hardware will be clearly represented in the exploration of this planet. Virtually every aspect of the system will benefit from and perhaps be dependent on yet-to-be-developed improvements in technology. Some of these might be revolutionary departures from the past, such as nuclear propulsion.

It is obviously premature to at-

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tempt to conceive hardware designs of interplanetary spaceships in any significant detail. Yet it is important from time to time to try to visualize future mission requirements so that today's research and development efforts may perhaps be better focused. Let us discuss some considerations which will shape the design of this mission and which should influence research and development efforts in the interim.

Mars has an orbital period of roughly 1.88 terrestrial years. Opposition presents the natural time for missions to be flown, and these occur at intervals of slightly over two years. Time and distance for the next several oppositions are as follows:

DATE	DISTANCE (mi.)
Feb. 3, 1963	61,700,000
Mar. 8, 1965	61,700,000
Apr. 13, 1967	56,200,000
May 29, 1969	45,300,000
Aug. 6, 1971	34,600,000
Oct. 21, 1973	40,600,000
Dec. 13, 1975	53,100,000

Eccentricity of the planetary orbits (primarily Mars) causes opposition distance to vary, and a 2-deg difference in the orbital planes of earth and Mars results in a variation of the energy (velocity) required to make the mission at each opposition. Low-energy transfers require approximately a half year each way, depending on the distance at opposition.

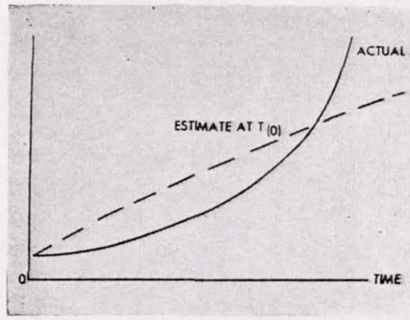
Then the nature of the mission—fly-by, orbital reconnaissance, rendezvous with a Martian moon, or planetary landing—naturally affects total energy requirement.

The fly-by mission will demand the least energy but will also have the least scientific value. It will give the crew an opportunity for a close-up observation of Mars. This mission may be feasible within Saturn launch-vehicle capability, especially if earth rendezvous were to be employed. Assuming only minor velocity adjustments would be made during the mission, the proximity of the fly-by to Mars would depend on the velocity in the vicinity of Mars and the amount the spacecraft path was to be deflected. For this reason, the proximity will be totally dependent on the manner in which the whole mission is planned. The shorter the total mission (higher energy), the closer will be the fly-by path. This is a necessary feature only of those missions in which it is desired that the spacecraft return to earth without a major propulsive thrust in the vicinity of Mars.

In addition to providing the crew with a close-up optical observation of Mars, a fly-by mission could take advantage of other means of probing for scientific information. Detailed surface-temperature measurements could be made with bolometers. Spectrographic analysis of the atmosphere could be obtained by observing the entry wakes of probes. Similarly, bombs might be used to analyze surface constituents. Other probes might be soft-landed on the surface for more sophisticated investigations. All these many measurements, however, would have to be made during a short period when the crew would also be concerned with the most exciting exercises of the spacecraft's navigation. Furthermore, these measurements are in many ways no better than those which might be obtained with a properly operating, rather sophisticated, unmanned probe.

Mars orbital reconnaissance would differ from the fly-by mission by using rocket power in the vicinity of the planet first to enter into and then depart from an orbit about it. The use of propulsion in the vicinity of Mars releases the mission from the proximity restraints which characterize the fly-by mission. Significant savings in the energy requirements could be achieved by using an eccentric orbit entered (and left) at perigee, rather than by using a circular orbit. This eccentric orbit would provide sufficient opportunity for close-up observation of Mars, as well as excursions through its magnetic field and radiation belts. In general, the same types of measurements would be made as those in the fly-by mission, except that there would be a much longer time for observation and an opportunity to approach much closer to the Martian surface.

Mars has two very small, apparently natural, satellites—Phobos and Deimos, conservatively estimated at 10 and 5 mi. in diam, respectively. Lacking significant gravity, these satellites can be landed on simply through orbital-rendezvous techniques. The satellites have orbits essentially in the plane of the Martian equator, and so inclined to the ecliptic by about 25 deg. Making this plane change in approach will require additional energy.



PROGRESS in an engineering development typically takes this form. Judging the future from t_o has historically been a difficult matter, full of pitfalls.

These two moons are in fairly low orbits, Phobos 3300 n. mi. above the surface and Deimos 11,000 n. mi. Besides being interesting in themselves, they should be ideal sites for long-duration research instrumentation set up to observe the planet throughout its seasons (assuming the near certainty that they have reached a fixed relative orientation to Mars through gravity-vector stabilization and magnetic damping).

The mission designed to land men on the surface of Mars will not only require the largest and most complex spacecraft system, but will also provide the greatest return in scientific data. This mission will require much more propulsive energy for velocity changes than those previously described, especially in any short-duration flight attempted. The Mars landing mission may have to await the development of nuclear propulsion to be considered practical.

A number of alternate mission schemes may be considered. The use of a separate landing module as will be used in Project Apollo is certainly an obvious contender. This landing module may be launched from either a close-in circular orbit or a highly eccentric orbit with a low perigee. The circular orbit would require the least performance from the excursion module, but would require more performance from the mother ship. Since the lander can use atmospheric braking for descent, this tradeoff would appear to favor the elliptical orbit from a performance standpoint. Operationally, however, the circular orbit would appear to be somewhat simpler.

In the Apollo mission analysis, clear-cut performance gains were shown for the rendezvous technique. Since Mars has an atmosphere, a new performance tradeoff study would be required for this condition to determine if the advantage would still remain.

Operationally, however, the direct-landing technique may be hard to justify. It would commit an extremely complicated and heavy spaceship to a landing on the far side of the planet without the benefit of updated reconnaissance. This vehicle would also have to make a successful hypervelocity atmospheric maneuver with the very awkward-to-carry propulsive capability necessary for return to earth.

The employment of earth-orbit rendezvous for assembly of the total mission capability will most likely be required. Reliability might be enhanced by use of a fleet of two or three vehicles, rather than a single spacecraft. The fleet approach may also be used to improve capability through resupply or refueling in transit. Perhaps the most significant manner in which in-transit rendezvous could be employed would be a pickup maneuver immediately after trans-earth injection. In this event, the pickup craft would trail the landing party by several weeks during the outbound journey. It would be guided along a fly-by trajectory and hence would require only modest propulsion capabilities. The spacecraft would leave the surface of Mars, on an orbit about Mars, at the proper time to rendezvous with the pickup vehicle on the homeward-bound leg. While this may be considered a high-risk operation, it may be favored as a scheme that would lie within practical chemical-rocket capability.

Many unknown environmental factors contribute to uncertainty in the design approach. It is hoped that more factual information on the environment can be obtained during the same period in which spacecraft technology is improving. Mission and system analysis can then be carried out with a minimum amount of guesswork when the time comes for the final design decisions.

More facts about the Martian weather need to be known. The velocity and direction of surface winds and the nature of any sandstorms are certainly important considerations. Apparently there are seasonal effects and, undoubtedly, there are variations with time of day and latitude. It might be mentioned that a special statistical study of wind and wave conditions in the Atlantic Ocean had to be made as part of the operational analysis that went into the landing system design for Mercury. Not as much will be known about Martian weather as about Atlantic Ocean weather. The result will undoubtedly be the use of design margins as a substitute for knowledge.

At this time, only conjectures can be made about magnetic fields and trapped radiation belts about Mars. However, a mission envisioning an extended period of orbiting the planet must include an estimate of the radiation dose. There is also a need for knowledge of the micrometeorite flux in the regions of space between Mars and earth. Although Mariner gave indications of decreased micrometeorite encounters as it left the vicinity of earth, this can only be considered a favorable sign. Mars is much closer to the asteroid belt and may also share the earth's apparent ability to concentrate micrometeorites.

The surface characteristics of Mars and its atmosphere are not as well defined as might be wished. It does not seem likely that a horizontal landing would be employed on the initial attempt. For this reason, improved knowledge will not strongly affect the design approach. The biological environment, on the other hand, will undoubtedly be an issue of concern from the standpoint of extra-vehicular operations.

From a communication standpoint, the possible existence of ionized layers that would block part of the transmission spectrum may be of interest. It would seem unlikely, however, that this would include part of the spectrum not already blocked by the earth's layers. Thus, this would only be a consideration in choosing a frequency for communicating between the landing vehicle and the mother ship.

More information can undoubtedly be obtained with improved observation of Mars from the earth's surface. It is safe to predict, however, that only modest changes in the total knowledge of the planet can be obtained in this manner during the next decade. The necessary knowledge of the environment of the mission must come from other sources. Primarily, improvements can be obtained by use of manned fly-by and Mars-orbital missions and possibly from unmanned probes sent to the vicinity and the surface of Mars. Because Mars has an atmosphere, the landing of probes, particularly those launched from a manned Mars-orbital spacecraft, should not be too difficult. The simplicity of atmospheric deceleration and aerodynamic stability, as opposed to rocket deceleration and black-box stability, will go far to overcome the difficulties associated with the remoteness of the planet. Such probes would not only be of immediate value to science, but would materially assist the manned mission.

Moreover, the knowledge of Mars might be greatly enhanced by observations from an earth-orbit space station or a lunar base. The almost continuous observation of Mars will be very valuable in assessing the seasonal and daily variations in the surface environment and will, perhaps, provide a means for interpretation and evaluation not otherwise available.

Special attention has been given here to the Mars-exploration mission as the most advanced mission on which conjecture is timely. Again, we direct the reader's attention to the fact that, in order to insure the proper planning of the utilization of the nation's resources that can be committed to the space program, the frontier for exploration must be both pointed and deep. A broad attack on this frontier could absorb the total resources of the nation. It is important that the program goals selected be those giving a very high return per unit effort in the short range while at the same time opening opportunities for similarly high gains during the next program generation. ••

Project Mercury experiences

BY WILLIAM M. BLAND JR. AND
LT. COL. CHARLES A. BERRY, USAF, MC

In its main objective, manned orbital flight, in its scientific and physiological operations, and in the main stream of spacecraft development, Project Mercury built a sharp measure for succeeding programs

Project Mercury has given the world certain self-identifying items—the astronauts, the Mercury spacecraft and its launch vehicle, and the Mercury control center—outward forms of a successful program. Here we will review some salient, if less obvious, features of the Mercury program, including its biomedical and scientific results.

The scope of Mercury operations since inception of the program in 1958 is charted on page 31. From these operations it has been learned how to use and direct effectively the capabilities, resources, and know-how of many diversified and sometimes widely scattered organizations to achieve a major national goal in a short time.

After performing hundreds of thousands of tests on the components for the spacecraft and launch vehicle, completing hundreds of systems tests, and making 140 full-scale tests, including the five manned spaceflights, certain things stand out that can be recognized as major reasons for the success of Project Mercury's space systems. Some were known and applied at the beginning of the program, others were developed as the program progressed, and still a few others have become evident in retrospect. All should have a bearing on future programs, and they will therefore be spotlighted here.

One of the significant technical facets of this program has been the careful and continuing attention given to detail in all phases of the program from design through flight training. It has taken major project management efforts to instill in industry the beginnings of the desire and conviction to provide this attention to quality and engineering detail that is required in order to achieve the necessary reliability for manrated space systems. At this time a beginning has been made; however, continued improvement in all levels of industry is still mandatory as evidenced by the examples discussed in this paper.

Early in the program, individual systems were carefully designed to fulfill the performance requirements in the real environment with as little weight cost as possible. At the same time, it was desired to utilize proven and qualified components in as straightforward a manner as possible to save development time. During the course of the program, however, it became clear that these objectives would not produce the reliability desired without some additional detailed modifications to accommodate the rigors of the environment, the interaction between components and systems, and the requirement for flight safety to be preserved even though failures occurred. Thus, qualified performance was accepted only after demonstration of successful operation of the systems under both the expected and the unexpected, but possible, environmental conditions. These demonstrations were accomplished with simultaneous operation of all the systems that make up the complete spacecraft. The qualification for the unexpected conditions can often spell the difference between failure and success from both mission and flight-safety viewpoints after the expected environment has been changed by the malfunction of some component.

On the other hand, it was a requirement from the beginning of the program that certain components would have to be able to operate satisfactorily for the length of time required to preserve flight safety even though other failures had occurred to prevent accomplishment of mission objectives. During a final part of the qualification program, an earthbound Mercury spacecraft completely equipped and fully instrumented was tested again and again through simulated missions in a space environmental chamber. It was during these tests, with all the systems operating as integral parts of the complete spacecraft as they would on space missions, that it was determined that the sensors of the automatic control system could not provide the necessary margin of safety after a loss of spacecraft cabin pressure. Complete evaluation of the problem disclosed that the interaction of all systems during this condition imposed a greater heat load on these particular components than analytical methods had predicted. Once the problem had been analyzed, corrective action was successfully applied by replacing some of the components with others that were more temperature-tolerant.

The tests with the earthbound

Mercury spacecraft disclosed deficiencies in many other areas when the systems were operated together in flight configuration under mission-performance requirements and near orbital environmental conditions. These deficiencies had not been detected previously because analytical methods had failed (as one might expect) to predict accurately the net integrated effects on particular components of systems operating together as they would during an actual Mercury mission. Extensions of the environmental tests to higher and lower pressures than anticipated have also been of great benefit to Project Mercury, particularly in the fields of electrical and electronic components and to higher pressures for mechanical systems.

Of course, using only proven components enhances the reliability of any program. But, even if it is possible to obtain proven components, constant vigilance must be maintained to prevent the sometimes catastrophic side effects of seemingly innocuous, straightforward "product improvement" changes that are often made in components without proper consideration of possible indirect effects. Changes so minor in process or material that they could not be visually detected were accepted by some as "the thing to do." Some of these changes were later found to produce either bad side effects or would not do the job at all. From Mercury experience it has been found best to use this statement as a guide line, "Do not make changes for changes sake. Change only those items which must be changed and then closely scrutinize each and every change from every possible point of view."

Maintaining launch preparations on a given schedule has always been important and becomes more important operationally as launch windows for some of our future manned space missions become smaller. In launch preparation, some schedule slippage occurs from component failures caused by excessive use, improper environment, or mechanical accident. It is important to recognize early in the design that some component failures can and will occur, and that therefore it is very important to provide easy access to components, particularly to those of limited lifetime, without causing many other systems or components to be removed, disturbed, or disconnected. Mercury experience has shown that even with careful planning and careful checkout procedures, some component replacements become necessary when the lifetime of the components is extended beyond allowable limits by some other, and possibly unrelated, difficulty which has taken unscheduled time in the launch preparation. For instance, replacement of the important and life-limited carbon dioxide absorber in the environmental-control system became necessary prior to the MA-6 launch, because more time than had been planned was required to check out the system. This replacement required eight major equipment removals and four revalidations of unrelated subsystems for an over-all delay of about 12 hr. By comparison, replacement of the carbon dioxide absorber itself took only 1 1/2 hr. Most of the time was used to gain access to the absorber and then to restore the spacecraft to the condition it was in just before replacement action.

Project Mercury experience demonstrates that qualification is not an end in itself. Instead, it is the beginning of the intended useful life of a component or a system. In a number of cases, components have passed qualification tests, but then others of the same type have consistently failed or exhibited below-specification performance during more fully integrated systems tests or in actual flight. These failures have been traced on occasion to the use of "hand built" pilot models in the qualification program surviving conditions which later production counterparts could not take.

An important part of the Mercury electrical power system successfully passed all the qualification tests, yet when installed in the spacecraft it exhibited numerous failures. These were laboriously traced to certain temperature-critical components. The closeness of these to the heat source was recognized as being a possible problem in the hand-built model which passed the qualification tests. Their location, however, presented a problem in production of the part. A slight change in location of the critical components to facilitate production was thought to be insignificant, but this change caused an increase in component operating temperature that rendered the part unusable in the spacecraft. The solution was obviously the proper relocation of components.

Failures have also been traced to qualification tests themselves being relaxed in some seemingly unimportant areas because of oversight or lack of test equipment. A prime part of the attitude-control system used for the Mercury spacecraft experienced many problems which could have been detected and remedied early in the program had adequate facilities been available for simulating the space environment. However, the vacuum-testing requirement was relaxed, and this, combined with the integrated spacecraft thermal balance, resulted in an unacceptable temperature feedback into this critical component of the attitude-orientation system. This problem was resolved by redesign of much of the hardware and the method of installation.

It has become most apparent, then, to those in the Mercury program that qualification programs must be properly aligned and executed and that the equipment being qualified must be identical with that to be used in the program.

It has also been learned, moreover, that it is important for the subcontractor, the contractor, and the user to understand mutually the particular function that each component contributes to the whole system, so that delivery inspection tests by the subcontractor, acceptance inspection checks by the contractor,

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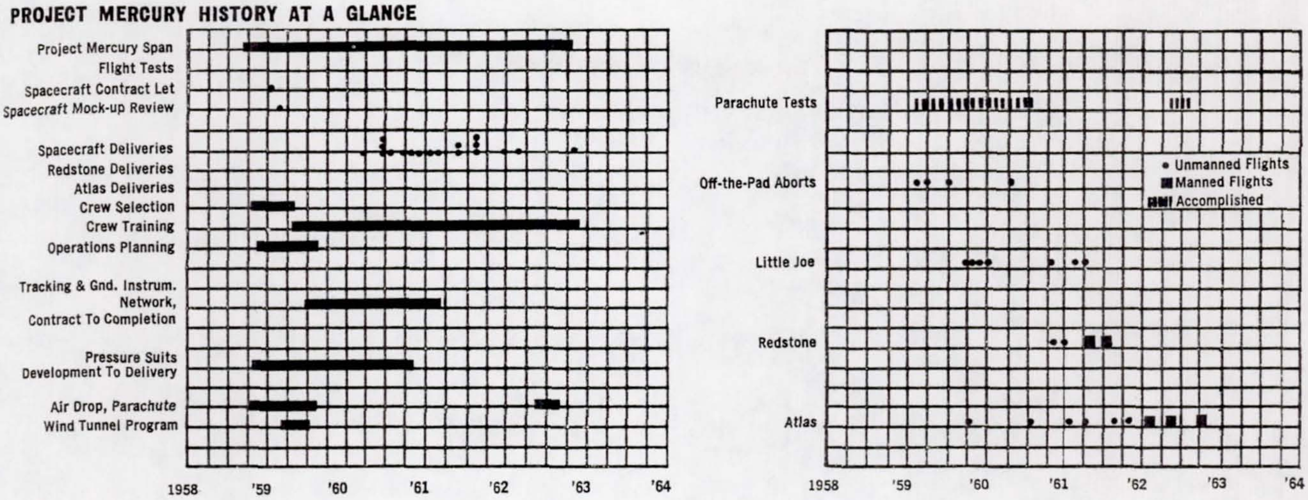
PROJECT MERCURY HISTORY AT A GLANCE



and pre-installation inspection checks are identical with one another and also detailed and severe enough to cause rejection of the units actually unsuited for use.

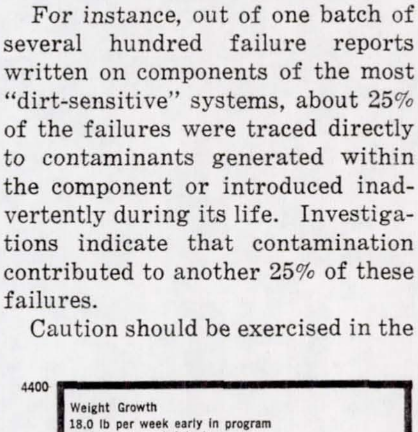
It is most important the procedures and checkout equipment used in the final plant assembly be identical with the equipment and procedures used in the final checkouts in the launch support area and at the launch site. Experience in Project Mercury verifies this. For example, after several spacecraft had been flown successfully, and a great deal of checkout experience gained, a decision was made to change the gas medium used to check a liquid-gas handling system for leaks. The change to an easier-to-handle gas was made to facilitate the checkout, because it was found that a large portion of the leaks detected by the first gas were, in reality, quite liquid-tight, and thus the tedious checks were overdone. The change to the new medium was adopted by the team that checked out the spacecraft early in the schedule, but the team that did the checking on the same spacecraft immediately before launch continued to use the first gas. The rest of the story should be obvious. That is, the first crew delivered a spacecraft that was leak-tight by one standard, but the second crew found many leaks with the first gas. Repair of these "leaks" caused additional work, more wear and tear in the associated components, and delays in schedule, all of which were unnecessary. Of course, the answer to such problems is to make the checkout procedures uniform.

Identical equipment and procedures serve their purpose only when there is also a complete and updated interchange of experience among technical crews at the different sites. Nothing should be left to chance, even though the system checkouts are designed to detect human errors as well as equipment failures. Because of human failings, systems must be so designed that they can be checked out, installed, connected, and operated in only one way if high reliability is to be achieved. Electrical connectors unintentionally interchanged in the spacecraft and in the ground-checkout gear often evidenced the lack of application of this principle. Similar mistakes can occur in hydraulic components and mechanical systems unless positive steps are taken to require connectors that *cannot* physically be mated incorrectly.

Because systems used in space vehicles have to be lightweight and have to operate efficiently to conserve consumable materials under very severe conditions, they have to be designed to very close tolerances. These close tolerances mean that particular care must be taken to see that shortcomings in cleanliness and storagability do not offset the carefully detailed design work, precision manufacturing processes, tedious qualification testing, and exacting checkout procedures. Experience in Project Mercury indicates that detail design, including the choice of materials, should provide comfortable margins in anticipated shelf life to accommodate schedule delays and extensions of programs. Handling methods and storage conditions also play such important roles in shelf life that they must be carefully chosen and proven in the qualification program and then be carefully monitored throughout the project. Equally important is the maintenance of chosen cleanliness standards throughout all phases of fabrication assembly, checkout, installation, and use. Many failures in Project Mercury have been directly traceable to contaminants finding their way into components. Lax standards at all levels of production, assembly storage, and checkout caused these failures.

For instance, out of one batch of several hundred failure reports written on components of the most "dirt-sensitive" systems, about 25% of the failures were traced directly to contaminants generated within the component or introduced inadvertently during its life. Investigations indicate that contamination contributed to another 25% of these failures.

Caution should be exercised in the



WEIGHT HISTORY of Mercury spacecraft.

application of such common things as lubricants. Even though qualified for use, with the particular materials and environment being considered, excessive amounts of lubricants can cause contamination problems as surely as can dirt from the floor. The trouble that Astronaut Schirra had in establishing a comfortable suit temperature during the Sigma-7 mission was traced to excessive lubricant contaminating a sensitive valve. Proper lubrication would have been provided by three drops of lubricant. Something like six to 10 drops were used and the excess above three drops became a contaminant, and caused considerable concern and almost had serious consequences. This experience illustrates the level of detail that must be controlled by quality specification and inspection to insure success.

We can mention one last interesting experience with the over-all spacecraft system. It concerns weight growth, a common phenomenon in an engineering development. The weight of the complete spacecraft has increased almost 800 lb above the initial estimate made late in 1958, as the graph on page 31 indicates. The greatest rate of weight increase occurred early in the program, as might be expected.

As the program progressed, the engineering became finalized and the weight control became tighter. This resulted in a decrease in the rate of increase to about 1.5 lb per week or to less than 0.04% per week. The significant fact is that even in the later stages of the program, the weight has continued to increase in spite of a determined effort to control weight. Thus, it is emphasized that ample allowances for weight growth must be made early in a program to prevent redesign and attendant requalification during the program because even though gross weight decreases are made on occasion through reevaluation of the design and the missions (events A through D in the graph on page 31), the weight does increase with time.

Concerning mission planning, it has been found important to freeze the flight plan well ahead of the scheduled mission. This gives the flight crew sufficient time to train with a particular plan. The preparation of this plan has started earlier and received more attention on each succeeding Mercury flight. Of course, preceding the actual freeze, care has to be taken to establish the desired plans and then to have them carefully reviewed so that the manner in which the basic mission objectives are to be accomplished is mutually acceptable to flight crew, the operational staff, the medical advisers, and the engineering staff. Naturally, freezing of the flight plan must be accompanied by a similar freeze in the spacecraft and launch-vehicle configurations, the launch-vehicle flight programming, and the recovery plans. Then, within the constraints established by these, by now, mutually acceptable plans, the flight crew and the operational staff must conduct the training exercise. From this point on, no changes should be made to the flight plan or flight hardware except those required to satisfy safety of flight measures, and then only after the representatives of the same organizations who agreed on the original plan have concurred with the changes.

It has also been found very important to avoid filling every available moment of the flight with a planned crew or ground-station activity. Time must be available to investigate any malfunction in a system and to observe and measure the unexpected. The importance of an extra margin of time can be appreciated simply by remembering the automatic-control-system troubles encountered by one astronaut and the suit-temperature-control problem encountered by another. The problems were circumnavigated in considerably different ways by each pilot. The first pilot resolved his difficulty by resorting to emergency techniques because he had no time to isolate the trouble on a rational basis. If he had had sufficient time, the chances are great that he woluld have made the retrofire maneuver within nominal tolerances in spite of the control system trouble. The other astronaut, on the other hand, largely as a result of experience obtained from former flights had a more leisurely flight plan and was able to devote almost all of his time to determining the proper setting for a comfortable suit temperature as long as it was a problem. He was able to resolve this difficulty in a straightforward manner. Thus, until more is learned about this new environment of space, it will be necessary to provide ample time for possible adjusting of systems after their initial exposure to the real space environment. Also, as in the case of all exploratory missions, time must be allocated for observing the unexpected and the new, such as the "fireflies" observed in all three missions and the luminous layer around the earth that has been the subject of observation and measurement on these flights. Time must also be programmed to allow the pilot to consider thoughtfully his reactions to weightlessness and its effects upon him. On the other hand, a delicate balance must be maintained in the flight plan so that the astronaut does not have extremely long periods without any planned activities.

Finally, the special equipment provided the flight crew for special tasks of measurement and observation should be carefully designed for human operation in the weightless environment and the confines of the suit and spacecraft. All equipment must be easily reached in the operational situation. The

PHYSIOLOGICAL RESPONSES OF MERCURY ASTRONAUTS

Flight phase	Heart rate, beats per min.		Respiratory rate, breaths per min.		Blood pressure[a], mm Hg Systolic		Diastolic		Body temperature, F	
	Max.	Min.	Max.	Min.	Max.	Min.	Max.	Min.	Max.	Min.
All preflight	160	42	40	5	155	91	120	44	101.5	97
Countdown	140	50	30	6	139	105	94	56	99.2	97
Flight	170	56	40	8					100.5	97
Launch	164	82	40	8						
Orbit	114	56	26	8	143	103	94	59		
Reentry	170	72	32	11						

[a]Three orbital flights, but no inflight data from MA-7; no determinations during launch and re-entry.

equipment must also be available long before the mission to give the crew an opportunity to become completely familiar with its characteristics and operating procedures, so that its use becomes second nature during the mission.

Training and simulator devices have proved valuable tools for preparing a man for spaceflight. But merely compiling hours in these devices does not accomplish the training task. The pilot must have, in association, detailed training on the basic systems and plans for the mission so that complete understanding is obtained to provide the basis for rational reaction to the unrehearsed situation.

Besides preparing the pilot for normal and emergency flight duties, training must prepare him to carry out successfully the special experiments assigned his mission. For certain of these tasks, the pilot becomes a laboratory experimenter, and each experiment demands certain training. So far, many different training modes have been used to good advantage. These include lectures by the specialists, discussion with the associated scientists, familiarization sessions with special flight gear before the flight, and parallel study in the field of the experiment. During Project Mercury, the special training given the astronauts has produced properly trained experimenters for each mission.

For a program to progress at a fast pace, it is necessary for management to have the tools and information necessary for making effective and timely decisions. Project Mercury has accomplished this objective by a continuing series of engineering and mission reviews, responsive to project development, that commenced soon after the contract was signed.

Engineering, technical, configuration, and mission reviews have been held within Manned Spacecraft Center on about a weekly basis to present up-to-date information on proposed technical changes, problem areas, potential problem areas, and test results. At these meetings, the necessary day-to-day decisions were made to keep the program moving alone the chosen path.

At other times, development engineering inspections were held at the spacecraft contractcr's plant as significant spacecraft approached delivery status. Also, composite testing and buy-off inspections were held at the launch-vehicle plant. These inspections were attended by top management and the best, most experienced supervisors, pilots, engineers, specialists, inspectors, and technicians. Results of these inspections after thorough discussions manifested themselves in requests for action that had to be satisfied before the spacecraft or launch vehicle was accepted.

Technical reviews, attended by top management, probably constituted the most significant management tool used in Project Mercury to insure that the proper attention had been given to necessary details. These reviews were held in the days just before launch, and preparations for them proceeded simultaneously with the launch preparations. In the process of ascertaining that the material required for presentation at the meetings would be acceptable, the technical work in progress was reviewed in great detail and updated and corrected wherever necessary. At the reviews, then, the questions relating to the flight readiness of the spacecraft, the launch vehicle, the crew, the network, the range, and the recovery effort could be answered in the affirmative. The following philosophy, believed to be the basic reason for Mercury's operational success, ruled the reviews: "Mercury launchings will not take place in the face of known troubles or in the face of unresolved doubts of any magnitude that affect mission success or flight safety."

A basic objective of Project Mercury was to evaluate the physiological and psychological reactions of man in a space environment. This objective has been accomplished to the extent of the Mercury capabilities. The longest U.S. spaceflight to date was Astronaut Schirra's mission of about 9 hr, in Sigma 7 on Oct. 3, 1962. The other manned spaceflights were of shorter duration: two which lasted 4 1/2 hr, and two which gave about 5 min of weightless flight (MR-3 and MR-4).

These flights have demonstrated that man is a useful, dependable—even a necessary—part of a space system. With this evidence, many redundant backup systems have been removed from the Mercury spacecraft to reduce weight, complexity, and checkout time for the manned one-day mission. And the experience gained so far has greatly influenced the design of other manned spacecraft (see page 35).

The five manned Mercury flights produced significant biomedical data, from which have been drawn medical trends and general physiological information. The biomedical monitoring, we should note, was provided in Project Mercury primarily to assure flight safety. Operations did not permit classical medical research and measurements, for the state of the art of many medical instruments do not render them operationally feasible. There follows some of the more significant medical information gathered in the project so far:

1. A great deal of experience has been gained in proper medical operational planning for preparation, in-flight monitoring, recovery, and debriefing.

2. The physiological responses to the acceleration of launch and reentry have been shown to be readily tolerable by the astronauts. The previously accepted "normal" range for pulse and respiration rate have been expanded, as the table on page 32 indicates.

3. Results of repeated preflight and postflight physical examinations have been within the normal range, and no changes directly related to the spaceflight experience have been noted.

4. Considerable knowledge has been derived regarding the use of biosensors and resultant telemetered data in the medical flight-control tasks.

As would be expected, numerous artifacts have been noted during observation of the telemetered electrocardiogram. Experience has produced monitoring confidence in the proper real-time evaluation of these artifacts. During the MA-7 mission, some apparent elevation of blood pressure was noted throughout the flight. A very detailed and time-consuming postflight evaluation of the blood pressure measuring system (BPMS) revealed the necessity for very accurate matching of cuff, microphone, controller gain setting, and astronaut if the telemetered reading was to be accurately related to the usual clinical blood pressure reading. Such matching produced excellent results in the MA-8 flight.

5. There have been no aberrant neuromuscular, vestibular, or mental reactions, and the astronauts have proved to be capable of completing spacecraft control and reporting tasks and of making vital decisions affecting flight safety.

6. The weightless state for the time periods studied has not shown cause for concern. Food consumption and intestinal absorption have successfully been accomplished during weightlessness. Furthermore, normal bladder sensation and urination have been reported. Solid, cube-type and semi-liquid foods have been eaten in flight without difficulty. The semi-liquid foods, packaged in tubes similar to those used for toothpaste, have consisted of applesauce, beef, and vegetables. The specially coated cubes were of chocolate, fruit, and cereal. In both MA-6 and MA-7, the astronaut took a xylose tablet in flight, and the intestinal absorption was determined by analysis of the amount recovered in urine.

7. Auditory, olfactory, and visual responses, which include tracking, color vision, and depth perception, have been normal and apparently unaffected by spaceflight.

The astronauts have reported color vision to be normal and almost identical to that experienced in observation of the earth from a high-flying aircraft. Scott Carpenter reported that colors on the panels of the balloon he observed during his flight looked virtually the same as they did on the same panels seen on the ground.

Drifting flight, and its attendant low rotary rates, has not altered the pilot's normal responses nor has disorientation occurred.

In summary, the astronauts have tolerated their limited spaceflights of up to 9 hr of weightless flight in a very satisfactory manner.

In the course of accomplishing their primary mission objectives, the astronauts have also contributed to our knowledge in the natural sciences through the following observations and measurements:

1. With the information obtained by the astronauts, three primary characteristics of the airglow (luminous) layer around the earth have been established. First, the strongest radiation of the airglow layer was found to be at 5577 Å wavelength. The layer was estimated to be about as bright as the moonlit horizon on that mission, or about 6×10^{-3} lux per steradian. Also, the height of the airglow layer was observed to be between 2.6 and 10 deg above the horizon. The smaller angle is believed most nearly correct.

2. Particles were observed in space and were discovered to have emanated from the spacecraft.

3. The apparent flattening of the sun's image through refraction of light rays by the earth's atmosphere was further substantiated.

4. Visibility from the orbital altitudes corresponded to that normally observed by the pilots from high-flying aircraft; earth colors appeared true and object definition was possible.

5. Experimental results confirmed behavior predicted by theory for liquid in a weightless environment.

6. A number of photographs of cloud formations were taken that have been of significant assistance to those who have been analyzing and interpreting pictures obtained by television from weather satellites. Also, some of these photographs have been used to determine the best filters to be used on cameras in later weather satellites.

To recapitulate, we have seen Project Mercury meet its primary objective—manned orbital flight—and moreover confirm these facets of the program:

1. The basic Mercury concepts established in 1958 were valid: Existing technology and off-the-shelf items used wherever possible; the most simple and most reliable approaches to system design followed; an existing launch vehicle, suitably prepared for manned flight, employed; and a logical and progressive test program used.

2. Mercury design, production, inspection, qualification, and checkout produced systems suitable for manned use in a space environment.

3. The flight crew and ground-network crews were trained to overcome, in real time, deviations from the normal with a minimum of realignment.

4. Data from a manned space mission can be analyzed, documented in detail, and distributed in an edited form within three weeks of the mission. This effort includes measured data, personnel debriefing information, hardware inspection, and trouble shooting and also the reproduction and distribution of the results. During this three weeks, most trouble areas can be detected and given initial corrective action.

The management structure that evolved has shown that it is:

1. Capable of successfully directing a program to maintain a tight schedule in spite of a very large and complex arrangement of supporting elements distributed over a large portion of the earth.

2. Capable of setting up the management tools that are required to produce success in a short time in a program where high reliability is the keynote in the face of a new and sometimes hostile natural environment.

The smooth and efficient way in which the last Mercury orbital mission, that of Astronaut Walter M. Schirra, was accomplished has left no doubt that man, the Mercury space systems, and the manned spaceflight operational organizations are ready for more extended missions. Work is already underway for accomplishing the next facet of the manned space program, the manned one-day mission, during the first half of 1963. This mission will require more consumable items but will otherwise differ little in procedure from previous Mercury flights. It will see continuing interest directed toward the body system, particularly the cardiovascular, and toward other areas, such as crew and flight-controller fatigue and crew waste disposal.

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Project Gemini design philosophy

BY JAMES A. CHAMBERLIN
AND ANDRE J. MEYER JR.

First of the second-generation manned spacecraft, just emerged from design, Gemini introduces major departures from Mercury— a new level of systems integration and operational potential

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Project Gemini, which introduces the second generation of NASA's manned spacecraft, has completed the design stage. Many individual pieces and components, moreover, have been fabricated, and the long and complicated process of assembling and checking out the parts—first as modules, then as major systems, and finally as a completed integrated spacecraft—has begun.

To understand the Gemini design philosophy, we should look first at the primary project objectives:

1. With a minimum of expense and time, to provide a logical follow-up to Project Mercury.
2. To subject two men and their supporting equipment to long-duration flights in space, a requirement for lunar trips and beyond.
3. To rendezvous and dock with another orbiting vehicle.
4. To maneuver a spacecraft in space after docking to a new propulsion system.
5. To experiment with men climbing out of the spacecraft for short periods while in orbit.
6. To perfect methods for returning and landing the spacecraft on a small preselected land site.

The first objective—providing a follow-up to Project Mercury—imposes many limitations on the design of the Gemini spacecraft. Although the objective resulted in following in the footsteps of Mercury in many ways, it also necessitated departures from the Mercury program to remove many of the limitations of the Mercury design, some of which were inherent in its objectives and some of which were revealed as the program progressed.

Mercury was designed with the sole purpose of placing a man in orbit in a minimum time. The main emphasis was put on solving problems—re-entry aerodynamics and thermodynamics, human tolerance to both high accelerations and zero gravity, etc.—which had never been encountered before. Consequently, great attention was not directed to the serviceability of the spacecraft. Hence, when guide lines were established for a follow-on program, it was assumed that solutions to all the basic problems had been obtained in Mercury and that the emphasis could be placed on serviceability and flexibility of detail design. The Gemini spacecraft would reduce flight itself to a relatively routine performance and put the emphasis on experiments in orbit, rather than just attaining orbit.

In Project Mercury, most system components were in the pilot's cabin; and often, to pack them in this very confined space, they had to be stacked like a layer cake and components of one system had to be scattered about the craft to use all available space (see page 37). This generated a maze of interconnecting wires, tubing, and mechanical linkages. To repair one malfunctioning system, other systems had to be disturbed; and then, after the trouble had been corrected, the systems that had been disturbed as well as the malfunctioning systems had to be checked out again. Only one technician could work inside the Mercury cabin at any one time.

In the Gemini craft, systems are modularized, all pieces of each system being in compact packages. Spare packages can be kept completely checked and ready for rapid replacement. The packages are so arranged that any system can be removed without tampering with any other system, and most of the packages ride on the outside walls of the pressurized cabin for easy access. This arrangement allows many technicians to work on different systems simultaneously. The illustration on page 37 shows clearly only one of several walls used in this way. The modular concept applies even to wire bundles, which are fabricated on special fixtures and then merely clipped in place.

Only the visual instruments, controls, and survival ingredients such as the food, water, waste-handling equipment, rescue aids, and breathing apparatus ride inside the pressure vessel.

Placing units outside the pressure compartment causes other problems. For example, cabin atmosphere can not convectively cool the units, and each must therefore be mounted on a cold plate to carry away heat electrically generated. The elimination of convective cooling has the effect of modularizing the systems thermodynamically. A space radiator has therefore been designed to unload system heat. Since the outer covering of the spacecraft re-entry section does not lend itself to radiator construction, the transition, or adapter section between spacecraft and launch vehicle has been made to serve this purpose.

Besides radiating heat into space, the adapter stores mission supplies. These supplies include breathing oxygen in supercritical form, fuels and thrusters for orbital maneuvering, communications equipment needed only in orbit, and the fuel cells and associated supercritical hydrogen and oxygen used to generate electric power and drinking water. The adapter, being unprotected against high heating, must be jettisoned before re-entry.

The second objective—a two-man crew and long-duration flights—introduced basic departures from Mercury. The first was the *two-man* crew. It was believed that, for really extended periods, it was most desirable to be able to alternate rest periods and generally to lighten the load on one man. It was obvious, moreover, that providing supplies and facilities for living in space for a long period represented a major step. The basic problems to be faced were made much more difficult by the small space available in the cabin.

In many cases, even the equipment that performed the same function in Mercury required considerable modification for Gemini to boost the mean time to failure to a level consistent with long-duration flights. Many provisions had to be incorporated in the circuit design and selection of electronic components to secure the required life. For instance, although there are not many vacuum tubes in Mercury, none could be tolerated in the Gemini spacecraft. In the mechanical area, gear drives on fans and horizon scanners had to be eliminated for Gemini. Special inverters were installed so that the correct fan speed could be obtained directly. Here the policy of separating systems, which resulted in modular power supplies within the individual system packages, proved a necessity, not just a virtue.

The long-duration flights planned necessitated special attention to the meteorite problem, particularly in the space radiator. The design evolved circulates fluid in a hollow bulb along the inner edge of the stiffener extrusions, as shown in the illustration below right, and hence secures a high degree of inherent mechanical protection. This design combined with the redundant paths gives acceptable reliability.

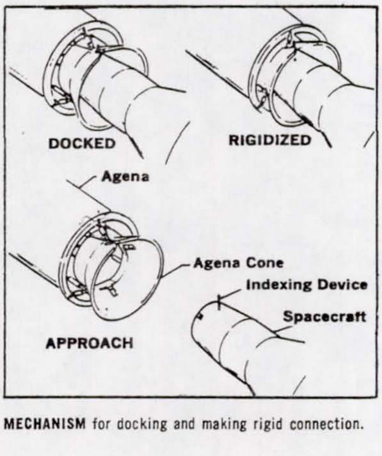
In general, a great deal of attention has been directed in all aspects of design to reliability. But the goals are so high that really meaningful demonstration testing is virtually impossible in the time available. The dilemma involved in this situation suggests that some new approaches to reliability testing must be devised.

The third objective—to effect rendezvous and docking with another vehicle—introduces new systems such as radar units, on-board computers, and propulsion systems for making small accurate changes in flight position. These systems will include the following new equipment: radar, Westinghouse; digital computer, IBM; paraglider, North American; space radiator, McDonnell; fuel cells, GE; docking mechanism, McDonnell; landing skids, McDonnell; inertial-guidance platform, Minneapolis-Honeywell; incremental velocity measuring unit, IBM; and supercritical oxygen and hydrogen systems, AiResearch.

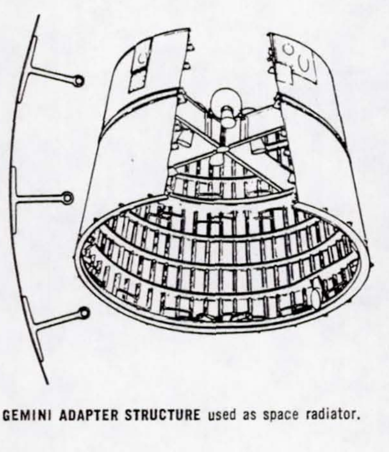
The third objective also requires launches to be performed within narrow periods of time, which means that holds on the launch pad and flight cancellations have to be minimized; this is where the benefits are realized from the emphasis on serviceability throughout the design. It introduces, moreover, the mating hardware on the spacecraft and target vehicle for docking, as illustrated below left. The Gemini spacecraft will rendezvous and dock with an Agena, initial contact being made with a floating cone supported on the front end of Agena by oleos. This cone absorbs energy when hit in any reasonable way, and will not cause a rebound, but will guide the Gemini in toward springloaded latches. After engagement, a mechanism snubs the whole cone to the Agena, making the combination rigid for space maneuvers.

The fourth objective—maneuvering the docked assembly in space—is almost implicit in the choice of target vehicle. It was considered necessary to indicate the "health" of the various target-vehicle systems to the pilots before doing any maneuvering. Accordingly, a series of parameters were chosen which could be used to activate lights and provide indications on gages as to the condition of the Agena systems. At first it was thought that these should be displayed on the pilot's instrument panel. After due reflection, however, it was considered that a much better and vastly simpler method, as to hardware, would be to display them on a panel on the outside of the *target* where either pilot could see them both before and after docking. This scheme eliminates a major requirement for hardline connections between vehicles. Much the same type of reasoning was applied to the command system. Since the system must operate by a microwave link before docking, this link might as well be used after docking. Hardline connections are retained for engine shutdown in parallel with the radio command.

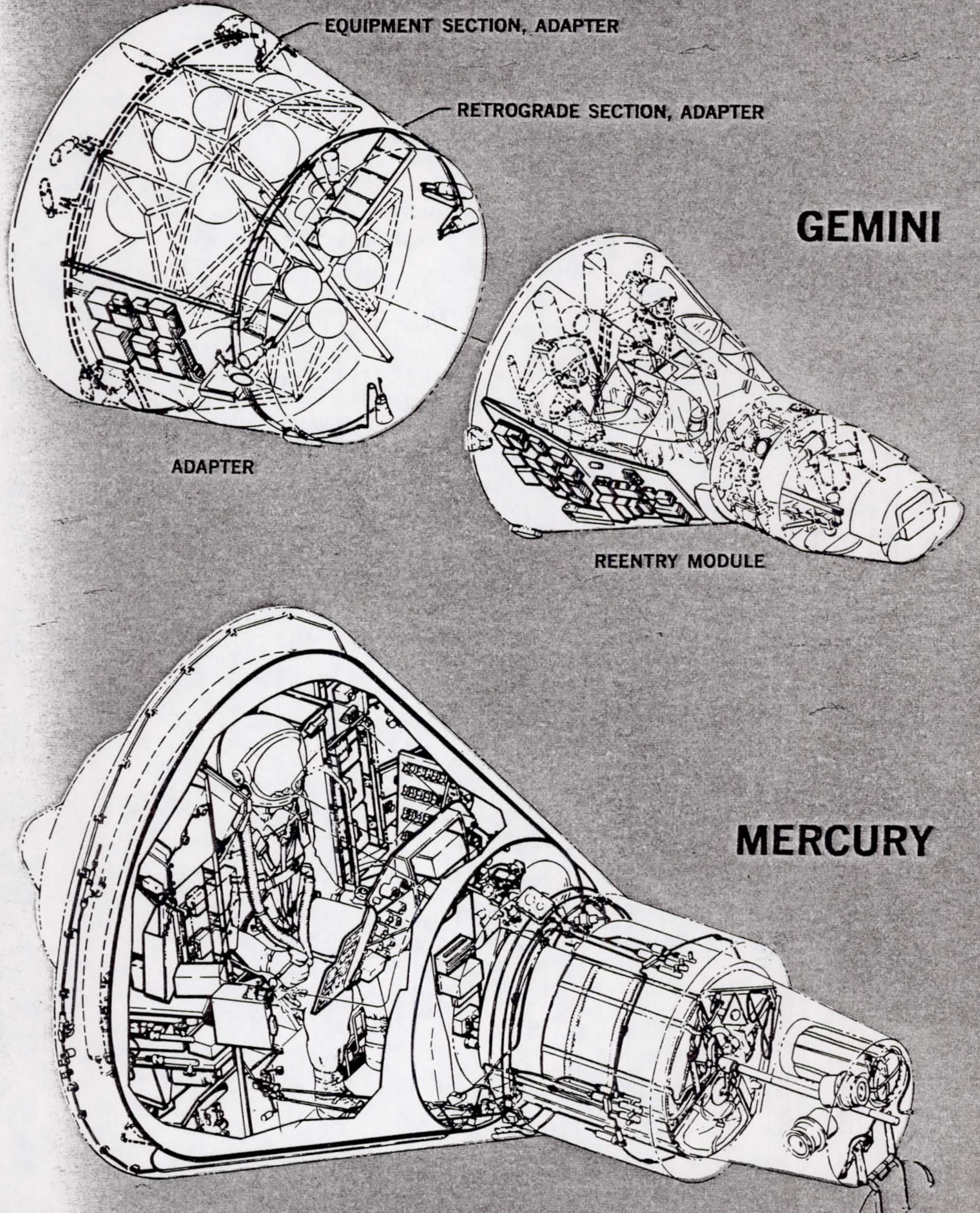
The fifth objective—extra-vehicu-



MECHANISM for docking and making rigid connection.



GEMINI ADAPTER STRUCTURE used as space radiator.



SIGNIFICANT DEPARTURES of Gemini from Mercury are illustrated in these cutaway drawings. In the Mercury spacecraft, the exigencies of the moment made it necessary to stow most systems in the cabin with the astronaut. The Gemini design moves most systems out of the pressure vessel and consolidates system components in easily accessible modules, an example of which is shown on the Gemini spacecraft.

lar experiments—also requires only minor changes to the spacecraft configuration. The hatch, instead of being bolted in place, is hinged and locks by mechanical linkages. Suitable personnel equipment is under development for the extra-vehicular experiments.

The sixth objective involves re-entry control and a paraglider for spacecraft recovery.[1,2] Re-entry control is obtained by using the lift generated by offsetting the center of gravity of the spacecraft and then modulating the roll during re-entry. An on-board inertial system and computer generates the required commands. The paraglider stows in the spacecraft's small cylindrical section. The sketches on page 39 show how it deploys by inflation to become a full-fledged wing. This can be flown by the astronauts much as a conventional light aircraft. The craft will have landing skids.

One of the most important differences in design philosophy between Mercury and Gemini has been greater reliance on the astronauts to control the Gemini spacecraft. This has been made possible not only by having redundancy in equipment (a common practice in Mercury) but also by having two pilots. Manual control, as opposed to complete automatic control, was selected to increase reliability by simplifying sequencing. The automatic abort modes in Mercury, for example, are very complicated and have caused the loss of complete spacecraft in the early developmental unmanned flights. In each instance, had a man been on board, he could have manually salvaged the situation. The Atlas is so instrumented that it will automatically abort the Mercury spacecraft if any one of a number of malfunctions is sensed in the launch vehicle. If a malfunction occurs, the propellants used in Atlas would react rapidly, causing a violent explosion. The storable propellants of the Gemini launch vehicle react more slowly and allow more time for pilot action.

In Gemini, a launch-vehicle malfunction activates lights and gages on the instrument panel and the astronauts exercise judgment as to the seriousness of the situation and the best procedure to follow during any special circumstances. With this sort of system, more than one cue can be used to verify an abort situation. Simulations reveal that, in many cases, much reliance is placed on the audio-kinesthetic cues for this purpose. These cues are not only very reliable but instill confidence in the pilots in the validity of the systems when they are checked by this means. Manual-control is used in many other mission phases, see table on page 39.

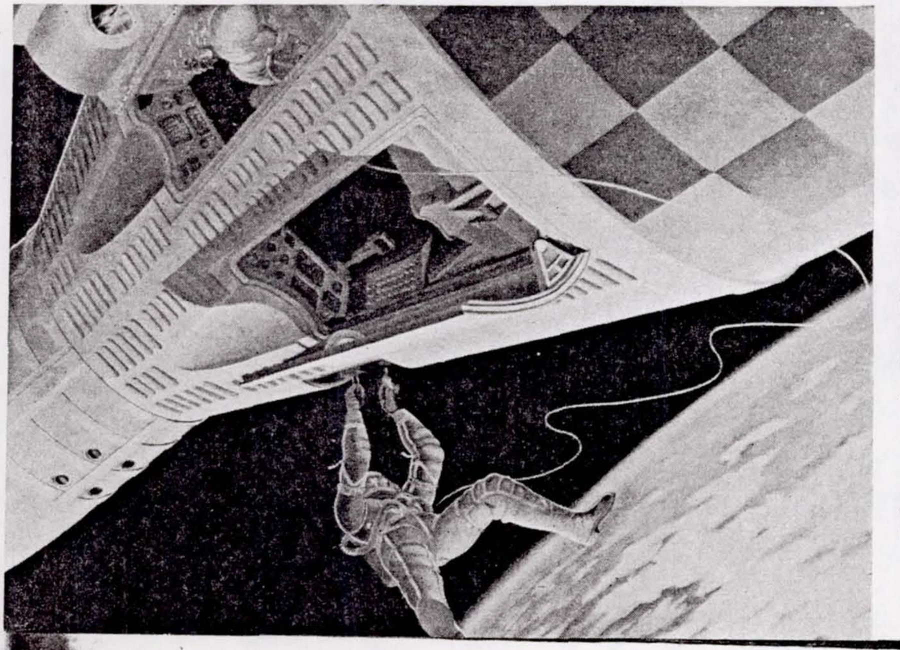
There are a few other differences in the design concepts of the two current manned space programs. Mercury uses an escape rocket that lifts the entire spacecraft, whereas Gemini uses ejection seats. There are advantages and disadvantages to both systems. The escape tower is only available up to staging. The ejection seats not only provide a substitute for a reserve parachute but also provide an escape mode both early in the flight and on landing. Ejection seats were favored because they are consistent with the modular concept, but they were really only made possible by the fact that there is no problem from blast pressures in the event of deflagration of the propellant used in the Gemini launch vehicle.

Another important design concept being pursued in the Gemini program is to retain a flexible universal spacecraft configuration. This effort is greatly facilitated by the modular design for the systems. In the Mercury program, much effort and money were spent in changing among unmanned (heavily instrumented), simulated-man, chimpanzee, manned, three- and six-orbit, and one-day configurations. In the Gemini program, mission variations are accommodated simply by replacing specialized modules.

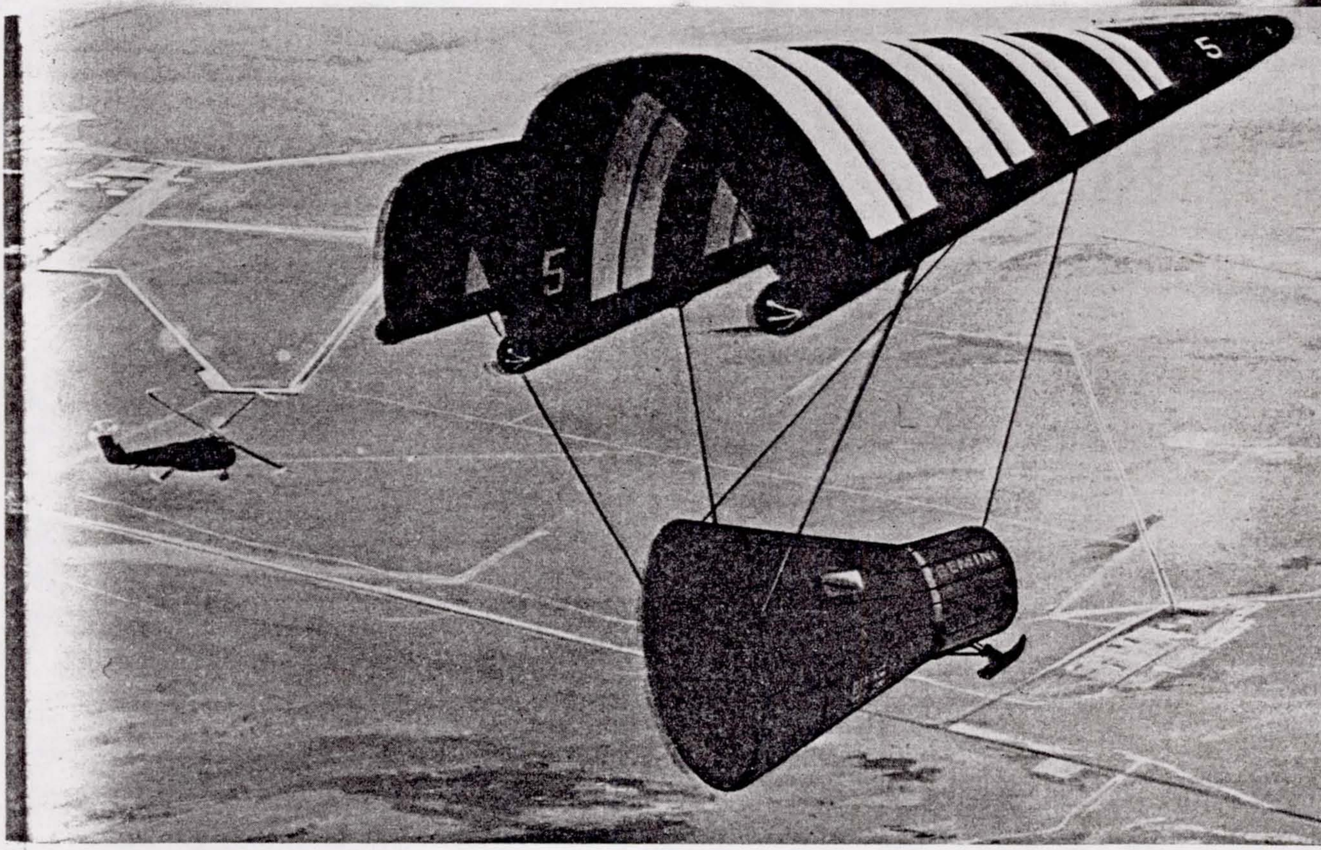
Finally, some hardware familiar from Mercury has been dropped. The periscope was eliminated because the benefits derived from it did not warrant the weight or the complications introduced by the need to extend and retract the main lens body. The landing bag is no longer a necessity when a paraglider and landing skids are used, or even if a large parachute should prove necessary instead of the paraglider. When a parachute is used, the spacecraft has been designed to land in water on the edge of the heat shield to attenuate the impact forces. Finally, the large reserve parachute has been omitted because the ejection seats allow emergency escape.

The objective behind all these changes and innovations has been to produce a spacecraft that will make manned orbital flight commonplace. Project Gemini is well on the way toward this goal.

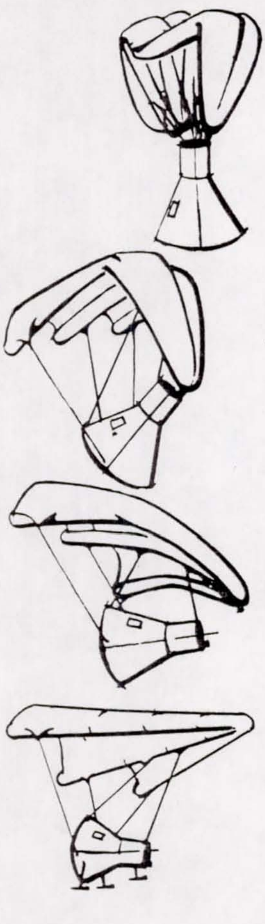
REFERENCES: 1. Rogallo, Francis M., "Paraglider Recovery Systems," presented at the IAS Meeting on Man's Progress in the Conquest of Space (St. Louis, Mo.), April 30–May 1–2 1962. 2. Rogallo, Francis M. and Lowry, John G., "Flexible Re-entry Gliders," Preprint No. 175C, SAE Meeting (New York, N.Y.), April 4–8, 1960. ••



HUMAN-FACTORS STUDIES such as the one illustrated here, involving egress and tools, will be possible with Gemini.

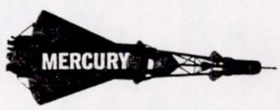


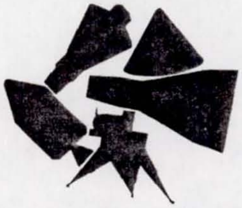
PARAGLIDER SYSTEM, being developed by NAA, will be used as the primary Gemini landing system.



PARAGLIDER DEPLOYMENT follows the sequence indicated by the sketches at left.

GEMINI AND MERCURY FLIGHT OPERATIONS COMPARED

SEQUENCE	 MERCURY	 GEMINI
Booster separation from spacecraft.	Automatic with manual backup.	Astronaut fires separation system.
Capsule turnaround to retro or orbit attitude.	Automatic with manual backup.	Astronaut turns spacecraft manually to proper attitude by watching attitude indicator.
Retro attitude before re-entry.	Automatic when signal is received by spacecraft.	Astronaut turns spacecraft manually to retro attitude, as displayed on attitude indicator.
Aborts, all levels.	Automatic with manual backup.	Ground-command lights, spacecraft-abort light, and astronaut control sequences manually.
Drogue-parachute deployment.	Automatic by 21,000 ft., barostat with manual backup.	Astronaut deploys drogue parachute manually at 60,000 ft.; automatic backup at 21,000 ft.
Landing.	Automatic from 21,000 ft. by parachute.	Manual control of paraglider by control stick.



Pilot safety and Mercury/Atlas

BY BERNARD A. HOHMANN

This program to protect the Mercury astronauts—based on quality assurance, intensive inspections, and ASIS—has important future implications

BERNARD A. HOHMANN recipient of various USAF commendations and of the meritorious Civil Service Award, is Mercury program systems engineering director for Aerospace Corp. An ex-test pilot, he now flies his own airplane.

When NASA chose the Atlas for Project Mercury, it was recognized that this launch vehicle had not been designed as a man-carrying vehicle, but for a ballistic weapon system. The design and development technology of ballistic launch vehicles, as well as their basic reliability, are far different from those of aircraft, which today are based on many thousands of hours of flight time and well-established operating experience and procedures.

NASA therefore established a requirement for the development of a highly reliable system to permit pilot escape. The space agency undertook design of a spacecraft-launch-vehicle separation system, and assigned to the Air Force the requirement to develop an automatic system to detect launch-vehicle failure. Recognizing the over-all safety requirements, Aerospace Corp. proposed a specific Project Mercury pilot-safety program, and this was implemented as a team effort of NASA, the AF Space Systems Div., contractors, and Aerospace Corp. The diagram on the facing page summarizes the key program efforts.

Let us now review this program and its chief results, beginning with the problem of design reliability.

Design Reliability. The pilot-safety program can best be viewed against the background of a typical launch vehicle's reliability as a function of time. The graph appearing on page 42 demonstrates the increment of safety needed for manned flight over the basic reliability of the launch vehicle itself. It is virtually impossible to obtain the high launch-vehicle reliability necessary in the time period scheduled for a given program—in this case, the Mercury program. It would have been desirable to improve the reliability of the Atlas launch vehicle to a somewhat higher level before a manned flight, but a major redesign and a very extensive test period would have been required to demonstrate that higher reliability actually could be obtained.

The basic Atlas reliability was consequently accepted and, to fill the gap between the basic reliability shown by the bottom curve and the desired higher level for manned flights, a special safety device was added—the abort sensing and implementation system, or ASIS, explained in detail in the discussion of reliability augmentation following.

ASIS automatically senses a malfunction of the launch vehicle and triggers the separation mechanism of the Mercury spacecraft to separate it before a major disturbance endangers the astronaut.

The over-all Mercury mission cannot be saved by the abort-sensing system, but it does give adequate pilot safety. As shown by the upper curve in the graph, it is not expected that 100% reliability can be achieved even with pilot-safety augmentation devices. Although ASIS is highly reliable, it is doubtful that it will provide adequate pilot safety for every possible malfunction. It does, however, provide the highest attainable degree of safety for the Mercury astronaut during the Atlas-powered portion of his flight, and it is believed he is at least as safe as he would be in a new, experimental-type aircraft.

To preserve the experience and reliability achieved in the Atlas ICBM program, the number of changes made to the Atlas to convert it to a launch vehicle were held to a minimum. The illustration on the facing page shows the major modifications.

Quality Assurance. A quality-assurance program was set up to guarantee the best quality, workmanship, and reliability possible for all hardware used in the Mercury/Atlas launch vehicle. It consists in part of an educational program for contractor and subcontractor personnel. Under this program, training courses, lectures, and presentations are given by GD/Astronautics to its engineering, inspection, factory, and subcontractor personnel to make them aware of the importance of the manned spaceflight program and its objectives. Literature pointing out key points and items of this program is also distributed.

The program also provides for selection of certain components and subsystems. Selection criteria include such considerations as clean inspection records and predetermined operating times before acceptance. Additionally, items with major repairs or refurbishment are not accepted. Spare parts are also selected to the same criteria and are specially allocated for use in launch vehicles for the Mercury program. Each selected or allocated component, part, or subsystem is identified by a special decal signifying it as an accepted Mercury component. All components identified by this decal are stored in a specially designated and controlled area.

End-Product Excellence. A factory roll-out inspection aims to assure that the Mercury/Atlas launch vehicle is complete in every respect, functionally acceptable, and ready for delivery to the Air Force. The inspection team consists of members of the AF Space Systems Div., the AF plant representative, and specialists of Aerospace Corp. for various technical areas (autopilot, pneumatics, ASIS, propulsion, electrical systems, guidance, etc.). Its members review progress on the launch vehicle on a continuing basis to identify potential problem areas, and evaluate all component records, subsystem test data, and composite test records.

The composite test, the final contractual AF factory-acceptance test of the launch vehicle, is made in the presence of AF inspection personnel with the various systems operating simultaneously under nominal flight-simulated conditions. Functional acceptability is based upon the evaluation of the data from this test.

Complete and satisfactory documentation of component and subsystem selection and of all test data, engineering change proposals, failure, consumption and data reports, etc., are required before end-product acceptance. The contractor must also submit a detailed report covering the status of qualification of critical items on the launch vehicle. No shortages are allowed; the launch vehicle must be functionally complete in every respect before delivery, so to guarantee it has been checked out as a complete launch-vehicle system.

Inspection-team members prepare a final report covering the assembly and test history, as well as all discrepancies uncovered and corrected on the launch vehicle up to the time of delivery to the AF and to the Atlantic Missile Range.

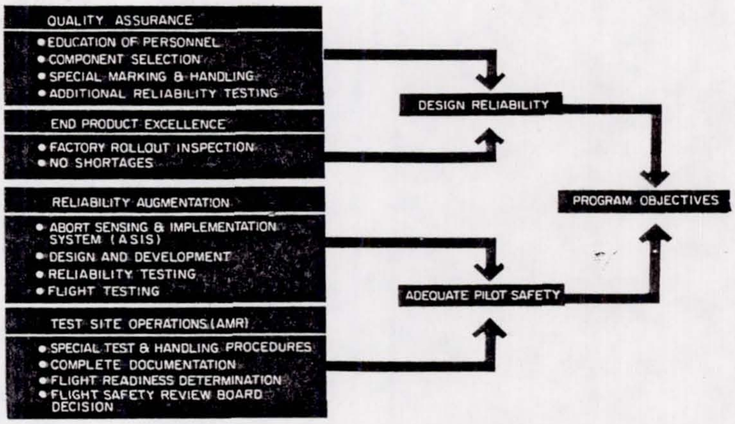
Reliability Augmentation. A means was sought to close the gap insofar as possible between Atlas reliability and the goal of 100% pilot safety. The approach taken was ASIS—a highly reliable system for sensing any impending catastrophic failure of the Mercury-Atlas launch vehicle and for automatically generating an abort command to shut down the propulsion system and activate the Mercury spacecraft's escape system before the astronaut might be placed in jeopardy.

ASIS continuously monitors certain critical launch-vehicle performance parameters in such a manner that, if preselected tolerances are exceeded, an abort signal will be generated and the spacecraft's escape sequence will then initiate automatically. To determine what ASIS should monitor, previous Atlas flight-test data were analyzed to locate parameters that indicated impending catastrophic failure during disastrous flights and that did not indicate failure on successful flights. Moreover, ASIS was designed to eliminate inadvertent aborts resulting from failure of its own sensing instrumentation or circuitry. Redundant wiring, sensors, and electronic components counteract the effect of any single component failure. The diagram on page 42 locates various electromechanical sensors throughout the launch vehicle which monitor the critical systems.

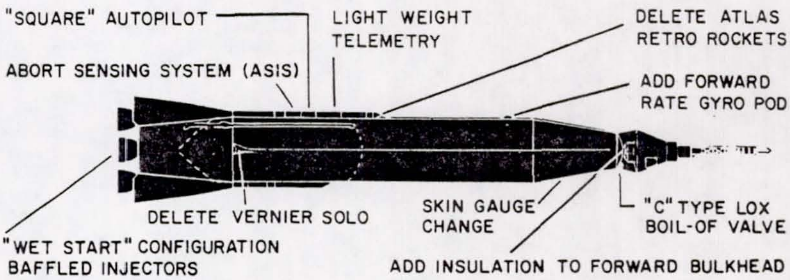
Various manual abort capabilities supplement the automatic abort system, as follows:

1. The test conductor can initiate an off-the-pad abort.
2. The NASA Mercury Control Center can initiate an abort.
3. The astronaut can terminate the mission at any time throughout the entire powered flight.
4. The range-safety officer can generate a manual engine cutoff command and thereby activate the automatic airborne abort system.

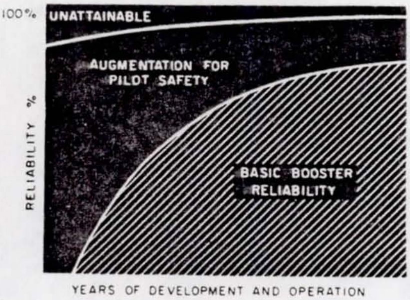
In addition to five successful ASIS development flights on the Atlas launch vehicle, a very exten-



KEY ELEMENTS in the pilot-safety program for the Mercury/Atlas launch vehicle.



MODIFICATIONS to Atlas, as indicated in this sketch, were made to qualify it for Project Mercury.



TYPICAL RELIABILITY GROWTH with launch vehicles.

sive reliability test program was conducted to assure its reliability under extreme environmental conditions. Extensive failure-mode analyses were conducted to select components whose failures, however unlikely, would be in the fail-safe direction. The complete system operated successfully in the open-loop configuration on MA-1, although the mission was not a success. MA-2, MA-4, MA-5, MA-6, MA-7, and MA-8 were successful flights in the closed-loop configuration. The MA-3 flight was prematurely terminated, but a successful abort was properly initiated and saved the spacecraft, which was flown again on MA-4.

Test-Site Operations. Factory roll-out inspection assures that the Mercury launch vehicles are in the best possible condition when they arrive at AMR. This condition must be maintained in the hangar and on the launch complex. Consequently, it is very important to have stringent control over the hardware configuration and to have complete and accurate documentation of any hardware changes. AF quality-control personnel closely monitor any necessary replacement of components and, particularly, selected components. A sufficient number of selected spare parts, components, and subsystems are stored in a specially designated AMR area. No hardware can be removed from Mercury launch vehicles to support other Atlas flights without specific approval of the AF. Only persons necessary to perform required tasks are permitted access to Mercury launch vehicles on the launch complex.

A Flight Safety Review Board (FSRB) determines whether the launch vehicle is ready for flight. For manned flights, participation on the Board is usually of high level, under the chairmanship of the senior AF representative. A team of four NASA personnel attends the final FSRB meeting, essentially a presentation by FSRB to the NASA Operations Director concluding with a recommendation on committing the launch vehicle for manned flight.

A team of technical representatives from NASA, the AF, Aerospace, GD/Astronautics, and the chief field representatives of Rocketdyne, General Electric, and Burroughs reviews for FSRB the entire history of the launch vehicle since its arrival at AMR and presents its recommendation on the technical flight readiness of the launch vehicle.

FSRB must determine that all possible efforts to insure a successful mission have been made, that the launch vehicle is in the highest state of technical readiness, and that any reservation on the part of participating agencies has been considered. It then conveys its recommendation to the NASA Operations Director for his consideration in conjunction with the corresponding recommendations from the Capsule Review Board, Tracking Network, and other agencies.

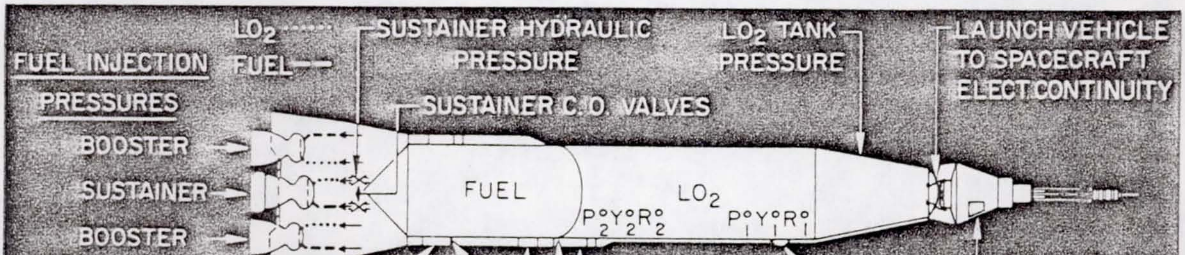
These procedures, plus ASIS, permitted NASA to begin its manned spaceflight program without the delay necessary to design and test a special launch vehicle, at no sacrifice to pilot safety.

Future Applications. The experience with the Mercury program clearly shows that future manned systems must incorporate a pilot-safety program. Even systems specifically designed for manned flight will require a pilot-safety program to assure that manrating actually is achieved as designed and that the manrating reliability can be and is maintained—a most important factor.

Through the efforts of the entire Mercury team, it was gradually recognized that the following factors govern the ability to maximize mission success:

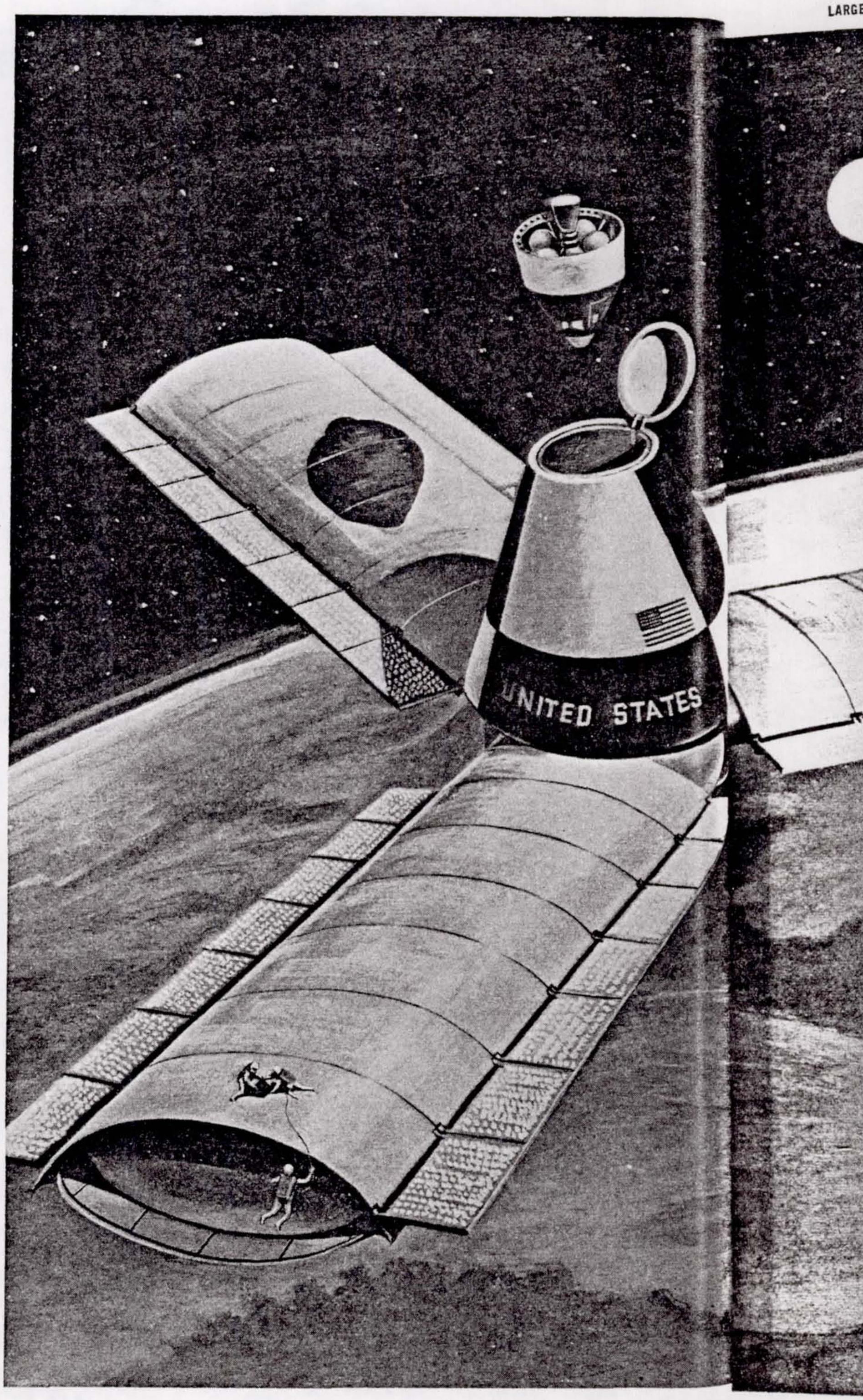
A team approach; systems engineering; an aggressive failure-analysis program; and a hardware quality-assurance program. These factors are sufficiently logical and general in content to allow their application to almost any complex development requiring a high degree of reliability. ••

ABORT-SYSTEM SENSORS

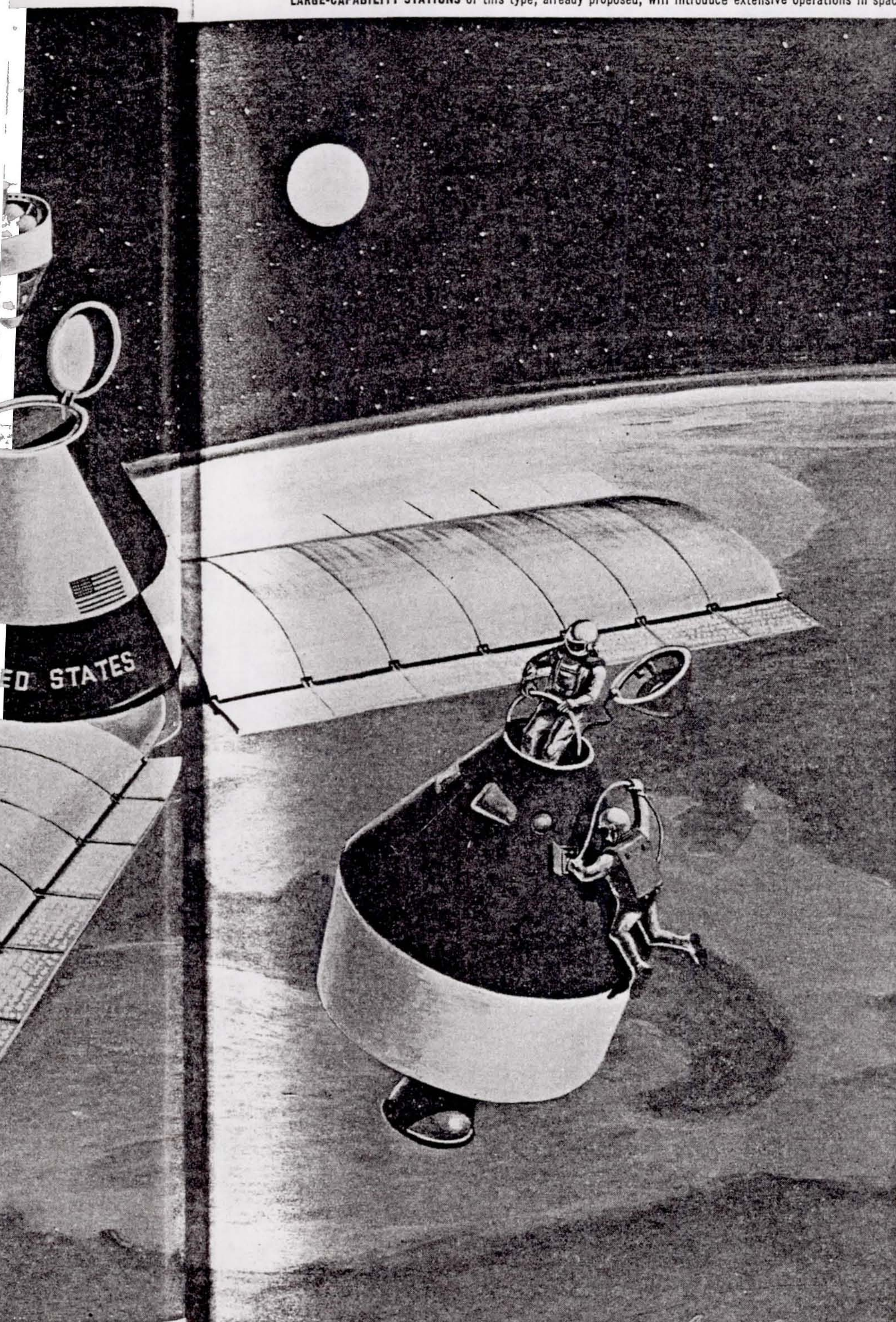


EARTH ORBITING SPACE STATIONS

Although not yet approved, space-station development looms as a logical follow-on to present manned spaceflight programs. This discussion considers the implications of missions, objectives, design configurations, logistics, and operations



LARGE-CAPABILITY STATIONS of this type, already proposed, will introduce extensive operations in space.



Missions, objectives, applications, and capabilities

BY EDWARD H. OLLING

Interest in space stations is not new. Space stations have intrigued men since the times of Nostradamus, Leonardo da Vinci, and Jules Verne, who, each in his own way, envisioned the potential applications of space stations related to the exploration of the frontiers of space. More recently, many serious scientists and engineers have proposed space-station systems not unlike those under investigation today. What is new with respect to space stations is our recent and immediate ability to place relatively large payloads in orbit.

After a manned lunar landing has been successfully accomplished, there are other continuing significant steps in man's exploration of space. Currently, the most immediate steps appear to be:

1. Placing a manned space station in earth orbit to conduct research and to perform a wide variety of space operations.
2. Establishment of a manned lunar base to assist in lunar exploration and exploitation.
3. Accomplishment of manned planetary flights to explore the near planets, e.g., Mars and Venus.

Although there is currently no approved national space station program, the potential role of the space station as the next major step and as a major contributor to the lunar base and planetary missions is currently the subject of intense investigation, analysis, and planning to determine the most effective and necessary flight schedules. In evaluating the potential capability of the space station as a major program in itself and in support of the over-all long-range manned spaceflight program, it is necessary to analyze space-station missions, objectives, applications, and capabilities with respect to the main goals of the national spaceflight program.

The manned exploration of space is a continuation of steps advancing progressively, based on the results obtained from preceding steps. The feasibility of the space station is based on the availability of technology resulting from Projects Mercury, Gemini, Apollo, and other manned and unmanned space programs. The primary missions of the space station reflect the support of future programs involving increased technical difficulty and longer duration of flight.

The space station is an essential and necessary component to the resultant success of manned planetary flight. The space station provides the research facilities to determine man's capabilities for sustaining long durations in the operational space environment. If it develops that man has serious physiological limitations, the space station could provide unique facilities and capabilities to investigate and determine criteria and solutions to allow man to overcome his limitations, thus permitting him to perform essential functions effectively and efficiently for the long periods necessary for successful manned planetary flight missions.

Personnel at the space station could investigate and establish the environmental criteria, such as radiation, meteoroids, and the other combined factors of the operational space environment which must be provided for in the design and development of the planetary craft.

The space station could provide support for planetary flights by conducting research on plants, animals, materials, finishes, processes, and equipment in space.

The engineering development, qualification, and reliability testing of planetary spacecraft subsystems, equipment, and mechanisms prior to the initiation of the flight could also be accomplished in the space station. In current missions, if a malfunction or system failure should occur, the flight could be aborted, and the crew can in many instances return safely to earth. In the planetary mission, the programmed flight plan may not allow time for the safe return of the crew to a recovery point if a malfunction should occur months or years in flight duration from earth. Therefore, it is absolutely essential that fully developed and qualified, reliable, proven, and tested systems, with adequate redundancy supported by service and maintenance provisions, be provided to insure mission success.

Another area of potential space-station contributions to the planetary program is the use of proven, tested, qualified, building-block segments containing the basic structure airlocks, and subsystems for the man-occupied part of the planetary spacecraft.

The space station could be instrumental in developing the crew qualifications for the planetary mission by providing the facilities for crew selection in the operational space environment and in the complete training of a competent, experienced crew before mission start.

The planetary mission may require an orbital launch operations facility. The space station will provide accommodations for the launch operations crew, and a base for shops and equipment necessary for the assembly and checkout of the payload segments of the planetary spacecraft in earth orbit.

The role of the space station in the lunar-base program is very similar in many ways to that described for the planetary mission.

First of all, the lunar-base crew could be selected and trained in the station based on information obtained from the Apollo manned landings on the lunar surface. The crew could be subjected to lunar gravity levels and other conditions for long durations by simulating lunar surface conditions in the near earth-orbital environment which cannot be duplicated on earth.

With respect to the lunar-base logistics transportation system, the space station could receive payloads from earth which could be stored and prepared for trans-shipment to the moon.

Orbital launch-operations facilities, essentially the same as those used for the planetary mission, could also be used to support the lunar-base logistics operations. The logistics transportation shuttle could be serviced, maintained, repaired, and overhauled at the space station with the facilities available there. In case of emergency, rescue missions might be initiated more quickly and flexibly from earth orbit than from the earth's surface.

In the general advance of science and technology to support the exploration of space, and particularly the long-range manned spaceflight program, the space station could make unparalleled contributions not possible on earth or by current spaceflight programs developmental.

The space station could be a national research facility in the operational space environment, conducting all types of basic and applied scientific research to further man's advances in space. In the accomplishment of such activities, much valuable data on the space environment could also be collected to establish design criteria for future programs.

Scientific probes could be launched to obtain data to meet the requirements of other programs.

Proposed advances in propulsion systems could be researched and qualified for future use, particularly where testing is not physically or economically practical on earth.

Another area of research where only the space station could provide effective facilities is in long-term investigations of biological, physiological, psychological, hereditary, and genetic factors related to man, animals, and plants.

EDWARD H. OLLING heads up MSC's Space Station Office, Spacecraft Research Div. A mechanical engineer, he has done research in thermodynamics, fluid dynamics, life support, and crew equipment problems. He joined NASA's science staff in 1962.

The infinite vacuum of space is available to the space station to be utilized in experiments or possibly in commercial applications requiring the attainment of very hard vacuum of unlimited capacity.

Many research and application projects are currently being conducted separately and independently of each other. While these operations achieve results, they are expensive in the consumption of funds, manpower facilities, and other resources. The space station could provide a common base and supporting facilities where many of these activities, such as meteorology, communications, navigation, astronomy, and earth geophysics, could be undertaken. Obviously, some applications could not be suitably accommodated in the station because of the requirements of orbital altitude or inclination.

Where the activity is appropriate, a far greater return for funds expended can be achieved when a number of applied functional operations are carried out in the space station. The research experiment or application project could be serviced, maintained, repaired, overhauled, and reoriented; and data could be acquired and evaluated by experienced, qualified personnel. This support would extend throughout the useful lifetime of the project and would enhance the successful fulfillment of objectives which currently are often unobtainable as a result of the lack of just such support as the crew in the station could provide. Combined installation of manned systems, or manned-supported systems, in many instances could yield greater returns and results for less expenditures than is possible with separate and independent unmanned projects.

In reviewing and analyzing space-station potentialities, it should be realized that the capability of developing the space station and achieving these objectives is now available. The technology to develop the space station is in existence, but it is still necessary to exercise the utmost ingenuity and intelligence in planning and designing the space station to achieve maximum results.

The feasibility of the space station has been demonstrated by numerous studies made by various NASA centers, by the Air Force and by a number of aerospace industrial contractors. The required technology in many ways is less demanding than that required for Project Apollo, for example, in such areas as navigation and guidance, re-entry heat protection, propulsion subsystems, and lunar-landing devices.

The interest in space stations has been restimulated by the availability of launch vehicles being developed by the Apollo manned lunar-landing program with large payload capability, such as the Saturn C-1 and C-5. For the first time, launch vehicles are being developed with adequate payloads to meet project objectives without serious limitations. With the C-5 launch vehicle, a space station with adequate volume could be orbited with a large crew and could stay in orbit for prolonged periods with sufficient electrical power capability to conduct a wide variety of tasks.

The space-station program as envisioned is based on existing technology, existing launch vehicles, existing launch sites, existing range and tracking networks, and possibly a modification of the existing Apollo spacecraft as the logistics spacecraft to transport the crew and cargo to and from the space station.

The basic guide lines defining space-station requirements are important factors related to the configurations which are being evolved. The space station could be launched from the Atlantic Missile Range at Cape Canaveral at approximately the same inclination angle as that used in Project Mercury. The orbital altitude would be approximately 300 n. mi. This approach would effectively utilize the existing checkout facilities, launch sites, and range and tracking network. Sufficient spacecraft would always be docked at the space station to accomplish normal crew rotation or to evacuate the crew in case of an emergency. The space station would be in continuous operation for a period of from one to five years. The space station would provide both zero and partial gravity capabilities and facilities. Lastly, the space station would be conceived so that it could become operational at an early date.

Several approaches have been proposed for executing such a space-station program, and many different approaches are now under investigation. The following examples appear to merit the most consideration at this time:

1. A large, rotating space station of wide, versatile, flexible capability would be launched with a Saturn C-5 launch vehicle as soon as possible, and would be supported by a modified six-man Apollo logistics spacecraft launched by a Saturn C-1B launch vehicle. Such a station is illustrated on pages 52–53.

2. An alternative approach is a two-step program which would use a Saturn C-1B launch vehicle to lift a modified Apollo spacecraft with an attached laboratory into earth orbit for a period ranging from several months to a year. The research results of such a laboratory would then be used to design a large-capability space station launched by a Saturn C-5 which would also be supported by a six-man modified logistics spacecraft.

3. Another approach is the sequential, progressively phased, building-block segment method which utilizes existing designs and tooling from various launch-vehicle stages to construct cylindrical and spherical building-block modules. The first of these building-block modules could be utilized as a bioastronautics laboratory. The results of laboratory research could be used to define the building-block segments required to make up a large modular space station. In turn, the space station could define requirements for planetary craft.

Then, appropriate segments could be used to build up the man-occupied portion of the planetary spacecraft from proven, tested, and qualified, building-block units.

In conclusion, the space station is a program which can accomplish a wide variety of major significant achievements. The operational schedule of such a program is limited only by effective program approval, funding, and launch-vehicle availability. The schedule for the achievement of future programs, such as the planetary mission, may well be dependent on the availability of research data and design criteria which can be developed and provided only by the space-station facilities and related capabilities.

SPACE STATIONS

Configurations and design considerations

BY OWEN E. MAYNARD
AND RENE A. BERGLUND

Recent accomplishments in manned orbital flights and the increased payload capabilities of planned launch vehicles have resulted in a wide variety of proposals for space-station configurations. While the parameters and principles that form the basic design criteria and the resulting proposed configurations are too numerous to describe, a review of space-station configurations proposed by NASA and industry indicates that two of the major areas of emphasis are assembly methods and rotational characteristics.

The following methods for assembling the space station have been proposed:

1. Earth-assembly configurations, in which the design of the station will be compatible with the launch vehicle and in which the station will be operationally ready when placed in orbit by a single launch-vehicle system.

2. Orbital-assembly configurations, in which numerous components of the station will be launched into orbit, brought together, and assembled. This method of assembly will require a number of launch vehicles to orbit the various components.

3. Configurations, assembled on earth and erected in orbit, which could be launched by a single launch vehicle and, when erected, will be operationally ready but will provide a geometric shape unrestricted by launch-vehicle constraints.

The rotational characteristics of the space station are identified as follows:

1. The nonrotating station, which will have no capability to provide artificial gravity by rotation and is generally associated with the earth-assembly method.

2. The rotating configuration, which will provide a capability for creating artificial gravity by rotation and is generally associated with the earth-assembly method.

A desirable feature of a rotating station would be the inherent stability resulting from the proper inertia distribution. This distribution would resemble that of a flat disk or fly wheel. This spin stabilization would assist in maintaining the station orientation and prevent the buildup of large oscillations. These oscillations could be detrimental to the crew, docking maneuvers, and the ability to abandon the station.

3. The partial-rotating configuration, which will be designed with nonrotating hubs for docking or for zero-gravity laboratories to which will be connected constantly rotating living and working areas. These are mostly associated with stations which are assembled in orbit or assembled on earth and erected in orbit.

In order to establish the necessary requirements to design an operational space station, it will be necessary to determine and evaluate parameters by using the mission profile, which includes a trajectory, related velocity requirements, associated astrodynamics, environmental conditions to be encountered, mission sequence, and logistics support. Essential supply support will be established by maintenance requirements, usage rate of consumables, life-support system requirements, and crew replacement. Since crew rotation will be dependent on psychological and physiological factors, the size of the station, type of mission, work-rest cycles, and environmental conditions will affect these factors and will increase or decrease the crew rotation cycles.

Two classes of launch vehicles are presently being considered in the design of space stations: The early launch-vehicle class, which has a payload capability of approximately 30,000 lb, and the later class, which has a payload capability of approximately 220,000 lb.

The launch vehicle of small capability imposes both geometric and weight constraints in designing a space station with which it would be compatible. Such a space station would have to be a highly sophisticated and complex structure. One concept for a space station of this type is a nonrotating, earth-orbiting laboratory which would be an adaptation of existing or proposed Apollo mission modules, placed in orbit by a Saturn C-IB.

The launch vehicle of greater capability would impose only a geometric constraint in designing a space station with which it would be compatible. Such a space station could be designed by utilizing standard materials and manufac-

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turing procedures and could result in a more reliable and economical structure at the expense of weight. This station would also have considerably greater capability for conducting scientific experiments. A reduction in the frequency of crew rotation and resupply is another advantage of the larger space station.

When the economics of the space station system are considered, the number of launch vehicles involved in providing an operational space station in orbit and the resupply launches necessary for the continued operation of the space station must be kept to a minimum. Feasibility studies of large erectable space stations indicate that they can be launched on a single launch vehicle. The operational procedure required for automatic erection in orbit should be a simple kinematic procedure with high reliability. All routine procedures should be as simple as possible, and there should be no requirements for the crew to do work outside the station in the space environment.

A space station of this possible design is discussed below. This type of station is more advantageous than the orbital-assembly type of station which requires a launch vehicle for each component.

In the establishment of design requirements for a manned orbiting space station, it will be necessary to consider a large number of human factors areas. These areas pertain to the man himself, the environment in which he must live and work, the tasks he must perform, the operations necessary, and his participation in these operations.

The space station must be operationally self-sufficient to support life for prolonged periods of time. The relationship between the man and machine operations must be considered in detail during design and development to attain this goal. A state of equilibrium must evolve between man and machine in the orbiting environment during these various time periods. This equilibrium must be closely predictable by a clear definition of crew tasks in order to optimize the probability of success of any given mission.

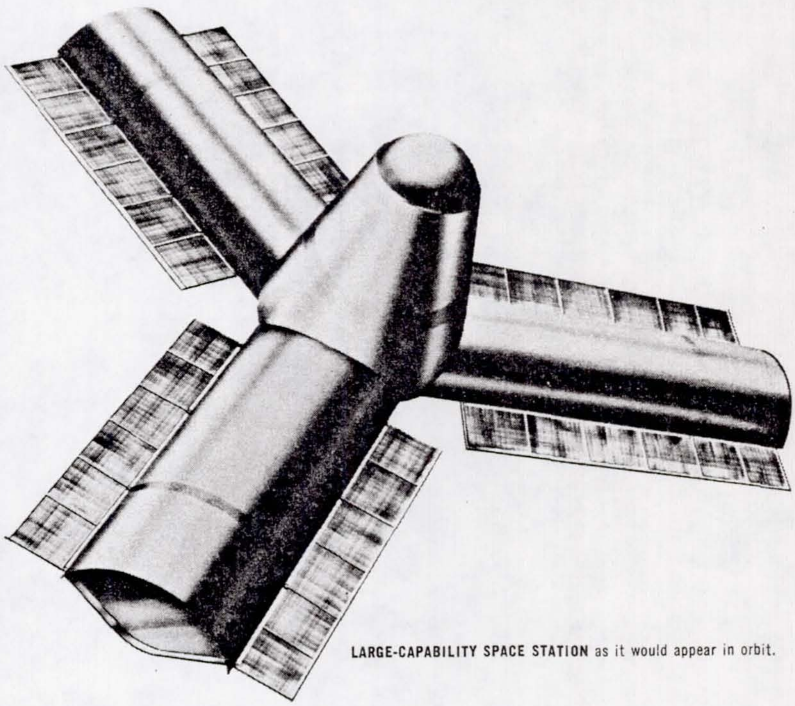
When human factors are considered, one of the most serious handicaps in the design of a space station is the inability to determine whether or not, or to what degree, an artificial gravity environment will be required for the well-being of the crew for the extended periods of time that will be necessary for planetary exploration.

When gravity research is considered, the partially rotating configuration would be perhaps the most desirable. A recently compiled list of uses for a space station and the experiments that could be performed on board indicates that few applications or experiments have a requirement for artificial gravity. The requirement would be on man himself and in learning more about his dependency on gravity. By having the nonrotating section in a rotating station, experiments at zero-*g* could be carried on simultaneously with experiments at various fractional gravity levels.

It would then be possible to determine with a single configuration the gravity requirements for future space missions.

The requirements for rotational effects to replace the gravity field are not well defined, nor are they likely to be without concerted effort in spaceflight operations.

The advanced Saturn will make possible launching a multiradial-module space-station such as the one illustrated below. This concept evolved from studies conducted to determine methods by which a space station of large capability could provide engineering and operational data that would be most useful in interplanetary spacecraft design. While it is generally desirable to limit a design to a minimum of design constraints and to embody a host of desirable features, there would be considerable difficulty at this time in separating the design constraints from the desirable features since the total concept is evolutionary in nature.



LARGE-CAPABILITY SPACE STATION as it would appear in orbit.

The total configuration was conceived to be launched by the first two stages of an advanced Saturn launch vehicle. A capability for shirt-sleeve operation in the central hangar area is incorporated to take maximum advantage of the crew's capabilities to service the logistics and other spacecraft and to minimize extra-vehicular activities. The three radial modules will be stowed axially for launch. Simple linear separation devices will permit deployment from a clean launch configuration to the radial arrangement for artificial gravity.

It should be pointed out that three radial modules will be employed for the space station, although two appear satisfactory for a spaceflight configuration where rotation about the minimum moment-of-inertia axis is desirable.

The space station could be launched with or without the manned logistics spacecraft. Docking would be provided for on the hub centerline at the top, with separation operations for earth-return conducted from the lower hangar area. The logistics spacecraft will be maneuvered along the axis of rotation to minimize the complexity of operations.

Each radial element will be deployed with a single degree of freedom. Breaking or making the major seals will be unnecessary. Each radial element may be independently deployed and secured after deployment. Elements such as the antennas, accumulators, and radiators could be extended or retracted as required.

An advantage attainable with this configuration approach will be the ability to study, over protracted periods, the effects of various gravity levels. The three peripheral compartments would, with spin rates believed to be acceptable, attain pseudo-gravitation thought to be in the comfort zone. Inboard compartments would have lesser levels approaching zero-*g*, thus permitting investigation of the possible extension of the comfort zone to lower radii. A near zero-*g* facility may be provided in or around the hangar area. It is recognized that an integral incorporation of a scientifically precise zero-*g* capability will constitute an operational complexity of a high order.

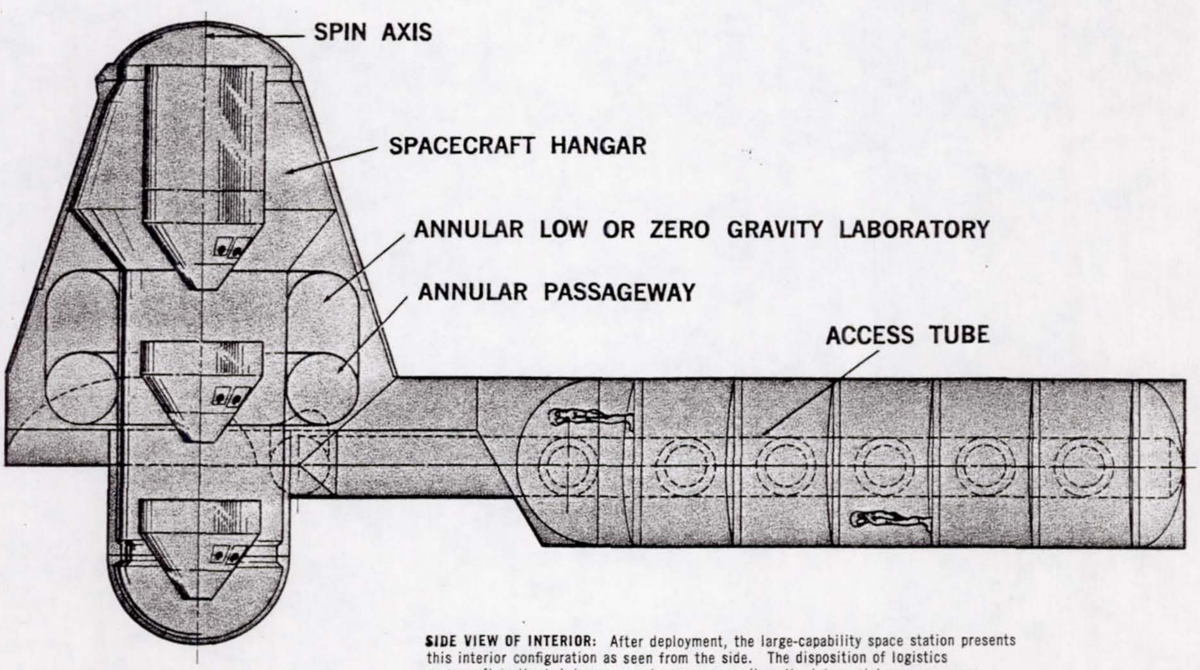
Access to all systems will be provided. Equipment, stores, or systems which do not normally require service may be stowed outside the areas provided with a shirtsleeve environment. Those which will benefit from frequent servicing and are not potential contaminators or hazardous will be stowed in the normal shirtsleeve compartments. Those which will benefit from frequent servicing but are potential contaminators or are hazardous, such as certain propulsion systems, will be stowed in isolated regions with shirtsleeve capabilities, where practicable.

Traffic patterns may be set up which would permit crew members to move about in the station in a shirtsleeve environment. A minimum number of hatches will be utilized to permit isolation of contaminated or otherwise undesirably conditioned regions, and the crew provided with alternate routes.

Crew transfer to and from the space environment, for maintenance or experimental purposes, will be provided through two redundant airlocks in each radial element. These airlocks may be located at the outboard ends of the access tunnels, as shown in the illustration, or normal to the tunnels inboard of the crew compartments. Simple hatches will be provided for direct access to crew compartments for earthbased operations or for emergency spaceflight operation.

Cargo transfer from the logistics spacecraft to the space station will be initiated in the hangar area. The operation will be made efficient by providing the crew with a shirtsleeve environment handling room, and by the inherent low gravity. Transfer from the hangar area to the radial elements will be initiated through access hatches in the annular passageway. General traffic routes and hatches will be sized to handle emergency conditions, and crews with appropriate logistics support items will be able to negotiate freely in routine operations.

A feature to be provided, which is unique to the radial element concept, will be the operational utiliza-



SIDE VIEW OF INTERIOR: After deployment, the large-capability space station presents this interior configuration as seen from the side. The disposition of logistics spacecraft in the hub hangar can be seen, as well as the intermodular passageways.

tion of all stations prior to launch, and prior to, during, and after deployment. The great benefit to be gained from this feature will be the capability for a flight crew to be trained or to take over the operation of the space station while it is on the ground being fitted out.

The direction of the gravity field is favorable; systems could be operated in the near-operational environment. Crew transfer networks throughout the station would be intact at all times. Equipment could be installed in the operational spaceflight location and be compatible with the launch environment. In general, the configuration will permit a highly efficient fitting-out operation which will save valuable flight man-hours.

Certain problem areas become immediately apparent. Perhaps the primary problem is that of sealing. In this design, which is conceptual only, a great deal of attention will need be given to the dome hatches at the top and bottom of the hangar area. A tradeoff will have to be made between balance while rotating and length of juncture to be sealed. Even if the length of the seal is minimized, it will remain very long. Only seals with the lowest possible leakage rates can be used. A simple but effective seal reinforcement will need to be applied by the crew. Neither seal nor reinforcement may preclude periodic opening and closing of the hatches with subsequent resealing.

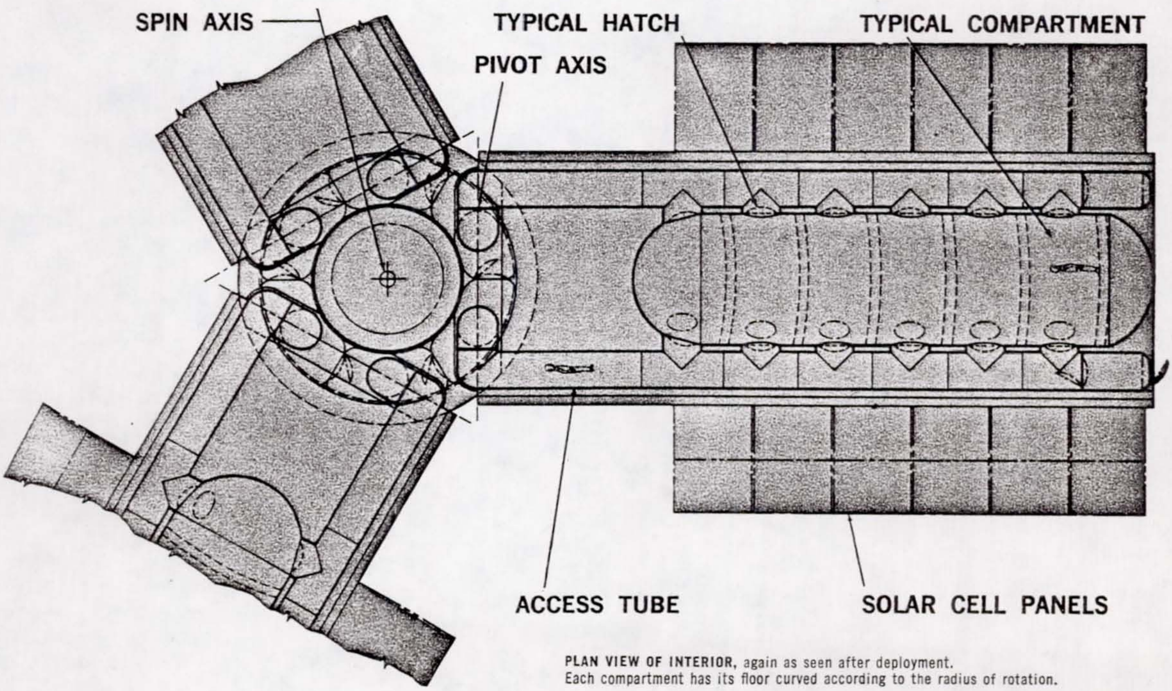
A corollary of the sealing problem is the necessity of conserving atmosphere in the hangar area when dome hatches are to be opened. Exhaust of this atmosphere would compound already complex logistics problems. Complete salvage by pumping the atmosphere to storage tanks would, if carried to completion, require heavy multistage pumps and extensive energy demands over too long a time. Necessary compromises between logistics demands and those of weight, energy, and time must be made.

Another problem area which will require much study by designers is the docking and storage mechanism for logistics spacecraft. The docking ring, which must mate with the forward part of the spacecraft, will have to be extended through the dome-hatch to provide a clear target for the spacecraft pilot. This extensible ring must be capable of taking impact loads, locking on, and then, in retracting, must pull the logistics spacecraft into the hangar to the position of best balance, which is a variable. There must be an auxiliary mechanism to align the spacecraft with the space-station axis. Each of these mechanisms must be demountable so that they may be used to control stowed spacecraft and yet be cycled to succeeding spacecraft.

Despite these major problem areas, it is believed the multiradial-module space station could be developed as a large-capability earth-assembled, and orbital self-erecting space station.

Here two general areas in establishing design criteria for earth-orbiting space stations have been presented. The first area has dealt with the design of a nonrotating space station which could be launched by a Saturn C-1B and which could be used as an interim configuration to provide valuable information on the design of larger space stations of extended lifetime, especially with regard to the desirability of the inclusion of zero-g capability in these larger stations.

The design criteria for a space station of greater capability which could be launched by an advanced Saturn have made up the other area of consideration. A large multiradial-module configuration of the type described here could provide the capability for conducting scientific experiments at various levels below 1 g.



PLAN VIEW OF INTERIOR, again as seen after deployment.
Each compartment has its floor curved according to the radius of rotation.

SPACE STATIONS

Logistics systems and associated operational requirements

BY CHARLES W. MATHEWS

CHARLES W. MATHEWS, who was chairman of the group which developed specifications for the Mercury spacecraft, is chief of MSC's Spacecraft Research Div. Formerly with NASA-Langley, his early work involved air-flight research and automatic-control devices. He later studied orbital manned spacecraft re-entry problems.

The problem of resupply is an extremely important aspect of the operations of a large space-station project. The station would logically remain in orbit for a period of a year or more and would have no re-entry and landing capabilities. The provision of sufficient stores on board for independent operation over a long period of time appears to be impractical and crew rotation will also be desired. Such activities will be done by means of a logistics spacecraft launched from ground bases.

A logistics spacecraft, launched periodically from earth, would rendezvous with and dock to the space station. It would carry crew replacements, scientific and engineering specialists for the conduct of experiments, and cargo for restocking the station and for setting up new scientific experiments. The logistics spacecraft would also provide the re-entry and landing capability associated with both normal and emergency operations. The latter requirement implies that a sufficient number of operational spacecraft would be docked at the space station at any one time to evacuate the entire crew.

To establish the practicality of this mode of operation, calculations have been made for launch requirements as affected by the size of station crew, the logistics spacecraft's transport capabilities, and the crew rotational interval. The results of these calculations are presented in the graphs on the facing page for various types of logistics spacecraft. Crew rotation alone is considered, although similar results would exist for other forms of resupply. Except for stations with small crews, spacecraft presently under development would require an inordinate number of launches even at a three-month rotation interval. This statement applies both to cost and to the more direct aspects of launch support.

It is possible to modify the Apollo spacecraft to provide a six-man capacity. The capacity of this modified Apollo spacecraft appears to provide reasonable operational scope. A 12-man spacecraft would meet such objectives even better, and the economics of the operational phase might justify such a development. The frequent operations also emphasize the desirability for reuse of the spacecraft. This conclusion would also apply to the launch vehicle, although the time period and cost required to develop large, recoverable launch vehicles may preclude their use in early station operations.

A typical logistics spacecraft operation based on a six-man Apollo spacecraft and an 18-man station crew might proceed as follows:

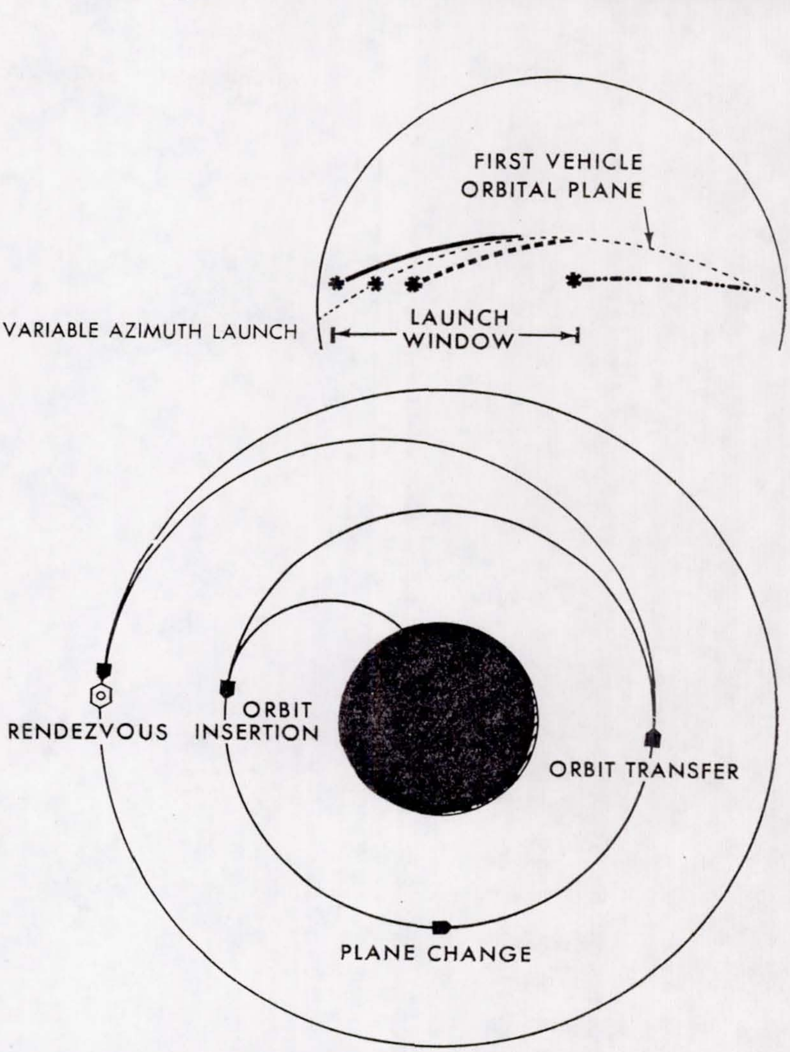
The station and one logistics spacecraft would be launched by a single launch vehicle with a crew in the logistics spacecraft only. Once in orbit, the skeleton crew of six men would transfer to the station, activate the onboard systems, and establish a duty cycle. Two men would remain on duty at all time to checkout systems and prepare to receive additional logistics spacecraft. The skeleton crew of six men is about the minimum required. Application of the station to actual experimentation would be on a limited basis during this time period. Build-up to the full-crew complement would then proceed by launching two additional logistics spacecraft at monthly intervals, or possibly sooner. Another launch would initiate the rotational cycle, and single launches at one-month intervals thereafter would establish a three-month crew rotational cycle.

The dominating factor in the launching of the logistics spacecraft is the time-critical nature of the launch window for the launch operations in order to achieve rendezvous proximity with the station within a reasonably short time period. Techniques for accomplishing rendezvous, allowing for reasonable launch-time windows, are being developed in connection with the Gemini program. Studies in support of Gemini indicate that, in order to limit propulsion requirements and obtain a satisfactory launch window for rendezvous, inclination of the orbit of the station should exceed the latitude of the launch site, but only by a few degrees. For a launch from Cape Canaveral, this situation will provide a launch opportunity on two or three consecutive passes of the station with a fuel expenditure for rendezvous of less than 1500 fps. It also allows operations to take place over the existing and planned Ground Operational Support System network.

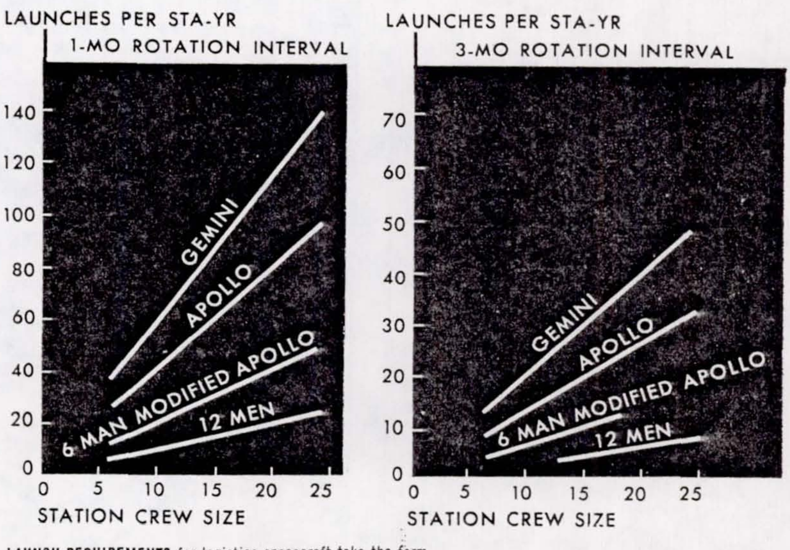
A possible procedure for rendezvous is presented on this page. The station is assumed to be in a 300-n. mi. orbit. The spacecraft is launched on an azimuth which varies as a function of launch time in order to minimize the plane change required on intersecting the orbital plane of the station. This intersection is 90 deg from the orbital insertion point of the spacecraft. The spacecraft is injected into a low orbit (perhaps 100 n. mi.) in order to provide a rapid reduction of the phase differences between spacecraft and station. On achieving the proper phase for Hohmann transfer (180 deg), a velocity maneuver of the spacecraft would produce the transfer up to the altitude of the space station, from which point a closed-loop interception operation would be initiated, ending with a maneuver to match position and velocity of the two vehicles.

With an on-time launch, the time spent prior to docking should be less than 1 1/2 hr. Launch delays of about 20 min will require about 6 hr of orbital operations prior to rendezvous. Even with the low catch-up orbit, approximately one day of orbital operations prior to rendezvous would be required for the least desirable phase relationship at launch. Allowing for this long catch-up would provide a continuous 3-hr launch window each day.

The terminal phase of rendezvous will utilize procedures developed in the Gemini and Apollo programs. They are quite similar to techniques developed in connection with interceptor aircraft. Docking, on the other hand, does present some unique problems, particularly if the operation is performed with a rotating space station. The docking fixture must be near the center of rotation of the station in order to provide a target which does not require extensive maneuvering of the logistics spacecraft and also to avoid disturbing the station after docking. Docking could be accomplished by using a rolling maneuver of the spacecraft to match the angular rotation of the station, or the station could have a nonrotating hub. Either approach appears feasible. Another factor to consider is



LAUNCH AND RENDEZVOUS OPERATIONS with logistics spacecraft can take the nominal form diagrammed above. Azimuth and inclination vary with time of launch. Launch goes parallel with the space-station plane, intersection occurring 90 deg from launch point.



LAUNCH REQUIREMENTS for logistics spacecraft take the form shown by these graphs for one-month (left) and three-month (right) crew rotations.

that all logistics spacecraft at the station must be deployed symmetrically about the hub in order to avoid producing an oscillation in the station. This condition must also be achieved during docking or separation of a logistics spacecraft.

Two types of docking and stowage arrangements for the logistics spacecraft are presented on the facing page. One involves external stowage and the other utilizes a hangar concept. For external stowage, the spacecraft is originally docked along the axis of rotation, but then is transferred to a position at right angles and in line with a transfer tunnel of the station. A connecting tunnel containing an airlock is deployed for the purpose of cargo transfer.

The hangar concept also involves external docking, but the docking ring would be on a retractable arm. Once inside the hangar, cargo, transfer, maintenance, and launch preparation would take place in a protected and environmentally controlled area. All spacecraft would be located along the hub and would proceed on a first-in, first-out basis. Operations in and out of the hangar area would be accomplished by pumping the hangar down, thereby saving the environmental gases.

The logistics spacecraft undoubtedly will be of modular design, with the velocity maneuvering module and the cargo module remaining in orbit. These modules would be sized in accordance with requirements yet to be developed, and they would be of a proper configuration to adapt the command module of the spacecraft to the logistics launch vehicle.

The external configuration of the passenger-carrying crew module would largely be dictated by reentry heating, maneuvering and landing considerations. Of two possible configurations that have been studied in recent years, one is the typical blunt-body configuration presently used in manned spacecraft operations. The other is a winged vehicle designed for hypersonic reentry but having reasonably good low-speed glide characteristics. Both vehicles have hypersonic maneuvering capabilities, with the lift-to-drag ratio (L/D) of the winged spacecraft exceeding that of the blunt spacecraft by a factor of 5. Many variations of re-entry spacecraft exist, most of them lying somewhere between these two extremes.

The chief advantage of the large hypersonic L/D is the sizable lateral maneuvering capability during reentry. The Apollo-type vehicle has a lateral range capability on re-entry from earth orbit of about 200 n. mi. With the winged vehicle, it is possible to increase this range by more than a factor of 10. For both vehicles, downrange maneuverability is more extensive than lateral maneuvering capability but is less important, because control of range to the landing site can be accomplished by timing the retrofire operations, and the downrange maneuvering capability is used only as a vernier control.

A question arises as to the importance and degree of lateral maneuvering during re-entry in the space-station application. One factor immediately apparent is that orbits that pass in closest proximity to a selected landing site may not pass over the site. (The site may be halfway between two orbits, for example.) For orbits of low or moderate inclination, the problem is circumvented by selecting a landing site latitude which is several degrees less than the orbit inclination.

This selection allows the site to be achieved within the re-entry capabilities of an Apollo-type spacecraft. Although this arrangement allows some flexibility in achievement of the landing selection, it does not permit immediate return to the site. The Apollo-type logistics spacecraft may have to stay in orbit for periods up to a day if a single landing site is selected. The acceptability of this philosophy for emergency return operations is the subject of much discussion. A more immediate return is certainly desirable. One alternative is to select a number of properly located and spaced landing sites.

The ability of the larger L/D spacecraft to achieve more immediate return is obvious, but this advantage must be weighed in terms of certain related disadvantages. Perhaps the most important consideration is the effect of increased weight of the high L/D vehicle. Such a vehicle incorporating the same passenger capacity as the blunt-body configuration is likely to weigh more than such a configuration. If this weight difference is put into orbit maneuvering propulsion for the blunt-body configuration, its lateral maneuvering capabilities may be competitive with the high L/D configuration for spacecraft weights now under consideration.

Another problem that has not been solved in connection with the winged spacecraft is the achievement of a satisfactory configuration at the nose of the launch vehicle. Aerodynamic loads during launch produce severe stability and structural problems. These problems can be alleviated through the use of large fins on the launch vehicle, structural strengthening and/or shrouds, at penalties to the payload capabilities of the launch vehicle. Launch aborts are also more complex, particularly in the early stages of the launch, while the spacecraft is in the atmosphere. For these reasons, it appears that the high L/D vehicle, although offering interesting and useful potentialities, will require a longer development time than the blunt body and must also pay some penalties in the form of reduced payload.

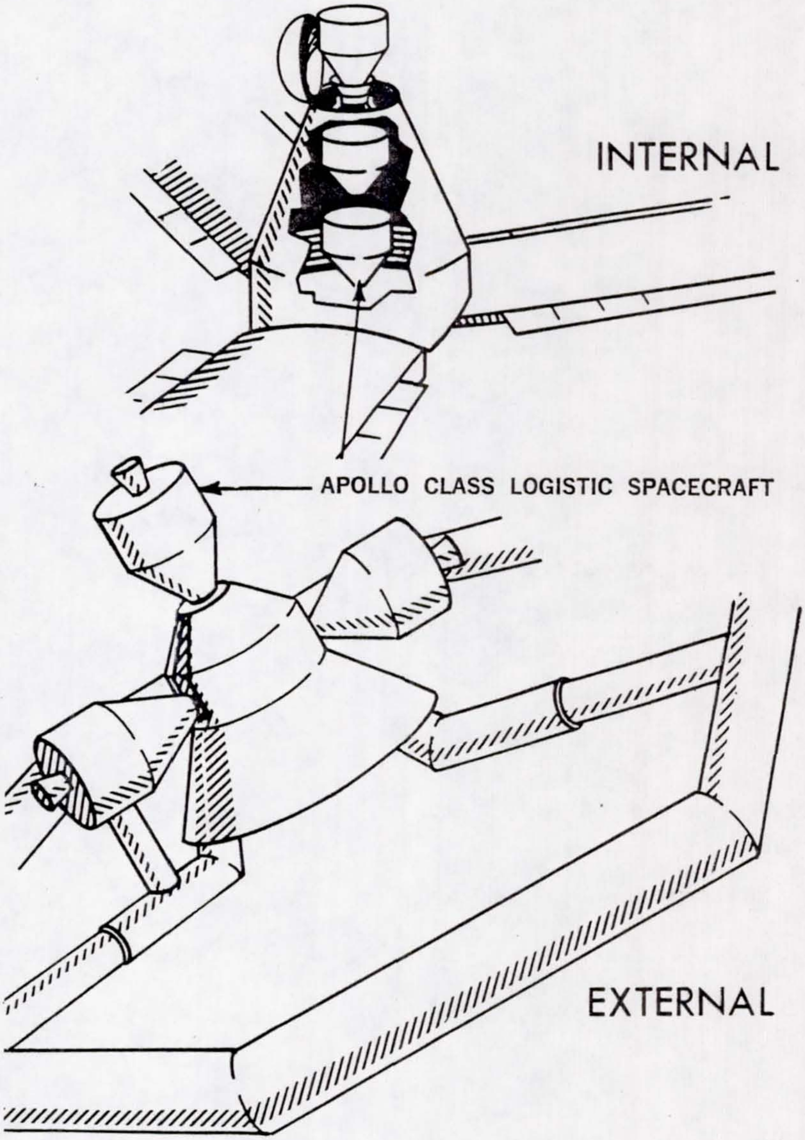
The terminal phase of the landing operation also is strongly influenced by the L/D and, therefore, the capabilities are sensitive to the choice of aerodynamic configuration. In any case, the requirement for re-use of the logistics spacecraft dictates that the landing system employed provide a means for landing with no significant damage to the spacecraft. The requirement for re-use, however, is often confused with a need for rapid turnaround.

Extremely rapid turnaround is not likely to be involved because of the nature of the vehicle, the launch operations, and the probability that the landing and launch areas are not the same. This situation gives rise to a possibility that a restricted area landing concept might be employed, provided the area is adequately limited by hypersonic maneuvering and that capability for windage corrections and obstacle avoidance in the terminal area is available.

The most elementary solution to this problem is a gliding-type parachute, several types of which have been successfully demonstrated. Because the L/D of such devices is low (approximately 1), maneuverability is limited and the ability to execute a landing flare is nonexistent. This characteristic requires the consideration of fairly high vertical velocities at impact (normally 20–30 fps). Thus, special impact alleviation features, such as a shock absorption system or a retrorocket system, are required.

Blunt-body spacecraft can utilize normal aircraft landing techniques through use of the subsonic paraglider. A stowable rotor system presents an alternate possibility. The paraglider can develop a subsonic L/D of 3 or more and has demonstrated a capability of performing a flare maneuver. It is therefore a system with a point-landing capability. The increased L/D also represents a no-wind maneuvering capability at about 20 mi. for normal deployment altitudes. The penalty paid for this increased capability is increased weight and complexity of the landing system, and compromises associated with emergency launch-abort situations involving ejecting the crew in certain flight phases and subsequent loss of the spacecraft.

The comments on the paraglider in general terms also apply to the winged configuration with several notable exceptions. Glide performance of the winged configuration is potentially much greater than that of the paraglider and may have subsonic ranges of several hundred miles. Further, deployment of auxiliary surfaces is more straightforward with an essentially rigid configuration. Lastly, control mechanization would more closely approach that developed for conventional aircraft.



TWO POSSIBLE DOCKING AND STOWAGE ARRANGEMENTS of logistics spacecraft in a space station. A more detailed view of internal stowage appears on page 58.

On the basis of the foregoing considerations, these points may be made:

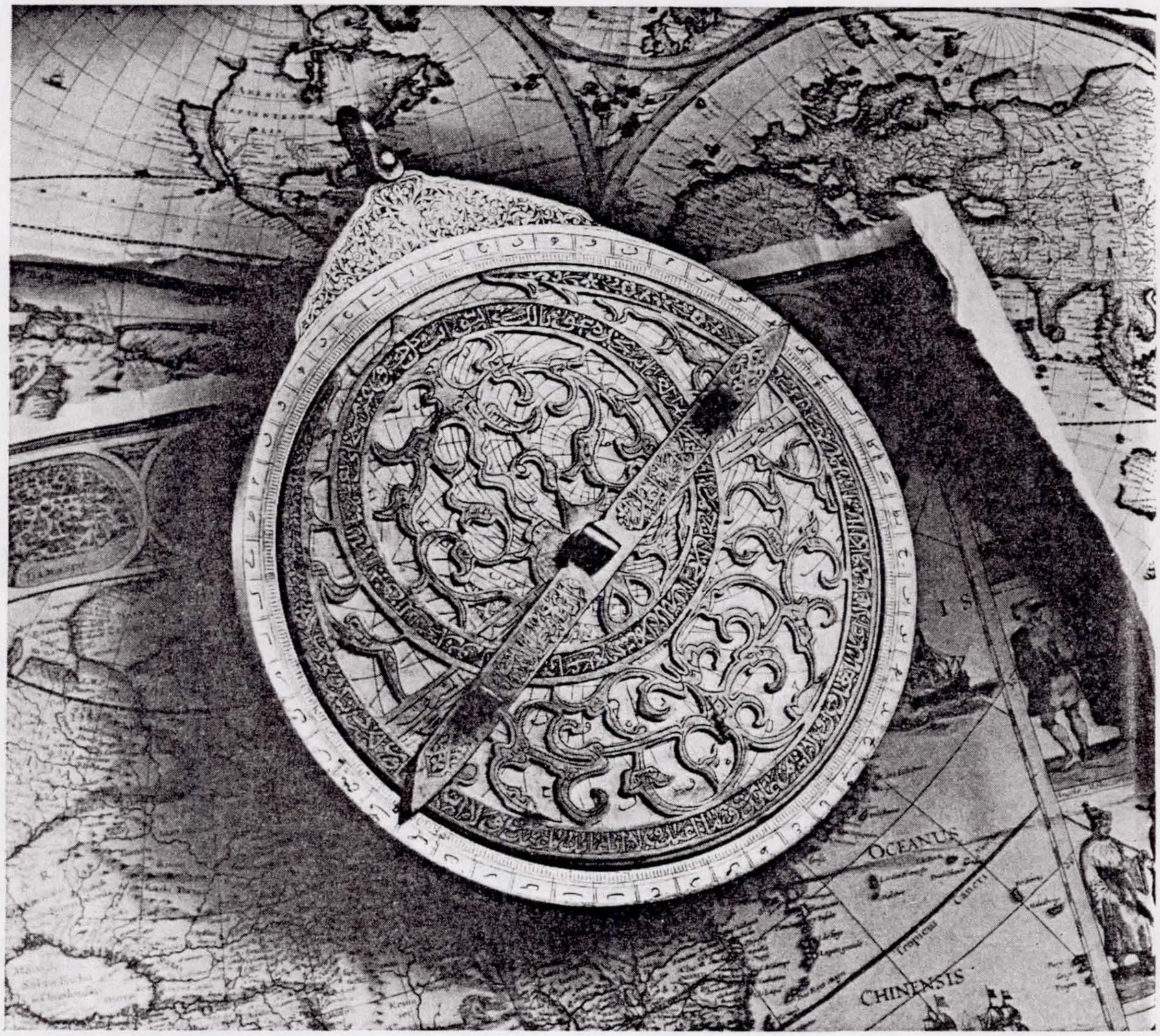
1. Logistics spacecraft will be required to support the space-station program when long-duration missions approaching a year or more in earth orbit are contemplated.

2. The economics of this support may dictate larger transport capabilities than those existing in manned spacecraft now under development. However, it appears that this capability might be achieved by modifications and addition of modules to spacecraft now under development.

3. Launch and rendezvous techniques for logistics support of the space station will closely approach those being developed for the Gemini and Apollo programs. Methods of docking, cargo transfer, and stowage of logistics spacecraft at the station are under study but require further development.

4. The blunt-body configuration of present manned spacecraft appears to afford sufficient operational flexibility to warrant its use as a logistics spacecraft. Winged-configurations and other configurations with higher lift-to-drag ratios may have improved operational flexibility during the return portion of the mission. However, their greater complexity, as well as the higher empty weights, may preclude their employment in the early phases of the program. ••

FROM ASTROLABE TO APOLLO



Astrolabe (1st Century B.C.)—A Greek invention long used by Arabs and Europeans to provide astronomical sightings. The Astrolabe is a predecessor to today's sextant. (Cranbrook Institute of Science)

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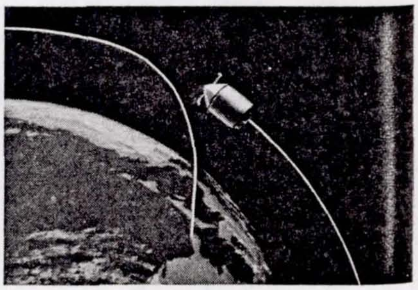

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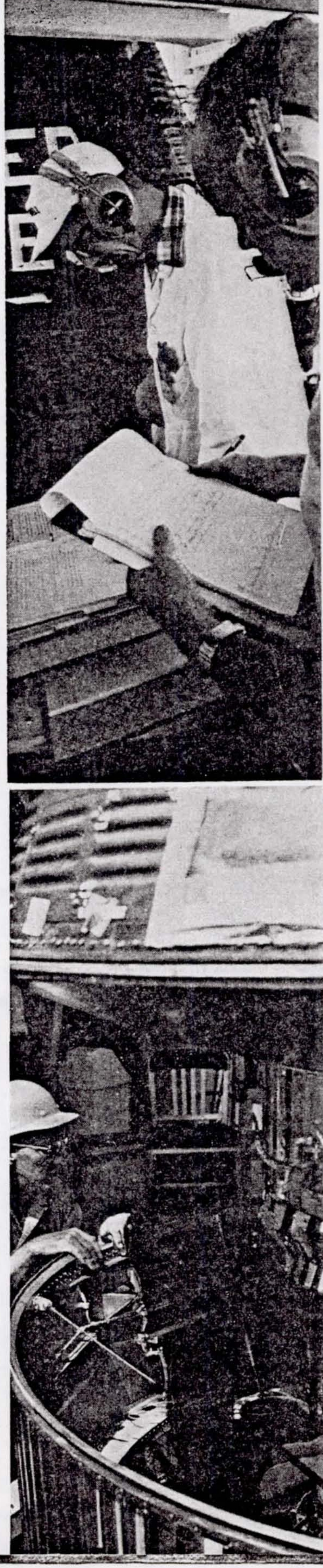
FLIGHT AND GROUND CREW OPERATIONS

Requirements for present and future NASA manned spaceflight programs, covering spacecraft checkout and launch operations; crew preparation and preflight training; flight control and monitoring and recovery for Mercury, Gemini, and Apollo



ASTRONAUTS Glenn (top) and Schirra (below) take part in preflight checkouts of their Mercury spacecraft.

Preflight crew operations

BY DONALD M. CORCORAN

In the early stages of Project Mercury, the MSC Preflight Operations Div. was charged with directing the checkout and launch of the Mercury spacecraft. Test approaches and procedures were to assure the safety of the astronaut and the readiness of flight-worthy spacecraft. What did these approaches and procedures entail, and how will they be extended to ground-crew operations in the Gemini and Apollo programs?

Because of the urgency of the Mercury program, nearly all spacecraft produced were used for flight testing, and few were available for developmental testing in the laboratories until late in the program. The preflight operations conducted at Cape Canaveral thus served not only to prepare a particular spacecraft for flight, but also formed part of its design evaluation.

This examination involved functional testing of the spacecraft systems. Tests, repeated often, duplicated as nearly as possible the different flight environments and modes. During the tests, any discrepancy, no matter how trivial, was scrutinized for its significance. Design changes, stemming from both the flights and these ground tests, were incorporated as rapidly as possible so that the optimum spacecraft configuration was flown. The astronauts participated in all system checkouts at Cape Canaveral and reviewed all design changes. This gave them an intimate knowledge of the spacecraft and its systems.

Mercury checkout thus represents the cumulative experience of launches over the past 2 1/2 years. It embodies six key points:

1. Building-block approach to testing.
2. End-to-end testing.
3. Isolation and functional verification of all redundancies.
4. Interface testing and verification.
5. Mission-profile duplication.
6. Astronaut as an integral part of the system during test.

These points represent a testing philosophy proved in practice. As they will be applied in future programs, let's look at each in detail.

As applied to the Mercury prelaunch checkout, the building-block approach meant that no assumption was made as to the operational status of any equipment or system when the spacecraft was received at Canaveral. The operational status of each system and each component in the system was verified functionally before that system was operated concurrently or in conjunction with another system with which it might have an interface.

This approach grew from an early review of missile-contractor checkout procedures at Cape Canaveral, and the known demands of the Mercury program in terms of time and safety. The review revealed that, at the initiation of most missile projects, the intention was to use only over-all systems test and simulated mission-checkout techniques, but that eventually individual component and system tests had to be used on almost all of these projects to assure flight readiness. Missiles under development clearly required a launch-site test program that established in detail the proper operation of a component or system. Over-all system tests and simulated flight tests that gave only "landmark, go-no-go type" checks did not pay off.

Then, there were other areas studied—potential damage to equipment in transit; the manner in which factory checkout procedures related to actual Cape Canaveral checkout requirements; and calculated risks and potential time lost if functional problems arose when the total system and simulated-mission test approach was used.

It was decided that the quality of performance of all components and systems would be established by individual tests before testing the whole spacecraft system. Practice since underwrites this approach as giving the maximum confidence level in preflight operations.

End-to-end testing means that during testing the initiating function and end function are introduced as they actually would be in flight (minimum of artificial stimuli). End-to-end testing has been used as much as possible. It can be seen clearly, for example, in the hangar-simulated flight test.

For this test, the spacecraft, placed on its adapter, has the escape tower installed and all internal components and wiring configured for actual flight. Test cabling is used only where T-connections to the system can be made. Thus, the process of signal monitoring in no way interrupts the flight wiring which carries the signal. Two-tenth-amp fuses are used as squib simulators. The electrical installation is at the same point of connection as is the actual squib. During the flight simulation, current is delivered to the fuse exactly as it would be to the squib in actual flight. The fuse is sized to experience an actual current slightly over the 3-amp "sure-fire" requirement on all Mercury pyrotechnics.

A launch-vehicle simulator in this setup delivers signals to the spacecraft system in the same manner and at the same place as it would in actual flight. The following example compares a sequence performed during a hangar-simulated flight with the same sequence in actual flight. The sequence selected for this illustration begins with the launch-vehicle-initiated signal of sustainer engine cutoff (SECO), just before spacecraft separates from booster.

The launch-vehicle simulator provides a 28-v DC signal to the spacecraft–launch vehicle interface wiring, as would the launch vehicle during flight. With the use of actual flight wiring, this signal, through relay action, causes a firing voltage to be applied to the main clamp-ring bolts. In actual flight, the bolts would fire and the clamp ring would mechanically separate. In simulation, the fuses used as squib simulators are blown to verify signal delivery and timing. Next, mechanical limit switches sense in-flight separation of the clamp ring and fire the posigrade rockets on separation. Again, during simulation the limit switches are energized mechanically as they would be in actual flight and provide a firing signal to the posigrade-rocket squib simulators.

The test data are monitored by instrumentation pickups, radiated by the spacecraft transmitters, and received and displayed at the ground station as they would be in a flight. Warning lights in the spacecraft cabin, monitoring the sequence, are observed by the suited astronaut and transmitted by him to the ground station by UHF voice link, as in a flight.

This end-to-end test process occurs during launch-complex testing also, the launch vehicle supplying the initiating signal instead of a simulator. RF command, voice, and data reception follow the test procedure similarly.

Such end-to-end testing must be performed to maintain the reliability of the total spacecraft, launch vehicle, and range combination at its highest level.

All redundant signal paths are isolated and proved out functionally by end-to-end tests. These paths include redundancies both between the spacecraft and launch vehicle and within the launch complex. For example, the Mercury spacecraft contains two command receivers which perform the identical functions of responding to RF command to initiate signals for abort, retrosequence start, change of orbital clock time, and R and Z calibration for the flight telemetry system. The RF signal can be generated during an actual operation by either of two low-power transmitters or one high-power transmitter located at the Cape Canaveral command building. The RF signal can be initiated from any of four stations at Cape Canaveral, three in the Mercury Control Center and one on the Test Conductor's console at Complex 14.

To verify redundancies, the ability of each command receiver to receive and react properly to the RF signal is tested *separately*. One receiver is turned off and *all* commands from *all* three transmitters are individually functionally tested. Then the first receiver is turned off, the second one is turned on, and the process is repeated. The whole process is repeated with both receivers on to show that no mutual interference exists.

Other redundant systems are tested in the same manner. This method contributes significantly to the achievement of a high confidence level in over-all system operational reliability.

There are two basic interfaces in Mercury—spacecraft to launch vehicle and space vehicle to range—and they include RF and hardwire links. These interfaces also receive end-to-end testing and testing of all redundancies. They have caused few problems. The spacecraft–launch vehicle hardwire interface transfers only six flight functions and several grounds. The RF equipment aboard the spacecraft was chosen in the design state to match the existing capabilities of range equipment. Interface simplicity should be a design goal.

Simulated mission tests involving the spacecraft, launch vehicle, and range are designed to approach functionally the actual mission conditions as closely as possible. These tests include simulating real-time functions all the way to orbit insertion. The astronaut occupies the spacecraft for these simulations and functions as he would be during the actual flight.

A total mission simulation is not possible during any one test because of restrictions imposed by environment and space-vehicle configuration. After all tests, however, the spacecraft has completed a series which, taken together, approaches total simulation. The life-support systems, including the suited astronaut, undergo a mission simulation in an altitude chamber which duplicates pressure environment and mission time. Mission simulations conducted at the launch complex with the spacecraft, astronaut, launch vehicle, and range operating as they would during flight assure that no procedural or functional interference will be encountered on launch day.

As practiced in the Mercury program, mission simulation includes all predictable abnormal flight modes as well as the normal. Abnormal flight modes include all abort configurations, all manual-override modes, partial power loss, parachute failure, etc.

The astronaut functions during systems tests and mission simulations as he would during the actual mission, to double advantage: The system tested comes closer to flight configuration and the astronaut becomes intimately familiar with the unique characteristics of his spacecraft.

This test experience developed in Project Mercury has proven practical and reliable, and will be extended to Apollo and Gemini.

The checkout and launch operations for the Gemini program are presently planned to be very similar to the Mercury operation. The Gemini equipment and mission will retain enough similarity to Mercury that direct application of Mercury procedures will be possible in most areas. Any new procedures needed will assume the basic tenets of Mercury philosophy.

The evolution from a single spacecraft in Mercury to a number of modules comprising the Apollo spacecraft, on the other hand, increases both the scope and complexity of launch preparation and checkout. New procedures and techniques have consequently been evolved to checkout Apollo and launch it. Notable has been the planned use of automatic checkout for certain systems. Automatic checkout equipment will make use of a pulse-code-modulation (PCM) flight-data system. The supporting ground network will have a digital data-aquisition and -display system built around a digital computer. Data display will utilize an alpha numeric system and cathode-ray tubes.

It is believed that automatic testing can be used without compromising basic Mercury philosophy. But automatic testing will be evaluated on a system-by-system basis. For example, an on-board computer system would be very adaptable to automated checkout; but the value of utilizing automated checkout on life-support systems is questionable.

MSC is now planning checkout and preparation of Apollo spacecraft at Cape Canaveral. This work rests solidly on Project Mercury experience.

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OPERATIONS

Flight control and monitoring and recovery

BY SIGURD A. SJOBERG

What constitutes flight control and monitoring and recovery operations for present and planned spaceflight programs?

The basic functions of the flight control and monitoring team are:

1. To increase flight safety and mission performance by providing the spacecraft crew with ground-based sources of information and data during normal or alternate missions and during mission emergencies.

2. To control and direct unmanned flights required in manned spaceflight programs.

The flight control and monitoring team obtains the information necber on its side and top, will irradiate the test article through ports in the chamber wall. An intensity up to 140 w/sq ft will be available, corresponding to solar-radiation intensity at earth-lunar distances from the sun. The initial irradiated areas will be 13 ft wide by 30 ft high from the side, and 13 ft in diam from the top. Ports allow expansion of solar simulation to areas 25 ft wide by 75 ft high from the side, and 25 ft in diam from the top. Present plans call for a carbon-arc solar simulator with radiation collimated within 1.5 deg and uniform in intensity within 5%.

A variable-speed turntable will rotate ($\pm$180 deg, 0 to 360 deg reversible) test vehicles weighing up to 150,000 lb at rates up to 1 2/3 rpm about the longitudinal axis. The turntable will be cooled with liquid nitrogen to 100 K and will rotate the spacecraft to obtain the desired orientation with respect to solar simulators. For lunar-surface simulation, the turntable may be heated as high as 400 K by electrical elements.

This chamber will be manrated. It will contain three entry locks, one at each of three elevations, through which astronauts and other qualified personnel may enter it under test conditions to perform, for example, simulated extra-vehicular operations in space and on the lunar surface, and work associated with the test in progress. The manrating of the chamber will permit real-time simulation of complete space missions (except the ascent phase) for up to 30 days. To protect men operating in it, the chamber will have emergency repressurization to 5 psia in 30 sec, the oxygen partial pressure being 3.5 psia.

The chamber will permit development and evaluation tests on integrated systems in full-scale spacecraft. Thermal-balance studies will be made with a crew on board and with all systems operating except propulsion and reaction control. These tests, conducted with astronaut participation and for extended periods, will closely simulate the actual space environment.

Chamber B, for astronaut training, is a vertical stainless-steel cylindrical vessel about 35 ft in outside diameter and 43 ft high. It will take the combined Apollo command and service modules. A removable hemispherical cap permits test articles to be inserted into the chamber.

The Chamber B vacuum system will contain mechanical and diffusion pumps able to maintain a pressure of 10^{-4} torr with a gas load corresponding to twice the leak rate estimated for an Apollo command module, a service module, and two space-suited personnel. Pumpdown from atmospheric pressure to the test condition will take about 3 hr.

The cold-wall heat sink provisions for Chamber B are the same as those of Chamber A in concept and performance, but do not involve cryopump panels. The chamber floor will support a 75,000-lb test vehicle. Like Chamber A, it can be cooled by liquid nitrogen to 100 K or heated by electrical elements as high as 400 K.

A solar simulator at the top of the chamber, much like that in Chamber A, will irradiate a 25-sq ft horizontal test area. Albedo simulation will not be provided initially. Ports in the vessel wall allow additional solar-simulator modules to illuminate a 13- by 14-ft side area and a 13-ft diam area from the top.

Chamber B will also be manrated, and will have one double man-lock. Closed-circuit television and windows in the chamber wall at the man-lock will cover operations inside. There will be emergency repressurization, as in Chamber A.

Chamber B will be used mainly to train astronauts, to develop spacesuits and other personnel space equipment, and to develop astronaut techniques for free-space and lunar-surface operation. Capable of continuous operation for 30 days, the chamber will allow astronaut training in simulated real-time missions.

The *Flight-Acceleration Facility*, illustrated on the facing page, will be able to subject an astronaut crew and operating equipment for a command module to accelerations simulating those encountered during the launch and re-entry phases of the Apollo mission. Designed primarily for work with humans, it will have commensurate safety characteristics.

Its rotating arm will initially measure 50 ft from the center of rotation to the center of the gondola. Provisions will be made to shorten or lengthen the arm by 10 ft to increase performance or reduce Coriolis effects.

Of truss construction, the arm will support a 12-ft diam gondola having two degrees of freedom, nominally called pitch and roll. Hydraulic motors will power the gimbal system. The outer gimbal will be able to rotate $\pm$300 deg at a rate that can move and return it 180 deg from a zero position in 3 sec. The inner gimbal will be able to rotate continuously in either direction at approximately twice the rate of the outer one. The payload capacity of the centrifuge will be 3000 lb. Total weight at the end of the centrifuge arm will be approximately 30,000 lb including the spherical gondola, gondola supporting fork, gimbal rings, slip rings, and other control and support equipment.

The facility will be powered by a single, large DC motor with a torque rating of 3.2-million ft-lb. It will be possible to go higher than this for momentary overloads. The payload can be accelerated from 2 to 20 *g* in approximately 6 sec—adequate performance for simulating accelerations of all normal missions. The centrifuge will be able to operate steadily at a level of 20 *g* and up to 30 *g* for short periods.

The flight-acceleration facility will have a comprehensive computer system, a prime function of which will be to insure the safety of test subjects. It will be able to control preprogrammed test runs by com-

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71-80

essary to accomplish these functions from a worldwide network of telemetry receiving, voice communications, and radar-tracking stations, all of which are integrated into a ground operational support system, the focal point of which is the mission control center. The paper on flight operations facilities in this issue (page 87) describes present and planned ground systems for manned spaceflight. This review will cover present and planned operations with these systems, beginning with Mercury control and recovery.

The charts on page 73 describe the Project Mercury flight-support team.

As the first chart indicates, the Operations Director commands mission operations, the Network and Recovery Commanders supporting him. The Flight Director assumes detailed responsibility for the flight at liftoff. The responsibility of his flight-support team can be divided into two areas—flight monitoring and control and support activities. The following paragraphs describe the functions of the positions indicated in the first chart on page 73.

Spacecraft Communicator—communicates with astronaut during powered flight and when the spacecraft is in radio range of the control center.

Flight Dynamics Officer—has over-all coordination responsibilities for the computing complex supporting the mission, monitors the trajectory displays indicating launch-vehicle performance, with aid of computer outputs makes the go-no-go decision on orbit suitability at insertion into orbit, and monitors orbital parameters.

Retrofire Controller—responsible for establishing time of retrofire for both normal and aborted flights so that landing occurs in desired areas.

Flight Surgeon—responsible for all aeromedical aspects of the mission.

Environmental Control Monitor—monitors spacecraft's life-support system.

Launch Vehicle Monitor—monitors launch-vehicle performance.

Spacecraft Systems Monitor—monitors performance of spacecraft flight-control systems, electrical power systems, etc.

In addition, the Network Status Monitor conducts the Mercury network countdown and coordinates network support during operations. The Recovery Coordinator keeps the Operations Director and the Flight Director informed on status of recovery forces and recovery conditions and keeps the Recovery Commander informed on mission status and landing-point predictions. The Support Control Coordinator manages the control-center systems support.

The second chart shows the organization at a remote site. The Spacecraft Communicator acts as the Flight Director for his particular station and is responsible for making decisions affecting the flight if time is not available to obtain direction from the control center. The Spacecraft Systems and Aeromedical monitors observe spacecraft systems performance and aeromedical aspects of the mission, respectively.

The training and simulation activities of this team include familiarization with spacecraft and ground systems through personal study, formal lectures, training time in a spacecraft procedures trainer, team training in operating procedures both within a site and for the entire network, and, for each mission, a series of simulations of both normal and aborted flights.

Two simulation facilities—the complete remote-site simulator linked with the procedures trainer and Mercury Control Center itself linked with the procedures trainer—have been particularly valuable. The latter allows simulating complete missions, launch through landing. Trajectory displays of the launch and of orbital flight are activated by taped inputs to the Mercury computing system. Also, by using taped inputs from the procedures trainer, orbital passes having the proper sequences are simulated at the remote stations.

Simulation of this kind has provided a means of developing good operational procedures between the astronaut and the flight-support team and between the flight-support team and a station's maintenance and operations personnel.

Mission rules were developed for each Mercury mission, from start of countdown through recovery, to establish a course of action for almost any malfunction or anomaly. Simulation of the powered phase of flight has been particularly valuable because, if malfunctions occur then, a decision to continue or terminate must be made very quickly.

For the manned one-day mission, the only significant change will be that two shifts of personnel will be required in the Control Center and at a few of the remote stations. These two crews will operate on about a 10-hr-on, 10-hr-off basis, with an overlap of about one orbit between shifts. Shifts will not be required at most of the remote stations because they will not be in contact with the spacecraft during many orbits.

Substantial changes are being planned and implemented in flight control and monitoring for the Gemini program. Beginning with the Gemini rendezvous missions, the Integrated Mission Control Center (IMCC) near Houston, Tex., will be used instead of MCC at Cape Canaveral. The ground computer complex and the communications center will be a part of IMCC.

For longer-duration Gemini missions, three shifts will be required at the control center and two at the remote sites. Mission organization will be much the same as Project Mercury's. An increase in systems and mission specialists supporting operations room personnel is planned, owing to increased mission complexities, such as rendezvous and docking.

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In the Mercury program, the computing complex is used only for generating trajectory information. In Gemini, however, the computing complex will also be used in performing systems analysis, for example, determining the best way to use on-board propulsion in rendezvous. Fewer remote stations will be used with Gemini (nine PCM telemetry stations, seven having digital up-data links).

Training and simulation activities for Gemini will be much the same as for Mercury. However, simulation capabilities available at IMCC will be substantially expanded over that available in the Mercury program. Two simulated remote sites will be provided in the IMCC. These sites will be tied in with a Gemini spacecraft simulator. The system will allow closed-loop simulations of orbital passes over remote sites. Also, trajectory displays and data will be supplied from computer programs rather than tapes and will be closed-loop, instead of open-loop as in the Mercury program. Actual remote stations will not have a closed-loop simulation capability.

Much planning for Apollo and Gemini is proceeding concurrently. Ground facilities, with few if any exceptions, will be compatible with spacecraft of both projects. The flight-support organization for Apollo will be much the same as for Gemini, but additional system and mission specialists will be required, owing to increased spacecraft and mission complexity.

It is highly desirable to reduce remote-site personnel requirements. The main problem consists of how to send adequate information back to the control center via links not adequate for data-flow requirements. One solution is to provide data processors at the remote sites, so that only processed data will be transmitted to the control center. Such a system is under study for Apollo. Training, simulation, and flight-control activities for Apollo flights will be much the same as Gemini.

Although there has been little specific operational planning for a manned space-station program, some general statements can be made. By far the largest part of the flight-support effort will be involved in the logistic-vehicle operations (launch, rendezvous, and return to earth). Operational techniques developed in present programs, for example, Gemini, will be directly applicable to a space-station program. Flight monitoring will require few, if any, additions to ground facilities planned through Apollo. It is also anticipated that flight monitoring support would not involve a large real-time effort. First, the orbit of the space station would undergo only slow changes. Second, it must be inherent in the space-station design that malfunctions that do occur will not endanger the crew or result in catastrophic failures in a short time period. Voice contact with the station crew must be possible at frequent intervals—perhaps once per orbit—and a ground staff must be ready to advise the flight crew on repairs, flight plans, etc. In essence, the mission scheme should minimize real-time monitoring and control from the ground except during logistic-vehicle operations.

The basis of the recovery planning for Project Mercury has been to provide a positive course of action for all probable recovery situations and to provide a recovery-force deployment which is commensurate with the probability of occurrence. In establishing recovery force requirements and deployment, detailed analyses of the mission are performed to establish the nominal mission landing areas and the probability of landing occurring in other areas. Among the most important considerations are launch-vehicle and spacecraft malfunctions, including the probability and time of occurrence, the desirability of daylight landings and the amount of daylight time available for search and recovery operations, spacecraft on-water endurance including the power available in the postlanding phase, weather, staging areas for ships and aircraft, and communications available. Therefore in planning recovery operations, account must be taken of both nominal and aborted flights. One factor which has dictated the need for a large part of the recovery force in the Mercury program is probability of abort during flight. This factor will continue to be important in future manned space-flight programs.

Based on an analysis of the mission, the following recovery-force requirements were established in Project Mercury. Forces were deployed near the launch site for rescue during the latter part of the countdown and for recovery from aborts early in powered flight. Helicopters, amphibious vehicles, and salvage ships were used to provide rapid access to the spacecraft. Forces were positioned for recovery in planned landing areas for aborts at various times during powered flight and, once orbital flight was achieved, on approximately a once-per-orbit basis. Airplanes for conducting search and location operations and ships, some of which had helicopters aboard, were deployed in the planned landing areas. The probability of landing in planned landing areas was high. Recovery could be accomplished in a relatively short time (maximum of 3–6 hr).

Forces for conducting search and location operations were deployed to strategic locations in case a landing occurred at almost any point along the ground track. A typical search unit consisted of two aircraft equipped to receive UHF-DF signals from spacecraft beacons and to conduct point-to-point and air-to-ground communications. Pararescue personnel (two per airplane) were included to provide on-the-scene assistance, on land or sea.

For the first two manned orbital flights, contingency recovery teams were deployed to 16 locations, so that, after a landing at any point on the ground track, a maximum of 18 hr would be required to locate the spacecraft. No retrieval forces were deployed for the contingency areas because of the low probability of a landing occurring in these areas. However, procedures were available for providing retrieval support if a landing occurred in such an area. Essentially, all recovery forces are provided by the Department of Defense.

Recovery forces for the manned one-day mission have been evolved on essentially the same basis as for Mercury. Planned landing areas for supporting recovery after an abort from powered flight are the same as for Mercury. After orbital flight is achieved, recovery forces will be positioned so that the spacecraft could land in a planned area at intervals of roughly 1 1/2 hr, this being reasonable from the standpoint of the number of recovery forces required and the confidence in spacecraft systems. All planned landings are in daylight in broad ocean areas. By properly selecting such areas (for instance, near the intersection of different orbital ground tracks), one recovery unit can support several areas. Moreover, a recovery unit can move several hundred miles during the course of one-day mission, and thereby support several areas.

Since the orbital ground track for this mission covers much more of the earth's surface than was covered on the previous three- and six-orbit flights, the number of search aircraft and crews required to support contingency landing areas will be increased. Recovery forces required for presently planned one-day missions are probably as great or greater than will be required for longer-duration missions in future programs.

The Gemini spacecraft, launch vehicle, and missions (see page 35) differ considerably from the Mercury, and this will influence recovery-force requirements to some degree. Requirements for the Gemini launch-site recovery force are similar to those for Mercury; but, if the Gemini pilots should land in water after ejecting from the spacecraft, timeliness in rescuing them becomes more critical.

The use of the paraglider on Gemini, with the attendant land landing, eliminates recovery forces associated with one planned landing area in the ocean. Land return would be in the southwest U.S. Many of the areas planned for landing at various intervals after the achievement of orbital flight will, in all probability, be in broad ocean areas.

Although land as opposed to water landings are highly desirable from the standpoint of the much friendlier postlanding environment, the difficulty exists in finding and/or preparing landing areas with a suitable terrain over an area large enough to allow for landings after deviations from nominal in re-entry guidance and control. As confidence in the re-entry guidance and control systems and techniques are developed with experience, additional land landing sites will be used instead of water landings.

For early Gemini flights, contingency recovery support will be similar to that required for comparable Mercury missions.

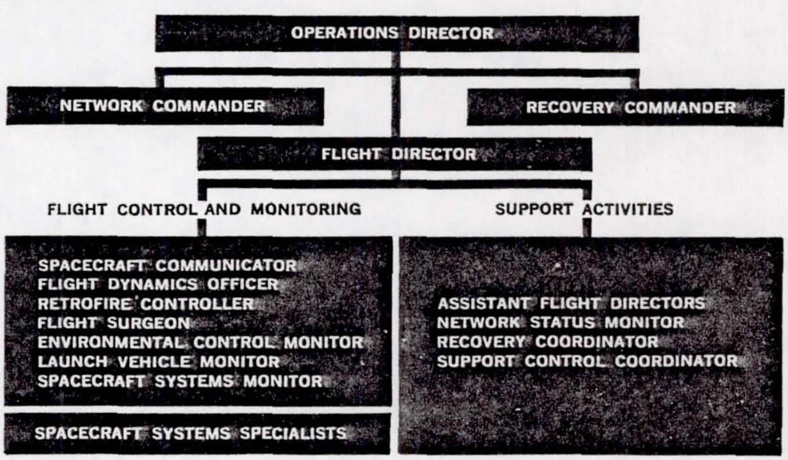
The Apollo spacecraft will be able to make either land or water landings by parachute. For Apollo earth-orbit missions, the basic recovery concepts developed for Mercury and Gemini will be continued. In the time period of Apollo orbital missions, some of the search and recovery equipment developments discussed below might be available.

Most difficult Apollo recovery problem involves malfunctions necessitating abort after translunar injection, primary re-entry guidance and control-system failures, etc. These might cause landings to occur anywhere over a large part of the earth's surface. No radar tracking would be available during re-entry as an aid for landing-point determination, since it would be prohibitively expensive to provide this capability.

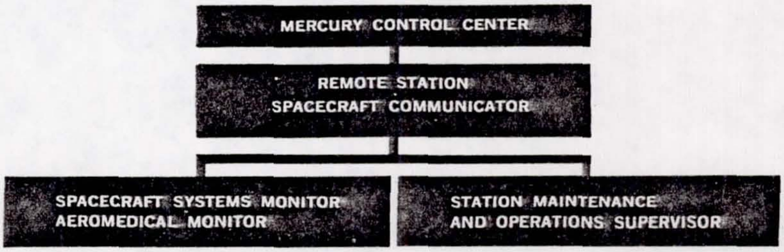
A worldwide HF-DF network for receiving signals from beacons in the spacecraft after landing could be used to determine its location. Such a system is already in existence for a considerable part of the northern hemisphere at latitudes of interest. Extension to areas not now covered, particularly in southern latitudes, would be highly desirable. Both ground and airborne systems could be incorporated into the network. If HF-DF systems were incorporated into high-speed long-range aircraft, perhaps 20 aircraft in all would be sufficient for search purposes after spacecraft landing in a contingency area.

A reliable technique for air pickup of a crew landed in a contingency area would be very advantageous. Both air-snatch and long-line pickup techniques have been developed to some degree. Incorporation of either into long-range high-speed aircraft is desirable for recovery in contingency areas, and

REMOTE-STATION ORGANIZATION



MERCURY CONTROL CENTER ORGANIZATION



NOTE: For outline of worldwide Mercury network, see page 88.

could also eliminate some "lower probability" planned landing areas. The recovery of the spacecraft could be accomplished by surface vehicles at a more leisurely pace after the flight crew had been recovered. Flotation gear would be attached to the spacecraft to insure its flotation if the landing should occur on water and location buoys would be attached as a location aid.

There has been little recovery planning for a space-station project, but concepts developed for the other programs discussed here are directly applicable. Major recovery-force requirements will be associated with logistic-vehicle operations, particularly if launch-vehicle stages are recovered. In a space-station program, contingency recovery forces would be required nearly full-time to support logistic vehicles. Other manned space programs, however, will likely be operational during the same period, and these could engage the same recovery forces.

OPERATIONS

Flight crew requirements

BY RICHARD E. DAY

The Manned Spacecraft Center selects and prepares the flight crews for present and future NASA manned space programs. What are their requirements and how are they prepared for space missions?

"The NASA Astronaut Program" by Walter C. Williams and Warren J. North (*Aerospace Engineering*, Vol. 21, No. 1, Jan. 1962, pp. 13–15) describes the selecting of astronauts in detail. Briefly, the men are selected for obvious physical endowments, a certain age range, and education and professional background that will minimize their training.

Nine new men joined the first seven astronauts in October 1962, early enough to gain full indoctrination and general participation in the engineering and development of the Gemini and Apollo programs. These 16 men, now in various stages of intensive training, will provide the primary and alternate crews for missions in overlapping programs. Additional men will be picked as hardware and program developments demand them.

The general training program for Gemini and Apollo includes science lectures, field trips to government and contractor facilities, systems briefings, engineering development participation, and environmental and operational training.

Astronomy, flight mechanics, computer theory and operation, rocket propulsion, communications, aerodynamics, guidance and navigation, atmospheric and space physics, selenology, meteorology, and medical aspects of spaceflight—these subjects form the science lectures, which are space-mission oriented. For example, the navigational tasks of the complete Apollo mission—from lunar insertion to earth-re-entry energy management —require knowledge of the first seven topics.

System briefings are presented and constantly updated as hardware develops. Systems trainers (see page 82) help develop systems knowledge and reduce training time.

As in the Mercury program, the flight crew will both follow and participate in engineering developments, and educate each other in specific areas.

Centrifuge programs will familiarize the pilots with launch and re-entry acceleration profiles under normal and emergency conditions and allow them to evaluate spacecraft systems, such as controls displays and restraints. They will be familiarized with pressure-suit characteristics and mobility. Each man will be trained to recognize his own symptoms caused by high concentration of carbon dioxide. Their survival training will cover landings in water, desert, or tropics.

The astronauts will gain operational training through a variety of fixed-base and free-flight simulators (see page 78), many simulating Gemini and Apollo missions from launch to near-landing. Instructors will be able to insert malfunctions, and the crew in turn will be able to perform in-flight tests and maintenance operations. Many early flights, moreover, will involve *training*. For example, lunar-excursion-module docking will be tried in earth orbit before a lunar mission. Throughout the program, the astronauts must maintain their flight proficiency in high-performance aircraft.

Specific flight-crew preparation requires practically full-time participation of the primary and alternate crews at the launch site. This preparation may begin three or more months before the scheduled flight, depending on the complexity of the mission. At that time, all spacecraft-engineering and flight-plan changes should be held to minimum, and so permit adequate checking of spacecraft systems, and allow the crew time to acquaint themselves thoroughly with the systems and planned operations.

During this period, the large number of operational checks in the white room, vacuum chamber, and vertical assembly tower, or on the launch complex, require part or all of the crew for participation or observation.

The crew will utilize the mission trainer to practice normal and emergency procedures, guidance and navigation, control-mode switching and tasks, and test monitoring and maintenance. In the final stages of preparation, integrated network simulations will be conducted with all ground and flight crews participating.

From this regimen the flight crew will emerge ready for its mission. ••

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CAPABILITY...17 YEARS OF EXPERIENCE

complete successes in 44 launches. Bullpup—the nation's only supersonic air-to-surface missile, so reliable it is handled like a round of ammunition without pre-flight check-out. Lacrosse—with pinpoint accuracy. All three on duty with Army, Navy, or Air Force.

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SYSTEMS MANAGEMENT. In the past five years, 1957 to 1962, Martin has delivered $3 billion in total contracts. $2.9 billion were completed at or under contract cost. $2.8 billion on or ahead of schedule.

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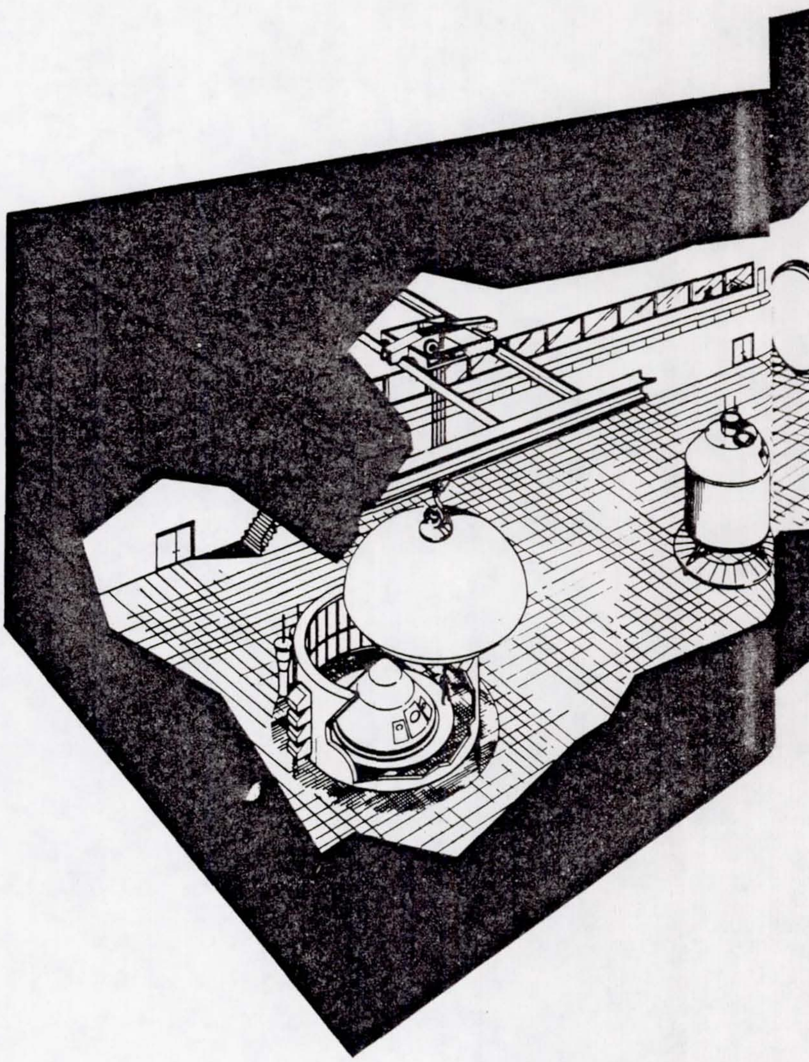
MARTIN MARTIN MARIETTA

FACILITIES FOR MANNED SPACECRAFT DEVELOPMENT

A discussion of major facilities now in planning or design phase for the manned spaceflight program, divided into three sections, covering environmental, acceleration and structural facilities; training and simulation facilities; and the integrated mission control center

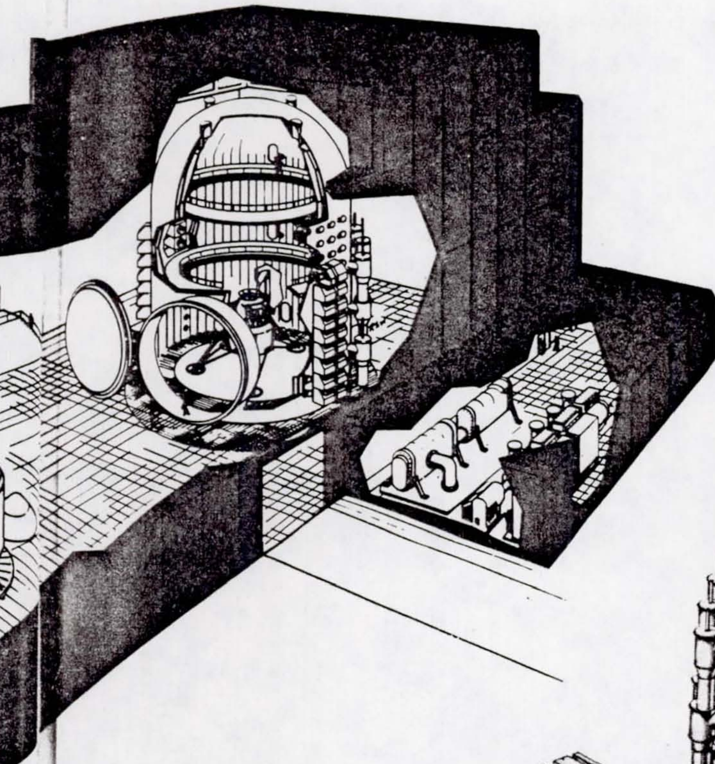


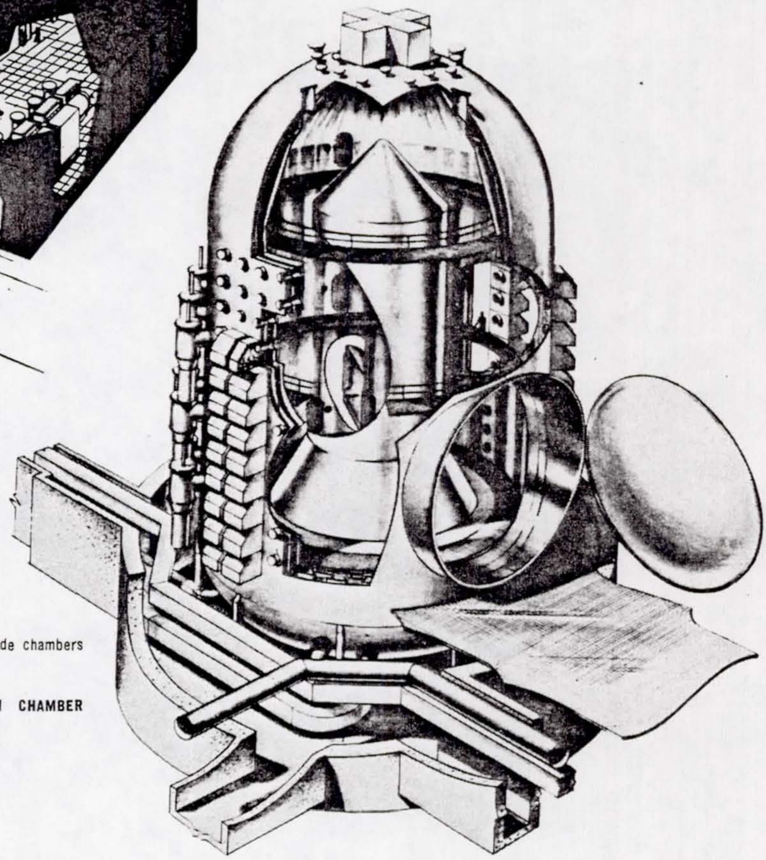
MSC Test Facilities

BY JOSEPH N. KOTANCHIK AND H. KURT STRASS

The Manned Spacecraft Center at Houston, Tex., is constructing a group of facilities essential to its mission of developing and operating manned spacecraft. The facilities, reviewed here, will have capabilities which, in general, do not exist elsewhere in terms of size, performance, manrating of equipment, or work loads. They will also satisfy requirements related to astronaut training, which will be centered at MSC.

The Space Environment Simulation Facility, illustrated above, will contain two space chambers, the larger for space and lunar-surface environment simulation, and the smaller for astronaut training. The two chambers will go in one building and share much auxiliary equipment and service—liquid-nitrogen refrigeration, mechanical pumps, data handling, astronaut preparation, etc. The area between the chambers will be used to prepare spacecraft for environmental tests. The construction schedule calls for partial operational capacity (vacuum only) in September 1964 and complete operational capability in March 1965.





SPACE ENVIRONMENT SIMULATION FACILITY (left) will provide chambers for both spacecraft testing and astronaut training.

SPACE AND LUNAR-SURFACE ENVIRONMENT SIMULATION CHAMBER (right) of this facility measures 65 ft in diam and 120 ft high.

The space and lunar-surface environment simulation chamber, designated Chamber A, shown above right, will produce the vacuum, thermal, and solar-irradiation environment of space and the lunar surface. A vertical cylindrical stainless-steel vessel approximately 65 ft in diam and 120 ft high, with a side access door 40 ft in diam, the chamber will accommodate not only the full-scale Apollo spacecraft, but also larger spacecraft—up to 75 ft high, 25 ft in diam, and 40 ft in base (landing gear) diam. Its hemispherical top will contain four lifting hoists for handling spacecraft modules inserted through the side door. These hoists will also suspend spacecraft for vibration tests.

Chamber A will employ a combination of mechanical, diffusion, and 20-K cryopumping equipment capable of maintaining a pressure of 10^{-5} torr under a gas load corresponding to double the leak rate estimated for an Apollo spacecraft and two space-suited personnel. Pumping down from atmospheric pressure to the test condition will require approximately 24 hr.

The chamber's interior will contain a shroud of black, nitrogen-cooled panels at approximately 100 K to simulate the heat-sink characteristics at the space environment. To the maximum extent practical, all surfaces in the chamber viewed by the test article will consist of such heat sinks. Cryopump surfaces cooled by gaseous helium will be shielded from the test vehicle by the nitrogen-cooled panels to minimize helium-refrigeration requirements.

Solar simulators of modular design, mounted external to the chambining the desired test characteristics with the applicable equations of motion. Another function of the computer will be to take control signals or other inputs from the test subject, combine these with the proper equations and limiting factors of the equipment, and compute, on a real-time basis, the proper input for the instrument panel and the correct signals to the arm and gimbal control systems to give the desired acceleration history.

The flight-acceleration facility will chiefly train astronauts in spacecraft operation. The simulation of spaceflights will be as realistic as possible. The gondola occupants will have full control of its motions (within limits of safety) as in actual spacecraft operation. The spacecraft instrument panel will be duplicated, and inputs derived from control will be displayed for the astronauts' use. Appropriate control equations will be fed into the instrument panel via a closed-loop computer to simulate actual missions more closely. The facility will also be used for physiological testing of crew personnel and acceleration testing of crew equipment and operating systems.

The Thermochemical Test Area will comprise a complex of five test facilities and a central laboratory for evaluation and qualification of spacecraft fluid and power systems. The nature of this testing requires some isolation from other MSC facilities. The five facilities are as follows:

1. A reaction-control test facility to evaluate hot firings of single thrusters or subsystems at sea level and at altitude conditions.

2. An attitude-control test facility to evaluate reaction-control and guidance and sensing subsystems and components on an air-bearing test bed with three axes of freedom, simulating spacecraft pitch, roll, and yaw "hold-limit" cycles, and maneuvers through a closed-loop control system.

3. A space power systems test facility to evaluate a wide range of dynamic electrical-power generation systems and components used in manned spaceflight. Initially, it will allow testing of hydrogen-oxygen fuel cells and dynamic engine-generator combinations utilizing hypergolic fuel-oxidizer mixtures at power levels up to 5 kw. Test-data recording and control instrumentation and equipment will support simulated mission-cycle tests of up to 20 days for fuel cells and up to 48 hr for hypergolic engines.

4. A fluid-components test facility to evaluate components under varying environmental and operating conditions. This will have four cells for testing individual components and subsystems with fuel, oxidizer, gaseous nitrogen, gaseous helium, and demineralized water. There will also be a cleaning system, and a clean room for the preparation and later examination of test items.

5. A pyrotechnics test facility to evaluate the performance of electro-explosive devices after or during exposure to the varying and combined environments of shock, vacuum, acoustic noise, vibration, temperature, and an assortment of gaseous vapors experienced during spaceflight. These devices will include gas generators, separation devices, ejectors, and initiators.

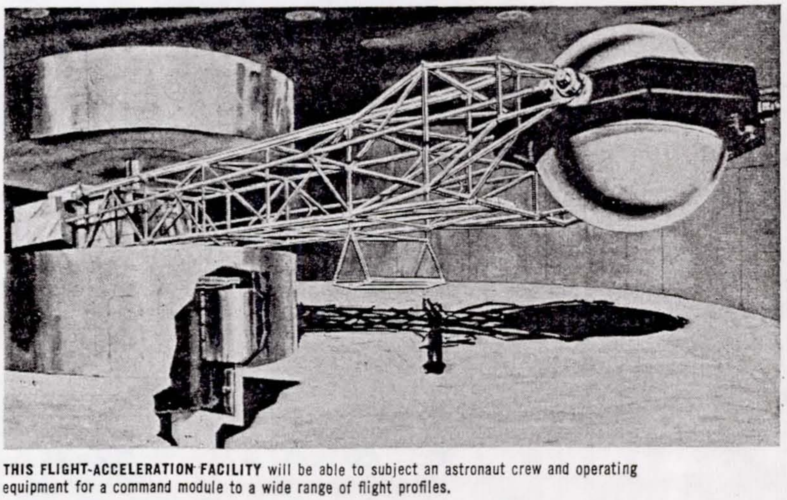
Besides these test facilities, there will be a thermochemical systems laboratory that will have a calibration lab, electronics lab, data-reduction office, and thermoelectric lab. This laboratory will house personnel who conduct tests in the various facilities in this area.

A Structures and Materials Laboratory will cover developmental and evaluation testing of spacecraft materials and spacecraft structural components ranging in size up to complete spacecraft. It will be concerned in particular with materials behavior at very low and very high temperatures. Low-temperature tests will involve cryogenically cooled equipment and vacuum environments. High-temperature tests will primarily be concerned with evaluation of thermal-protection materials for spacecraft, and will employ electric arc-powered equipment and radiant heaters.

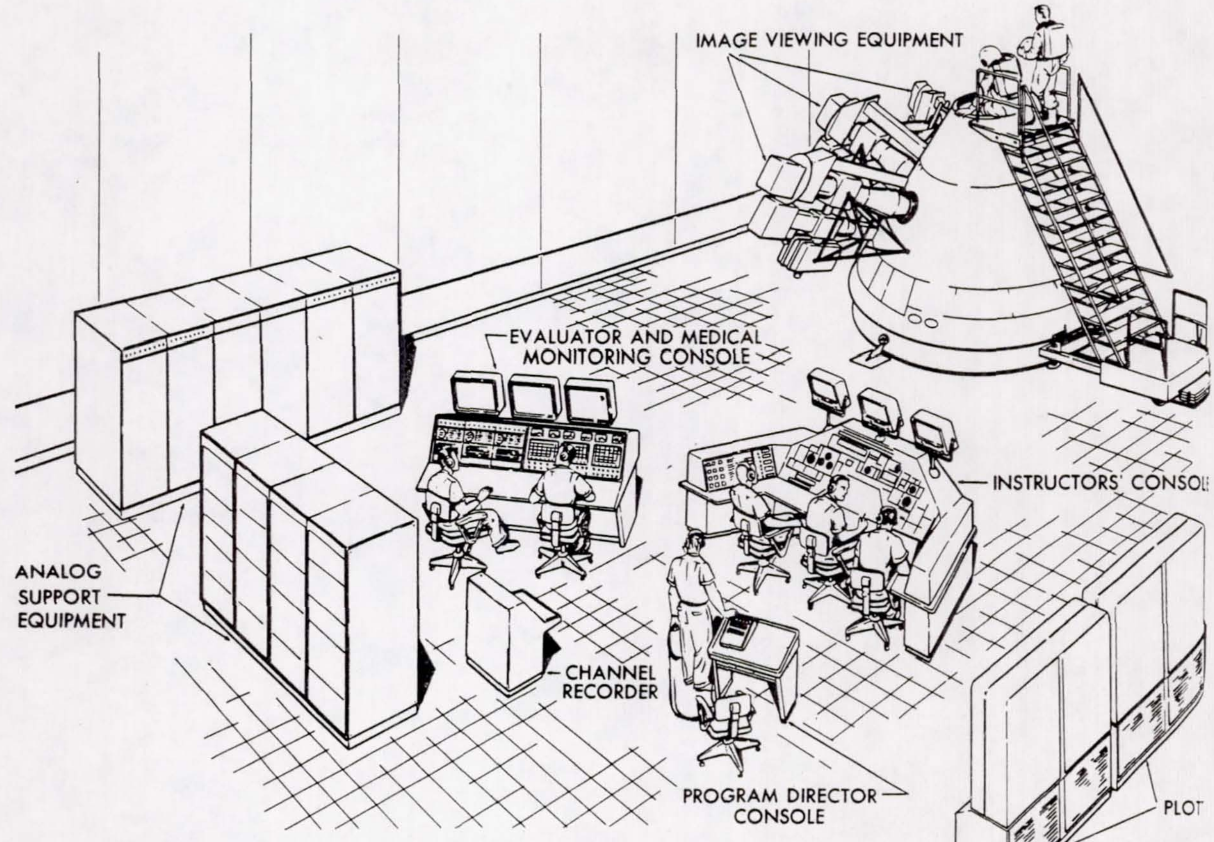
For investigating spacecraft structural problems, the laboratory will be equipped to subject components, modules, and, in some instance, assemblies of modules, to static, acoustic, or vibration loading. Conventional testing equipment will be employed for static loading tests. The type of acoustic loading equipment has not yet been decided on. Vibration test equipment will consist of six 10,000-lb capacity electromagnetic thrusters, which can be operated independently or from a central control, and which will be able to apply sine-wave or random loading to the test article.

The primary function of the Structures and Materials Laboratory will be to conduct tests and studies directly related to problems of manned spacecraft.

As has been indicated, the major test facilities planned for the NASA Manned Spacecraft Center constitute a valuable addition to the capabilities of the country for successful execution of current and future manned spacecraft programs.



THIS FLIGHT-ACCELERATION FACILITY will be able to subject an astronaut crew and operating equipment for a command module to a wide range of flight profiles.



APOLLO COMMAND-MODULE FLIGHT TRAINER will take this form.

FACILITIES

Simulation and training facilities

HAROLD I. JOHNSON, inventor of the ALFA trainer for Mercury program astronauts, is head of MSC's Spacecraft Operating Br. He has been at Langley since '41.

As we move toward future manned spaceflight programs, there will be brought into play a broad range of training and simulation facilities in NASA, other government agencies, and private industry. We use the term "facility" in the broadest sense—from an airplane pressed into use for zero-gravity familiarization to a large moving-base simulator housed in its own permanent building.

Here, we will look only at the facilities for future manned spaceflight projects— Gemini, Apollo, and others as yet unapproved. Knowledge being generated in Project Mercury or in early project-oriented engineering simulations may, of course, require facilities besides those covered here.

Gemini and Apollo will be discussed together at first because of the great similarity of many of the facilities being planned for these two projects.

The sketch on this page shows the approximate configuration of the Apollo command-module flight trainer, an example of one of the proposed facilities. A lunar-excursion-module (LEM) flight trainer for Project Apollo, corresponding to this simulator in functions, will probably be proposed.

The table on the facing page presents the simulation and training facilities proposed for projects Gemini and Apollo. Since items 1 to 5 are almost identical, they can be discussed together. The Gemini and Apollo flight trainers will:

1. Familiarize the flight crew with the appearance and operational modes of all the instrument displays, switches, and control systems in the spacecraft.

2. Acquaint the crew with many expected out-the-window views.

3. Train the crew to detect and correct anticipated systems failures.

4. Acquaint the crew with the dynamic response characteristics of the spacecraft, as shown by both flight instruments and out-the-window displays for spacecraft motions caused either by automatic or manual control systems.

FACILITIES FOR TRAINING AND SIMULATION

	Project Gemini		Project Apollo
	Gemini flight trainers	1	Apollo flight trainers
	Gemini part-task trainer (modified Mercury procedures trainer)	2	Apollo part-task trainers
	Gemini centrifuge simulations	3	Apollo centrifuge simulations
	Gemini egress trainer	4	Apollo egress trainer
	Gemini paraglider trainer	5	Free-flight lunar landing simulator
	Gemini docking simulator	6	Lunar landing facility (Langley Research Center)
	Gemini systems trainers	7	Three-dimensional lunar trajectory simulator
	Miscellaneous training facilities	8	Miscellaneous training facilities
	a. Mock-up trainer		a. Mock-up trainer
	b. Zero-gravity airplane trainers		b. Zero-gravity airplane trainers
	c. Manned aerospace flight simulator		c. Mid-course navigation simulator (Ames Research Center)
	d. Static and dynamic docking simulators (Langley Research Center)		d. Static lunar landing simulations
			e. Static earth landing simulations

5. Familiarize the crew with over-all mission timing and allow crew members to practice tasks in a specific flight plan.

This wide range of objectives makes these flight trainers the most important, and probably the most costly, of the training devices which will be used in either project. Only one factor allowing appreciable cost reductions can affect these flight trainers: The actual physical translational and rotational motions of the spacecraft will not be simulated, even as "washed out motion."

The justification for this omission is based on comments of all the astronauts who have flown the Mercury spacecraft. Apparently, rotational motions in space more nearly resemble a motionless state than a rotating motion in a 1-*g* environment, insofar as body sensations are concerned. At any rate, the astronauts report that display motions alone in fixed-base simulators constitute a surprisingly good approximation of combined instrument and body movements under weightlessness in space. Linear accelerations during launch and reentry can be reproduced satisfactorily as a part-task on the centrifuge and therefore need not be provided in the flight trainers.

Flight-trainer interiors will be replicas of the spacecraft. All interior displays will be activated and will operate as an analog of the actual spacecraft equipment, even though actual spacecraft equipment will usually not be used. Hybrid computers (a combination of analog and digital) will be employed to simulate both on-board systems and spacecraft response characteristics. The long-term navigation calculations requiring low, or no, computer drift will be handled by the digital computer, as will most simulation of on-board systems for which discrete actions or failures require certain discrete responses. Analog computers will be used for short-term high-response applications and for analog instrument readouts where small errors are not particularly important or where the analog readout can be periodically corrected either by the digital computer or by an instructor.

Out-the-window displays for the flight trainers have not been decided upon yet. The Gemini trainer will probably have a fairly accurate visual display of the celestial sphere, a display of the rendezvous vehicle, and a display of the earth, partially covered by clouds. The Apollo trainer will probably have, in addition to the foregoing, a display of the earth and moon from a wide range of distances and a detail display of the known part of the moon from close orbits. The LEM trainer will probably have a celestial display and a display of the moon ranging from close orbits to certain selected landing sites.

Providing good out-the-window displays for these trainers appears to be one of the most difficult tasks. Many display systems are being studied at the present time. Some of the most promising techniques under study include new methods of producing virtual images (which, by nature, have extremely high resolution), high resolution closed-circuit television systems, beam-splitter techniques for combining various scenes easily, and methods of generating scenes electronically which eliminate the need for actual models.

It is expected that two each of the Gemini and Apollo flight trainers will be procured. One of these, for early generalized training, will be located at the Manned Spacecraft Center. The other, for specific preflight training, will be at Cape Canaveral. These flight trainers will be fundamental components of the mission-simulation complexes, allowing all the operational personnel associated with the space mission to train as a team.

The Gemini part-task trainer (Item 2 in the table) will actually be a modification of the Houston-based Mercury procedures trainer. The plan is to reprogram the computer to represent Gemini dynamics and to set up a rendezvous task based on a new out-the-window display system. This system, now being procured, may conceivably be developed into a complete out-the-window display system for both the Gemini and Apollo flight trainers.

The Apollo part-task trainers

actually refer to three possible static part-task trainers: the earth-launch and re-entry trainer, the navigation and trajectory-control trainer, and the orbital and rendezvous trainer. A study under way will determine whether these three trainers can be combined into two, or possibly only one, to reduce costs.

The part-task trainers are similar in many points to the respective flight trainers, except that they cover only a certain portion of a mission. Part-task trainers will be used for one or more of the following reasons:

1. They can be obtained much sooner than flight trainers, and therefore astronaut training can start much sooner.

2. Time available on the flight trainers will be insufficient to train all the astronauts.

3. The particular mission phase needs a more accurate simulation than is provided in the flight trainer.

4. It is necessary to study crew operational problems involving variation in simulation parameters that cannot be accomplished with the flight trainers, because of either their unavailability or their design features.

The Gemini and Apollo centrifuge simulations (Item 3 in the table) refer to possibly two or three different programs for each project, all of which will utilize the U.S. Naval Air Development Center centrifuge at Johnsville, Pa. These programs will probably follow closely the pattern set in Project Mercury—that is, the first program will be a combined engineering-feasibility and astronaut-familiarization program, and succeeding programs will thoroughly indoctrinate the flight crew with the expected mission profile before the manned flights. These centrifuge simulations usually run open loop (programmed accelerations) during the launch phase but closed loop (with the astronaut in the control loop) during re-entries, whether from normal missions or simulated aborts. The centrifuge programs verify the capability of the crew to perform the tasks expected of them during the high-acceleration portions of the mission.

Egress trainers (Item 4) are normally boilerplate mockups of the respective spacecraft having accurate reproductions of recovery equipment and escape paths. Additionally, because of the ever-present possibility of landing in water on the return from any spaceflight, these trainers must be accurately scaled in center-of-gravity location and total weight, so that their hydrodynamic behavior will match the actual spacecraft. In fact, later in the programs, it is probable that actual spacecraft, already flown, will replace the egress trainers. Egress training is necessary because the extremely cramped spaces associated with early spacecraft make escape a difficult crew task which must be practiced a great deal. The egress trainers will be used in conjunction with the recovery forces to determine optimum recovery techniques and then to perfect these.

Item 5 trainers in the table apply to terminal-phase training in each of the two projects. The Gemini paraglider trainer will consist of a boilerplate spacecraft attached to a prototype paraglider. Normal provisions for pilot control of the paraglider will be provided so the crew can practice control during descent and landing after being released from a helicopter. The paraglider will be in the deployed position throughout the operation; the deployment equipment will be qualified in other unmanned test programs.

The Apollo free-flight lunar-landing simulator will probably consist of an accurate LEM mockup to which will be added an automatically controlled jet engine able to produce at all times a pure lift equivalent to 5/6 the weight of the spacecraft. The braking and control rockets of the LEM will be accurately simulated. The astronauts will fly this free-flight vehicle from the ground up to various altitudes and practice attitude control, translational control, and spot landings. Before attempting free flight with the LEM simulator, the astronauts will get primary training in a large lunar-landing facility employing a tethered spacecraft (Item 6). This facility is being procured by the NASA Langley Research Center.

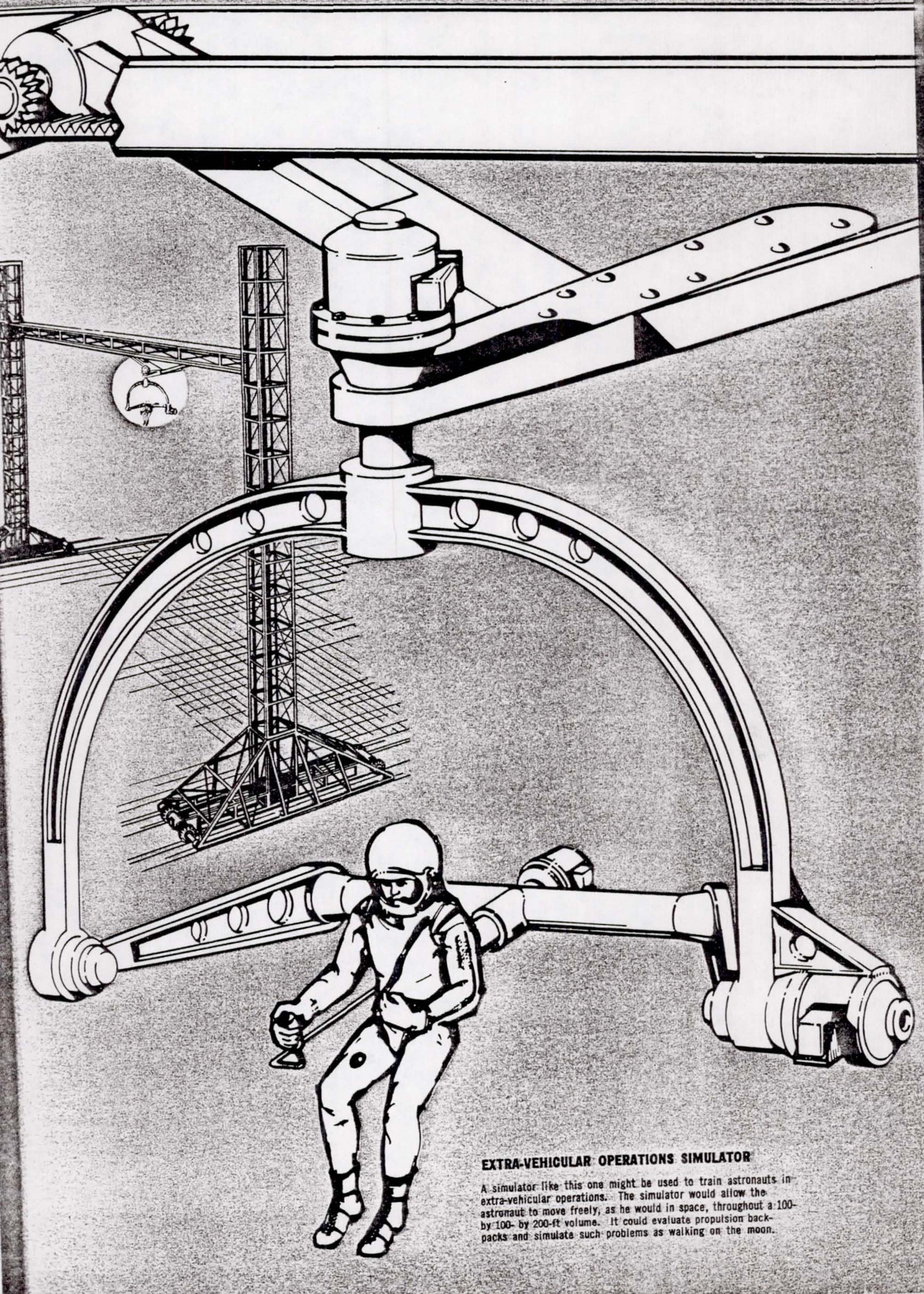
The Gemini docking simulator (Item 6), employing a large six-degree-of-freedom moving base, will be entirely enclosed in its own building or by temporary walls. It will include a Gemini mockup, having four of the six degrees of freedom (rotation ±45 deg in pitch, ±45 deg in yaw and ±45 deg in roll and ±24 ft in lateral translation). The Agena will also be simulated accurately, and will possess the remaining two degrees of freedom in translation (±16.5 ft of vertical and about 102 ft of range translation). The simulator will first be used by the Gemini prime contractor to solve actual hardware problems associated with the docking and latching phases. Later, it will be released for astronaut training for Gemini, and then probably converted to an Apollo docking simulator.

The Gemini systems trainers (Item 7), animated breadboards of some of the spacecraft's actual onboard systems, will give the astronauts a detailed understanding of the systems in as short a time as possible. The systems will be functionally represented by using flow-path indicating techniques to show which components are activated by any particular actual input control.

Item 7, then, covers simulators for the electrical system, including sequencer and pyrotechnics circuits; the environmental system; the ejection seat system; and the attitude maneuver control systems.

Miscellaneous training facilities are cited in the table on page 83. They are either well known or their characteristics are evident from the names.

The remaining Apollo simulators include the three-dimensional lunar trajectory simulator and miscellaneous training facilities. The three-dimensional lunar-trajectory simulator, as yet only in the early proposal stage, will show in true scale, except for spacecraft size, the relative dynamics of the earth, moon, and spacecraft system. The earth model will be about 1 ft in diam, so that a building approximately 70 ft in diam will be required to house the orrery. The earth will rotate about its own axis, and the earth-moon system will rotate about its total center of gravity, which will be placed at the center of the building. The lunar orbital plane will always be horizontal



EXTRA-VEHICULAR OPERATIONS SIMULATOR

A simulator like this one might be used to train astronauts in extra-vehicular operations. The simulator would allow the astronaut to move freely, as he would in space, throughout a 100- by 100- by 200-ft volume. It could evaluate propulsion back-packs and simulate such problems as walking on the moon.

at eye level. Collimated light from the proper direction will fall on the earth to represent the sun as seen from the earth, but an internal light shining past a hemispherical movable shade will represent the sun's illumination of the moon.

As the full-scale spacecraft would be much too large, this facility will probably employ a model in the form of a cone, the apex of which would represent the actual position of the spacecraft. The spacecraft will have three degrees of translational freedom so that the trajectories of actual Apollo missions may be traced. Motions of the "spacecraft" would be in response to computer or taped commands. These commands could also be speeded up by a factor of 1000 to make possible the demonstration of a complete Apollo mission in about 10 min.

Under this setup, if the observer merely walked out to the spacecraft and looked at either the earth model or the moon model, the device would represent an "inside-out" display. That is, the observer would see earth and moon in the same geometric aspect as would be seen by the occupants of an Apollo spacecraft en route to the moon. The lunar-trajectory simulator will be used to educate all types of personnel associated with spaceflight projects. During an actual Apollo mission, moreover, the orrery would be driven to keep pace with the flight and could be viewed by worldwide television to report progress in clear and dramatic fashion.

The last item for Apollo in the table covers other facilities which might be used in the Apollo training program. As before, the mockup trainer refers to the detail mockup used by the prime contractor to decide on the design of the command module or LEM spacecraft. The NASA Ames Research Center mid-course navigation simulator, still in a developmental stage, consists of an Apollo command-module spacecraft mockup mounted on an air bearing in a very large hemispherical domed room with a radius of about 50 ft. Simulated stars will be fixed in the surface of the dome to give the extremely high accuracy necessary for the practice of sextant operations during the mid-course navigation phase.

The static lunar-landing and earth-landing simulations listed in the table on page 83 are temporary setups involving either or both simplified instrument and out-the-window displays animated by general-purpose computers. Several such simulators are currently in use both in NASA and in private industry, and, undoubtedly, there will be many more before Apollo missions are completed. The astronauts normally have opportunities to use some of these simulators on an informal basis, and in so doing increase their control skills and background knowledge of the lunar mission.

Possible facilities intended for training flight crews for future, and as yet unapproved spaceflight projects include: Extra-vehicular operations simulator; spacecraft dynamic support cradle; MSC human centrifuge. If obtained in time, some of these will be used for Gemini and Apollo. Their feasibility and design features are now being studied by industry under NASA contract.

The illustration on page 85 gives an idea of one possible configuration of the extra-vehicular operations simulator. A moving-base simulator, with minimum necessary environmental protection, it would allow a man to move freely within a prescribed large volume (100 by 100 by 200 ft), just as he would move in this same-sized volume in space without restraints. As the mechanism would be computer-driven, it should be possible to simulate partial-gravity tasks, such as walking on the moon, as well as weightless maneuvers outside a vehicle. It would evaluate extra-vehicular propulsion devices, two of which industry now has in advanced hardware developmental stages. Also, flight crews could use it to train for duty inside nonrotating space stations as well as outside a spacecraft—that is, it would allow practice of movements in space generated by muscular power or strapped-on propulsion units.

The spacecraft dynamic support cradle will involve a hollow spherical structure having six steerable driving wheels normal to its surface and spaced 90 deg from one another. The wheels will drive against the inside surface of a hemisphere to give the cradle unlimited angular rotation about any axis without encountering such problems as gimbal lock. The cradle's inner space will accept any spacecraft of reasonable size, including an Apollo command module. The cradle will have on board a gas-turbine engine driving a hydraulic pump. The wheels will be driven and steered by hydraulic motors.

The spacecraft will be integrated with the cradle to the extent that thrust commands from any of the spacecraft control systems can be fed to the cradle computer. The computer, operating on these signals, will cause the cradle to rotate exactly as the spacecraft would in space. This simulator will thus be able to check out spacecraft control systems (including manual backup controls) and to familiarize the flight crew with their dynamic response and acquaint crew members with any idiosyncrasies of attitude-indicating display systems.

The human centrifuge planned for the Manned Spacecraft Center (see page 81) is expected to be about the over-all size of the NADC centrifuge at Johnsville, but will have a somewhat lower maximum acceleration level, not more than 30 *g*. Major differences will be the increase in maximum payload and gondola volume, which will be about three times larger. This size increase will allow at least three men and necessary equipment to be tested simultaneously.

The MSC centrifuge will probably also incorporate an energy-storage device to increase its capability for acceleration-onset. Even with this device, its acceleration-onset rate, fully loaded, may be less than that of the Johnsville centrifuge. The MSC centrifuge would be used primarily for astronaut mission-oriented training, and to a lesser degree for development and qualification of spacecraft equipment, and for certain physiological and medical experiment having a direct bearing on manned spaceflight.

Many of the coming trainers and simulators, it can be seen, derive directly from Project Mercury experience. Others, such as the extra-vehicular operations simulator, and the spacecraft dynamic support cradle, present interesting and challenging developments.

Flight operations facilities

BY JOHN D. HODGE
AND TECWYN ROBERTS

Early in the Mercury design stages, it became evident that this project would require an extensive tracking and data-acquisition network. The presence of man in a satellite demanded that considerably different requirements be placed on the tracking network than had hitherto been necessary for unmanned vehicles. The most significant of these requirements was that it was now imperative that the system respond rapidly to contingency situations to insure adequate safety of the astronaut. The following design criteria were therefore established:

1. A central control facility would be provided.
2. Maximum use would be made of existing facilities.
3. Continuous monitoring of the powered phase of flight would be provided, culminating in a real-time decision of satisfactory insertion into orbit.
4. A minimum of voice, telemetry, and command would be required at the time of retrofire for the planned re-entry.
5. Re-entry tracking would be provided where feasible.
6. A high degree of reliability had to be provided by selection of equipment, redundancy, and diversification of communication links.

The Mercury network, consisting of a number of stations encircling the world, took shape in terms of these aims. The network is shown and outlined on page 88.

Two ships were originally provided with voice and telemetry capability and one of these has since been modified to incorporate a command system. All sites have voice and teletype communications with the control center.

The computing and communications center at NASA Goddard Space Flight Center, Greenbelt, Md., acts as the data-processing system for the network. Centralized control of the entire network during an operation is provided at the Mercury Control Center (MCC) at Cape Canaveral.

The most complex remote site consists of a UHF/HF transmitting and receiving system, an S-band and/or C-band radar, a telemetry receiving system, an acquisition system, and a command system. Radar data are automatically transmitted to the Goddard computers, where they are used to determine the orbit ephemeris. The spacecraft air-to-ground voice communications during contact with a network station may be monitored at MCC via the ground communication links. At a remote site, telemetry data are displayed to three flight controllers who prepare summary messages for transmission to MCC after each pass. Commands are sent by the flight controllers, as required, to reset the onboard clock and to start the retrofire sequence.

MCC's two basic functions are to provide mission control and to monitor the powered phase of flight directly. Downrange sites are used to extend the receiving range of the instrumentation facilities at Cape Canaveral. In particular, high-speed radar data are passed from Bermuda to Goddard computers to confirm initial cutoff conditions.

The role of the Bermuda station has changed somewhat since the program began. Since no high-speed data lines were available there until recently, Bermuda required an IBM 709 computer to perform the necessary calculations for orbit determination. In this respect, it acted as a back-up control center. This situation tended to complicate the flight control operations, and led to development of extensive command handover procedures. These have been simplified considerably since high-speed lines have become available.

The drawing on page 89 shows a layout of MCC's control room. Each console position shown has communications control (112A key equipment). The monitoring positions for the aeromedical and spacecraft systems utilize meters and eight-pen Sanborn recorders for display of significant parameters, since these are primarily analog quantities. The retrofire and flight-dynamics consoles employ projection-type digital readout and the four X-Y plotters for analog quantities. The consoles for the Flight Director and Capsule Communicator are equipped with closed-circuit television for monitoring spacecraft activities during the countdown and the initial phases of powered flight.

In addition to these two positions, the front row of consoles has re-

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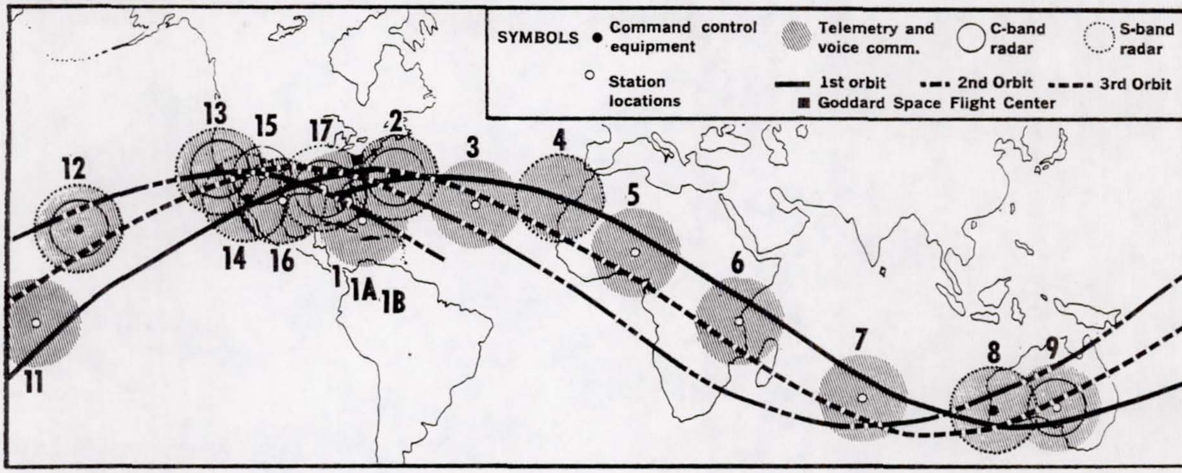
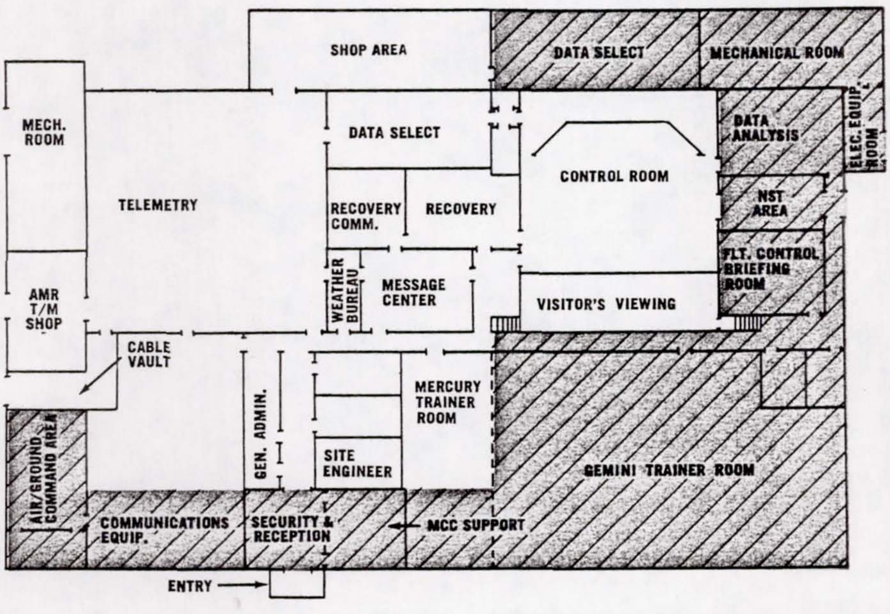
WORLDWIDE PROJECT MERCURY NETWORK STATIONS



GROUND COMMUNICATIONS IN NETWORK

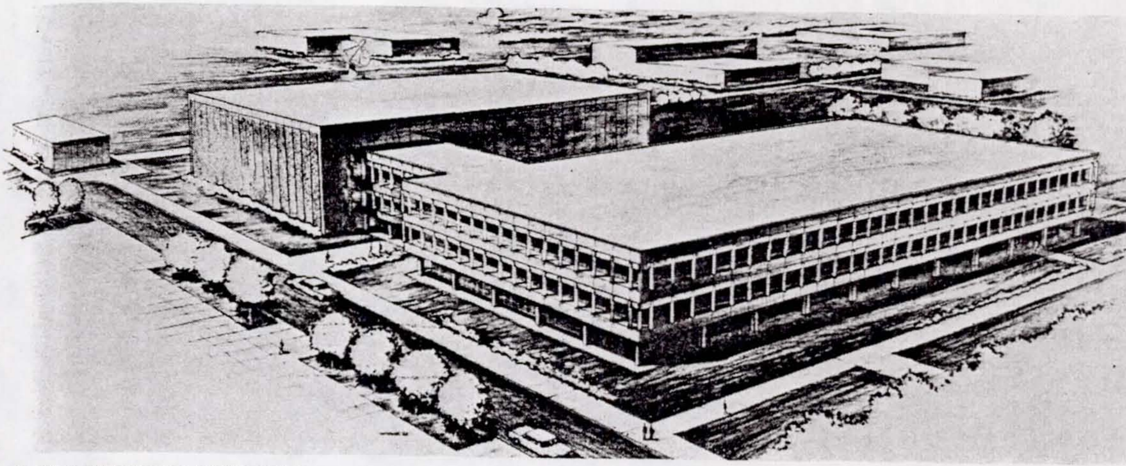
Station code	Station number	Name	Orbital-pass coverage	Radar S band	Radar C band	Telemetry reception	Communication Spacecraft	Command control	Acquisition* FA	Acquisition* SA	Acquisition* M	Ground communication Voice	Ground communication Teletype
CNV	1a	Canaveral	1, 2, and 3	(x)	x	x	x	x	x	(x)		x	x
GBI	1b	Grand Bahama	1, 2, and 3		(x)	x	x				x	x	x
GTI	1c	Grand Turk	1, 2, and 3			x	x				x	x	x
BDA	2	Bermuda	1, 2, and 3	x	x	x	x	x	x			x	x
ATS	3	Atlantic Ship	1, 2, and 3			x	x		x				x
CYI	4	Grand Canary Island	1 and 2	x		x	x		x			x	x
KNO	5	Kano, Nigeria	1 and 2			x	x			x		x	x
ZZB	6	Zanzibar	1 and 2			x	x			x		x	x
IOS	7	Indian Ocean Ship	1, 2, and 3			x	x		x			x	x
MUC	8	Muchea, Australia	1, 2, and 3	x		x	x	x	x			x	x
WOM	9	Woomera, Australia	1 and 2		x	x	x		x			x	x
CTN	11	Canton Island	1 and 2			x	x			x		x	x
HAW	12	Kauai Island, Hawaii	2 and 3	x	x	x	x	x	x			x	x
CAL	13	Point Arguello, Calif.	2 and 3	x	x	x	x	x	x			x	x
GYM	14	Guaymas, Mexico	1, 2, and 3	x		x	x	x	x			x	x
WHS	15	White Sands, N. Mex.	1, 2, and 3		x				x			x	x
TEX	16	Corpus Christi, Tex.	1, 2, and 3	x		x	x		x			x	x
EGL	17	*Eglin* Air Force Base, Fla.	1, 2, and 3	MPQ- 31	x				x			x	x
		Goddard Space Flight Center		Ground Communications									

*Site functions: FA=Fully Automatic SA=Semiautomatic M=Manual

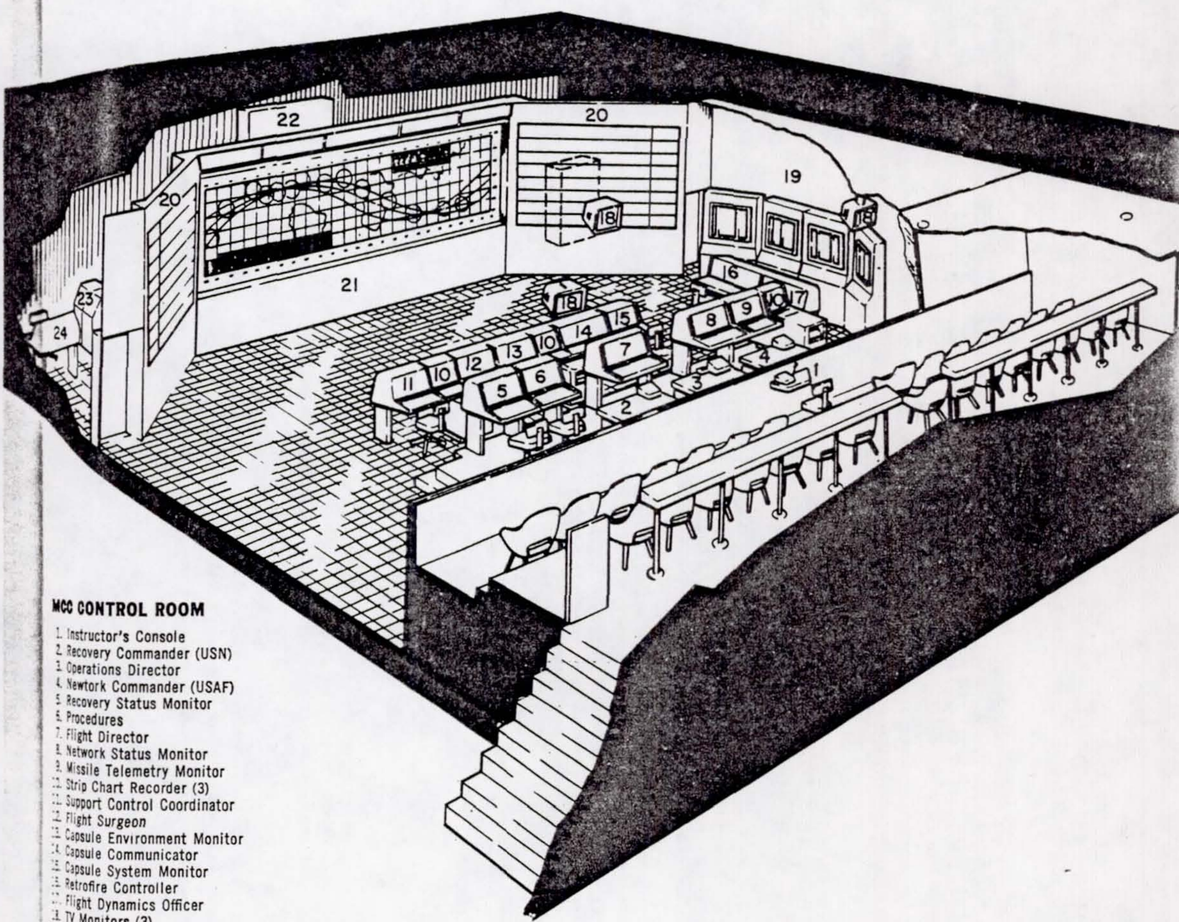


MODIFICATIONS PLANNED IN MERCURY CONTROL CENTER

(Additions are shown in color)



INTEGRATED MISSION CONTROL CENTER (IMCC) *proposed for Houston.*



MCC CONTROL ROOM

1. Instructor's Console
2. Recovery Commander (USN)
3. Operations Director
4. Newtork Commander (USAF)
5. Recovery Status Monitor
6. Procedures
7. Flight Director
8. Network Status Monitor
9. Missile Telemetry Monitor
10. Strip Chart Recorder (3)
11. Support Control Coordinator
12. Flight Surgeon
13. Capsule Environment Monitor
14. Capsule Communicator
15. Capsule System Monitor
16. Retrofire Controller
17. Flight Dynamics Officer
18. TV Monitors (3)
19. X-Y Recorders (4)
20. Trend Charts (16)
21. Operations Summary Display and Alphanumeric Indicators
22. Signal Distribution Panel
23. Teletype Printers
24. Data Entry Console

cently been equipped with small portable television monitors. These monitors are used to display teletype summary messages received directly from the communication center, and to display trends and analyzed data derived in the support areas outside the operations room.

The world map in the front of the room displays a network status summary. It has a computer-driven replica of the spacecraft indicating in real time the spacecraft's position on the ground track. The recovery-operations area is adjacent to the main operations room and is separated by an observation window. This area has direct communications with the recovery forces via military communication channels, completely independent of the Mercury network communication system.

MCC and the Mercury network have very adequately supported Project Mercury. The manned one-day missions will utilize the Mercury spacecraft, and very few modifications are required to support these extended missions. Some changes are needed in the computing program, and the ship positions have been changed to allow at least one contact per orbital pass.

Extensive modifications to the network will be required, however, to give adequate support to the Gemini and Apollo projects, which are now in the design stage. The most significant difference between these projects and Project Mercury is that in both projects, Gemini and Apollo, the spacecraft has an onboard propulsion system besides the normal retrofire system. The orbital ephemeris may thus be modified, because there will be several powered-flight phases.

The Gemini rendezvous task, moreover, will require the capability of monitoring two vehicles, and the Apollo lunar flights will require a lunar-range capability.

Flight-operations facilities for the Gemini and Apollo programs will be provided in several phases—the first for support of the Gemini orbital flights, the second for Gemini rendezvous and Apollo orbital flights, and the third for Apollo lunar flights.

The demonstration of man's capabilities in Project Mercury has allowed some relaxation in the extensive coverage of the Mercury network.

Consequently, planned MCC network modifications for support of the Gemini program, shown on page 88, will include the implementation of PCM telemetry and digital command systems at Cape Canaveral, Bermuda, Grand Canary Island, and Hawaii (Sites 1, 2, 4, and 12) and telemetry only at Guaymas, Mexico (Site 14). Telemetry and command will be provided at one site in Australia and on one ship to be stationed in the South Pacific. Powered flight will continue to be monitored at MCC, which will remain the central facility for command of the mission.

MCC modifications will be limited to what is needed for Gemini orbital command. The present building will be expanded to house a Gemini flight-crew trainer and associated simulation equipment. The systems modification will include the addition of a PCM telemetry receiving station and a digital command system. The operations room, shown on page 89, will remain basically the same, although the Gemini spacecraft, being more complex than Mercury, may require addition of one or two positions to monitor the mission.

The computer programs and flight dynamics displays will be modified to allow for the differences between the Mercury and Gemini launch vehicles.

The major change for future missions will be a new Integrated Mission Control Center (IMCC) at the Manned Spacecraft Center in Houston. Sketched on page 89, IMCC will essentially combine the functions of MCC and the Goddard Computing Center and will provide communications control during the operational periods.

IMCC's prime function will be to direct operations in manned spaceflight programs in all aspects of ground support, from the beginning of countdown through flight operations to recovery. This centralized operation will maximize both the safety and technical achievements in each flight.

During preflight operations, the IMCC functions will be primarily concerned with planning, simulation, testing, and checkout. During the mission, the primary functions will be direction of the ground operational support system, determination of mission status, prediction of contingencies, and determination of the best contingency response.

The IMCC will consist of four major systems: Real-time computer complex, communications, display, and simulation, checkout, and training. A fifth, the operational instrumentation system for RF receiving, may be added at a later date. These systems will be housed in IMCC's mission operations wing. The support wing will contain office and laboratory space.

The mission operations wing will be a three-story building, the first floor containing the real-time computer complex and the communications system, and the second and third floors, essentially the same, consisting of a central operations control room surrounded by support staff rooms.

To provide flexibility, in view of the stringent schedules which must be met, IMCC will be designed to support a mission and a simulation simultaneously. The systems will also be designed for rapid changeover from one mission to another.

The later stages of the Apollo program will require a ground operational support system with lunar-range capability. It is presently envisaged that a complex of three sites spaced equally around the globe will form the basis of this capability. These three sites will operate at a single frequency in the S-band range. In general, all information flow between the spacecraft and the ground will be provided by modulating S-band.

This is the present picture of available and planned facilities for the operational support of manned spaceflight programs. Future facilities will emphasize the flexibility required to support the variety of planned missions. Some compromise will, of course, be necessary, and some design requirements will be placed on future spacecraft. The concept of central control has been emphasized. The present limitation of this concept is associated with available communications bandwidths between the remote sites and the Mission Control Center, and this limitation will remain until the advent of suitable communication satellites. ••

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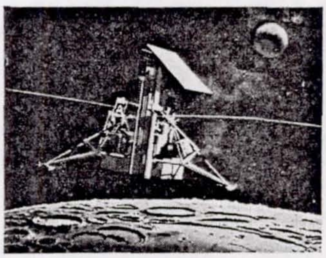

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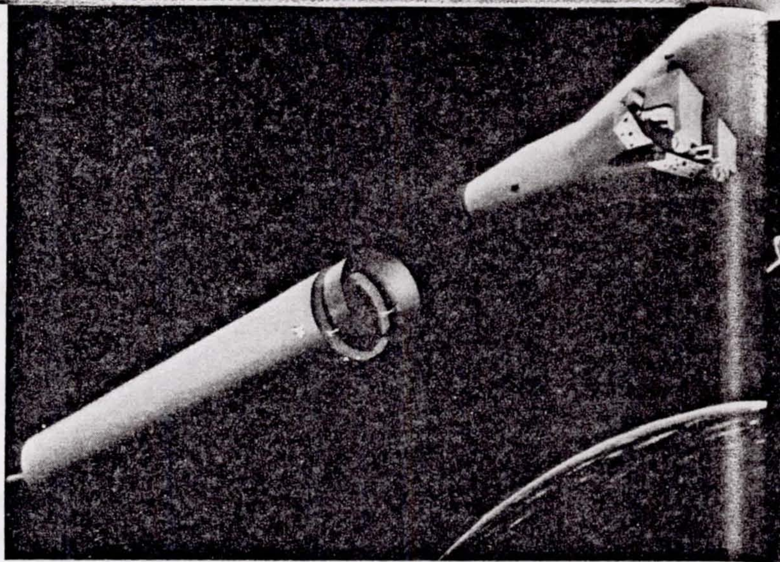
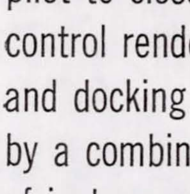
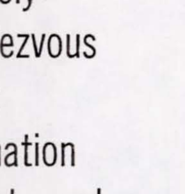
Visual capability in rendezvous

BY JACK E. PENNINGTON
AND ROY F. BRISSENDEN



Both research background and recent experiments strongly back the capability of a human pilot to closely control rendezvous and docking by a combination of instruments and visual sightings





J. E. PENNINGTON is an engineering physicist in NASA Langley Research Center's Aerospace Mechanics Div., now engaged in simulation studies.

R. F. BRISSENDEN an engineer in Langley's Flight Research Div., has done extensive work in automatic tracking and simulation systems.

NASA Langley Research Center's studies of man's capabilities in space operations have, among other things, covered piloted space-rendezvous maneuvers, including the principal human factors involved. In particular, it has been demonstrated that efficient visual rendezvous can be performed, and the conclusion reached that visual rendezvous make maximum use of man's capabilities, thereby tending to minimize system requirements and increase the probability of mission success.[1,2]

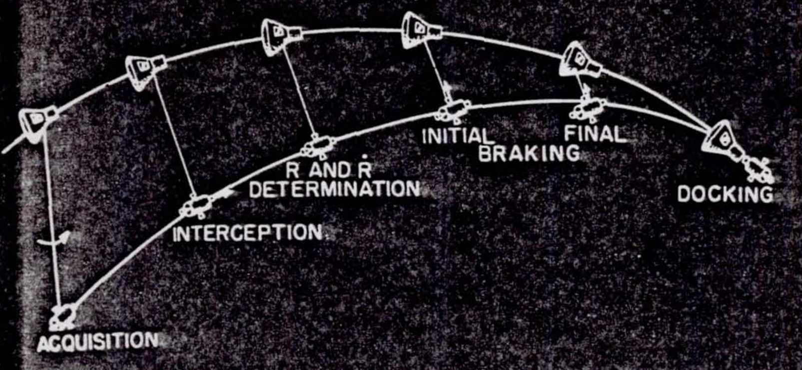
An efficient rendezvous in space requires the performance of the visual tasks with precision not generally required in everyday living. While many studies of visual perception have been made, they have often been general in nature, and do not pertain directly to the rendezvous situation. For example, object perception has been determined for size and shape of the object, lighting, color, and range; but these studies concerned objects relatively close to the viewer, so do not apply directly to the visual rendezvous problem.[3-5] Other basic studies of visual acuity concern themselves with the mechanisms by which the eye performs.[6,7]

Our purpose here will be to summarize these previous studies, as applicable to visual rendezvous, and to present the results of several experiments made at Langley to fill gaps in our knowledge. These results support an active role for the pilot in rendezvous maneuvers.

The sketch at the top of page 97 depicts the various phases of a visual rendezvous operation, as presently conceived. Control data needed for the maneuver are line-of-sight angles and range, and the rates of these. The sketch at the top of page 97 depicts a constant-bearing intercept along which a schedule of braking can be made. The pilot must first acquire the target, perhaps at distances of 200 mi. or more, and then attain the path of intercept by arresting the angular motion of the line of sight, seen as the motion of the target against the star background used as an inertial reference. Before braking can begin, the pilot must know the range and range rate along the intercept course. These can be derived through visual information.[2] The braking operation has been investigated.[1] It proceeds until closure speed is slow enough, by the time the shape of the target becomes visible, so that the pilot can complete the docking maneuver visually.

The table on page 97 outlines specific important visual tasks associated with various rendezvous phases. Let us consider each task separately.

The first objective is to acquire and recognize the target. Sun-illuminated or carrying a beacon, the target must *emit light* the pilot can spot. How intense the light must be will be determined by the separation distance at which acquisition



PHASES OF RENDEZVOUS	VISUAL TASKS	TARGET PARAMETERS
Acquisition	Detection	Intensity, color signal sequence
Establishment of intercept	Angular-rate discrimination	Motion cues
Range and range-rate estimation	Angular-rate discrimination	Motion cues
Braking operations	Distance and closure-rate judgment	Size, shape, motion cues
Docking	Attitude	Aspect

GEMINI-AGENA RENDEZVOUS and docking phases and, right, nature of task.

is required (inverse square-law relationship) and the physiological threshold of visual perception, this being affected by such factors as night adaption, background brightness, and color of received light.

Given sufficient time, it is possible to see an isolated 8.5-magnitude star,[8] equivalent to 1 candle at about 13 mi. However, in a field of stars the threshold of perception is closer to a fifth- or sixth-magnitude star, fifth magnitude being equivalent to 1 candle at 4 mi. Moreover, if the time for dark adaptation is limited, the brightness of the target must be increased.

The graph on page 98 shows how the visual threshold tends to vary with the time allowed for the eyes to adjust to darkness.[9,10] Because pre-exposure brightness strongly influences the situation, data of this nature are difficult to interpret. If we accept the data in the graph on page 98 without qualification, about 10 min or more of adaptation time would be required to see a fifth or sixth magnitude source. More definitive tests appear desirable.

For the acquisition task, both the color of the target and contrast with the background are important considerations. Baker and Grether have studied the effect of background illumination on the required intensity of colored signal lights at long distance.[11] The graph on page 98 shows that, for positive identification, yellow requires the greatest intensity and white the least, with red or green about as effective as white. Then, in studies at Langley, Martin showed that various treatments of the target surface, causing it to diffuse and reflect incident light, can increase the definition of the target surface and outline.[12]

Although the target may be bright enough to be visible, the pilot may not detect it against the star background—particularly if its motion is very slow, as at the initiation of rendezvous. A flashing light, much more easily detectable than a steady one,[13] will lessen this problem. But we must establish the optimum flash rate and flash duration.

Around 1911, Blondell and Rey studied the effect of flash duration on the apparent intensity of a light seen by the human eye.[14] Their results are summarized in the graph on page 98, taken from Baker and Grether's report. In this graph a steady light, just barely discernible, is used as a datum reference, with a relative intensity level of unity. The graph shows that little increase in relative intensity is required down to flash durations approaching 0.2 sec. For flash durations less than 0.1 sec, however, the required relative intensity increases as an inverse function of time.

For example, if the flash duration is about 0.003 sec, the intensity relative to the steady light must be increased by a factor of about 100. The Blondell-Ray relationship can be closely approximated by the equation known as Talbot's law[15]

$$E = E_0\left(\frac{t + a}{t}\right)$$

where E = apparent intensity of flashing source, E_0 = intensity of steady source, t = duration of flash in sec, and a = curve-fitting constant equal to 0.21 sec.

While the flash duration influences the apparent intensity of the light, the flash rate influences the ease of acquisition, and these two factors influence the power consumption required for a beacon. The rate must be slow enough to permit a flash duration not requiring excessive power, but still fast enough so that at least several flashes will occur during the pilot's search time of the target area. For example, in evaluating a flashing light for use in a proposed orbital acquisition and tracking experiment, it was found desirable to flash at about 1 cps. This appears to be a good, representative flash rate.

The second and third phases of rendezvous are concerned primarily with angular rate perception and correction. The pilot must detect the rotation of his line of sight as the target moves across the star background and then must thrust to bring the rate to zero. After reaching the intercept course, the pilot will perform the braking maneuver to bring the vehicles together.

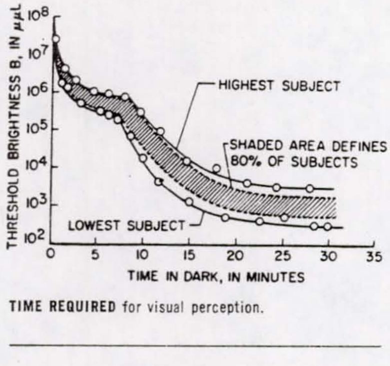
Any rotation of the line of sight

must be detected and measured precisely to perform these phases efficiently. In an instrumented simulation study made at various levels of rate resolution,[16] it was determined that 0.1-millirad/sec resolution was necessary to control a completely visual rendezvous, including the computation of range and range rate from angular measurements.

At the time of that study, there were no data to indicate whether pilots with unaided vision could achieve this precision. Subsequent experiments explored the smallest angular rate that a pilot could be expected to detect. These used the same basic equipment as the rendezvous simulations. Briefly, results show that a pilot can detect line-of-sight angles and angular rates accurately enough to perform the tracking portion of visual rendezvous.[17]

Important in the detection of angular rate is the angular separation between the target and the nearest reference star. Separation between target and nearest background star used in this visual-acuity study ranged from zero (superimposed condition) to as much as a 60-millirad angle. For the superimposed condition, the pilot could readily detect within the 0.1-millirad/sec rate.

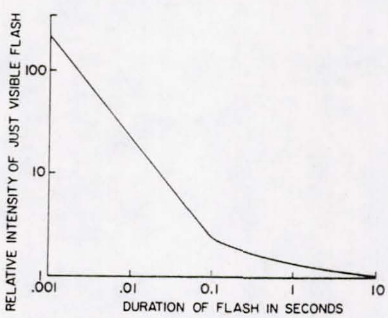
The graph on page 99 summarizes test results for 12.5- and 34-millirad angular separations between target and nearest background star. It shows that at 12.5 millirad the



TIME REQUIRED for visual perception.

pilot needs less than 10 sec to detect the desired 1-millirad/sec rate established in the previous study. His ability deteriorates somewhat at 34 millirad. Readings made at 60 millirad were not consistent, and gave no usable results.

Star-chart study shows a pilot can expect to have a visible (sixth magnitude or brighter) star usually within 2 deg of target. This means he may have to delay line-of-sight correction until the target comes close to a visible star or, alterna-



SIGNAL NATURE for equal apparent intensity.

tively, to use an optical aid, either to superimpose the target on a star or to make more stars visible by magnifying them. For instance, an average density of 4–16 stars eleventh magnitude or brighter per square deg could be obtained with a 3-in. telescope, and this would place the target within 12 millirad of a star and permit detection of 0.1 millirad/sec.

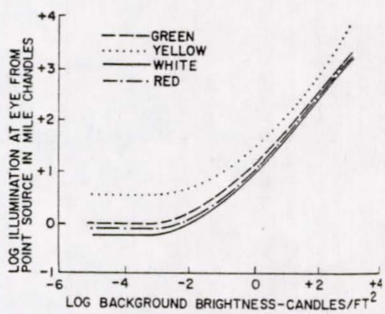
In these visual tests, two procedures were used for presenting target motion to the pilot. One began with the slowest angular rates, which required more intense concentration, and proceeded to the faster rates. The other began with the fast, easier rates; here the pilot felt fatigue, as indicated by a decrement in visual performance, by the time the slower motions were presented to him some 10 min later.

In an actual mission involving visual rendezvous, however, observing time needed to effect the initial intercept would be much less than 10 min and only occasional subsequent observations should be required to correct residual errors. Pilot fatigue can be minimized by a little training to avoid preoccupation with the tracking task.

After acquisition, correction to an intercept course, and initial braking, leading to something less than 2-mi. separation between chaser and target—the final braking and docking phases take place. The pilot observes aspect and closure rate to complete the maneuver. The visual conditions here are entirely different from those of the early phases, where the target was seen as a point source. Now the target-vehicle's shape and orientation are visible; and this information enables the pilot to control the range and range rate by eye, while orienting his vehicle for the docking and final latching maneuver.

The pilot uses the apparent size of the target to estimate the distance between the vehicles. Tests have been made at night inside Langley's 3000-ft-long hydrodynamic model basin to determine a pilot's ability to judge separation distance with no visual cues except the apparent size of the target. After adaptation to darkness, subjects estimated the range of several randomly placed models of known size.

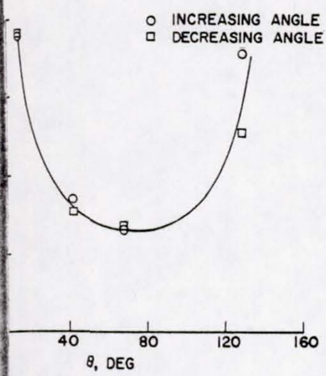
The graph on page 99 gives results, the solid line representing perfect estimates. Estimates were better than expected at ranges beyond 500 ft, but with a tendency towards overestimating the distance of large objects and underestimating the



FLASH DURATION'S EFFECT on visual threshold.

distance of small. All estimates, except for the balloon, were fairly accurate out to 500 ft.

In the Aurora flight, Scott Carpenter overestimated the range of a balloon similar to the one used in the model-basin tests. Both overestimations may have been caused by poor balloon outline, resulting from nonuniform brightness of its i-colored sections. The model- tests indicate that even this over-estimation diminishes in st 400 ft to provide reasonable useful values. The effects of ination, color, and aspect are ntly being studied further.



ANGULAR-RATE perception.

rate of change of size can be to determine closure rate be- the vehicles. This, too, has studied at Langley, primarily ning vertical descent to the surface; but the results may applied to visual contact in g. The relationship between e and rate of closure is given

$$\frac{\dot{S}}{S} = \left(\frac{1 + \tan \theta/2}{2 \tan \theta/2}\right)\dot{\theta}$$

S = separation distance, $\dot{S}$ = tion rate, θ = visual angle, = visual angle rate.[18]

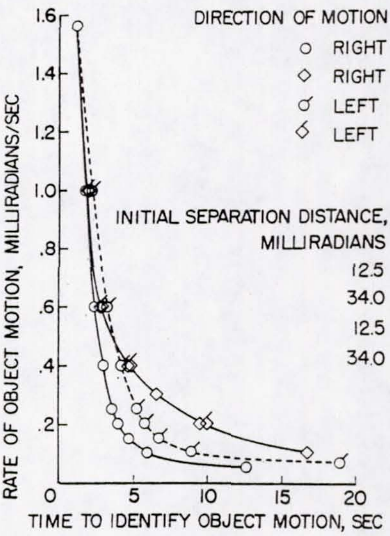
n from this study, the graph ove summarizes results for server over the range of considered, and defines (for st subject) the minimum ld of $\dot{S}/S$. This threshold ed on a 2-sec reply time dic- time lags inherent in the cedure. The graph shows e maximum perception of occurred between a visual f 80–90 deg, as subtended by get outline. This agrees edictions from the equation d.

visual angles will occur in al rendezvous and docking r. For the subjects tested, it was found that a representative value for the closure threshold ($\dot{S}$ min$/S$) falls between 0.013 and 0.016.

Relating these results to docking, the pilot should be able to judge the closure rate to about 0.15 fps from a distance of 10 ft, a value which agress with some preliminary visual docking simulation studies conducted at Langley using closed-circuit television techniques.

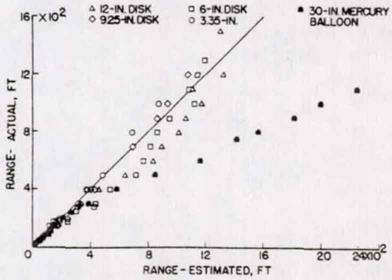
Further experiments, during actual space missions, would be desirable to complete these studies as well as to confirm them. But the results presented here should be sufficient to design space-rendezvous missions to the extent of man's capability, and should be applied.

For the conditions considered, the studies have shown that man can perform rendezvous efficiently. The optimum use of both man and machine, however, will be necessary for maximum efficiency and reliability. In making this integration, it should be kept in mind that man, if not overloaded, can be the most reliable and versatile element in the system. Optical and electronic aids can considerably reduce the workload of



THRESHOLD of velocity perception.

the pilot, making it easier for him to discharge his prime responsibility to exercise judgment and control. Devices selected on this basis, not subordinating the pilot to the equipment, would be desirable.



DISTANCE judgment.

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Designing helicopter and VTOL transports

BY E. F. KATZENBERGER,
H. D. ULISNIK,
P. J. WILLIAMS
AND A. A. WOLF JR.

Current design information points to advantages of VTOL aircraft for intercity transport, but serious gaps exist in research that would make such vehicles practical for commercial use later in the decade

The authors are all members of the Sikorsky Aircraft Div. of United Aircraft Corp. **E. F. KATZENBERGER** is chief of advanced design, **H. D. ULISNIK** is a senior design engineer for advanced concepts, Commercial Operations, **P. J. WILLIAMS** heads the Research Section, and **A. A. WOLF JR.** is a senior preliminary design engineer in the Advanced Design Branch.

The economic potential of rotary-wing and non-rotary-wing VTOL transport has yet to be adequately explored in all its aspects. This is not surprising, since it is an exceedingly broad and complex problem. An entirely adequate solution calls for the pooling of disciplines which cut across the spectrum of aircraft - industry capabilities. While the problem is not yet open to definitive solution, we can examine its more important aspects, taking the period of interest for projected applications as beginning in 1967.

The question might be asked at the outset: Why a helicopter or other VTOL air transport system? Fixed-wing aircraft are generally simpler; and, if not necessarily faster as optimum solutions for the stage lengths of primary interest in this context, they at least possess higher payload to gross weight ratios. The VTOL could not be considered potentially competitive were it not for the fact that its vertical flight capability is assumed to provide economic advantages; and these, if properly exploited, outweigh the disadvantages of greater complexity and poorer payload ratio.

The economic potential of the VTOL depends on two elements. Any analysis of this potential is essentially an investigation of the degree to which these two elements make possible an economical transportation system, and of the state-of-art capabilities of providing the required vehicles and subsystems. The two elements are:

1. *Relative freedom from fixed-wing traffic patterns.* This makes it feasible to consider the VTOL as having an economic potential for inter-airport operations regardless of the development of fixed-wing instrument landing systems, since the number of aircraft which can be handled by a given runway in unit time is limited, and saturation has already been reached for some airports.

2. *Considerably greater freedom afforded in the location of landing and takeoff areas* as determined by the required size, proximity to obstacles, and alternative real-estate costs. For VTOL economic feasibility, this element is the far more complex of the two; but it also appears to represent the more significant source of air-traffic growth since by this means the points at which the air traveler in fact becomes airborne and where he ceases to be so may be brought closer to his actual points of origin and destination. The significance of this source of traffic has been broadly dealt with by several studies.[5]

Concerning traffic patterns, it is largely necessary to consider, for a given route, only the possible gain resulting from a drastic revision of the delay-time situation created by nonadherence to fixed-wing traffic patterns.

Much more complex analyses are required on airports. These break down into three phases. The first of these might be termed the *economic-industrial-cultural,* and includes a coordinated study of traffic sources; qualitative and quantitative growth trends of urban, suburban, and industrial areas; travel habits and incentives affecting probable choices of alternate means; existing and foreseeable ground-transport networks; city planning and even local politics. These define the market potential by defining the problems the proposed system is expected to solve more conveniently or more economically, or both, than existing means make possible, either in their present state or through foreseeable modifications or additions, such as the monorail railroad. The most significant existing means for most areas at present is the private automobile. For many areas, however, this does not make sense economically or as a matter of real convenience, either for the private person or for the vehicle-choked community, but exists primarily in the absence of an alternative solution.

The second phase, which might be called the *systems analysis-synthesis,* aims at identifying system characteristics which provide economic solutions. Since no solutions are optimum in every respect, a variety of solutions must be compared to determine tradeoffs. The tradeoffs adopted are, in the last analysis, matters of judgment. This phase includes determination of several alternative sets of vehicle parameters; coordinated determination of vehicle subsystems and ground systems; customer acceptance factors, such as convenence of location and physical aspect of loading areas, baggage handling, ticketing, system response to delay situations, probable reliability, travel comfort, noise, and psychological factors.

None of these factors is unique to the VTOL, and all are the subjects of continuing study in the air-transportation industry. But the VTOL poses unique problems. If, for example, the systems study points strongly toward the direct-lift jet as the best approach, what techniques, equipment, procedures, and local structures in the loading area will be required to solve the problems of jet noise and blast effects?

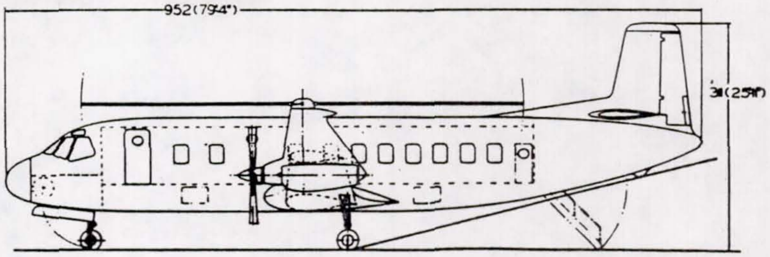
Having defined the problem and determined the parameters of the system required to solve the problem more conveniently or economically, or both, than existing means or their modification make possible, we must now consider the design and development problems of a system having these parameters.

Here again, the problems are not different in kind from those confronting fixed-wing aircraft; but in almost all, the fundamental areas of structures, aerodynamics, and propulsion are substantially more difficult, largely because VTOL research in these areas has thus far been very small compared to concurrent research into fixed-wing design problems. An excellent summary of this situation was given at the MIT Symposia of February and July 1961.[1]

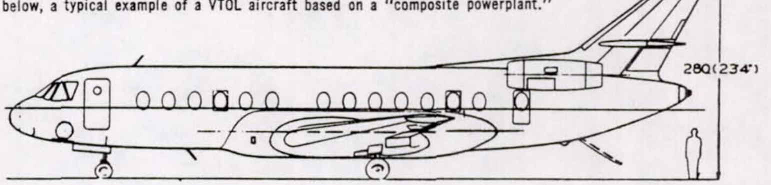
For a long time, the talents and resources of the aviation industry have been successfully directed to the improvement of long-haul transportation. Yet, in astonishing contrast, comparatively little has been done to improve short-haul transportation, with the result that the total travel time between cities a few hundred miles apart has decreased but slightly during the past 25 years.

Despite its imperfections, short-haul air transportation is a significant portion of the total travel market, as evidenced by the fact that the median journey is only 410 mi.,[2] with every prospect of the relative size of the short-haul market being increased as the travel time is rapidly decreased.

If the urgent need for an im-



SIXTY-PASSENGER Jet Liner Design Concepts: Above, a compound helicopter; below, a typical example of a VTOL aircraft based on a "composite powerplant."



ROUTE, PASSENGER, POWER, SIZE, AND COST RELATIONSHIPS: Top, routes considered (1) and passenger distribution from city centers (2); middle, effect of engine-out capability with lateral-compound design (3) and cumulative distribution of city-to-city distances (4); and bottom, side effect on direct operating cost of lateral compound (5) and direct operating cost vs. cruise speed for lateral compound (6).

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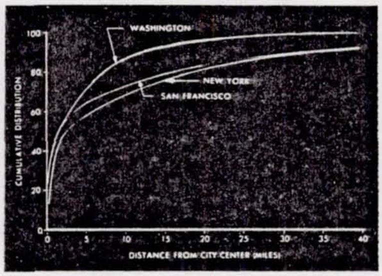

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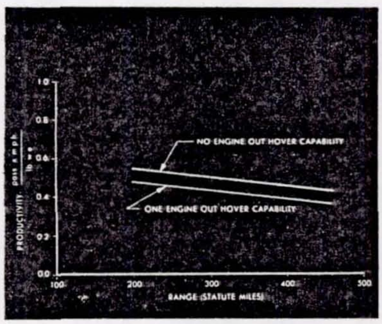

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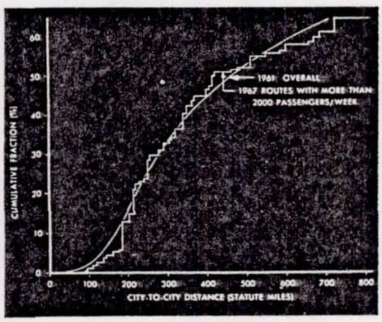

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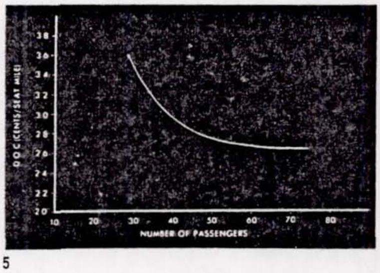

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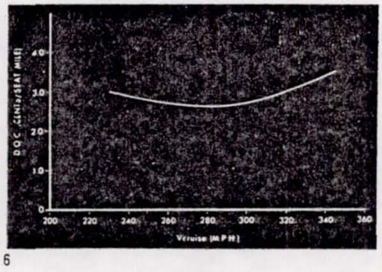

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proved short-haul air transportation system is obvious, the solutions are not. Our analysis here will but attempt to define the nature of the problem and the operational aspects necessary to determine the aircraft requirements, and to present the comparative merits of a number of different solutions. The preliminary aspect of this paper cannot be overemphasized, for it is but an initial step in the conquest of an intriguing problem.

Selected for study have been all domestic routes of less than 420 st. mi., which have a predicted 1967 volume of traffic greater than 2000 passengers per week. There are 36 routes in this group, involving 32 cities as shown in the map on page 101; and together they serve approximately 19% of the total number of air passengers. These routes were selected because they provide a valid sample and a solid basis for operational analysis, and because, if operational compromises are to be made, they should be made to benefit the high-density routes.

It is not sufficient to know the origins and destinations of passengers in terms of terminal cities. It is also necessary to know the distributions of origins and destinations within the metropolitan area. An excellent study of this problem has been made by the FAA,[3] the results of which were used to determine the cumulative distribution of passengers versus the distance of their origins and destinations from the city center, as shown on page 101. Three cities—New York, San Francisco, and Washington—together account for 38% of the traffic in the study.

The median distance from the origin or destination to the center of town is about 2 mi. This concentration of origins and destinations makes it desirable to land the aircraft as close to town as practical. Landing as close as possible to the center of town, plus the need for operational flexibility, dictate the requirement for a VTOL aircraft. Thus, the solutions presented in this paper are VTOL's, although the final comparison includes competitive fixed-wing aircraft.

The selection of suitable sites for VTOL terminals is a complex subject, deserving of further study, and its ramifications should be more thoroughly explored before they can be integrated into the transportation system. One factor which must be considered involves the economic penalties associated with central or near-central locations, as contrasted with the increased value of the service offered. The penalties associated with central terminal locations can be divided into two categories—increased cost of terminal facilities (which will be manifested in higher landing fees) and penalties which must be imposed on the aircraft, such as noise suppression and engine-out hovering capabilities, which for the compound helicopter results in a 15% decrease in productivity, as shown on page 101, and somewhat greater penalties for the nonrotary-wing types.

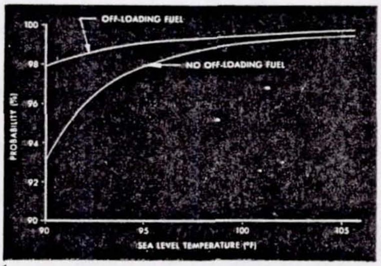
However, if the terminal allows an overwater approach, the need for noise suppression and engine-out hovering capability is reduced. Fortunately, 87% of the passengers in the sample of this study arrive or depart from cities with waterways within 1 mi. of the center of town.

A significant factor in determining the location of a terminal is the number of aircraft which will be loading or unloading simultaneously. The table, page 104, shows number of flights per hour during the peak daily period and the number of boarding areas required for those cities which have a peak load of more than six departures per hour. These data are based on all short-haul flights (not limited to those of the sample used in this study) and on the anticipated 19— volume of traffic.

Before the design and selection of an actual aircraft, there are several basic operational needs that must be set in advance—design range, optimal payload, desired cruise speed, design hover requirements, engine-out climb requirements, and passenger amenities.

Various criteria can be used to select the design range of an aircraft, most of which depend on t—

HOVERING AND LOAD FACTORS: Top, probability of adequacy of a given hovering temperature (1); middle, probability of hovering vs. load factor for design sea-level temperature (2) and probability of being on an aircraft with or over a given load factor (3); bottom, passenger fraction that must be unloaded at various temperatures (4) and temperature-altitude environment (5).



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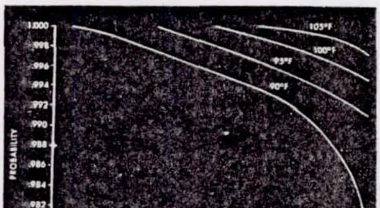

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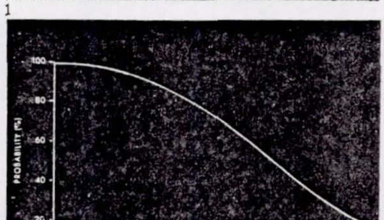

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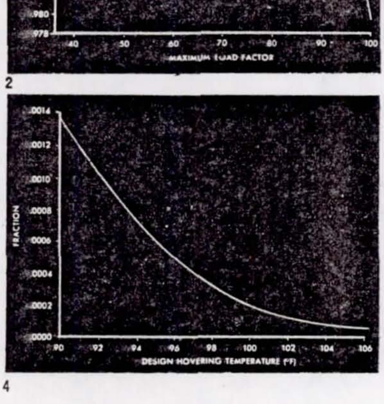

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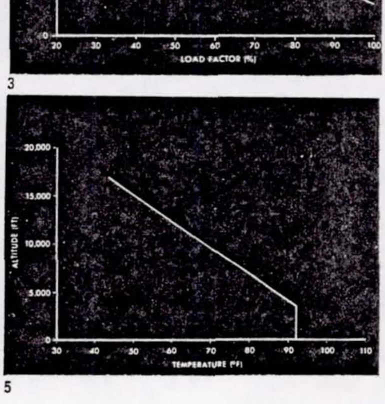

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variation of the performance and economics of a given configuration with range. The aircraft presented in this paper, however, are designed to fill a void in the market, and hence market considerations should predominate in selecting the design range.

All city pairs with an anticipated 1967 volume of traffic greater than 2000 passengers a week have been determined, and the cumulative volume of traffic has been plotted against the distance traveled, as shown on page 101. There are 81 routes in this category, and they comprise 38% of all passengers carried—a good sample.

A perusal of the diagram on cumulative volume of traffic reveals that there is a marked decrease in the rate of increase in passenger traffic at ranges greater than 420 mi. Hence, a design range greater than 420 mi. would not generate an amount of traffic which could justify the decrease in productivity associated with higher ranges. Thus, a tentative design range of 420 mi. has been selected, with the possibility that this range will be modified when it is considered in combination with the other design parameters.

The design optimal payload was determined by a number of different criteria, all of which fortunately resulted in similar results. On the basis of these considerations, a uniform payload of 60 passengers has been used.

The effect of payload on the direct operating cost was determined for the lateral-compound design shown on page 101. It was found that the optimum payload of the lateral compound is 60 passengers, whereas the optimum payload of the direct-lift aircraft is between 70 and 80 passengers, with little difference in the direct operating cost between a 60-passenger version and an optimum version. Further study into the economics of large-size direct-lift aircraft will be necessary before the effect of sizes in excess of 80 passengers can be determined precisely.

The effect of size on the frequency of service significantly influences the optimum size of the aircraft. Assuming a minimum of six flights per day in each direction will be necessary to provide satisfactory service between cities with a volume of traffic of 2000 passengers per week, 200 passengers going in each direction each day during the peak-travel days, an aircraft capacity of 56 passengers is dictated.

Another factor which determines the size of the aircraft is the effect of size on the number of flights per hour and the number of boarding areas required. From the table on page 104 it can be seen that a 60-passenger aircraft does not impose unreasonable requirements on terminal size and facilities.

The desired cruise speed is the speed for minimum direct operating cost, if the speed is compatible with the power available from reasonable numbers of predictable engines. The cruise speed of a current state-of-art helicopter would be approximately 190 mph, or near the maximum speed achievable without encountering severe design difficulties that would up direct operating cost. The cruise speed of the lateral-compound helicopter considered here would be 288 mph—the speed for minimum direct-operating cost as shown by the graph on page 101.

The cruise speed of the direct-lift aircraft is 500 mph—the maximum attainable without undue compromise in the low-speed handling characteristics and without encountering severe developmental complications, with resulting increases in the direct-operating cost and decreases in reliability.

The maximum temperature at which the aircraft should be required to hover at sea level represents a compromise between the penalties associated with high-temperature hover capabilities and the loss of income and good will caused by off-loading passengers to reduce takeoff weight. Arbitrarily, but reasonably, it was decided that a reasonable sea-level hovering temperature will require not more than 0.1% of all passengers to be off-loaded.

The probability of exceeding a given equivalent sea-level temperature at each of the 32 cities in the sample was determined. These data, weighed with respect to the volume of traffic in each city, are given in the graph on page 102. An associated graph there shows, for each city pair, the probability of being able to hover over a given equivalent sea-level temperature and load factor. This result has been combined with the probability that a passenger will be on an aircraft with a load factor equal or greater than a given value, as another graph there shows. The curve is a standard probability one generated on the assumptions that the average load is 60%, a load factor of 100% is attained 10% of the time, and the probability of being on an aircraft with a load factor less than 20% is negligible.

Still another graph in the group combines these data to form a relationship between the fraction of passengers which must be off-loaded and the design hovering temperature. It shows that the amount of passenger off-loading, based on the annual spectrum of temperatures in various cities, can be held to 0.1% if the design hovering temperature is 92 F.

It is axiomatic that IFR capability is necessary, and thus the aircraft must be able to maintain a best rate of climb of 150 fpm with one engine out. A temperature-altitude envelope must be determined for this climb requirement. The 92-F sea-level temperature requirement is maintained to an altitude of 3500 ft to allow for temperature inversions, and then the standard lapse rate followed beyond this point, the envelope shown on page 102 being generated.

The aircraft is designed for short-haul service in which the total block time is relatively low and in which an efficient operation demands rapid loading and unloading with consequent high utilization. Therefore, the following design conditions, presented here without argument, are dictated:

1. The minimum seat pitch shall be 36 ft for two abreast and 37 ft for three abreast.
2. Two passenger doors are necessary.
3. Air-stairs are desirable.
4. Carry-on baggage racks are manditory.
5. Large cargo and baggage compartments are not of great necessity, and their inclusion should not compromise the design.
6. A single lavatory is adequate.
7. No galley is necessary.

The justification of one mode of

transportation over another, as far as the passenger is concerned, is based primarily on the time, convenience, and cost of the journey. Convenience is an intangible thing, but it does, in part, comprise such things as the number of transfers which must be made between vehicles and the dependability of arriving at the desired destination at the predicted time.

A VTOL aircraft operating between city centers, but also having flights which serve existing airports to accomodate transfer passengers, eliminates the large number of transfers currently necessary.

Furthermore, there will be a high degree of operational dependability because a VTOL aircraft can operate outside of fixed-wing traffic patterns and can, if properly designed, make steep unaccelerated approach paths under IFR conditions. The exact design criteria for assuring this capability—which involve flight handling characteristics, instrumentation, navigation equipment, and the coordinated design and development of ground and airborne guidance equipment—are not sufficiently well known at present. But the work in this area currently being done by NASA should provide many of the answers necessary for preliminary design and for definitions of development program targets.

In addition to this, the ability of several VTOL aircraft to make simultaneous landings at the same terminal would largely reduce the need for stacking and thus increase the probability of on-time arrival.

The greatest value of a VTOL aircraft to the traveler, however, will be the saving of tedious ground-transportation time, with resulting savings in the total block time over that of fixed-wing aircraft, as indicated in the graph on page 105. This time saving has been given an arbitrary but conservative value of $3.00 per hour.

It is difficult to find an accurate value of the premium that travelers are willing to pay for a certain saving in time, but the estimate of $3.00 per hour seems reasonable and conservative in the light of the following factors:

1. The median income of air travelers is between $5.00 and $6.00 per hour.
2. Approximately 80% of all air travelers are traveling on business, for which they will be reimbursed.
3. Approximately 45% of all arrivals occur during working hours, and an additional 20% arrive in the early evening.

The existing rates for scheduled ground transportation have been analyzed and have been found to average about $1.50. On the other hand, the taxi rates associated with the typical 2-mi. trip necessary to arrive at the VTOL terminal average $0.75. Plus the time value of $0.50 for the 10-min trip, this gives an equivalent cost of $1.25 for VTOL travel.

Hence, the justifiable difference in the fare charged a VTOL traveler, as compared with existing coach fares, comes to $3.00 per block hour saved plus $0.50 saving in ground transportation.

At this point, it might logically be expected that there are sufficient data for the computation of direct operating costs and profit potentials for the types of vehicles considered, and therefore also for a comparison with a representative fixed-wing aircraft. Unfortunately, this is not so. The past and current gaps in VTOL research alluded to earlier make it extremely difficult to predict capabilities at the present time and still have the necessary assurance that the risk is reasonable for anything but the two ends of the spectrum where experience is already considerable—namely, for the rotary-wing VTOL and for the fixed-wing aircraft.

This characterization of the situation unfortunately appears accurate despite the excellent work being done, on limited funds, by NASA and other organizations in the VTOL field. The Air Force–Army VTOL transport should yield valuable developmental information, as should the projected Navy developmental vehicle and the Army lift-fan research program.

But the time span of interest here begins in 1967, only slightly removed beyond the minimum period required to develop vehicles based on *current* state-of-the-art knowledge. The situation is analogous to what would have happened if the attempt had been made to design the project programs for fixed-wing commercial jet aircraft when the designs of the first experimental jet aircraft were preliminary.

This situation with respect to VTOL research, and its significance with respect to the ability to project realistic designs on a timely basis, is admirably set forth in the MIT symposium.[1] Although this symposium concerned specifically Army requirements, the problems dealt with were fundamental. It is beyond the scope of this paper to consider these problems in detail, but it is necessary to an understanding of the problem of the commercial VTOL transport to note the important gaps in the funding of current research needed for VTOL applications.[1,4] Examples of various problem areas that deserve attention and research effort include—

VOLUME OF VTOL-AIRCRAFT TRAFFIC AT PEAK TRAFFIC PERIODS

CITY	DEPARTING FLIGHTS PER HOUR	REFUELLINGS PER HOUR	BOARDING AREAS REQUIRED
New York	65	38	16
Chicago	51	47	14
Los Angeles	27	24	7
San Francisco	20	20	6
Washington	19	17	5
Miami	17	17	5
Boston	14	4	4
Cleveland	14	14	4
Pittsburgh	13	11	3
St. Louis	12	12	3
Philadelphia	12	8	3
Minneapolis	11	11	3
Cincinnati	10	10	3
Detroit	9	9	3
Kansas City	9	9	3
Dallas	8	8	2
Seattle	7	0	1
Houston	6	8	2

All aircraft have 60 seats; a non-refuelling stop takes 10 min.; a refuelling stop takes 15 min.; and the aircraft boarding area is unoccupied for not less than two min. between flights.

MATERIALS

Ordered load spectra to replace random for fatigue testing and design.
Refinement and extension of cumulative damage theory.
Mechanism of crack propagations.
Influence of residual stress on fatigue.
Mechanism of origin of fatigue.
Fracture of dynamically loaded materials.

STRUCTURES

Experimental structures development.
Structural dynamics problems: rotating beams, VTOL props., rotors, lift-fans, primary structure.
Elastic and plastic instabilities: slender shell beams under combined torsion-bending—rotors, props., wing and fuselage structures.

AEROELASTICITY AND LOADS

Control effectiveness as a function of elastic characteristics of rotors, propellers, supporting structures.
Aeroelastic stability of blades including influence of pressure of duct walls, flutter of unloaded rotors at high speed.
Effect of flow instability over duct walls on fatigue loadings of elastic blades.
Reduction of rotor, prop., fan induced vibrations by modification of blade stiffness dist.
Effects of mass and stiffness distribution on fatigue loading for new rotor, prop., fan configurations.
Determination of steady and harmonic airloads for new configurations of lift systems.
Determination of transient airloads in gusts and maneuvers. Determination of dynamic response.
Maximum control effectiveness as limited by (h) above.

AERODYNAMICS

Development of low-speed aerodynamics for optimum wing designs at very low speeds.
Development of aerodynamic theory applicable to design of ducted propellers.
Investigations of wing-rotor interaction effects; wing angle of attack distribution as affected by presence of propellers, ducted propellers, lift-fans, lifting jets.
Definition of and provision for satisfactory control and handling qualities.
Provision for space stabilization in low-speed flight and hover.
Reduction of drag of aircraft configurations necessary for vertical flight capability.
Investigation of three-dimensional boundary-layer flows.

GUIDANCE AND CONTROL

Development of fundamental control techniques to satisfy optimum automatic guidance and control requirements.
Development of equipment for position and orientational references specifically to VTOL requirements.
Extension of work in adaptive control systems to VTOL area.

NOISE

Determination of effect of shear flow stability on noise.
Determination of parameters affecting noise output of rotors, propellers, ducted configurations.

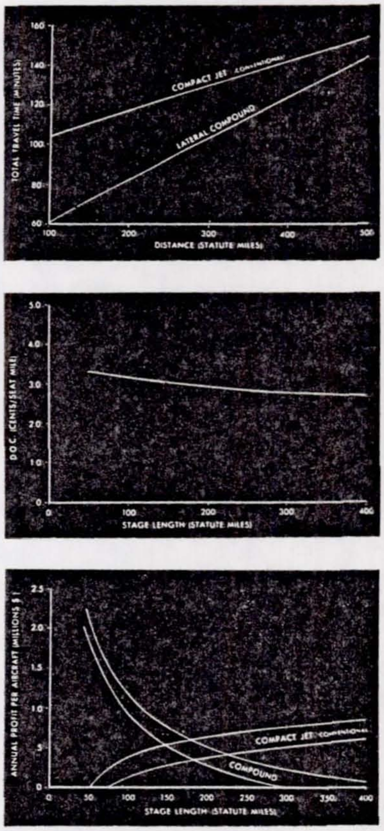
PROPULSION

Augmentation of research data on internal flow phenomona.
Investigation of problems related to possibility of short-duration high peak-power engine ratings.
Investigation of standby power-augmentation means.

The economic evaluation at this point, therefore, can yield only tentative results. The foregoing operational analysis indicates preliminary systems requirements, at least in terms of size and performance. The present level of VTOL technology, for the reasons given, can as yet yield no reasonably certain answer as to how these requirements may best be met. At best the two ends of the spectrum can be compared, as in the graph on this page, where the lateral compound is compared with a representative fixed-wing jet as to profit potentials. An associated graph gives the direct operating cost of the lateral compound.

By definition, the specific VTOL designs presented here represent state-of-art design capabilities; by implication they also represent extrapolations of these capabilities in the light of current knowledge. The lateral compound is sufficiently well based on available research information and on an in-house Sikorsky and United Aircraft Corp. research programs extending over the past 10 years, plus the byproducts of three high-performance helicopter weapon-system programs during that period, to be programmed with a high degree of confidence for 1967 commercial availability. Other approaches are open to various degrees of qualification as bases for forecasted programs for this period in view of the significant gaps in current research.

The particular solutions considered here do not by any means exhaust the possible varieties of VTOL configurations which warrant serious consideration and which, in fact, are currently being given consideration.



TRAVEL TIME, COSTS, AND PROFIT POTENTIAL: Top, total travel time city center to city center; middle, direct operating costs of lateral compound; bottom, profit potential of various aircraft.

Shown on page 101, the compound helicopter of 60,000-lb gross weight studied would carry 60 passengers with maximum comfort for a range of 420 st. mi. with 100 mi. of fuel reserve. It would cruise at 250 knots at 5000 ft on a standard day and hover at 2500 ft O.G.E. on an 83-F day. The design meets all applicable CAR7 requirements.

The flight regime may be divided into three modes. In the helicopter mode, the craft would hover and accelerate as a conventional helicopter; in the transition mode, the main rotors slow down and are partially unloaded by the wing. Propellers give thrust for further acceleration. In the compound mode, the main rotors are in autorotation and provide approximately 28% of the lift at 250 knots, the wing providing the remaining 72%. Main-rotor governors, sensing rotor rpm and controlling longitudinal cyclic pitch, automatically maintain rotor-speed stability. Conventional airplane controls will be provided for the pilot.

The rotor-tip speed was selected at 650 fps for the helicopter mode. This tip speed, however, is too high for the airplane mode because of the rotor power required and advancing blade Mach number limitations. Calculations for the forward flight mode using strip theory, which eliminates simplifying assumptions, indicates a 423-fps tip speed for this mode. Care must be taken in the selection of the rotor parameters to assure that autorotation is possible and that, for a given collective pitch setting, the slope of torque-vs.-rotor angle-of-attack curve is linear over a wide enough range about the zero torque value. This is to assure reasonable operation of the main rotor governors.

Under the above conditions the rotor system is flying at an advance ratio of $\mu = 1.0$ and, if the rotors were not partially unloaded, severe rotor stall and high control load feedback would result. As it is, the rotor will be partially stalled, but it appears that performance and stability will not be severely affected. This has been analytically verified for a single-rotor configuration based on studies conducted with the analog facilities at UAC Research Laboratories. However, these studies did not indicate the vibratory stress levels acting on the rotor blades and controls, nor did they take account of the side-by-side system being mounted on a relatively soft spring-mass system—that is, the wing. These areas need further study to determine their effect on rotor-blade and control-system life and reliability.

The wing structural design also needs further study. Preliminary dynamic analysis indicates that the normal quasi-static wing load criteria will not be sufficient. The large dynamic masses of the rotor system mounted on the wing tips require that the wing structure be structurally adequate to avoid resonant dynamic response. The analytical methods in conjunction with high-speed computers can now provide the required solution, but more work will be necessary to determine the tradeoffs of varying the design parameters—that is, section modulus, mass distribution, number of rotor blades, etc—to find the optimum combination. More than the usual amount of dynamic structural testing, moreover, will be required to correlate and verify the analytical work.

Assurance of accurate predictions of performance, flying qualities, and control requirements will necessitate further studies of methods of reducing drag, mutual interference between rotor and wing, safe flight-transition techniques from helicopter to airplane flight and back, reliable main rotor governor designs, minimum-pilot-effort control systems, and accurate weight-prediction techniques.

Continuing studies will also be required to determine interior arrangements that provide maximum crew and passenger comfort and efficient handling of passengers and baggage.

As to the direct-lift jet, the 60-passenger design shown on page 101 typifies a VTOL aircraft embodying the principle of the "composite powerplant," in which special lightweight lift-engines power takeoff and landing and separate engines power cruise. It has been designed to the preliminary requirements given in the foregoing analysis.

The 10 lift-engines ride in a wing root fairing and are symmetrically disposed relative to the normal center of gravity. The four "corner" engines have thrust-vector control, and thus supply the X-Y forces required in vertical flight and transition. Conventional power-assisted aerodynamic surfaces and pressure-balanced nozzles at the extremities of the wing and fuselage control roll, pitch, and yaw. The nozzles, normally partially shut, are fed lift-engine-compressor bleed air from a central gallery duct.

The design cruising speed of 500 mph at 20,000–30,000 ft was chosen in order not to compromise the low-speed handling characteristics. Designed within the state of the art, the wing has a maximum Mach number for drag divergence of 0.82 and will not require stability augmentors. The installed sea level, standard-day thrust-to-weight ratio of approximately 1.5 results from designing the aircraft to hover at sea level on an 92-F day with 10% lift engine compressor bleed for control and one lift engine dead. The design range with a vertical takeoff and landing is 420 st. mi., and the craft would have an additional fuel allowance for 100 st. mi. at the normal cruise speed of 500 mph.

The desirability of a direct-lift transport depends in part on achieving the lowest possible weight in lift engines and their installation. Three basic approaches have thus been examined: All convertible engines similar to the Bristol Siddeley BS. 53; part convertible engines and part lift engines; and lift engines plus propulsive engines. For a transport configuration, the composite use of lift and cruise engines gave the best arrangement and lowest installed thrust-to-weight ratio when thrust for control, the match between hover and cruise thrust requirements, engine failure, elevated temperature, and airport altitude had all been taken into account.

The tremendous possibilities opened up by a VTOL direct-lift transport are not without challenging problems.[1] With one possible exception, however, the problems facing the designer of a direct-lift commercial transport appear susceptible of straightforward solution, given adequate research support, for a more extended time-frame than considered here.

The one possible exception appears to be the problem of noise. This is particularly significant in view of the fact that economic potential of the VTOL, as previously noted, depends largely on its ability to operate close to the centers of populated areas. If the direct-lift jet is to be seriously considered, a very intensive research effort in this area will be required.

To recapitulate, from the passengers' point of view, there is a definite need for an air transportation system with a lower ground-travel time, fewer transfers, fewer delays, greater dependability, and greater convenience than now available.

From the airlines' point of view, there is a need for an aircraft which is profitable throughout its designed spectrum. Furthermore, the availability of an economical short-haul aircraft will provide a substantial increase in the volume of air travel.

Current design and systems studies indicate the feasibility of rotary-wing and non-rotary-wing design solutions for the short-range air transport problem.

The rotary-wing solution appears economically advantageous, for stage lengths of 200 st. mi. or less, over the state-of-the-art fixed-wing jet.

A rotary-wing solution may be programmed for commercial delivery in 1967.

Non-rotary-wing solutions cannot be programmed for 1967 and their economic potential is presently indeterminate, because of important past and continuing gaps in VTOL research.

REFERENCES: 1. Symposium on Research Related to Army Air Mobility, Massachusetts Institute of Technology, Feb. 14, 1961 and July 24, 1961. 2. "Origin-Destination Airline Revenue Passenger Survey," Air Transport Association of America, 1961. 3. "Airport Transportation, A Study of Transportation Means between Airports and the Metropolitan Areas They Serve," prepared for the Federal Aviation Agency by Human Sciences Research, Inc., February 1961. 4. NASA Conference on V/STOL Aircraft, Nov. 17–18, 1960. 5. "Economic and Safety Aspects of Short Haul V/STOL Aircraft on High Density Routes," George Ray and Robert Steiner, April 3, 1962.

Highly underexpanded exhaust jets against adjacent surfaces

BY LEONARD E. STITT
AND WILLIAM T. LATTO JR.

In stage separation, rendezvous trajectory corrections, and lunar takeoff and landing, the impingement of flaring exhaust jet and adjacent vehicle surfaces can cause stability problems and structural damage

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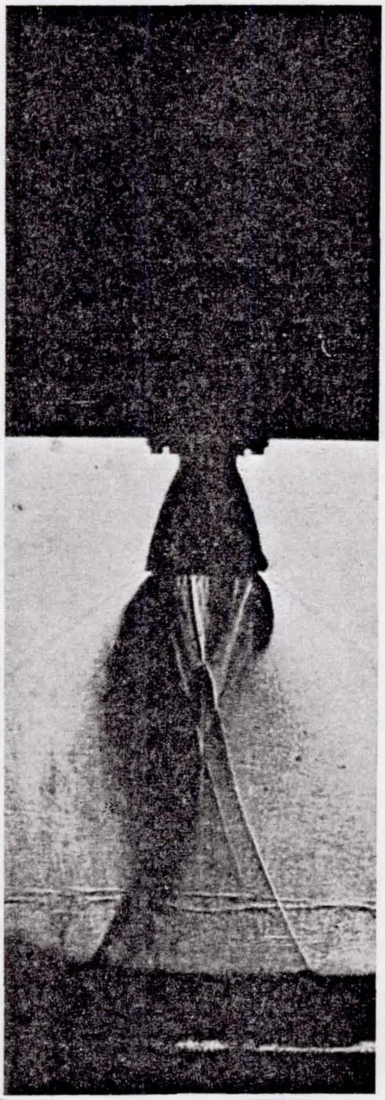
The use of hot, highly underexpanded exhaust jets during high-altitude stage separation, rendezvous of vehicles in orbit, attitude control or trajectory correction, and lunar soft-landing or takeoff could cause various problems, depending on the characteristics of the billowing jet and its proximity to nearby surfaces. For example, the impingement and reflection of the hot gases from an adjacent surface back onto the vehicle could cause stability problems or structural damage from the resulting pressure and temperature increases.

Theoretical calculations of the diffusion of a jet into a vacuum require an accurate prediction of the conditions at the exit of the exhaust nozzle. Experimental verification of these methods is desirable, since the Mach number and specific-heat ratio are difficult to compute, being dependent on the complicated combustion and expansion processes occurring within the rocket.

SHOCK STRUCTURE within an expanding jet—cold-flow nozzle; $j = 1.4$; $\epsilon = 25$.



Most of the available information on surface pressure and temperature magnitude and distribution has been limited to low nozzle pressure ratio. The interaction of highly underexpanded cold-air jets with nearby surfaces has been studied at a total pressure ratio of 288,000, pressure distributions on hard, flat, and discontinuous surfaces, as well as the interference effects on two arbitrary vehicle shapes (a sphere and a cylinder), being obtained over a range of descent heights with nozzles of various contours and area ratios.[1]

Presented here are further studies that have included the effects of a hot exhaust jet and the resulting temperature and pressure distributions on a cylindrical vehicle and a hard, flat surface.

Experimental Technique. Experimental determination of jet-surface interactions in a space environment has not been satisfactory. It is difficult to simulate high pressure ratios because of the inability to maintain a vacuum in a test chamber with the addition of propellant flow into the system.

The following experimental technique has been used at the NASA Lewis Research Center to obtain and to maintain a high pressure

SYMBOLS

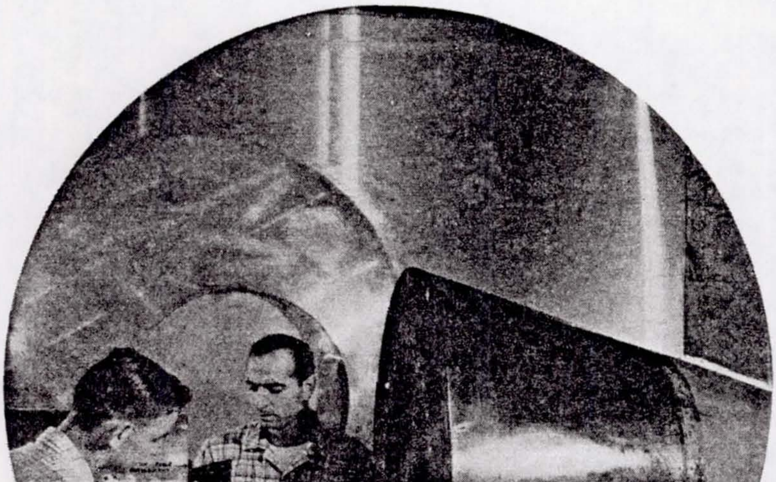
- d_t = nozzle throat diameter
- h_s = target-surface heat transfer coefficient
- P_c = combustion-chamber pressure
- p_b = vehicle-base pressure
- p_s = target-surface pressure
- T_b = vehicle-base temperature
- T_c = combustion-chamber temperature
- T_s = target-surface temperature
- X = distance measured along the surface of the target from the center of the jet
- y_b = distance from target to vehicle base
- y_t = distance from target to nozzle throat
- ϵ = exhaust-nozzle area ratio
- γ = ratio of specific heats

ratio for reasonable test times. The 10- by 10-ft supersonic wind tunnel is operated at its highest Mach number and altitude to provide a low ambient pressure. Simulated space vehicles are then mounted behind the 40-in-diam base of a cone-cylinder-flare body, as shown below left. This body shelters the configuration from the supersonic flow and provides a target for the expanding jet. With a chamber pressure of 2000 psia and a measured ambient pressure of 1 psf (abs), a nozzle total pressure ratio of 288,000 is obtained and is maintained as the tunnel flow scavenges the exhausted flow.

COORDINATES FOR CONTOURED EXIT NOZZLES

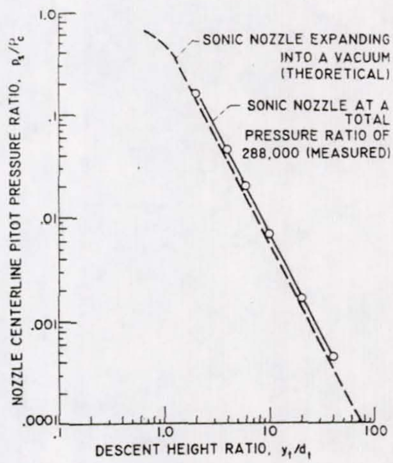
Cold-flow (d_t = 0.50") γ = 1.4; ϵ = 25		Hot-flow (d_t = 0.25") γ = 1.28; ϵ = 40	
l/d_t	d_l/d_t	l/d_t	d_l/d_t
0	1.00	0	1.00
.10	1.05	.48	1.37
.22	1.19	.84	1.81
.37	1.35	1.14	2.15
.52	1.53	1.54	2.59
.68	1.70	1.94	2.99
.84	1.87	2.16	3.20
1.09	2.12	2.61	3.60
1.21	2.22	2.94	3.86
1.33	2.34	3.34	4.16
1.59	2.57	3.74	4.44
1.91	2.83	4.22	4.76
2.12	3.00	4.54	4.85
2.38	3.19	4.94	5.18
2.63	3.37	5.48	5.48
2.90	3.54	6.17	5.82
3.18	3.71	6.90	6.14
3.47	3.88	7.32	6.32
3.79	4.05		
4.12	4.22		
4.48	4.40		
4.86	4.56		
5.20	4.73		
5.73	4.90		
6.00	5.00		

The quality of space pressure simulation achieved in the experimental programs can be seen from the adjacent graph which compares measured Pitot pressure on the centerline of a sonic nozzle at a finite pressure ratio of 288,000 to the theoretical Pitot pressure variation of a similar nozzle expanding into a vacuum.[2] The degree of simulation is apparent even though condensation of both the nitrogen and oxygen would be expected at these pressure ratios. The measured pressures were always higher than the theoretical values, and the experimental results presented here should be considered in this light.



SIMULATED LUNAR-LANDING VEHICLE installed for experiments in Lewis[1] 10- by 10-ft supersonic wind tunnel.

Measured Pressure Effects. One of the convergent-divergent nozzles used in the cold-flow program had a



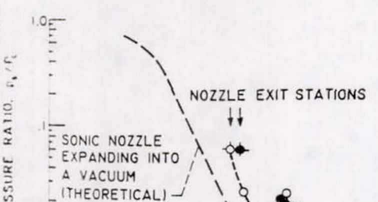
LUNAR SIMULATION met in experimental program.

length 80% of an equivalent 15-deg conical nozzle, an area ratio of 25, and an exit Mach number of 5.0.[1] Its contour is a parabola tangent to the throat radius and exit angle.[3] The nozzle used in the hot tests also had a parabolic shape, but it had an area ratio of 40 and a corresponding exit Mach number of 4.4. (Coordinates for both of these nozzles are listed in the table at top left.) Gaseous oxygen and hydrogen propellants were burned at combustion-chamber pressures from 500–1200 psia.

A graph on page 109 shows the Pitot pressure distribution obtained on the centerline of these two bell shaped nozzles for a range of descent height. The flagged symbols on the figure locate the nozzle-exit stations (or minimum descent height) at the corresponding theoretical value of Pitot pressure. Each of the two nozzles had some type of internal shock system within the expanding jet that extended about two exit diameters downstream, as shown in the schlieren photograph on page 107. These compression waves produced the discontinuity in the centerline pressure variation shown in the graph on page 109 at descent heights between 10 and 20 throat diameters. Similar pressure trends have also been obtained with other bell-shaped nozzles during wind-tunnel tests of high-altitude stage separation when the target was close to the jet exit. In the cold-flow investigation, a conical and a long isentropic nozzle were effective in eliminating the discontinuity in centerline pressure variation. Apparently the shorter bell-shaped nozzles overturned the flow, and internal compression waves were generated.

In general, the centerline pressure measurements indicate that the jet diffused rapidly downstream of the nozzle exit. At a distance of 25 throat diameters the peak Pitot pressure was only 1% of the chamber pressure, and at a distance of 85 throat diameters it had decreased to 0.1%. A comparison of the data obtained in the hot- and cold-flow tests indicates general agreement in centerline pressure variation resulting from these bell-shaped nozzles.

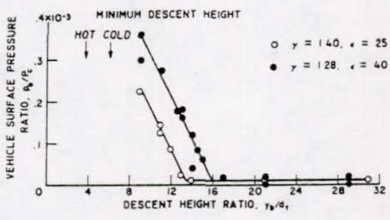
The graph at far right shows measured pressure distributions along the target surface at nominal separation distances of 40, 20, and 10 throat diameters. The correlation of the data between the two tests again was good. At the greater distance the surface pressures were relatively low in magni-



LANDING-SURFACE PRESSURE on jet centerline.

tude and extended over a large area. At closer separation distances the high pressures were contained within a circle with a radius of 8 throat diameters. The annular pressure distribution at a distance of 10 throat diameters resulted from the internal shock system that was associated with these bell-shaped nozzles, as discussed previously.

In the cold-flow test, a pressure increase was measured on the base



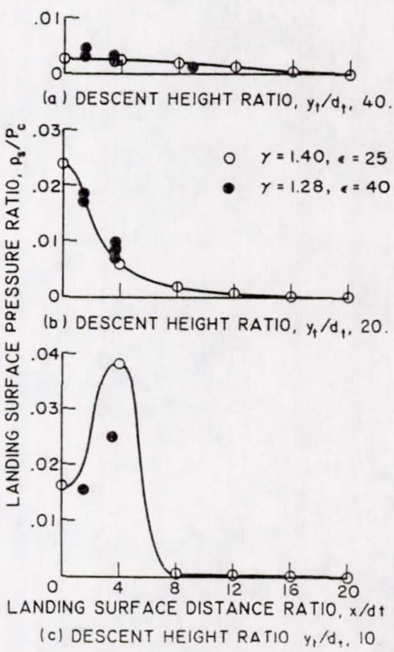
SURFACE PRESSURE on vehicle.

of the cylindrical model at separation distances less than 14 throat diameters, as shown by the graph just above. In the hot test, the pressure rise was first detected at 16 throat diameters because of a higher vehicle base diameter ratio; but the rate of increase was about the same. The pressure distribution in the base was symmetrical, and no moments were imparted to the vehicle when the target surface was flat and hard.

Measured Temperature Effects. The temperature distributions on the surface of the target and the base of the vehicle were obtained from thermocouple measurements. The peak target temperature was divided by the combustion-chamber temperature. The graph on page 110 presents results for a range of descent height and combustion-chamber pressure. The combustion-chamber temperature varied from 4800 to 5100 R over the range of chamber pressures presented. The peak target temperature increased linearly with decreasing separation distance and also increased with higher chamber pressure. At a distance of 40 throat diameters, the peak temperature was about 80% of the combustion-chamber temperature.

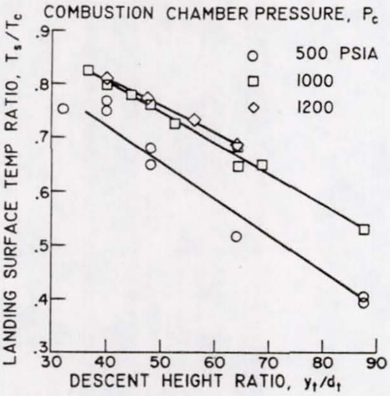
The graph on page 110 gives the temperature distribution measured along the target surface at descent heights from 37 to 69 throat diameters. The temperature increased near the center of the target as the separation decreased, as would be expected. At surface distances greater than 30 throat diameters, the opposite effect was noted; that is, the surface temperatures decreased with decreasing descent height. Similar trends were also noted in the surface pressure distributions during the cold-flow investigation. At a radius of 70 throat diameters, the target temperature was about 25% of the centerline temperature.

Heat-transfer coefficients, obtained from measured heat inputs to a copper-disk calorimeter located at a radius of 15.5 throat diameters from the center of the target, are presented in the graph on page 110 for a range of descent height ratios. The measured recovery temperature of the calorimeter was low—for example, 2300 R for $y_t/dt = 40$ and $P_c = 1000$ psia—because of the heat-sink effect of the target surface. The large influence of the



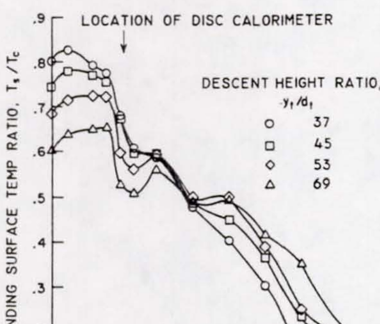
LANDING surface pressure distributions.

heat sink in maintaining a cool boundary layer resulted mainly from the low gas density in this area. The difference between the measured heat-transfer coefficient and one that would be obtained with a higher recovery temperature would be primarily a function of the change in the physical properties of the gas. The heat-sink effect will be present to some degree in all tests of jet impingement, depending mainly on the surface conditions and the gas density.



MAXIMUM landing surface temperature.

The heat-transfer coefficient increased at a linear rate with decreasing descent height and also increased with combustion-chamber pressure, as expected. As shown, top right, values of heat-transfer coefficient varied from 100 to 280 Btu/hr-ft^2-R over the range of variables presented.



LUNAR SURFACE temperature distributions.

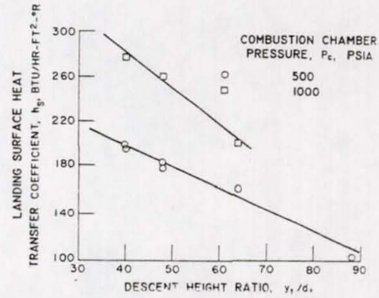
The temperature on the base of the cylindrical vehicle increased initially at a separation distance of 20 throat diameters, as shown in the graph at the right, reaching a value near 60% of the combustion-chamber temperature at a descent height of 8 throat diameters. The base temperature also increased slightly when the pressure ratio was doubled.

A copper disk calorimeter was also used on the base of the vehicle to obtain heat-transfer coefficients. The calorimeter was located at the edge of the base surface. At descent height ratios greater than 20, the base was heated only by radiation. At lower descent heights the base was heated further by convection from the hot jet reflecting back from the landing surface. At a descent height ratio of 12.5, a vehicle base heat-transfer coefficient of 100 Btu/hr-ft^2-R was measured, increasing to 200 Btu/hr-ft^2-R at a descent height ratio of 10 for a chamber pressure of 500 psia.

Summary. Although exploratory, this study indicates trends and first-order effects pertinent to the jet-surface interactions associated with a rocket-powered vehicle operating in close proximity to nearby surfaces. Results of these studies, a summary of which follows, are applicable to the areas of stage separation at high altitudes, orbital rendezvous, attitude or trajectory control, and lunar operation.

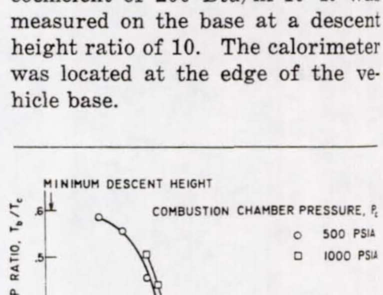
Pressure effects. First, space pressure simulation obtained in the experimental studies was reasonable, the Pitot pressure measured on the centerline of a sonic nozzle at a finite pressure ratio of 288,000 agreed favorably with that predicted theoretically for a similar nozzle expanding into a vacuum. Second, the jet diffused rapidly downstream of the nozzle exit. At a distance of 25 throat diameters a peak surface pressure of 1% of the chamber pressure was measured. This pressure decreased to 0.1% at a distance of 85 throat diameters. Third, interaction between the surface and the vehicle only occurred at separation distances less than 16 throat diameters.

Temperature effects. First, the peak target temperature increased linearly with decreasing separation distance, reaching a value of 80% of the combustion-chamber temperature at a distance of 40 throat diameters. Then, the temperature on the base of the cylindrical vehicle increased at descent heights less than 20 throat diameters, a peak base temperature equal to 60% of the combustion-chamber temperature was recorded at a distance of 8 throat diameters. And third, heat-transfer coefficients were obtained on both the vehicle base and target surface from the measurements of copper-disk calorimeters. Measured values of the heat-transfer coefficient on the target surface at a distance of 15.5 throat diameters from the center varied from 100 to 280 Btu/hr-ft^2-R for a range of descent height ratios from 40 to 90 and combustion-chamber pressures from 500 to 1000 psia.



LANDING SURFACE heat-transfer/coefficient; calorimeter at x/dt = 15.5.

At descent height ratios less than 20, the vehicle base was heated by radiation and convection from the hot gas reflecting back from the landing surface. A heat-transfer coefficient of 200 Btu/hr-ft^2-R was measured on the base at a descent height ratio of 10. The calorimeter was located at the edge of the vehicle base.



VEHICLE surface temperature.

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Wind tunnel studies of booster base heating

BY MILTON A. BEHEIM AND LEONARD J. OBERY

This survey of rocket-booster base-heating studies shows the care necessary in designing of small-scale model tests and the needs such tests will continue to meet in the design of future boosters

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LEONARD J. OBERY joined Lewis in 1948 and now heads both its Rockets and Aerodynamics Branch and the 10- by 10-ft Supersonic Wind Tunnel Branch and advises the director of Development Problems on power problems.

The analytical and experimental studies of rocket-booster base flow discussed here originated with afterbody and base pressure studies of turbojet installations, progressed to base-heating studies of single- and four-engine rocket boosters, and most recently has included base heating of an eight-engine rocket booster. Models of the Jupiter, Thor, Atlas, Polaris, and Saturn vehicles and generalized configurations were investigated experimentally in several Lewis Research Center wind tunnels—the 8- by 6-ft SWT, the 10- by 10-ft SWT, and small (18- by 18-in.) tunnels.

Some results have been reported, some are being prepared for publication, and the results of some studies were so greatly affected by the specific model performance that they were not sufficiently precise or general to warrant publication. Our purpose here will be to summarize this over-all program, to illustrate some of the considerations and limitations of model testing, and to present an integrated interpretation of the results. The future role of small-scale base-heating tests will be indicated.

Base-Flow Studies. Numerous studies of base-flow aerodynamics were undertaken during the late 1940's and early 1950's to optimize the aerodynamic design of turbojet- and ramjet-powered configurations.[1] However, knowledge of the fundamental parameters affecting the base-flow phenomena was inadequate until Korst, et al., at the Univ. of Illinois presented a basic flow model which has been very useful in understanding the aerodynamics of jet mixing.[2] Using this analysis, Kochendorfer developed a technique for theoretical prediction of base pressure resulting from the jet-stream interaction of an axisymmetric configuration.[3] A variety of base and nozzle configurations employing a cold gas jet were investigated in small (18- by 18-in.) tunnels, and results showed good correlation with theory for convergent nozzles but poor correlation for convergent-divergent nozzles.

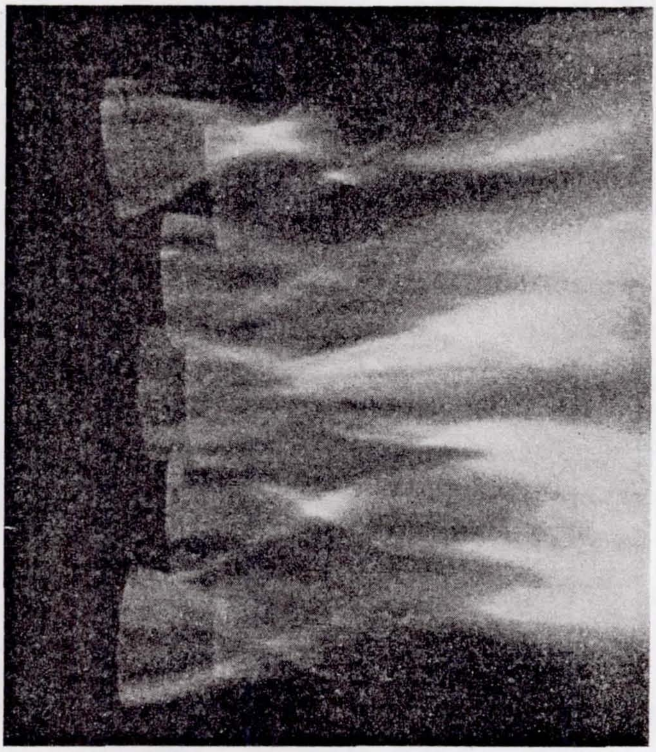
Shortly thereafter, base-heating studies were begun in the 8- by 6-ft tunnel of single jet configurations. A lox-JP rocket motor was employed, and results showed serious effects of base burning when fuel from the rocket jet or from the turbine-exhaust discharge were entrained and ignited in the base region.[4–9] A similar study of single-jet base heating at a high Mach number was conducted in the 10- by 10-ft tunnel,[10] and again serious effects of base burning occurred as a result of entrainment of fuel from the jet.

The theoretical analysis of the base-flow phenomena had been interrupted because the poor correlation theory and experiment for the convergent-divergent nozzles of Reference 3 could not be explained. To investigate a possible scale effect on the mixing phenomena, a similar study of axisymmetric base pressure was conducted in the 8- by 6-ft tunnel with large-scale models. A cold-air jet again was used, and effects of base bleed were also determined. These results showed good correlation with theory for the convergent-divergent nozzles.

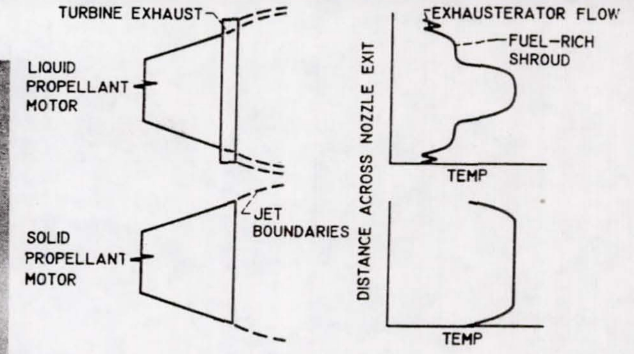
In addition, the analysis was extended to determine base gas temperature and pressure of an axisymmetric configuration with a hot jet.[11] To assess the accuracy of the theory, the base flow of a hydrogen-peroxide rocket was investigated in the 8- by 6-ft tunnel. Correlation of base pressure with theory again was good, but base gas temperatures were somewhat less than theory. The results of both these studies have been presented.[12]

A third general study in the 8- by 6-ft tunnel investigated the aerodynamics of the base flow of axisymmetric rearward-facing steps.[13] Because it had been determined that clustered nozzle configurations also have a serious base-heating problem due to mutual jet interaction,[14] studies of four-jet geometries were initiated in the 8- by 6-ft tunnel during 1959 with hydrogen-peroxide rocket motors. During the test, it was observed that the nozzle-wall temperature had an important effect on base gas temperature; and, since the wall temperatures of the nozzles used in this study did not simulate the wall temperatures of a full-scale booster engine, results were not reported.

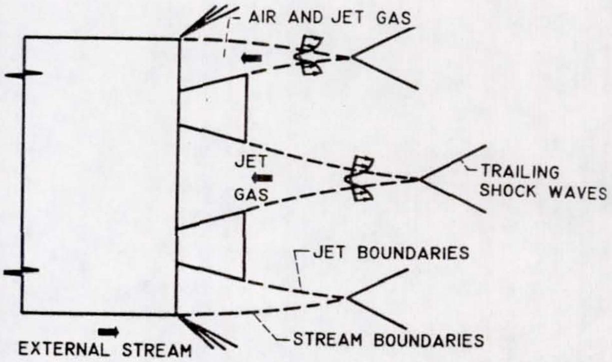
Also in 1959, a four-jet configuration employing lox-JP rocket motors was investigated in the 8- by 6-ft tunnel. In this test it was determined that base-heating data



LUMINOSITY of exhaust from Saturn model.



NOZZLE-EXIT temperature profiles, shown schematically.



BASE-FLOW aerodynamics at high altitudes.

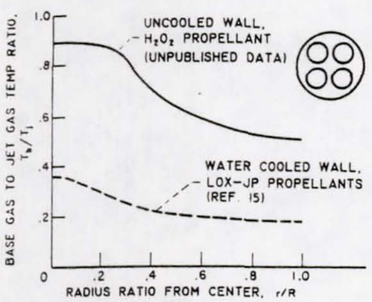
were greatly influenced by erratic base burning, resulting from the use of engines with relatively low efficiencies ($c^* = 4700$ fps). These results were not considered to be sufficiently representative to permit publication. A similar study of the base heating of a four-jet configuration was also conducted in the 10- by 10-ft tunnel with high-efficiency engines ($c^* = 5400$ fps), and results showed the effects of geometric variables on base heating.[15]

To investigate the base-flow aerodynamics of the eight-engine Saturn configuration, a cold-air-flow model of an early version of the configuration was tested in the 8- by 6-ft tunnel. Results showed a high degree of entrainment of exhausterator gases in the base region.[16] A similar study of the Saturn base heating was conducted in the 8- by 6-ft tunnel during 1960 and 1961. The model employed eight lox-JP rocket motors, and the effects of configuration variables (such as cooling-air scoops and turbine-exhaust discharge ducts) were determined. Over 500 test firings of the model were made in the wind tunnel, and interpretation of results was seriously hampered by extensive effects of base burning. A report is being prepared on this.

Interpretation of Results. The effect of altitude on the aerodynamics of base flow has been described in other publications. The illustration just above shows the basic-flow model with a high-altitude condition for a multijet configuration. The causes for base heating inherent in this flow phenomena include radiation from the rocket jet to the baseplate, recirculation of rocket or turbine exhaust gases into the base region as a result of jet-stream and jet-jet interaction, and combustion in the base region of fuel, which originated from the fuel-rich rocket jets and/or turbine exhaust discharge, with oxygen entrained from the free stream.

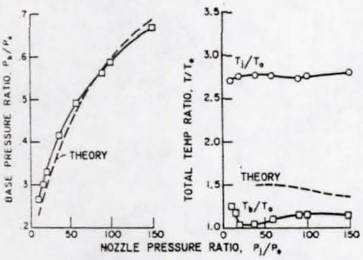
Since base heating is an important consideration in missile design, it is desirable to determine the design heating rates with small-scale models, to assure the success of the flight vehicle. Base heating is affected by a very large number of parameters, however, all of which need to be considered in the design of a small-scale model and in the interpretation of model results. Providing the small-scale simulation of all of these parameters is the crux of the problem.

To simulate the radiant heating of the baseplate by the rocket jets, the simulation parameters that need to be considered are the jet temperature, the jet emissivity, and the jet-to-base form factor. The emissivity of the jet is such a complex phenomena that it appears necessary to use the same propellants for the small-scale model as will be used in the flight vehicle. This choice also approximates the correct jet temperature. However, simulation of the form-factor effects is more difficult to achieve, since much of the radiation originates from the afterburning muff located some distance downstream of the nozzle exit. The illustration at the left shows the muff which occurred with the Saturn model in the 8- by 6-ft tunnel. The distance from the nozzle to the afterburning region depends on the mixing process of the jet with free-stream oxygen, on the excess-fuel distribution in the rocket jet, and on the chemical reaction kinetics; therefore it cannot be scaled easily. As a result, this distance (in terms of model scale) is generally too long with a small scale model and the form factor (and hence radiant heating) is low.



EFFECT OF NOZZLE-WALL COOLING *on* base gas temperature—four-jet configuration, choked center star.

To simulate correctly heating effects produced by hot-jet gases recirculated into the base region, it is necessary that all parameters affecting the jet-stream or jet-jet interaction be correctly duplicated. The flow angularity and velocity requirements in the region of the trailing shock have been adequately discussed in the literature; and these requirements can be achieved by using the same propellants as



THEORY VS. EXPERIMENT—single jet, $M_o = 2$.[12]

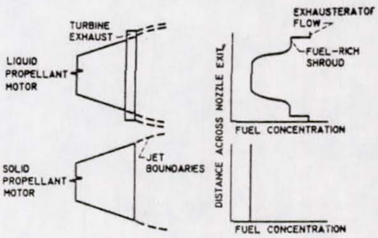
the flight vehicle, a carefully scaled model and nozzle geometry, and the correct nozzle pressure ratio. More difficult to achieve, however, is an additional requirement, that the portion of the jet recirculated into the base region be at the correct total temperature.

The total temperature profile of an exiting jet from a full-scale motor, illustrated schematically on page 112, will depend on the propellants, oxidant to fuel ratio, and engine efficiency; on the nozzle-wall cooling effect, which will be greater for regeneratively cooled nozzles than for other types; on the fuel-distribution pattern of the injector, particularly if a very fuel-rich region is created around the periphery to alleviate the wall heating problem; and on the effect of turbine exhaust gases discharged into the periphery of the jet. As illustrated in the figure on page 112, this temperature profile generally will be flatter for solid than for liquid propellant engines.

Obviously, it is not easy to determine the jet temperature profile in the region of the trailing shock and then of duplicating it on a small scale, so that gases recirculated into the base provide the correct heating effect. The illustration at the left shows the significant effect that the nozzle wall-temperature has on the base gas temperature distribution of a four-jet configuration. The nozzle walls of the lox-JP rocket motors were watercooled, whereas the nozzle walls of the hydrogen-peroxide motors were uncooled. Although nozzle-wall temperatures were not determined, the rocket firing was of sufficient duration that the uncooled wall temperature was approximately equal to the hydrogen-peroxide decomposition temperature (1400 F). The geometry of the two models was not identical, but the differences were small and would have little effect on base gas temperature for the choked center star conditions illustrated at top left.

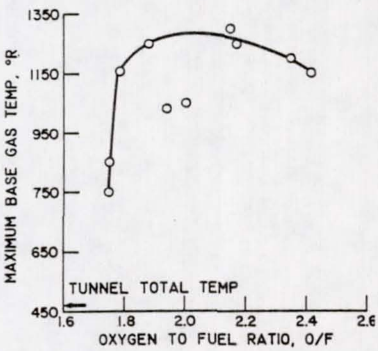
A large reduction in base gas to jet gas temperature ratio with the cooled nozzle wall is evident. The drawing at left shows additional indications of nozzle-wall cooling effects for a single-jet configuration.[12] Although the experimental and theoretical base pressures were in good agreement, the experimental base gas temperatures were somewhat less than theory. This difference may have been due to inadequacies of the theory; but, since the nozzle wall was water-cooled, it is more likely that the lower temperature ratios resulted from nozzle-wall cooling effects.

Adequate simulation of the heating resulting from base burning does not appear to be feasible, since combustion characteristics of the type encountered in base flow generally cannot be scaled satisfactorily. For the base-flow phe-



EXCESS-FUEL PROFILE at nozzle exit, shown schematically.

nomena, simulation parameters that must be satisfied simultaneously (in addition to the requirements discussed earlier) to determine if base burning occurs, and to determine the heating rates which result, include the following:



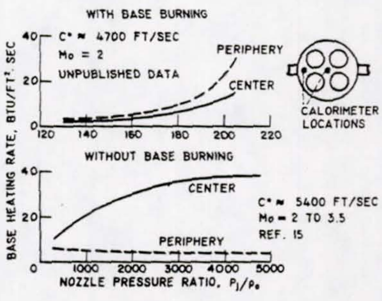
EFFECT OF OXIDANT-TO-FUEL RATIO on base burning—single-jet configuration.[5]

1. The correct quantity of fuel must be recirculated into the base region from the rocket jet and from the turbine exhaust. This requires that the excess-fuel concentration profile across the jet ahead of the trailing shock must simulate that of the flight vehicle. In a manner similar to that for the jet total-temperature profile, the excess-fuel profile of the full-scale vehicle, illustrated schematically at top here,

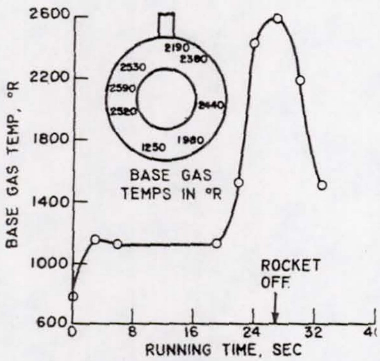
depends on the propellants, oxidant-to-fuel ratio, and engine efficiency; the fuel-injector distribution pattern; and the effects of discharging fuel-rich turbine exhaust gases around the jet periphery. As indicated in the figure, these profiles will generally be more irregular for liquid- than for solid-propellant motors. The task of determining the full-scale excess-fuel profiles in the region of the trailing shock and of duplicating them on a small scale is again very difficult. Special consideration must be given to the problem of achieving the high engine efficiencies of large motors with small-model rocket engines, and of simulating the diffusion characteristics of the turbine exhaust into the rocket jet to provide the correct fuel distribution in the jet at the trailing shock.

2. The local mixing in the base region of entrained fuel with entrained air should be simulated to provide the correct local fuel-air ratios. In spite of the extensive work on combustors for ramjet and turbojet engines, little can be done to control or scale this mixing. Obviously combustion will not occur if the local mixture is either too rich or too lean.

3. Since flammability limits depend on temperature, this temperature level must be correctly duplicated in the base region. As discussed earlier, this requires correct simulation of jet recirculation and of the jet-temperature profile at the trailing shock. Additional effects on the temperature of the base gases that should be considered are quenching effects of structural surfaces located in the base region. The base temperatures of a flight vehicle increase as it flies its trajectory and in general are substantially higher than those of a wind-tunnel model operated intermittently at discrete points of the trajectory. Hence quenching effects are greater with the model. A further complication arises from protective ablative coatings used on portions of the flight-vehicle base; effects of such ablative material on base burning are difficult to duplicate with a model.



ENGINE-EFFICIENCY EFFECTS on base burning—four-jet configuration.



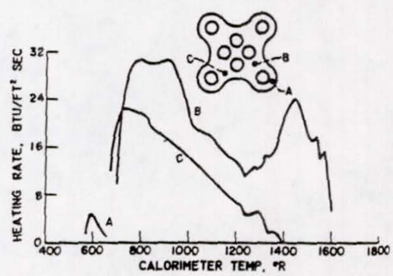
INTERMITTENT AND ASYMMETRIC NATURE of base burning—single jet configuration, $M_0 = 3$.[10]

4. The flame speed and ignition-delay characteristics of the entrained fuel should be correctly scaled relative to local velocities and residence time in the base region of the model. Again, the work with ramjet and turbojet combustors is evidence that satisfactory simulation of these parameters cannot be provided on a small scale.

It is apparent that existing base-heating models do not provide a rigorous simulation of these parameters affecting base burning; the model results, however, are greatly influenced by base burning effects. Hence, there is no assurance that the model results will be related to the conditions encountered in full-scale flight. Several situations resulting in a major influence of base burning on model data have been encountered in the tests at Lewis. An illustration of a typical effect of rocket-engine oxidant to fuel ratio on base burning of a single-jet configuration appears on page 113.

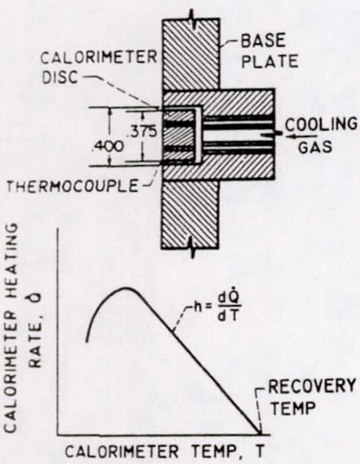
At oxidant-to-fuel (O/F) ratios below 1.7, the entrained combustion gas-free-stream mixture was apparently too fuel-rich to ignite, and measured gas temperature and visual observations indicated no base burning. However, for higher values of O/F ratio, ignition occurred and relatively unsteady combustion was observed and measured in the base. The range of O/F ratio for which base burning will occur will depend on, among other factors, the efficiency of the particular engine being employed.

The intermittent and asymmetric nature of the burning is shown at far left for a single-jet geometry. For the first 18 sec of the rocket firing, measurements indicated a steady flow of entrained gases into the base regions at a relatively constant temperature of about 1100 R. Suddenly, shortly before rocket shutoff, the gas mixture ignited and combustion temperature reached 2600 R. The rocket-engine operating conditions and the tunnel flow were constant during the entire firing until engine shutdown at about 26 sec. The typical distribution of base gas temperature shown in the figure indicates that, for this particular firing, some regions of the base experienced intense combustion (left side) while in other regions (such as the lower portion) there was virtually none.



BASE-BURNING EFFECT on calorimeter heating—Saturn model, $M_0 = 0.8$.

Effects of base burning on the base heating of a more complex four-jet configuration is shown at the top for two tests—one with high-efficiency engines and very little base burning;[15] and one with similar model geometry but lower-efficiency engines giving extensive base burning. The trends of the data with high-efficiency engines can easily be explained by considering recirculation effects; that is as pressure ratio increased, center



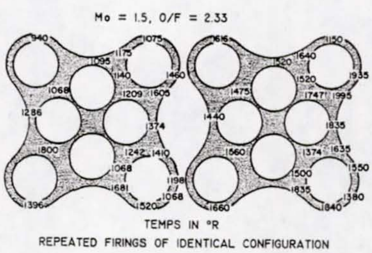
CALORIMETER DESIGN used in experiments.

heating exceeded peripheral heating. However, contradictory results occurred with the low-efficiency engines; increasing pressure ratio caused greater peripheral heating than center heating. This peripheral heating was chiefly a result of base burning centralized in the base regions directly downstream of the model struts.

The effect of localized, intermittent base burning on calorimeter heating of the eight-jet Saturn model is shown on page 114. Heating traces for the calorimeters in positions designated "A" and "C" are typical of those where local base burning effects were minor, while the irregular nature of the "B" calorimeter trace is the result of base burning at that position. These effects of intermittent burning are difficult to interpret, because it is not certain that the results of a single firing can be repeated, or that these results yield the maximum heating rates which should be considered in the full-scale vehicle design.

Typical effects of base burning on temperature distributions in the Saturn-model base region are illustrated here below. These temperature distributions resulted from repeated firings of an identical configuration, and lack of reproducibility is evident. Movies taken of the model base clearly showed the intermittent and random nature of the base burning.

Model Instrumentation. Extrapolation of small-scale base heating data to a full-scale vehicle requires a knowledge of the heat-transfer coefficient and the temperature of the base gas. The design of the calorimeter which was most satisfactory for the Saturn-model test conditions appears at left. The calorimeter disk was supported only by the two pairs of thermocouple wires to avoid corrections for conductance losses to the surrounding metal. The disk thickness and material were varied depending upon the magnitude of the heating rate being measured; greater disk thickness and metals with higher specific heats were employed for the higher heating rates. A cooling gas was discharged through the annulus about the disk during engine startup to minimize the deposit of carbon and combustion products of the tetraethylalumina used for engine ignition and to insure that a substantial disk temperature rise occurred during data acquisition,

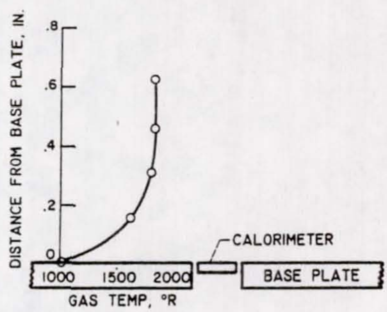


BURNING EFFECTS on Saturn-base gas temperature distribution.

thereby improving the accuracy of heating-rate computations.

The flow of cooling gas was interrupted during data recording. The graph above left presents an idealized trace of calorimeter heating rate versus disk temperature. The heat-transfer coefficient, h, is obtained from the negative slope of the curve, and the intercept at zero heating rate is defined as its recovery temperature. However, the heat-sink effect of the model baseplate surrounding the calorimeter creates an uncertainty in this analysis of the data because of nonisothermal wall effects.[17]

If the flow in the base region passes over the baseplate before passing over the calorimeter, a substantial temperature gradient can

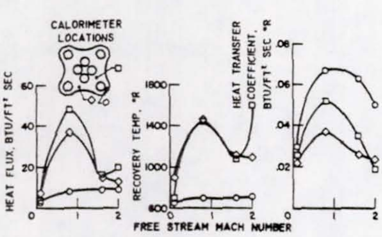


HEAT-SINK EFFECTS of Saturn-model baseplate.

be imposed upon the flow by the heat-sink effect of the baseplate. The graph here above gives a typical temperature profile normal to the baseplate of the Saturn model. Since the baseplate surface approaches an equilibrium temperature much more slowly than does the calorimeter disk, the recovery temperature of the disk may be less than the true value of base gas temperature. The magnitude of this discrepancy will depend upon the flow angularity approaching the calorimeter (which may differ in various regions of the base), on the magnitude of the temperature gradient produced by the heat-sink effect of the baseplate, and on the calorimeter dimensions. It will be greater if the flow approaching the calorimeter is parallel to the baseplate, and if the temperature gradient produced by the baseplate is large, and if the diameter of the calorimeter disk is small. A large difference in temperature between the baseplate and the calorimeter can also influence the magnitude of the heat-transfer coefficient; but it was determined that, for the temperatures encountered in normal model operation, this effect on h was small.

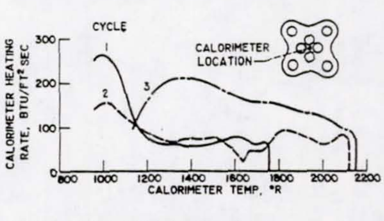
During the Saturn test, it was determined that the cooling purge was not entirely successful in preventing the formation of the deposits encountered during engine startup, and that these deposits had an appreciable effect on the calorimeter data when high heating rates were encountered. This effect is illustrated at right for the center star calorimeter of the Saturn model. The deposits sharply reduced the calorimeter heating rates, and only by repeated cycling of the calorimeter disk to high temperatures (by means of the cooling gas purge) during a given firing of the engines could useful data be obtained, for example cycle 3 in the graph above right.

Saturn Base Heating Results. Peripheral and center star base heating data for one of the Saturn-model configurations are shown in



PERIPHERAL HEATING of Saturn model.

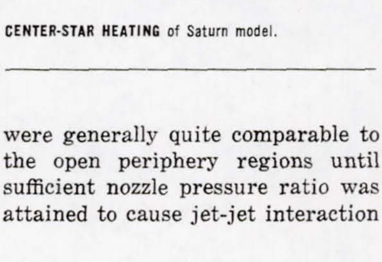
the two adjacent graphs. Heat-flux rates all correspond to a calorimeter disk temperature of 560 R. These results show reasonably consistent trends. To obtain data of this nature, however, many lox-JP rocket engines were tested prior to tunnel installation, and only those with sufficiently high engine efficiency (c^* greater than 5200 fps) were installed in the model. In addition, the injector pressure drops of the engines were closely matched, since several engines were manifolded to a lox and a JP supply line. If these requirements were not carefully observed, the base heating results were greatly influenced by base burning peculiar to that specific combination of eight engines installed in the model. More than 200 engines were used in the program.



EFFECT OF CALORIMETER CYCLING during continuous rocket operation: Saturn center-star location.

The reliable measurements indicated peak heating rates of about 40 Btu/ft^2/sec occurred in the open areas of the peripheral region of the base at Mach 0.8. This condition corresponded to the highest aerodynamic q region of tunnel operation, and hence the highest base pressure. As indicated by the recovery temperatures in the graph at the left, base burning occurred in these regions. The sheltered region between the outboard engine and the fuselage fairing experienced low heating rates and low recovery temperatures at all conditions. The restricting affect of the curved passage between engine and wall probably minimized flow circulation; but the quenching effect of the watercooled motor must also have contributed to this low heating rate.

The center star heating rates



CENTER-STAR HEATING of Saturn model.

were generally quite comparable to the open periphery regions until sufficient nozzle pressure ratio was attained to cause jet-jet interaction and intense reverse flow of the jet gases. As shown in the graph at bottom the heating rates increased with free-stream Mach numbers (actually corresponding to an increased jet pressure ratio) and would be expected to reach a constant value at some higher Mach number when choking of the passage between the four jets would occur, thereby isolating this region from further ambient effects.

Future Role of Small-Scale Base Heating Tests. Although Lewis has not been engaged in full-scale flight tests of missile base heating, several small-scale models of flight vehicles have been tested in the Lewis tunnels. Correlation of model and flight heating data generally have not been good, particularly for the four- and eight-jet geometries. A part of this discrepancy may be due to instrumentation difficulties, which obviously are even more severe on a flight than on a wind-tunnel model; but it is apparent that the major differences were due to inadequate simulation of the many parameters affecting base heating.

Future small-scale base heating tests must therefore be planned very carefully, and the limitations of small-scale tests must be recognized. A carefully planned model test can determine recirculation heating rates, with due regard to simulation of the jet temperature profile and the baseplate heat-sink effects; effectiveness of aerodynamic techniques of preventing base heating

SYMBOLS

c^*	= characteristics velocity
h	= heat transfer coefficient
M	= Mach number
O/F	= oxidant to fuel weight flow ratio
P	= total pressure
p	= static pressure
$\dot{Q}$	= heat flux
R	= radius of afterbody
r	= local radius
T	= total temperature
SUBSCRIPTS	
0	= free stream
b	= base
j	= jet

from occurring, for example, cooling-air scoops or turbine-exhaust discharge stacks; and effects of model configuration on base pressure distribution. Small scale models are extremely limited in establishing flammability limits of base burning resulting from fuel entrainment, base heating rates in the presence of irregular base burning, or radiant heating rates.

The need continues for small-scale data to aid in the design of flight boosters. Small-scale tests, however, must be carefully planned and results carefully interpreted to avoid misleading information. Although improved testing techniques are necessary, much can be achieved through the careful instrumentation of full-scale flight vehicles.

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