2nd Manned Space Flight Meeting

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2nd Manned Space Flight Meeting

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BANQUET SPEECH

by

LYNDON B. JOHNSON

Vice President, United States of America

You may recall the circumstances of a football game in Dallas which went into Southwest Conference records as a one-to-nothing victory for SMU over TCU. On a rainy autumn Saturday, the bus carrying the Horned Frogs from Fort Worth, 32 miles away, couldn't make it up Chalk Hill, west of Dallas on the old Dallas Pike. The result was a forfeit--with the official score of one-to-nothing.

I mention this to contrast our horizions--and our capabilities. Many of you can remember when Chalk Hill was a major obstacle and the supreme test for every new automobile. Motorists could brag if they climbed Chalk Hill with no more than two or three rest stops to permit the engine to cool. Yet now, we are able to meet to talk seriously about explorations and journeys 26 million miles away to the planet Venus--or 47 million miles away to the planet Mars.

We have come a long way. In the new Age of Space which brings us together, we are destined to go a long way further.

The Age of Space is less than six years old. It is no insignificant detail that this important scientific forum is convened in this city and this state far removed from the older established centers of economic, commercial and industrial enterprise of our nation. On the contrary, this emphasizes one fact about the Age of Space which many have neglected:

The fact that the Age of Space is--for all regions of America--a second industrial revolution--a revolution which is bringing a new future, a new degree of participation, a new scope of contribution for each region of our country--and for every segment of our society.

Some of our countrymen suspect that space is a gimmick or a gamble. Others regard it as forced upon us solely by considerations of national prestige and international competition--the necessity of "keeping up with the Russians".

You and I believe such conceptions are false.

Space is not a gambit.

It is not a gimmick.

Our national activities in space research and exploration are no longer in the category of a gamble.

"Keeping up with the Russians" is not our primary motivating force. In fact, we want to pass them, not just keep up with them. Moreover, what we are doing--and what we hope to do--would still be required of us in our own self-interest if we were the only nation on earth to possess space capabilities.

Yet, if misconceptions do exist among the public, the fault is less theirs than it is the fault of those who have understanding of space but have not taken the time to explain fully its meaning or the necessity for our national activities in this new realm.

The Age of Space reflects and is supported by a revolution in science and engineering.

Ninety per cent of all the scientists who have ever lived are living today. More mathematics has been created since the beginning of the twentieth century than in all the rest of history. Ninety per cent of the drugs being prescribed by our physicians now were not even known ten years ago. In many fields of science, the research literature published just since 1950 is several times greater than all that published in the centuries before.

This exploration of science and engineering is not just American. It is not just in the Free World. The sudden burst forward of man's scientific knowledge and capability is world-wide -- on both sides of the Iron Curtain.

In this context, when we speak of competition between the Free World and the Communist World, the competition in space is merely symbolic. The real competition between our systems is not a race for the moon--it is a race to see which system will do the best job of developing and applying new scientific capabilities for the betterment of life on this earth for all mankind.

If this is the competition, I am sure some will ask why go to the moon? Why be concerned about Venus and Mars? Why send men up to orbit the earth? Why not keep both feet on the ground and spend our money here instead of in outer space?

These are sincere questions and valid questions. They cannot be dismissed impatiently, If the public is to be asked to support the cost of space research, the public is entitled to understand the justification for these large--and still growing--demands.

There are several parts to the answer.

First--and foremost--is the fact that the full value of science to the improvement of life on earth cannot be achieved, if we artificially limit our horizions to the outer edges of the earth's atmosphere. If we are to make science the servant of man, we must go into space to put science to work for us on earth.

Two examples help illustrate this.

First, there is the field of weather. Scientists tell us that if we could accurately predict the weather five days in advance, it would mean cost savings of $2.5 billion annually in agriculture, $45 million in the lumber industry, $100 million in transportation, $75 million in retail marketing and more than $3 billion in management of our water resources. This is for the United States alone. The total world-wide benefits are beyond calculation.

We will be able to do these things--these and many more. We will be able to predict the weather, not just five days in advance, but perhaps for a whole season. In advance, we will be able to eleminate the high toll of lives and property from hurricanes and tornadoes and typhoons. We will some day perhaps be able to modify and exercise control over our weather--eleminate drouths and floods, bring rains to the deserts and control the deluges of the jungles. But we can do none of this without developing space capabilities on which we are at work today.

Second, there is the field of communication. Our first trans-Atlantic telegraph cable opened in 1866. Sixty years elapsed before the first trans-Atlantic radio circuit in 1927. Thirty more years passed before we had the first reliable trans-Atlantic telephone cable in 1956. It took us ninety years to increase our capabilities a thousandfold. But today we have the prospect through communication satellites of achieving another one thousandfold increase in less than a decade.

Perhaps before the 1960's end, we will be transmitting hundreds of individual messages by satellites in one second. Books the length of the Bible will be transmitted in written form in less than one minute. Dallas business mem will be able to check their inventories in London or Tokyo in a matter of seconds. Giant business machines and computers will be tied together across the oceans. Scientists in distant, underdeveloped nations will be able to solve their equations on computers in Texas or New England or California.

Again, if we are to achieve these things--if we are to put our new technical competence to work for us to the fullest extent possible--we must be capable of utilizing space as the avenue for these advances. Earthbound science cannot serve us. A ceiling on science would be a ceiling on our progress and our prosperity--and eventually on our freedom itself.

That last point is one we must not forget.

The United States has chosen as a matter of firm national policy to commit its efforts to the development of the peaceful use of space. It was my privilege in 1958 to present the United States policy to the United Nations at the request of President Eisenhower while I was Majority Leader of the Senate.

We have invited and encouraged all nations on earth to join with us in this common peaceful endeavor. We can be proud that more than sixty nations--from Argentina to Zanzibar, from Iceland to Thailand--are working cooperatively with us in the greatest international program of scientific information exchange ever known.

Our first and continuing objective is to develop the peaceful uses of outer space. But we are not unmindful of the threats to peace on earth which would result from the exclusive mastery of space by any power seeking to perpetuate earthly aggressions.

When we send our probes on missions into space, near and far, we are not engaging in idle adventures. As some say, "What man can conceive, man can do". As just one example, it is conceivable that an unfriendly power might use space for arms storage, or for the stationing of an offensive weapon, or for other hostile purposes.

If we are to be reasonable and prudent, we must anticipate today what the Soviets or others might have or might develop to threaten our freedom. We cannot wishfully and unrealistically assume that no nation will extend its objectives of world domination by means of space weapons.

If we are to realize the full capabilities of our technology for the betterment of American life--if we are to meet fully our national requirements--we cannot arbitrarily put a ceiling on the domains of science.

Responsibility to ourselves--responsibility to the cause of freedom which we lead--requires us to explore the realms of space as we are doing today.

There is another part to the answer about our purposes in space which is too little understood. This is the fact--as I mentioned at the beginning--that this Age of Space is the star performer in this industrial revolution.

Our space program is creating and helping to create new basic industries for our economy. The number of private companies and research organizations participating in our space programs has grown in less than six years to more than five thousand. A single orbital flight by one of our astronauts last year required the supporting services of more than four hundred thousand individual jobholders. Out of our space research programs have already come more than 3,000 new products and methods available for the use of private enterprise.

Technology of space research is just now beginning to be felt in our daily lives. By-products of the space program are appearing in the automobiles we drive, in our office equipment, in the air-conditioning and heating equipment in our homes, in the medicines we take and even in the pots and pans our wives use in their kitchens.

Many of the Buck Rogers symbols of this age have unexpected values. For example, the space suits worn by our astronaut heroes--such as Colonel Glenn--may soon be used widely in medical centers to relieve the discomfort of stroke victims. By-products of the research on the propellants which send our missiles aloft will provide effective means of treating tuberculosis.

It took us 112 years to develop photography, 56 years to develop the telephone, 35 years to perfect the radio, 15 years to make radio workable, 12 years to make television feasible and 6 years to develop the atomic bomb. But the outpouring of space research into our lives and businesses is coming after less than six years--and this is only the beginning.

Equal in importance to these developments is the impact of our space effort on the economy of every region of our nation. As might be expected, Texans have said much about the significance of this State--and of the Southwest--in the Age of Space. But I would emphasize that this is by no means exclusive to the Southwest alone.

Space research has brought a major rejuvenation to New England.

It is generating new growth and prosperity in the Deep South and Southeast.

Space will be an increasingly important source of contracts and jobs in economic activity for the Great Lakes, the Northwest and the Rocky Mountain States.

Space-related activities have already become the number one industry for the nation's most populous State, the State of California.

For the first time in our nation's history, an historic new boom is beginning in which all regions will share--in which the strength of every region will be marshalled and its promise realized and the people permitted to prosper.

This city, this state, this region have known boomtimes in the past. But the promise of the future now unfolding dwarfs the dimensions of the past. Scientists and engineers--as well as public officials--have a duty to bring the reality of this future home to the American public now.

To reach for the moon is a risk. But it is a risk we must take. Keep in mind that failure to go into space is even riskier.

In less than a lifetime, Chalk Hill--and many obstacles like it--have been mastered and young generations have forgotten where it is. So it will be with outer space. Where the moon is a major goal today, it will be tomorrow a mere whistle stop for the space traveler.

We go into space as pioneers came into this West, for one purpose only: to find ourselves and our families a better life on earth and to assure the ultimate success of the cause of freedon we uphold.

If we do not succeed in these efforts--as one great American has put it--we will not be first on the moon; we will not be first in space; and one day soon we will not be the first on earth.

SYSTEMS ENGINEERING FOR MANNED SPACE FLIGHT

by

JOSEPH F. SHEA
Deputy Director, Systems
Office of Manned Space Flight
National Aeronautics and Space Administration

(Luncheon Speech)

The manned lunar landing program presents not only the most difficult technical challenge which the nation has undertaken but also one of our most complex management problems. We are now almost two years into the program. I would like to review with you today how we are attacking the task; particularly the systems engineering function within the OMSF and some of the systems problems with which we are still wrestling.

Before discussing our organization, I would like to reminisce a bit. The first task thrust upon us after the establishment of the OMSF Office of Systems was the selection of the lunar mission mode. I recall we were all quite enthusatic about the job for it was a wonderful technical problem. That was, incidentally, before we recognized all the other ramifications of making such a decision.

In order to develop perspective for the studies, we did a bit of historical research to see how the wisdom of the past might be brought to bear on our present problems.

Since Apollo sprang originally from Greek civilization, we started there, and found, in the story of Phaeton what is probably the first description of a nominal space mission and the difficulties of astronaut selection and training. Although a few of the details are lacking, I was struck by the grasp of the overall problem.

Phaeton was the son of Helios, the sun god. One of Helios' duties was to drive the chariot of the sun from dawn to dark. Phaeton desired to drive his father's chariot. As recorded in Bullfinche's Age of Fable, Helios, in warning Phaeton of the difficulty of the task, said:

"Your lot is mortal, and you ask what is beyond a mortal's power. In your ignorance, you aspire to do that which not even the gods themselves may do. None but myself may drive the flaming car of day. Not even Jupiter, whose terrible right arm hurls the thunderbolts. The first part of the way is steep, and such as the horses when fresh in the morning can hardly climb; the middle is high up in the heavens, whence I myself can scarcely, without alarm, look down and behold the earth and sea stretched beneath me. The last part of the road descends rapidly, and requires most careful driving. Tethys, who is waiting to receive me, often trembles for me least I should fall headlong. Add to all this, the heaven is all the time turning round and carrying the stars with it. I have to be perpetually on my guard least that movement, which sweeps everything else along should hurry me also away. Suppose I should lend you the chariot, what would you do? Could you keep your course while the sphere was revolving under you? Perhaps you think that there are forests and cities, the abodes of gods, and palaces and temples on the way. On the contrary, the road is through the midst of frightful monsters. You pass by the horns of the Bull, in front of the Archer, and near the Lion's jaws, and where the Scorpion stretches its arms in one direction and the Crab in another. Nor will you find it easy to guide those horses, with their breasts full of fire that they breathe forth from their mouths and nostrils. I can scarcely govern them myself, when they are unruly and resist the reins".

Even the dangers can be identified. The Archer obviously fires the micrometeorites we must guard against, and the poison of the Scorpion is the lethal radiation from solar flares.

The "clause" of the Crab is possibly what makes our contracts hard to negotiate.

In selecting our approach to systems engineering for the manned space flight programs, historical perspective was also important.

The concept of systems engineering for complex research and development programs has evolved in this country over the last two decades. By this time, the name has been applied to so many different variants of technical management that it may now call to mind only a generalized impression of responsibilities and functions. Indeed, I often think that systems engineering is at least as difficult to define as it is to do. In general, the management of research and development programs today recognizes that two broad sub-divisions of the program team exist: the research and development organization, often called the project, and the systems engineering organization. It is almost universally recognized that systems engineering is not responsible for the development of the "hardware" elements of a program. The generation of the systems concept and the overall system specifications is usually accepted as the system engineering task.

The gray area is the extent to which the system engineering group participates in the development phase of the program, and the organizational relationship, both explicit and implied, which it holds with respect to the development team.

This area is also the one where personal relations become most important. I gave a talk last week at the University of Kansas and got into a bull session with some of the undergraduate engineering students. They had just taken a course in personnel relationships, and several were apparently thinking of giving up detailed engineering in favor of going into technical management because they felt that working with people was so important ---- and that engineers were usually insensitive to personal relationships.

Although I objected strongly to their premise ---- for how can you manage well things which you don't understand in detail --- they certainly had identified one of the major factors which must be taken into account in establishing a smoothly working program structure.

In the spectrum of approaches to the task which have been employed, two historic examples come close to defining the extremes of the distribution. One is the role which systems engineering plays in the Bell Telephone Laboratories; the other the role which it played in the Ballistic Missile Program.

At Bell Telephone Laboratories, the major responsibility of systems engineering is the determination of new specific systems and facility development projects -- their operational and economic objectives and the broad technical plan to be followed. Systems engineering controls and guides the use of the new knowledge, obtained from research and fundamental development programs, in the creation of new systems. In determining new development projects, systems engineering considers the content of the reservoir of new knowledge awaiting application, and the opportunities for its use. It attempts to insure that the technical objectives of the development projects undertaken can be realized within the framework of the new knowledge available in the reservoir and present engineering practices.

As the development organization proceeds with the project, systems engineering maintains close contact, continually appraises the results, and amends the objectives and plans as required. Service trials are generally needed during the course of development. It organizes the trials in cooperation with operating engineers and participates in the testing and evaluation of results. When the system is standardized and placed in manufacture, systems engineering follows service performance of first installations and coordinates the "growing pains" engineering that is ever present on new systems as they enter service. It finally participates in the evaluation of the system and economic worth that are experienced.

In addition, early in a developmental program, the development organization may carry responsibilities to systems engineering in areas where they have a particular competance. Throughout the development program they are available to the project people and are often used as consultants.

Although stresses and strains undoubtedly existed in the initial phases, management has woven the systems engineering and development departments into an efficient working team, aided by the "espirit de corp" of a single corporation.

Perhaps the other extreme is the addition of a technical direction function to the systems engineering responsibility which the Air Force employed in the ballistic missile program. The development team in that case was a number of associate contractors, each responsible for major hardware elements which had to be integrated to provide a system.

In addition to developing the systems concept and the specifications for the major system elements, the systems organization technically directed the activities of the major contractors to assure that both design and manufacturing procedures would result in satisfactory operational systems.

The technical direction function explicitly included responsibility for design engineering of subsystems; direct assistance to associate contractors when necessary; planning and developing ground support systems for research development and operational phases of the program; directing test programs for systems, subsystems, and selected components; directing the associate contractors to implement such research and development operational requirements as are approved by the Air Force and directing the contractors in implementing reliability programs.

Hence, in a sense, the system engineering organization carried the responsibility both for the development of the system concept and the details of its execution. Implicit in this concept is a very close personal identification of the system people with the details of the development program. In addition, the implementation of the concept obviously required detailed direction of a number of industrial concerns by another, essentially competitive, industrial concern. This approach, justified by the exigency of the program, did result in difficult working relationships.

Much can be learned from these two examples. In evolving the systems engineering concept within the manned space flight program we have charted a course between the two extremes cited, coming somewhat closer to the first than to the second.

The Nature of the NASA organization, and the magnitude of the task, dictates that the implementation of the program be done through the NASA Centers. They combine both the administrative and technical competence to contract with and technically direct the industrial firms selected to execute the hardware developments. All direction of industrial firms is done by government people.

The systems engineering role in OMSF is intended to provide program wide technical analysis for management to insure that functional and performance requirements placed on all elements of a system are within the present or projected state of the art and can be developed within the scope of the project. The specific nature of this generalized responsibility has evolved over the last year as the systems organization has come into being.

The overall nature of our work is illustrated by the mission mode decision and the planning of the possible future manned space flight programs, such as the Lunar Logistic System, the Space Station and the Planetary Programs. Our people concentrate on the requirements for such systems and the study programs which must be conducted to answer the eternal questions: what, why, when and how. The studies are usually implemented by the Centers and contractors working either for the Centers or OMSF. We then pull together the study results, select the system concept and develop the system specification and the program plan. The object is to have available, at the start of any future program, the blueprint from which the development team can proceed.

On the existing programs, the relationship with the Centers is more intimate. The Apollo program was underway by the time our organization was formed. Since the mode decision, the details of the system concept have been evolving in parallel with the development of the early hardware. Our formal interface with the development teams is the Systems Specification, which we prepare, and which defines the functional and performance requirements on major systems elements. The specification also ties down the interfaces between hardware being developed by different NASA organizational elements.

In the case of critical systems, we specify the design approach but not the details of the design. We have spent considerable time arriving at the proper level of specifying the design approach. Obviously, the specific design decisions are properly the prerogative of the development team. As an example, for a digital computer we might specify memory capacity, arithmetic speed and input - output functions, but not the type of memory, circuit design or input - output mechanization.

Split of responsibility at this level is one of the keys to an effective relationship between the system people and the development team. In addition, the fact that the system group does not specify design details provides a measure of impartiality which must be maintained by the system group to perform its other function, a technical consulting service and problem audit capability for either center or ONSM management.

Technical impartiality is one of the major keys to effective operation of a systems engineering organization. It is also one of the hardest states of mind to achieve. In an era when any individual function can be implemented in a number of different ways, it is difficult, if not impossible, to specify uniquely which approach is best. The proper frame of mind is to ask whether the approach selected can do the job within the system constraints, not whether it will do the job in accordance with an individual's preconceived notion of the proper implementation.

For the last year we have been carefully recruiting an in-house team of scientists and engineers to do this systems job. Frankly, the task has been much more difficult than we had expected, because of the restrictions inherent within the Civil Service structure. In order to provide a quantum jump in our capability, last February NASA requested the American Telephone and Telegraph Company to assist us in the system engineering effort. We asked them to provide an organization of experienced men able to develop the factual bases needed by responsible NASA officials to make the wide range of decisions required for the successful execution of the manned space flight program.

AT&T responded to this request by forming Bellcomm, Inc., in March 1962.

It is important to note that the Bellcomm organization assists and supports a technical office within the Government rather than an administrative office. This important distinction makes it possible for the Government to properly carry out its total responsibility (utilizing the contractor for assistance) to maintain the final technical, engineering and procurement judgements within the Government itself. The key to our success will be our ability to attract and keep extremely competent men on our in-house staff, which intend to keep relatively small. Present plans call for the group to total no more than 125. Bellcomm presently has approximately 100 technical people on board and we expect it to level off at about 200 such employees

Frankly, we have not been without growing pains over the past year. The task of building a team, setting the standards for performance and establishing working relationships within an existing program structure has been a full time job, and it is far from over. I can report, however that I am convinced we have turned the corner and are beginning to measure up to the task thrust upon us.

So much for organization. I would like to dwell for the remainder of my time on some of the system aspects of the program. The task we are undertaking is so complex that we must design the system to maximize the probability of success and safety. To me, this means continuing to examine all possible ways of performing a particular function, and selecting the subset which provide the highest degree of flexibility and reliability. One of the pitfalls which must be avoided is the dogma of preconceived ideas. The man-machine relationships for space flight has been one area in danger of being overwhelmed by dogma.

For a while I was afraid that Apollo might be one of the last battlefields on which human race took up arms against the encroachment of machines. Catch phases such as "man in the loop", "man out of the loop", the middle ground of "man across the loop", and, I suppose, even man just "looped", have purported to represent the proper solution to one of the more subtle system problems facing the program.

Some of the extreme protagonists in the field would return the Space Age to the good old days of the Wright Brothers when we did not have all that electronics cluttering up the airframe. Others would wrap the astronauts in a cocoon and, through the wonders of the age of automation, deliver him rested and safe to the surface of the moon.

Either extreme is obviously absurd. As you probably could judge, the human factors design of Helto's chariot left much to be desired. Phaeton finally did get to drive his father's course and perished in the attempt when the firy steeds ran out of control. The epitaph which Ovid had dedicated to him has proved immortal:

"Here lies Phaeton, Driver of his father's chariot, which, if he failed to manage, yet he fell in a vast undertaking".

Despite the immortality, I suspect he would have welcomed a "fly by wire" mode or even a little closed loop guidance and control.

Both man and machine have many functions to perform in a complex space mission. Guidance and control requirements dictate the presence of inertial platforms, digital computers, auto pilots and radio links with earth. The man can best control sequences such as docking, selection of the landing site from the hover point and, perhaps, lunar touchdown. Although these functions are important, his essential role is in monitoring the systems, and, in the case of a malfunction, selecting and setting up the alternate operating modes so that the full range of possible redundancies available in the system can be used.

From this point of view, the terms "manual" and "automatic" carry more emotional than technical content. For example to term an inertial guidance system manual because the output of the digital computer is sent to a panel display and then, through hand controller to the autopilot, rather than directly, is stretching things a bit. However, the computer output may be the best point for the man to monitor the system, and having his hand on the controls may place him in the best position to compensate in the event of a malfunction. And these things are subject to analysis. In such a mode, the question of whether man is in the loop or out of the loop becomes one of semantics.

If I attempted to come up with my own trite phrase, the best I could do might be "man optimized in the system" which is sufficiently general and ambiguous. It indicates no particular mechanizations, but, hopefully, indicates an open mind about studying the problems, which is exactly what we need.

The role of the astronaut in the system is an interesting technical problems --- but the reliability of the systems is essential. This is the area in which the greatest strides must be made. Bob Gilruth emphasized it this morning, I'm adding my own two cents worth now, and we must all continue, henceforth and forever, to be aware of the problem and bend every effort to solve it.

If we predict, based on failure rate data available today, how reliable our missions will be, the answers are discouraging. We, within NASA, are convinced that the past need not be extrapolated forward. The success of the last string of Mercury Atlases and the first four Saturn I launches is, we believe, tangible proof that this nation can indeed exercise the design skill and attention to detail so essential to successful space flight programs. But the discipline so far demonstrated has been in a narrow sector of the team. The test, retest and test again philosophy which we use at the Cape and the refusal to accept any unexplained anomolys in the data has been the major difference. This philosophy must extend back to the factories so that the operation at the Cape becomes, truly, the redundant check it is intended to be.

The solution to our reliability problem is not masses of statistically significant data. It is people, their attitude and their competence. And we have a long way to go before all the many tiers of people who work on our programs, ---- any one of whom may, in a number of subtle ways, maime a part which may later fail in flight -- have the training and devotion to their job required. Reliability means that a contractor's attitude should not be one of getting the equipment out the door, "by the inspectors" if possible, and let the follow on checks take care of any deficiencies. Rather, the contractors must be more rigorous, if possible, than the customer. I'm sure many organizations give lip service to such standards -- especially at proposal time. Would that there were nearly as many who actually perform that way.

Ovid's account of Phaeton's story contains the phrase which, I feel, best summarizes the accolade which the people who work on our program must deserve: "Materiem Superabat Opus". Freely translated, it means "the workmanship surpasses the material."

The road to the moon is long. Much of the road is hard and unglamorous. One pundit has noted that the quantity of paper generated during the project, if piled up, would reach the moon before Apollo. We've studied the suggestion and determined that, although it is quantitatively true, the resulting structure would be very unstable and hard to climb. The medical people said we would have to substitute page boys for astronauts. We concluded that the real hardware had to be developed.

Although each succeeding year will bring increasing evidence of accomplishment on the program, we must remember that the national effort, and interest, must be sustained over many years for this project to be successful. Continued support can only come from public understanding of the nature of complex technical programs. It is hard to get across the story of the early phases of a program the things which must be done ere we take to television for the spectacular missions.

The nation must understand that last year was the time of definition for the program. This year will be the year of detailed design and the early phase of the ground test program. The spectacular milestones will be few. In May, Astronaut Gordon Cooper will circle the earth for one day. December will bring the first unmanned test of the Gemini spacecraft, which will later sustain two men in orbit for two weeks.

This summer, the first two-stage Saturn I will be launched. This vehicle was developed by the Marshall Space Flight Center and will place over 20,000 pounds in orbit. This will be the largest payload injected into orbit from the surface of the earth in a single launch. With this milestone accomplished, the United States will be second to none in booster capability.

Perhaps the answer is to let the public participate more. I understand that when the citizens of Texas heard about the 20,000 pound payload, they devised an experiment which, they felt, would demonstrate to the world the capability of the United States and provide a little free advertising to Texas as well. They proposed to place a group of their prize cows in the payload compartment. They argued, logically, that this would be the first "herd" shot around the world and would "beef" up our image. Furthermore, they continued, we would then "milk" the shot for all it was worth, and the world would understand that there was no "bull" about it.

After careful study, it was decided that there was already too much at "steak" on the shot, and the experiment did not "meat" all our requirements. Our guidance people were worried about getting a bum steer. But Brainerd finally said the whole idea was "udderly" fantastic.

Perhaps, in your idle moments you can turn your thoughts to somewhat more concrete solutions to the problem. Ideas will always be welcome.

Thank you.

MILITARY ASPECTS OF MANNED SPACE FLIGHT

by

GENERAL THOMAS S. POWER
Commander in Chief
Strategic Air Command, Offutt AFB, Nebraska

(Luncheon Speech)

President Kennedy has made it clear that the United States has one primary objective in space -- its peaceful conquest. But he has made it equally clear that this country will take every step necessary to protect itself against any threat from space that may be posed by other nations.

Toward this end, the President has assigned the Department of Defense certain tasks which, in his words, entail "increasing responsibilities and burdens.... to make sure that space is maintained for peaceful purposes, and that no nation secure a position in space which can threaten the security of the United States and the Free World."

Some people claim to see an inconsistency in the fact that this country exhibits an interest in military space applications while, at the same time, professing its desire to ensure the peaceful use of the space medium. But there is no inconsistency at all, because our military space effort is an integral part of a single national space program designed to benefit all mankind.

Space potentially offers unique military advantages, and we must anticipate that some nation or nations will endeavor to exploit such advantages to help them attain their political objectives. This leaves us no choice but to protect ourselves against such a contingency, and our military space effort is, therefore, essentially a matter of self-defense which is the right and duty of all sovereign nations. Our main problem in this respect is the fact that we cannot foresee the exact nature of the threat which we will have to meet. Therefore, as Secretary of Defense McNamara pointed out, "the requirements for specific military operations in space are not completely clear."

It should be emphasized that any military space capability which we may develop would not be directed against a particular nation. It would be directed against any potential aggressors who, at some time in the future, might pose an offensive threat in space or attempt to deny us the space medium for peaceful pursuits. I submit that this concern on our part is well warranted. Just as we are now witnessing gradual proliferation of nuclear capability, there is the possibility of future proliferation of space capability, This would enable a growing number of countries to use space for aggressive pruposes unless we are in a position to prevent them effectively from doing so.

The question arises as to what we can do today to meet any space threat of the future. To answer that question we have to choose among three different approaches. The first approach requires that we try to anticipate the type and scope of military threat from or in space we may have to face, both in the immediate and more distant future, and then take all steps necessary to cope with such threat or threats. I call this the "defensive approach".

The second approach entails the expeditious development of a military space capability which is so advanced that it would discourage any attempt to use space for aggressive purposes and, at the same time, augments our present retaliatory deterrent. This may be called the "deterrent approach".

The third approach which has been suggested is based on the realization that, while we have made great strides in space technology, we still know too little about future military space potentials to establish parameters for an operational capability in either the defensive or offensive areas. Therefore, the proponents of this approach maintain that we should direct our military space effort primarily toward basic research in all the scientific disciplines and fields which, in one way or another, can contribute to the development of military space systems, both manned and unmanned. The point is that, once we can determine definite operational requirements for such systems, we would have the knowledge and techniques or, in other words, the "building blocks" to develop them speedily and economically.

The choice among these three approaches is not only a most difficult one but also very critical because we cannot afford to make a mistake. If this country should suddenly be confronted with a "Space Cuba," and have the wrong or perhaps no means to deal with such an emergency, our very survival might be at stake.

What makes the choice so difficult is a complex combination of a variety of factors, such as political considerations, limitation of resources and technological problems. Most of all, the lead time required to bring a new weapon system from original inception to operational readiness has generally been in the order of years and can be expected to be much longer for military space systems even if we should have all the essential "building blocks". Moreover, the state of the art advances at such a rapid pace that any space system under development may be obsolete before it becomes operational and, hence, would no longer suffice to cope with the more advanced weapon systems of an aggressor.

It has been said that military space technology is now at the stage where aerial warfare was in 1908 when the War Department accepted its first airplane from the Wright Brothers. I would go further than that and say that it is at the stage of the very beginnings of the military utilization of air, namely, the French Revolution when balloons were first used for battlefield observation. Then as now, no one could predict the ultimate potential of the new operational medium, let alone speculate on how best to exploit that potential or how an enemy might exploit it.

In trying to illustrate that point let us assume that this was the year 1938 instead of 1963, and you had asked me to talk about the evoluation of aerial warfare during the next 25 years. I might have told you that, by 1963, we would have airplanes flying at speeds of five hundred miles an hour and at altitudes of 50,000 feet, and that these airplanes might carry bombs with the explosive power of ten to twenty tons. You probably would have accepted these predictions as being in the realm of possibility. But what would your reaction have been if I had been able to forecast what we really have today -- combat airplanes without propellers, flying at more than twice the speed of sound and at altitudes of some 17 miles, each carrying the equivalent of millions of tons of TNT? And what would you have said if I had dared to predict that, within less than 25 years, we would have unmanned rockets which could take pay loads of similar magnitude to targets six thousand miles away in half an hour?

By the same token, it is just as impossible to foretell today what lies ahead for us in space. Considering the ever accelerating pace of technological progress, I am convinced that the next decade will bring even more dramatic advances than the past 25 years. For this reason, it would seem unwise to project our programs for military space systems on the basis of present knowledge and present weapons concepts unless we allow ourselves sufficient flexibility and latitude to adapt these program to any future developments.

Therefore, this country endeavors to find ways of meeting the demands of both the immediate and more distant future without committing itself to a rigid approach. Toward that end it may be well to analyze the potential military uses of space, recognizing that this is necessarily a matter of conjecture. Within that frame of reference. I want to discuss some potential areas of strictly military space applications which have a direct bearing on the subject of your Meeting -- the role of man in space.

One such area concerns future means for command and control of our global strike forces. Effective command and control of these forces is an integral component of our overall retaliatory capability, and its survivability in case of a surprise attach is, therefore, a vital element of a credible deterrent. There must be reliable two-way communications between the authorities in command of all combat forces in the field, be they underwater, on the ground, in the air, or, ultimately, in space. Because of the immense scope and world-wide deployment of these forces, there must also be extensive electronic equipment for rapid processing of all information received from them so that the command element can make instant and appropriate decisions.

The Strategic Air Command now has in operation the most advanced and extensive communications network in existence. Among the many measures taken to enhance the survivability of command and control of SAC's far-flung bomber and missile forces are: back-up or "redundancy" of our communications, alternate headquarters and, for over two years, an airborne command post equipped to assume command in case all other command facilities should be put out of commission.

While these measures should insure the survivability of SAC's command and control system for some time to come, continuous improvements will be needed to keep up with any new developments that might impair the effectiveness and survivability of that system. Communication satellites offer a variety of possibilities in this respect. However, we may find that, eventually, the only really survivable command and control structure - not only for SAC but all our military forces - would be one employing a maneuverable command post in space.

Should such a spaceborne command post become necessary, it would have to be large enough to carry all electronic gear required to gather, process and disseminate operational information on a global basis. Also, it would have to be capable of defending itself against any interference or attacks from the ground and space. It is inconceivable to operate such a central command post, especially one deep in space, without a skilled crew to operate and maintain its complex equipment and without competent officers fully qualified to assume command of the strike forces whenever necessary. Here, then, may be the first major requirement for military men in space.

Another area of military space applications conceivably could be means for inspecting suspicious satellites. A large number of manmade objects are now orbiting the earth. In addition to instrumented satellites launched by both ourselves and the Soviets, there is also considerable space debris, that is, components of the rockets employed to put the satellites in orbit. Because of the steadily growing number and small size of all these objects it will become increasingly difficult to keep track of them, let alone determine their nature.

The time may well come when our security will make it necessary to ascertain whether and which of these space objects constitute an offensive threat. They might be orbital ballistic missiles or, perhaps, employ some radically new and still more potent weapon technique.

Future developments may permit conclusive inspection of potentially hostile satellites and, if need be, their neutralization from the ground or by means of some type of unmanned space vehicle. However, it would appear more feasible to use maneuverable spacecraft, manned by crews who can search out suspected weapons carriers, board them and, if required, neutralize them. And this could be another important role for military men in space.

This brings me to the next area of possible manned military space applications, namely, that of space defense. The main problem in projecting defensive systems against any future military threat from or in space stems from the fact that we cannot predict, with any degree of assurance, the nature of the threat against which we may have to defend ourselves. As I indicated earlier, we must expect the discovery of new phenomena and techniques which, by the time there will be operational space systems, may have revolutionized all current concepts and tools of warfare. This means that we must anticipate dramatic advances in the state of the art and technological breakthroughs with which we must be prepared to cope in both their defensive and offensive applications.

There is, however, one definite trend which is indicative of what we can expect, and that is the continuing compression of time in the application of firepower. It used to take months and even years to carry firepower to a military objective and additional weeks or months to apply that firepower in sufficient amounts to achieve the desired results. The airplane compressed the total time for reaching and destroying a military objective to hours; the missile with its nuclear warhead has compressed that time to minutes. It seems reasonable to expect that future weapons will reduce it to seconds and even less. We must bear this in mind as we try to visualize future defensive systems which doubtless will present a still greater challenge than we face today.

As of now, there is not even an effective defense against ballistic missiles although it is safe to assume that the Soviets are just as intent on developing such a defense as we are. While we have limited knowledge about Soviet efforts and progress in this field, except for propaganda statements, indications are that they are pursuing the development of anti-missile missiles. Our present efforts are along similar lines but, even if we should succeed in producing a reliable anti-missile missile, it would be a stop-gap measure at best.

As the missile inventories of both sides continue to expand, the "shotgun method" of missile defense would become increasingly ineffective against an all-out missile attack, with hundreds of warheads and decoys flashing through the skies. The ultimate solution, therefore, may lie in the development of space-based manned systems capable of destroying enemy missiles during their boost phase or in mid-course.

Assuming such a system should become technically and economically feasible some time during the next decade, any nation whose strategic capability were to rest primarily on ballistic missiles would no longer have a deterrent against aggression employing other types of weapons. We have taken this contingency into account by continuing to program our strategic forces on the basis of the mixed-forced concepts. This concepts entails a balanced mix of both manned and unmanned weapon systems in which the advantages of both can be exploited to the fullest, providing invaluable flexibility and optimum effectiveness in their employment.

Most of the Free World's nuclear fire power - between 80 and 90 per cent - is presently concentratcd in the Strategic Air Command which, in addition to its fleet of strategic bombers, has a rapidly growing inventory of Atlas, Titan and Minuteman ICBMs in widely dispersed and well hardened sites. The nation's missile inventory is being augmented by the Navy's Polaris weapon system which, because of its mobility and underwater operation, has good survivability and is, therefore, an effective deterrent. The targeting and operational planning for all these strategic forces, as well as for other nuclear strike forces under the control of the Unified Commands, has been fully integrated through the Joint Strategic Target Planning Staff, an agency of the Joint Chiefs of Staff located at SAC Headquarters.

As a result of this joint effort, there is now assurance that all our nuclear strike forces, being committed to a single strategic target list and single operational plan, fully complement and supplement each other rather than compete with one another as was possible in the past. This has not only strengthened the nation's overall nuclear deterrent immeasurably but also made it easier to maintain that deterrent in the face of any new developments that, otherwise, might impair it. Among such developments would be an effective defense against ballistic missiles along the lines I mentioned. In this case we would merely revise the present concept of our common war plan by assigning a different role to SAC's manned aircraft.

At the moment and for some time to come, the bombers still carry the bulk of the deterrent load, a fact which is not too well understood. As our missile inventories increase, the role of the manned strategic weapon systems will decrease proportionally until we have reached the relative strengths deemed necessary for a well balanced mixed force. We cannot anticipate today what that balance should be five or ten years from now because we cannot predict the factors that may influence it. But if it should become possible to perfect effective defenses against ballistic missiles, we will undoubtedly have to place more reliance on manned weapon systems again.

In fact, manned aircraft may have to serve as a penetration aid to our missiles by seeking out and destroying an enemy's missile defenses, at least those based on or controlled from the ground. This task can be greatly enhanced by employing air-breathing air-to-ground missiles, similar to the Hound Dog with which many of SAC's B-52s already are equipped. In contrast to ballistic missiles, the Hound Dog can fly close enough to the ground to be rather immune to any presently foreseeable missile defenses. Its penetration capability, accuracy and firepower make it such a promising weapon against these defenses as to warrant the development of long-endurance aircraft which can carry a number of Hound Dog missiles on extended airborne alert, without itself ever approaching enemy territory.

I have dwelled on the subject of manned bombers because it has considerable bearing on the last area of potential military space applications I want to discuss, namely, space-to-ground offense.

As a nation, we abhor the very prospect of offensive weapons in space, and if it were possible to conclude enforceable international treaties for banning offensive space weapons, we would no doubt take the lead in promoting such treaties. On the other hand, we must be realistic enough to accept the possibility that, regardless of any future treaties, some hostile nation may succeed in placing offensive weapons in space which would gravely impair our deterrent.

Preservation of our military deterrent is a vital prerequisite for our statesmen in their continuing endeavor to maintain an honorable peace. The value of the "strategic umbrella" represented by SAC, as the nation's principal military deterrent, was demonstrated again during the recent Cuban crisis when, as President Kennedy stated, it "provided a strategic posture under which all United States forces could operate with relative freedom of action". I am confident that, if our survival should demand extensive of our strategic capability into space, we will be ready and willing to do so.

This may require the addition of manned strategic spacecraft to SAC's inventory so as to provide a truly mixed force, capable of accomplishing a wide range of missions across the entire spectrum of strategic operations. No one can predict today what these operations will entail, but I do know that SAC has the competence and flexibility to adapt itself to any new requirement and any new weapon system, no matter how revolutionary. Therefore, I have no doubt that it can meet any future demands of the space age, just as it is now meeting the demands of the missile era.

From what I have said so far, it should be evident that none of the three approaches to which I referred in the beginning - the defensive, deterrent and "building block" approaches - will by itself suffice to counter the overall military threat from space. Instead, we must select that combination of all three approaches which makes the most economic use of our human and material resources on one hand and, on the other, assures us an adequate military space capability for both the immediate and distant future.

Next, we must continue our intensive non-military effort along the entire spectrum of space and space-related sciences. The primary responsibility for this effort rests with the National Aeronautics and Space Administration. Its close cooperation with the Department of Defense will not only further its own objectives of the peaceful conquest of space but also help create the building blocks for the future military systems which may be required, to repeat President Kennedy's words, "to make sure that space is maintained for peaceful purposes".

Finally, we should keep adding to the select group of men who are being trained to live and work in space. I have pointed out the reasons why certain types of military space systems may have to be manned. If and when the time comes to employ such systems operationally, we must be certain to have sufficient numbers of highly qualified and motivated officers to man them. That is a requirement we know we will have to fill, no matter what technological surprises the future may have in store for us.

In this age of computers and automation we tend to forget that, at the beginning and end of the chain, the most important element is still man himself. And regardless of how advanced and complex our weapon systems may be, man has always been and will always remain the only true ultimate weapon. Therein, I believe, lies the real military significance of manned space flight and, therefore, the greatest challenge for the men of the Strategic Air Command.

FLIGHT EXPERIENCES IN THE MERCURY PROGRAM

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INTRODUCTION

At this time, the National Aeronautics and Space Administration (NASA) has successfully completed five manned space flights in Project Mercury. Two of these were suborbital, and of approximately 15 minutes duration, and each of these subjected the pilot to about 5 minutes of weightlessness. The remaining three manned flights were orbital. Two of these flights involved the completion of three orbital passes, or about 5 hours of weightlessness. The last orbital flight was extended to six orbital passes with nearly 10 hours of weightlessness.

As a result of these flights, we have gained a wealth of information about space and man's ability to function in this new environment. There is, however, much which is yet to be learned regarding both of these aspects.

In April of 1959 when the astronauts joined the NASA, we were told that, in our flights, we would be subjected to a frightening combination of stresses resulting from vibration, heat, cold, humidity, noise, acceleration, weightlessness, high concentration of carbon dioxide, immobility, disorientation, radiation, and hopefully, landing shock. Our eyes, inner ears, cardiovascular and respiratory systems, and even our very intellect were considered suspect by many who contemplated man's flight in space. On the other hand, we all felt that this step was a natural one, albeit a big one, in the progress of aviation. We were concerned that there was nothing really mysterious or insurmountable lurking beyond the nebulous division between earth and space. We did recognize that some physiological unknowns would be encountered, and we attempted to familiarize ourselves with them as best we could.

The training program that we put together proved to be of tremendous value. The simulators that were developed for Project Mercury, by and large, provided accurate and representative experience. The most valuable of all the simulators is the procedures trainer, which is a full-sized replica of the Mercury spacecraft with its controls and displays animated by a computer. Except for weightlessness and acceleration, all aspects of the flight can be simulated. There were repeated training periods on the centrifuge. Desert and water survival instruction was given. Zero-g flights in various types of airplanes, star recognition, SCUBA diving, egress training, and systems study were among the more notable phases of this unique education program.

In retrospect, the only unknowns that existed in my mind after this very thorough training program was completed were: what would be the effect of prolonged weightlessness, and what reactions might be caused by various untried combinations of these stresses?

We had been unable to simulate either of these adequately, but, again, we were convinced that, if the machines we had designed could make the flights, then we who had been trained could also accomplish the mission.

Many, many times during the training period and particularly on the centrifuge, I have been filled with wonder at the ability of the human body to withstand and even combat automatically, but intelligently, these strange stresses with which it has had no evolutionary experience. We in Project Mercury think that we have designed a very good spacecraft, but I submit that there is great talent and foresight evident in the design of the pilot as well.

As the title of this paper indicates, it is to be a discussion of flight experiences in Project Mercury. In order to give you a chronological progression of events I have excerpted from the flight reports in their proper order, salient features and experiences. I hope that the results will give you a reasonable appreciation for what it has been like to fly with the Manned Spacecraft Center team which is under the direction of Dr. R. R. Gilruth.

HISTORY OF MERCURY FLIGHTS

Our first ballistic flight was made by Alan Shepard in Freedom 7 on May 5, 1961. He lifted off at 9:30 a.m. after spending a little over 4 hours in the spacecraft. The flight progressed almost exactly as planned with only one minor problem in vibration during powered flight. Al said that his vision was blurred slightly around the maximum dynamic pressure period. This condition has been avoided in subsequent flights by the addition of a sponge rubber pad between the couch and helmet, and by a small modification to the Redstone Launch Vehicle.

Al's flight was a tremendous boost for us all. As you may remember, Yuri Gagarin made his flight about a month earlier and we were feeling a little low, since we had not even made a ballistic flight. Not only was Al's flight a morale booster but it confirmed the adequacy of all elements of our system, including the computers, booster, spacecraft, men, and procedures. The attitude control system worked well and verified the accuracy of our simulations, ignition of the retro-rockets proved to be smooth, and attitude excursions produced by retrofire appeared to be easily controlled by the pilot.

The reentry started on schedule and the associated deceleration peaked at the expected 11g, which is higher than our orbital reentry of 8g, but of shorter duration. A helicopter had hooked onto the spacecraft within 2 minutes after landing. Al made his egress through the side hatch and was hoisted aboard the helicopter, which carried them both back to the carrier, Lake Champlain.

Our second ballistic flight by Gus Grissom on July 21, 1961, in Liberty Bell 7, was similar to the first suborbital flight except for the flight plan procedure and the spacecraft configuration. The flight plan was designed to investigate spacecraft systems not used on the first flight; and the spacecraft had a large window which was directly in front of the pilot, instead of the small portholes at either side. This window permitted a much better examination of the earth's surface and horizon than was possible through the portholes. Maneuvers were made by using alternate control systems and the window as a reference rather than the periscope and instruments which Al had used.

The flight proceeded normally in all respects until shortly after impact when the explosive side hatch, which was being flown for the first time, detonated prematurely for reasons still undetermined; and Liberty Bell 7 began to ship water. Gus made a rather hasty exit and although the helicopter did manage to hook onto the spacecraft, the added weight of the onboard water coupled with an engine malfunction in the helicopter made it necessary to release the spacecraft, which sank immediately. Gus had neglected during his rapid egress to seal the oxygen inlet to his suit, so he, too, lost a great deal of bouyancy and, by the time the secondary helicopter reached him, was nearly exhausted.

Once during egress training, I made the mistake of jumping into the water with my gloves on but not sealed to the suit at the wrist. I lost the trapped aid through the sleeves much the same as Gus did and I can assure you that swimming with the Mercury suit on and no air inside is a mighty tiresome task. I think I have a good feel for how tired Gus was when he finally got into the recovery sling.

Although the onboard data were lost, telemetered information was available for postflight analysis and the flight was a success. On the strength of the data from these two flights, it was decided that the next manned mission would be an orbital flight.

John Glenn's epic three-orbit flight in Friendship 7 on February 20, 1962, was a tremendous technical and personal triumph. Since the news media have given this flight such complete coverage, I will attempt to cover only those details which I think you might not be aware of.

This flight also went substantially as planned, but some minor malfunctions did occur. The first of these was the inability to control the suit temperature precisely. Although the temperature was not excessively high, John was not able to control it accurately. The next problem involved the failure of two of the 1-pound yaw thrusters. Although yaw attitude was controlled by the large 24-pound thrusters, fuel consumption was excessive. So John took over and controlled manually for most of the rest of the flight.

The failure of a limit switch later in the flight gave the telemetry indication to the ground that the landing bag had been released. If this were true, the only thing holding the heat shield in its proper position would have been the retropack retaining straps. For this reason, a decision was made not to jettison the retropack. The reentry progressed normally, however, except for the brighter than average reentry glow which resulted from the vaporizing retropack.

The heat shield was deployed normally and on schedule after the main parachute, indicating that the telemetry signal had been erroneous. Despite these minor problems much information was gained from the flight and it was a decided success.

The destroyer USS NOA had the spacecraft aboard approximately 20 minutes after landing and John made his egress via the side hatch directly onto the deck of the ship.

This flight was a boost to morale for the whole country as well as to all of us in Project Mercury. It had demonstrated man's usefulness and ability to function in space. Many intriguing scientific observations were made, including the presence of the airglow layer and the discovery of the luminous particles. Many color photographs were taken but those of the sunsets and cloud formations are the most striking.

We had, after this flight, first person testimony that prolonged zero-g was not bothersome; on the contrary, John felt that it was a very comfortable state, and the rest of us who have followed concur with him in this. He confirmed that there was no undesirable vestibular effect, no nausea, no vision problem and, subjectively, no reduction in his ability to withstand acceleration after long periods of weightlessness.

As a result of Astronaut Glenn's experience, minor, but important, changes were incorporated into the spacecraft. A modified limit-switch circuit was incorporated into the heat shield jettison system, and many other minor modifications were made to the control system and parachute sequencing. In addition, equipment with which to carry out several scientific experiments was added.

On May 24, 1962, Aurora 7, in which I was fortunate to be the pilot, lifted off on this Nation's second orbital flight. The spacecraft systems operated perfectly except for the horizon scanners which caused an intermittent error in pitch attitude. This error existed at the time of retrofire and necessitated manual attitude control during this period. A yaw error existed which was not corrected and this was the major factor in the 250-mile overshoot.

During this flight, the same inability to control the suit temperature accurately gave me some concern, but, again, after a short familiarization period, the temperature was stabilized at a comfortable level.

Measurements and observations were made of the airglow layer, including its composition, brightness, and wave length, which was strong in the 5577-Å region, and its altitude which is roughly 80 km. The luminous particles were again observed, and it is of interest that these particles seen by John have never been observed on the dark side. This fact leads us to believe that, if they are luminescent, they must be excited by solar radiation. It was also determined that they almost assuredly emanated from the spacecraft because rapping the side of the spacecraft released clouds of them.

Cloud height determination and horizon definition were attempted by photographic means.

I also undertook to determine whether any unusual sensations such as Cosmonaut Titov had reported would result from head motions under zero-g. But I noticed no effect at all even after very violent head motions.

An attempt was made to determine the drag due to the residual atmosphere at the orbital altitude by means of a towed balloon and tensiometer, but this experiment was unsuccessful because of the unsatisfactory inflation of the balloon. A sphere and standpipe installation was included so we could observe the effect of the weightless state on confined fluids. As one might expect, in the absence of gravity effects, surface tension is predominant and it appears that it alone may be sufficient to facilitate the efficient transfer of fuel from one tank to another at zero-g, without the use of bladders.

On October 3, 1962, Wally Schirra, in his Sigma 7, lifted off on our third orbital flight and flew a near perfect six-pass flight, almost putting Sigma 7 down on the deck of the carrier.

Wally, in an effort to prove the capability of the Mercury spacecraft to fly a day-long mission, made extensive use of drifting flight in which no attitude control is attempted, thereby conserving fuel. Although the constraint on fuel consumption precluded an extensive scientific program on this flight, he was able to show that we could indeed fly 18 orbits with the equipment on hand.

During his first pass, Wally also found it difficult to obtain the proper suit temperature, but by the beginning of the second pass, he had it well under control and no further problems were encountered.

This spacecraft again had some minor modifications. Included were a better HF antenna, a yaw reticle on the window to determine whether or not we could establish yaw attitude without the periscope, and the capability of disabling the high fly-by-wire thrusters which helped to conserve fuel.

FUTURE PROGRAMS

Next month, the Mercury program will culminate in the 1-day mission of Gordon Cooper. Gordon will take advantage of the experience we have gained with drifting flight to stretch the spacecraft fuel supplies to cover the 30 hours he will be in space. The area formerly occupied by the periscope has been used for storing extra oxygen for this longer mission. Gordon will continue the modest program of research which John and I began by taking special photographs for the Weather Bureau to determine the relative clarity of cloud formations in the infrared spectral region. Further photographs of the earth's limits and the night airglow are also planned, together with a study of the visibility of a flashing light which may be used as an aid in making a rendezvous in our future space programs, Gemini and Apollo.

Thus, already our activities during these last Mercury flights look forward to our future programs - programs in which manned space flight will really come of age. They will give us the capability of putting several men in space with increased amounts of scientific equipment and of transporting them to our nearest neighbor - the moon.

As a prelude to the other papers of the conference, I would like to add a few words about these future manned spacecraft programs, in which all of us will be vitally concerned. Specifically, I would like to talk for a few moments about astronaut participation in these programs, as a topic that may not be covered so intimately by other speakers.

The Mercury program is nearly wrapped up now and has given us skill, confidence, and knowledge of space flight that we started out to acquire 4 years ago. We are eager to push ahead and use our experience to solve some of the more complex problems that we will face in future flights.

We expect to start the first manned Gemini flights in 1964. The Titan II launch vehicle is rapidly approaching a man-rated state with eight successful flights out of twelve tries thus far in the program. The first production models of the Gemini spacecraft are on the assembly line at the McDonnell plant and we at MSC are turning a large part of our attention to the Gemini systems.

The Gemini missions are going to require a great deal of astronaut control, from the launch to the recovery phase. Although we will be aided and backed up by the same flight operations team that has made our Mercury flights so successful, pilot decision is going to play a larger part in the space flight missions of the future.

The critical time of launch and insertion into orbit will be monitored onboard by the pilot and the exact moment and method of abort will be his decision. The delicate maneuvers necessary for space rendezvous must be accomplished by the pilot, aided as he will be, by high-speed computers, radar, and beacons.

The landing and recovery phase is the third area of pilot control. Developmental problems in the controllable paraglider will require water recovery using a parachute system for early Gemini flights. The beauty of the paraglider system is that it can be put into the Gemini program at any time with a minimum of trouble and little redesign in spacecraft systems. The paraglider has been redesigned to give an increased wing area and a smaller angle of sweep. With these and other lesser design changes, the maneuvering ability of the paraglider has been increased, providing a positive control landing system for the Gemini pilot.

One of the greatest benefits we expect from Gemini flights will be the familiarization they provide with the problems of rendezvous which are vital to the lunar landing scheme now being pursued. These flights will also provide excellent orbiting scientific laboratories from which we stand to learn many, many things about our space environment.

Although the lunar orbit rendezvous technique provides for landing only two of the three astronauts in the Apollo spacecraft, from the astronaut's aspect, it continues to make use of the team concept in space flight. Since the Lunar Excursion Module will have no long-distance communications capability, the astronaut remaining in the lunar orbiting Command Module will serve as a monitor and coordinator for the men on the surface, relaying their status and observations to the deep space tracking stations on earth. We plan to have three of these powerful communications stations for the Apollo mission in order to keep in constant contact with the voyagers. However, no matter how close the contact between earth and spacecraft will be, the decisions on orbit injection, midcourse correction, and landing must be made by the astronauts.

CONCLUDING REMARKS

I think this philosophy sums up our attitude toward manned space flight. It is the increasing importance of the man in the system that is going to characterise our space flights of the future. We will need skilled pilots to fly these missions, and we are in the process of developing those skills now.

The era of manned space flight is just beginning, however, and Project Mercury is only the first step. We are convinced that it has been a very profitable step and that succeeding programs in manned space flight will continue to expand our knowledge of the universe, hopefully for the benefit of all mankind.

RESEARCH REQUIREMENTS FOR FUTURE MANNED SPACE FLIGHT COMMUNICATIONS

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Introduction

The research requirements for manned space flight, in many respects, are "staggering to the imagination," and yet the Nation is embarking on just such an undertaking. Research requirements are large because prior to 1961, when President Kennedy established as a national space goal a manned lunar landing for this decade, few dared to think seriously in terms of such a project. It is not the enormity or possibility of success of such a goal that creates research requirements, but rather the need, via research, to lower the cost required to continue manned space flight beyond the moon. Once triggered, the imagination transcends the original concepts and pushes to even higher goals. To achieve these goals, such as manned explorations of Mars, Venus, Pluto, and some of our nearer stellar neighbors, many scientific breakthroughs will be required to make these explorations feasible.

Many technological areas of research too numerous to list must be pushed simultaneously to achieve the required capabilities. This paper will discuss communications research activities required for future manned space flight missions.

Requirements

There are many different factors involved in manned space flight that have either a direct or indirect bearing on communications requirements. Any listing of such factors would include the following:

1. Public interest
2. Available resources
3. Mission objectives
4. Size and weight of spacecraft
5. Booster capabilities
6. Mission time
7. Experiments carried
8. System configuration

Needless to say, it would be unrealistic to expect the communications system to meet all requirements, or for this paper to discuss all the factors bearing on these requirements.

On the top of the list is public interest, since, to a large extent, I believe this to be of the utmost importance and should be considered more seriously by the scientific community than it has yet seen fit to do. It is up to the scientific community to convince the public (i.e., the customer) that manned space flight and exploration is not only a good, useful, and worthwhile undertaking in itself, but an investment in the future as well. To be somewhat trite--at the minimum, we must put on a good technical show or demonstration for the public, as well as show the everyday benefits that befall the public as a result of our space effort.

To maintain public interest in space exploration places a much more severe requirement on communications than the sum total of scientific experiments. For instance, a minimum requirement should be real-time television for close-in manned missions such as exploration of Mars and Venus and a reasonable amount of picture transmission from the more distant planetary missions. Coupled with this requirement is that the size and weight of the communications system must be no greater than present equipments, and the reliability at least an order of magnitude better.

As a result, the nominal communications requirements for future manned space flight are:

Bit Rate	= 10^6/sec.
Size and Weight	= no increase
Reliability	= order of magnitude improvement
Primary Power	= no increase

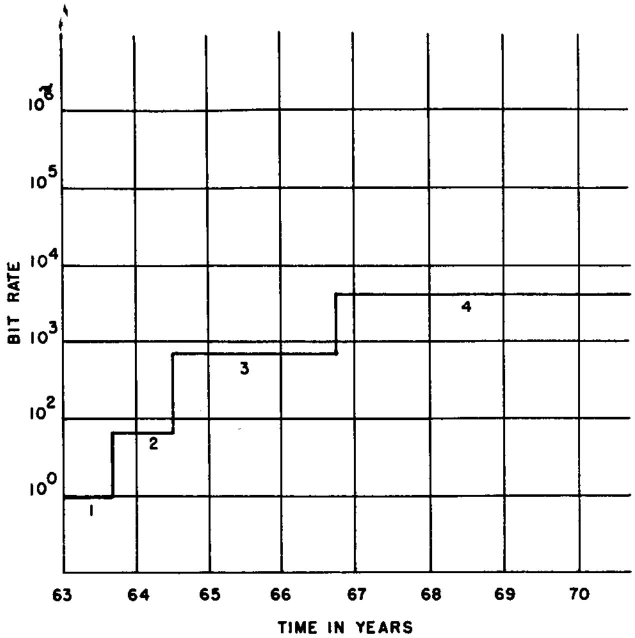
for systems transmitting from our immediate planetary neighbors and somewhat less in bit rate for the more distant missions.

The preceding argument for a communication bit rate of 10^6/second does not include any reference to unmanned space probes which must precede manned exploration. If a comparable increase in the communications capability could be incorporated into the unmanned missions, which are designed to provide the information required prior to undertaking manned exploration, then a considerable savings could be realized in national resources. Communications capability for presently scheduled lunar and planetary probes limits both the type and number of experiments contemplated. It limits the type of experiment in that the data rate is too low; and the number, in that the power consumption and weight of the communication system is an appreciable percentage of the spacecraft.

It is also important to note that an increased communications capability of the scale previously cited would allow a striking demonstration of the technological advancements that our free enterprise system is capable of achieving; and if this capability were achieved first, would emphatically demonstrate the superiority of our system of government in achieving technological advancements.

Present Capabilities

Our present communications capability, while a source of pride to both the Nation and the NASA, falls far short of the aforementioned requirements.



	COMMUNICATIONS PARAMETERS				
	FREQ.	GND. ANT.	VEH. ANT.	XMIT. PWR.	NOISE TEMP.
1	960 MC	85'	4'	3 W	70°
2					20°
3	3.84 KMC				
4		210'			

Figure 1 shows the capability of the Deep Space Instrumentation Facility for communications to the vicinity of Mars, as well as the projected capabilities resulting from improvements which can reasonably be expected to be incorporated into the system.

The expression for the theoretical bit rate for a digital communications system limited by gaussian noise is:

$$I_{max} = \frac{4.6\, P_t\, G_t\, A_r}{4\; R^2 K T}$$

P_t = S/C transmitter power

G_t = S/C antenna gain

A_r = Receiving aperature

K = Boltzman's constant

T = System noise temperature

R = Range of communication

which gives for a system with the following parameters:

P_t = 3 watts

G_t = 20 db

A_r = 85 ft. receiving aperture (25.5 meters) (5680 ft^2)

a nominal bit rate of 200 bits/second from the vicinity of Mars. This bit rate is considerably higher than could be achieved in practice for two reasons: first, the hypothetical system is an infinite bandwidth system; secondly, no communication system to date has been built that comes closer than 6 db to the theoretical limit. As a result, the operational bit rate for a communications system with the given parameters is nominally 10/second. To meet the desired requirements of 10^6 bits/second will require an improvement of 50 db.

For a given range, the bit rate is proportional to:

$$\frac{P_t A_t A_r f^2}{T}$$ where A_t is area of S/C antenna, f is frequency

The design engineer can improve the system capability by varying these parameters. It is reasonable to expect that the following improvements could be made without changing the weight carried aboard the spacecraft.

frequency	- increase by a factor of 4	- 12 db
T_{eff}	- decrease by a factor of 4	- 6 db
A_r	- increase by a factor of 6.31	- 8 db
	Total	- 26 db

The remaining 24 db improvements must be made onboard the spacecraft. It therefore adds to the weight and complexity of the spacecraft. It is reasonable to expect that advances in the component technologies, such as higher efficiencies, better structures, energy storage, and large self-erecting spacecraft antennas, will allow as much as 5 db improvements without increasing spacecraft weight. In summary, without a major "break through" in technology, about 20 db improvements at the expense of spacecraft weight must be made in order to achieve real-time television reception from the vicinity of Mars. Of course, the capability from the vicinity of Pluto would be approximately 20 db worse.

Research Approaches

There are many methods by which you can improve the performance of the communications system. Some of these are:

1. Increasing antenna sizes
2. Increasing transmitter power
3. Increasing the frequency
4. Lowering the system noise temperature.

Other methods which come to mind for improving the communications system are by improving the modulation techniques to closer approximate Shannon's limit, employing on-board data processing to eliminate redundancy and thereby increasing the information per bit, and by carefully designing the experiment. The current efforts in these approaches will be discussed in the following sections.

Antenna Size

NASA is presently negotiating with Industry for the construction of a 210-foot-diameter parabolic dish which should be available for incorporation into the DSIF in the mid-to-late Sixties. Studies have shown that for larger antenna apertures, the arraying of smaller antennas to achieve the same effective aperture is economically competitive. Presently, NASA is exploring techniques to more effectively realize this equivalence. (It is interesting to note that the Russians employ the arraying technique to achieve high antenna gains and large effective apertures for their space communications.)

NASA efforts in the arraying of antennas is presently limited to an in-house effort with emphasis on a system of signal processing which has promise of improving performance at low signal levels.

Considerable engineering effort must be undertaken in the arraying of antennas before a decision can be made concerning this approach as opposed to larger single dish apertures. In a theoretical sense, the approach appears rather straightforward--detect and lock-on in phase the received signal in each of the antennas and then coherently add the signals from the individual antennas. As simple as this appears, there are a significant number of problems that must be investigated before such a system will be a reality. These are:

1. Propagation effects of the atmosphere
2. Optimum separation of antennas
3. Modulation techniques
4. Antenna size
5. Operating frequency
6. Number of antennas that can be arrayed.

In fact, attention must first be directed toward the parameters and configuration of experiments that will identify the technical and operational problems involved in arraying antennas to achieve apertures greater than 500 feet in diameter or gains of 75-85 db.

Space Antennas

Space offers essentially a gravity-free environment, and, possibly, lightweight structural techniques could be employed to achieve antenna apertures in space considerably in excess of 500 feet. Such an antenna must await more economical boosters and advances in space and structural materials technology; however, it is not too early to perform design studies on such antennas to clearly identify the important areas of research.

Optical Techniques

The laser or optical maser has opened a new realm of possibilities in space communications. There have been many papers written concerning the possibility of employing lasers for deep space communications. In general, the papers either appear to be the type that allow the possibility of employing optical techniques in space communications or point out that optical techniques are not applicable to space communications. Oddly enough, most of these papers have been written by microwave experts and one is led to history to draw a parallel. In the early part of the 20th Century, there was a similar lack of optimism concerning the automobile, with people continually pointing to the advantages of the horse and buggy. Certainly microwaves will never take the type of backseat that horses were forced into as a result of the automobile; but it might well be that optical techniques will replace microwave techniques in deep space communications.

The research and development that must be performed before optical communication becomes a reality is considerable, and a listing of the major advancements required are:

1. Stable frequency oscillator
2. Efficient conversion of electrical power to coherent optical radiation
3. Reliability of optical power oscillator
4. Pointing accuracy
5. Acquisition technology.

To indicate the improvements possible using an optical communications system, a comparison will be made between microwave and optical systems with both systems having the same power supply drain. The calculation is based on the following assumptions:

1. Optical oscillator efficiency is 1/100 that of the microwave and operates at 6000 angstroms.

2. Spacecraft aperture of the optical system is 1/100 microwave aperture.

3. Optical ground receiving aperture is 1/400 the aperture of the microwave receiving aperture.

Effective systems temperature of optical system is given by:

$$T_{eff} \text{ optical} = \frac{hf}{k} = 2.4 \times 10^4 \ {}^oK$$

where h = Plank's Constant

k = Boltzman's Constant.

Comparing the bit rates by the expression:

$$\left(\frac{P_t A_t A_r F^2}{T} \right)_{MW} \bigg/ \left(\frac{P_{to} A_{to} A_{ro} F_o^{\ 2}}{T_o} \right)_O$$

it is found that for these parameters the optical communications system is 22.6 db better in performance than the microwave system. The 6 db future improvement in the microwave effective noise temperature as was mentioned earlier cannot be matched by a like improvement in the optical system, reducing this advantage to 16.6 db. Research in the efficient generation of optical frequencies has the possibility of obtaining a further gain of 20 db, thus an overall 36.6 db advantage for the optical system. This magnitude of system improvement allows serious consideration of real-time TV reception from the vicinity of Mars.

Possibly the greatest advance required in technology for application of optical communications is that of beam pointing. In the previous comparison, the optical aperture was 4.8 inches in diameter which results in a one-second of arc beamwidth. As a result, the pointing accuracy of the spacecraft system must be within a few tenths-of-a-second of arc in order to realize this 16.6 db advantage. Let us now consider an unconventional approach to improving the operational laser system that is to extend the microwave techniques of coherent signal mixing to higher and higher frequencies. Such advanced technology would permit the arraying of optical telescopes. This increase in effective aperture would provide an improvement in received signal of well over 10 db or the same magnitude of improvement possible by arraying of microwave antennas.

Whenever optical communications is proposed as an operational possibility for future space communications, the "opposition" immediately puts a cloud over the system. Admittedly, a cloud seriously impedes any optical system, but this impediment normally represents no more than 12 db degradation to a camera. The most degrading feature of a cloud appears to be physiological since it prevents man from physically seeing or forming an image with the light received. This does not preclude the use of optical frequencies but points to an area where further understanding is needed.

S-66 Laser Tracking Experiment

As an initial step in providing many of the answers needed before optical technology may be brought to bear on the challenges of space communications and tracking, an active optical system will be employed to track a satellite. The experiment will be performed by Goddard Space Flight Center and Wallops Station personnel using the S-66 Polar Ionosphere Beacon Satellite as a reflector. Initially, the NASA effort will be directed toward achieving high precision satellite tracking data.

The equipment for this experiment is composed of three subsystems: transmitter, satellite retrodirective optical system, and receiver.

1. Transmitter: The transmitter is a Q-switched ruby laser operating at 4.32×10^{14} cycles per second (or a wavelength of 6940°A). Energy output is a one-joule 200-nanosecond pulse per second; transmitter beamwidth is three minutes of arc, or an effective transmitting antenna gain of 70 db.

2. Satellite Retrodirective Optical System: The satellite portion of the equipment is an array of retrodirective optical elements or corner reflectors. The individual elements are fabricated of fused silica and have a 2.5 cm hexagonal pupil. Reflected energy is constrained to a one-second beamwidth which is equivalent to 93 db gain. The satellite reflector assembly consists of nine panels having 40 elements each, for a total of 360 elements. This assembly is the frustrum of an octagonal pyramid whose base width is 45.7 cm, top width is 20.3 cm, and 17.8 cm high. The effective reflecting area is approximately 800 cm^2.

3. Receiver: A modified IGOR tracking telescope, which has a 45 cm diameter aperture. Telescope modifications consisted of replacing the camera with a 10A° filter and a photomultiplier. In operation, a satellite slant range of 1,500 km is expected. Actual satellite altitude is 1,000 km. Its velocity is 7.4 km sec^{-1}. These operational parameters will create a relativistic velocity aberration which will be manifested by the beam center of the reflected laser pulse being returned to a ground point 38 meters forward of the transmitter. However, the returned beam spread is sufficiently wide to permit the transmitter and receiver to be mounted together.

Accuracies to be expected with this system are:

Range - ± 3 meters

Range rate - ± 15 meters sec^{-1}

Angle - ± 30 seconds of arc.

However, the returned pulse has sufficient signal-to-noise ratio to permit the satellite to be photographed against a star background. The photographic mode of operation can give angular accuracies of two seconds of arc.

The satellite reflector assembly was designed to last for over a hundred years--this limit is imposed by a darkening effect on the corner reflectors as a result of the radiation belts.

Ground equipment, which will be continually updated, can be compared to its predecessor. In addition, those qualified individuals who wish to participate will have a common target, thus facilitating the reduction and correlation of the experimental data.

The spacecraft optical system is a type of laboratory equipment and as such will allow many different kinds of experiments to be performed. Experiments are being planned to investigate the following classes of problems:

1. Signal-to-noise: How much energy is lost in transmission? How is this energy lost? What affect does background energy have on laser operations?

2. Signal adulteration: How does the aerospace environment affect the intelligence being carried and the recovery of information on a coherent optical beam?

3. Physical phenomena: What are the refraction characteristics of a coherent optical beam in an aerospace environment, the polarization properties, and even the coherence properties?

There are, of course, many other important experiments unmentioned and even not envisioned that can be performed using the satellite as a reflector. Those listed above represent the initial efforts of a continuing program of investigation.

Reentry Communications

Presently configured communications systems do not provide any capability during one of the most critical phases of manned space flight--during reentry into the earth's atmosphere. Project Mercury flights experience greater than four minutes of "blackout," and manned lunar missions will experience even longer times of reentry communications blackout. The reentry blackout gets progressively worse the farther into space man goes and returns.

There are many research approaches to minimizing the blackout period created by the shock-induced plasma surrounding the spacecraft. Examples of possible approaches are:

1. Aerodynamic shaping: This is an obvious approach but has limited applicability due to need for aerodynamic breaking.

2. Increase frequency: For deep space missions, the frequencies needed to provide reliable communications during reentry extend beyond the frequency where efficient power generation is possible.

3. Magnetic field: Strong magnetic fields applied to the plasma act to constrain electrons motion and thus make the plasma appear as a dielectron. Need for strong magnetic fields limits usefulness of concept.

4. Materials addition: Materials can be injected into the plasma which effectively cool the plasma and thus decrease ionization. Applicability of this approach has not been fully evaluated; however, an active research program is being pursued in this area.

5. Plasma antennas: Since the plasma is essentially a conductor, some means should be possible to use the plasma as the antenna. To date, no promising concepts exist.

The same solution to reentry blackout will be applicable to manned fly-by exploration where a probe is sent from the spacecraft to the surface through the planetary atmosphere.

Component Technology

Continuing efforts are needed in component technology to increase the reliability of the communications systems on spacecraft and to reduce the size, weight, and complexity. Research efforts in these areas should be striving to increase lifetime and efficiency of transmitting devices, as well as obtaining semiconductor devices which have long life in the space environments; i.e., radiation resistant semiconductor devices.

Operation of present day semiconductor devices, because of their dependence on the crystal structure and high degree of purity, are vulnerable to radiation damage. Identifiable radiation damage mechanisms are:

1. Displacement effects: where the atoms are displaced from their normal lattice sites.

2. Surface effects: where the residual gas in the semiconductor package is ionized, affecting the surface of the device.

Research efforts are currently being pursued to achieve an active thin film device which will be radiation resistant. With such devices, the radiation should easily pass through the thin film without seriously affecting the physical structure of the material. On the other hand, since such a device would not depend on the crystal structure for its operation, the radiation displacement effects are not present.

Microelectronic technology is an outgrowth of the transistor technology where the material and fabrication processes are well understood, well controlled, and generally involve fewer steps than the manufacture of conventional electronic components; so one might conclude that reliability is inherent in microelectronics.

One of the limiting factors in modern day technology involving the reliability of electronic systems is the number of connections or interconnections which must be made. It is widely recognized that the reduction of these connections, either within integrated elements or between integrated circuits, is of great importance in achieving reliability. It is in this area where further study is required, since the application of microelectronic techniques without major revisions in interconnecting circuit elements and system components will yield only part of the potential benefits from use of these techniques.

Studies are being initiated to determine pertinent applications of the microelectronic technology as it applies to specific NASA spacecraft electronic systems.

Component technology must advance to the point where on-board data processing can be used extensively without degrading overall system reliability. Pre-processing of experimental data will allow an increase in the information per-bit of communications and thus indirectly increase the communications capability. For instance, much of the TV picture is redundant and substantial reduction in the bit rate needed to transmit real-time TV can be realized by pre-processing the picture data.

Much has been said about component technology and reliability and the promise the future holds in this area. It remains to be seen whether or not these advances will materialize and be a panacea for spacecraft reliability.

Speech Compression

Considerable effort is being expended by the Government, Industry, and the universities in speech processing. The objectives of this ongoing effort vary from that of obtaining secure voice communication to assisting the blind. This research effort provides a broad base of technology that can be exploited for manned space flight communications; however, the speech compression systems evolved to date are not directly applicable.

The following are major constraints imposed on previous work that are not applicable to manned space flight.

1. Universal applicability. Previous speech systems have been designed to work with any speaker. There are a limited number of astronauts and it is conceivable that a speech compression system could be tailor-made for each astronaut.

2. Two-way. The requirement for a speech compression system for manned space flight is to limit the spacecraft complexity and power consumption and essentially imposes no such restriction on the ground system; in fact, the earth-space link could be a conventional voice channel.

3. Bandwidth. In manned space flight, the primary effort is energy conversion as opposed to conserving bandwidth.

4. Real-time. Space distances preclude real-time in that propagation times are measured in minutes.

5. Speaker identification. Due to the complexity of placing a man in deep space and the equipment required to receive voice communication from deep space, it is not necessary to place the burden of speaker identification on the voice channel.

Study effort is currently underway to determine the proper orientation of a NASA research program to take advantage of these differences in designing deep space voice communication. This study effort should indicate whether or not there is sufficient pay-off in this area to warrant NASA research.

Summary

Considerable research effort will be expended by the NASA, as well as the electronics industry, in increasing the capability of communications systems for future manned space flight. This research effort will pay dividends in both the manned and unmanned space missions. All the research activities touched on in this paper are oriented toward increasing the information received from a spacecraft. Some of these efforts are aimed at increasing the bit rate; whereas, others are aimed at increasing the information content. The reentry communications applies to the reliability and safety of manned space flight as well as making possible certain types of missions such as a fly-by with a landing probe.

There are a large number of operational research problems associated with future manned space flight which have not been discussed, such as below-the-horizon communications on the lunar surface, communication between the orbiting spacecraft and excursion vehicles, and the affects of planetary atmospheres on communication. Many of these answers and research efforts must await more information about the environment and types of missions contemplated.

4

GEMINI PROGRAM AND MISSION

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SUMMARY

Gemini is the manned space vehicle system following on from Mercury. It is planned to use this system to lay the groundwork of experience required for future manned space programs. Flights of up to 14 days in duration will be attempted, and techniques for rendezvous and docking with another space vehicle will be established. Some details of these and other associated mission objectives are discussed.

INTRODUCTION

The Gemini program has been established to provide a flexible space vehicle system which can be used to explore the many problems of manned operation in a space environment in much greater detail and depth than could be done with the pioneering Mercury vehicles.

Although there are many objectives that may ultimately receive attention, the following specific objectives have been selected for the present program:

1. Long duration (up to 14 days)

2. Rendezvous

3. Maneuvering in space (before and after docking)

4. Extravehicular activity

5. Landing at a preselected site

6. Provision of a platform for scientific experiments

The specific configurations of the Gemini vehicles have been designed to accomplish these objectives. Although more detailed descriptions have been given elsewhere[1,2], it is believed that a very brief summary of the basic characteristics should be given at this point.

The spacecraft is shown in figure 1. The reentry vehicle is similar to Mercury in overall shape, but it is enlarged to accommodate 2 men. The equipment installations have been revised for ease of assembly and checkout. The Gemini configuration also departs from Mercury inasmuch as the Gemini retains in orbit the adapter section which mates the reentry vehicle with the launch vehicle. In this section, many of the supplies and systems are carried which are used in orbit but not for reentry.

Although landings will initially be made with a conventional parachute, a paraglider, shown in figure 2, is under development and is expected to be available for later flights. The paraglider is a wing which is designed to be stowed in the after end of the spacecraft, to be deployed at the appropriate time, and inflated so that a landing can be made similar to that of a conventional airplane.

The launch vehicle used to insert the spacecraft into orbit is a Titan II which has been modified in some respects, especially by the addition of those systems required for pilot safety.

The target used in the rendezvous program is a version of the Agena D adapted to the rendezvous requirements. The Agena will be launched by an Atlas Standard Space Launch Vehicle.

In order to assist the reader in understanding the terminology of this paper, a list of terms and definitions is presented in the Appendix.

GENERAL OBJECTIVES

The long duration flights will be designed to provide much information on man's ability to live and work in a space environment. The crew's physiological condition will be monitored, and their ability to perform tasks will be evaluated. Experience will be gained in the problems associated with eating, sleeping, and waste disposal under prolonged periods of weightlessness. In general, this part of the program will use the same types of instrumentation and methods that were used on Mercury. These long duration flights will also provide a very critical test of the reliability of the spacecraft's systems, which must be specially configured to meet very severe requirements in this regard.

One of the features of the present program is the experiments in rendezvous. Since rendezvous has many planned and potential uses in other space programs, considerable thought was put into designing these experiments so that a groundwork would be laid that would be useful in all applications to which rendezvous might be applied. Because of the complexity of these missions, a detailed discussion will be given later.

The ability to maneuver in space and perform orbital corrections after insertion is also fundamental to many space programs. During the rendezvous operation, maneuvers of the target and spacecraft under both ground and onboard control will be accomplished. However, after the spacecraft has docked with the target, it will be possible to check out and light up the main engine of the Agena. This operation will exercise all the safety monitoring features necessary when a man is involved. Similar precautions are also necessary with respect to the navigation system, which is responsible for making it possible to make a safe reentry.

On some of the flights, it is proposed to depressurize the cabin and open the hatch and allow one of the astronauts to climb out of the cabin using a 30-minute supply of oxygen strapped to his suit. In this experiment, it is proposed to test the astronauts' mobility and ability to perform tasks. The feasibility of repair and assembly of equipment in space will be explored during these excursions. Answers to the many questions in this area are vital to the concepts that will be used on future space vehicles and space stations.

The Gemini spacecraft has been designed to land in a preselected area, and after the introduction of the paraglider, at a preselected point on land. This is intended to simplify the recovery problem in the primary landing areas, as compared with recovery at sea. It should be noted that it is anticipated that the use of alternate landing sites will be necessary when the weather is below the minimums established for a safe landing. The establishment of these minimums is a very important part of the paraglider program.

Initially, it is planned to deploy recovery forces in much the same way as was done in the Mercury program. A great part of this effort is devoted to covering the principal abort cases which almost invariably involve water landings and large Naval recovery forces. As the program matures and confidence is established in the reliability of the systems, it is hoped that it will be possible to reduce greatly these recovery forces and rely almost entirely on a relatively small contingency effort.

In consonance with the basic objective of the Gemini program, it is anticipated that there are many scientific observations and experiments that can be included in the Gemini program. Although many features are included in the basic design that should facilitate the introduction of a variety of experiments, detailed planning of these experiments has not yet passed the very preliminary stage.

PROGRAM OUTLINE

The Gemini program has established a series of flights to develop the required techniques, gain experience, and provide information on the earliest time schedule for long-duration missions and rendezvous operations in the areas discussed in the previous section. The first phase of the program will be unmanned flights to qualify the spacecraft and launch-vehicle systems. Because of the complex nature of the programs and procedures unique to rendezvous, the first group of manned flights are categorized as long-duration missions; nevertheless, preliminary experience in terminal maneuvering on some of the first manned missions will be accomplished with a small target carried in the adapter section of the spacecraft and ejected in orbit. The final missions of the program will be devoted to the development of rendezvous techniques and, ultimately, postrendezvous exercises.

Since a practical rendezvous program requires the development and optimization of a number of important areas, some of the basic requirements for rendezvous and the way in which the Gemini program will be implemented to solve these problems are discussed in the following paragraphs.

RENDEZVOUS OBJECTIVES AND METHODS

As was previously stated, the objective of the Gemini rendezvous program is to develop and to demonstrate techniques which will be practical for the majority of possible space programs which will utilize rendezvous. This requires a more comprehensive program than one in which a specific type of rendezvous is accomplished a few times. The program must include the use of different techniques and a wide range of variables, so as to arrive at the optimum and most practical procedures. Since the performance and accuracy of systems used in rendezvous are initially only based on estimates and even the feasibility of some features of these systems is not completely established, it was necessary to adopt a very conservative assessment of ability of these systems to meet their design performance, at least in the early stages of the program. For this reason, it was felt that the hardware should be designed so that alternate methods of rendezvous could be provided both in the spacecraft and in the target. In this way, the possibility of accomplishing at least some of the objectives of any given mission are maximized. As experience is accumulated, techniques will be refined and developed to the point where practical operating methods are established. In order to accomplish a flight program involving these concepts, all the rendezvous vehicles will have the same configuration and hence the same capability. Although experiments will be designed to exploit this capability progressively, the planning is such that if, for instance, the launch window actually achieved and the insertion accuracy are substantially better than anticipated, this fact can be exploited even in real time in optimizing the subsequent maneuvers to take advantage of this circumstance and to minimize the steps required in achieving the final objective of arriving at practical operating techniques.

Rendezvous will be an essential part of a number of future manned space flight programs. For instance, rendezvous has been selected as a primary phase in the Apollo lunar landing mission. Types of programs that will require the use of rendezvous as one phase in the achievement of final objectives are as follows:

1. Lunar orbit rendezvous (Apollo)

2. Earth orbital rendezvous (deep space)

3. Space stations

4. Satellite inspection and repair missions

In order to develop general rendezvous techniques that will be valid for future space applications, it is necessary to determine similarities among the various possible rendezvous programs.

The analysis of the rendezvous requirements for these different types of programs points out that a number of basic characteristics are common in each program when the basic premise of minimizing the operations and fuel required is used. Some of the important characteristics are as follows:

1. Target vehicle would be established in orbit prior to spacecraft launch.

2. A launch window of sufficient length would provide the necessary confidence level for operational use.

3. Launch techniques for the manned spacecraft would limit the out-of-plane error to within the terminal maneuvering capabilities of the chaser vehicle.

4. One vehicle would maintain a fixed orbit unless an emergency condition arose.

5. The initial orbit of the two vehicles would be of different periods for overcoming phase differences.

6. Terminal guidance techniques would be capable of overcoming dispersions generated through the launch and midcourse phases of the mission.

From these basic characteristics, Gemini has established a general operational rendezvous technique that will apply in the different rendezvous programs. This operational technique is shown in figure 3, which is a flow diagram showing the possible steps in an operational mission from the launch phase to the docking phase.

The target vehicle would be in orbit prior to the initiation of the manned spacecraft launch sequence. The spacecraft would be launched by using a variable-azimuth launch technique. This technique is presented in more detail in a subsequent section. The variable-azimuth launch technique controls the out-of-plane errors for an acceptable period of time within the maneuvering capabilities of the terminal guidance techniques. This allows the out-of-plane corrections to be determined and applied during the on-board controlled phase of the mission, making relatively complex and expensive plane adjustments unnecessary in the midcourse phase of the mission.

If launch of the spacecraft occurs at a time in the launch window when the phase relationship between the target vehicle and the launch site is optimum, then the mission sequence will go directly from the launch phase into the terminal phase. By definition, this transition is called an immediate rendezvous. Should spacecraft launch occur at any time in the window when the phase relationship is less than optimum, then the period difference created by the altitude difference of the target and spacecraft orbits would provide a fixed initial catchup rate for correcting the phase error. If a different catchup rate is required to reduce trailing displacement dispersions in the terminal phase of the mission, the chaser vehicle, which in the majority of cases will be the spacecraft, would adjust its orbital period to provide the desired catchup rate. The ground computer complex can determine the adjustment and provide the chaser vehicle with the appropriate information through a command link or by voice communications. It is also possible to determine this adjustment at the time of spacecraft launch; and with suitable programs in the spacecraft's on-board computer, the adjustment can be determined on board.

There are two primary methods of conducting a terminal maneuver. One method uses orbital mechanics to optimize the intercept course of the chaser vehicle. This method requires the use of a radar with angular measuring capability and an on-board computer. The method minimizes the fuel required for the maneuver, and thus allows relatively large initiation ranges between the target and chaser vehicle. Approximately 30° to 330° of orbital travel would be required to complete the maneuver, depending on the accuracy of the launch and midcourse maneuvers.

A second method for conducting the terminal maneuver would use optical guidance for establishing the intercept course. This method is more expensive from a fuel consideration since it does not use orbital mechanics as such to optimize the maneuver. The initiation ranges would be more restricted, but the hardware requirements would be minimized. In order to use an optical guidance method, ranging information must be provided to the astronauts. This information can be provided by a radar with range and range-rate capability or by cruder optical measuring devices such as a reference grid[3] or simple sextant.

As the requirements of different rendezvous programs will vary, so will parameters, such as time, altitude, inclination, and velocity, which form the restraints and boundaries of the different steps shown in figure 3. In order to determine these values of a given program, five major parameters must be evaluated and interrelated until they are compatible. These major parameters are as follows:

1. Operational launch window

2. Fuel allotted for rendezvous

3. Target inclination

4. Orbit altitudes

5. Mission time

The operational launch window is the actual time period which is required to provide an acceptable confidence level for operational use for the particular launch vehicle and spacecraft under consideration.

Control of the out-of-plane error requires the greatest expenditure of rendezvous propulsion. Using controlled or variable-azimuth launch techniques in conjunction with the target inclination relative to the launch site will control the out-of-plane errors between certain limits for a period of time. For a given amount of fuel, or velocity, to be used for maximum out-of-plane conditions in the terminal phase, a plane launch window is established. This launch time period with respect to out-of-plane errors must be as large as the acutal operational launch period required by the vehicle. If the plane window is smaller, either the target inclination must be lowered with respect to the launch site, the fuel allotment must be increased, or countdown and launch procedures must be improved so that a smaller operational launch window is required.

The difference in the orbit altitudes of the two vehicles will provide a fixed catchup rate between the target and spacecraft for correcting phase differences. The mission time sets the amount of time in which the midcourse catchup maneuver must be completed. Catchup time and rate regulate the number of degrees of phase difference that can be allowed when the spacecraft is launched. This limit of phase difference provides what is called a "phase launch window." Again, as with the out-of-plane consideration, this phase launch period must be as large as the operational launch window. If the phase window is smaller, either the catchup rate must be increased, the mission time extended, or the required launch window reduced.

Depending on the relative sizes of the phase and plane windows, either one or both of which may be much larger than the operational window, the optimum time of launch might be with respect to the plane or with respect to the phase. In the case of the Gemini program, as presented in a subsequent section, the optimum launch time is with respect to phase.

If both the plane and phase windows are equal to, or very near, the size of the operational launch window, additional evaluation will be necessary since the phase and plane windows do not necessarily begin near the same local time on the number of acceptable launch days. A study of the phasing times with respect to the position of the plane on the possible launch days will determine if adjustments to the plane or phase launch windows would be required.

The point to reemphasize is that though the values for the five major parameters just discussed will vary from program to program, the operational technique which uses these parameters can be the same.

The acutal purpose of the Gemini rendezvous mission will be to gain, through actual operational experience, the optimization of these parameters so that this information and experience can be made available for future applications.

ACTUAL RENDEZVOUS PARAMETERS

In order to describe the actual Gemini rendezvous mission, the important parameters for the operational mission are presented first. Then, the use of the Agena to provide increased launch-window tolerance, if that is required in the preliminary experiments, is described.

The orbits which have been selected for the Gemini target vehicle and spacecraft are presented in figure 4. The target vehicle will be placed in a 160-nautical-mile circular orbit, and the spacecraft's orbit will be an elliptical orbit having a perigee of 87 nautical miles and an apogee equal to the orbital altitude of the target. These orbits were selected primarily from launch vehicle performance and orbital lifetime limitations. The period difference in these two orbits provides the Gemini spacecraft, which is the chaser vehicle, with a 5.5° per revolution catchup rate for correcting phase differences.

For another performance reason, the spacecraft systems will have operational capability of 2 days; therefore, the rendezvous will be completed within a 2-day period after spacecraft launch. Present plans indicate that 29 spacecraft inertial orbits measured from the first apogee can be used for completing the catchup maneuver. With a catchup rate of 5.5°, an out-of-phase difference of approximately 160° will be a maximum. This phase difference corresponds to approximately 43 minutes of phase launch window. On a number of the Gemini missions, postrendezvous maneuvers with the spacecraft and target in the docked configuration are planned, so it will be desirable to limit the rendezvous time to 1 day in order to conduct these maneuvers during the second day. This time limit would reduce the launch-phase window to approximately 20 minutes.

The target vehicle will be placed into orbit prior to the time of spacecraft launch. The systems of the Agena target will be capable of operating for 5 days; thus, the spacecraft would be provided 3, 4, or 5 possible launch days. A 1- or 2-day mission completion time, the start of the spacecraft launch window at the beginning of the target's second orbit or second day, or postrendezvous requirements of the Agena will cause this variation in the number of possible launch days.

For the Gemini missions, the particular target orbital inclination of 28.87° has been selected in order to optimize the plane launch window and rendezvous fuel requirements.

Using this inclination and variable-azimuth launch techniques, the out-of-plane errors can be restricted to relatively small values for an extended period of time. One variable-azimuth launch technique is shown in figure 5. The maximum out-of-plane error that must be accommodated by the Gemini spacecraft in the terminal phase will be the perpendicular distance between the launch site and the plane of the target vehicle. This accommodation to error is accomplished by launching the spacecraft on an azimuth parallel to the plane of the target vehicle at the point of launch. Intersection of the two planes will occur approximately 90° from the launch site. In order to provide a continuous plane window as shown in figure 5, the target's maximum latitude point, which is established by the inclination in relation to the latitude of the launch site, must not exceed the spacecraft's terminal maneuvering capabilities. If the maximum latitude point of the target's plane relative to the latitude of the launch site exceeds an established spacecraft out-of-plane capability, the plane launch window will be broken in two sections. One section would be located around the ascending arc of the target plane and the second around the descending arc. An example is given in figure 6 which demonstrates this point. By using 0.6° for the upper limit of out-of-plane error, the plane launch window is reduced from one 141-minute to two 37-minute periods when the target inclination with respect to the launch site at Cape Canaveral is changed by only 1° from the near optimum 29°. A further increase of 1° reduces the launch window to two 28-minute periods. This example shows the extreme sensitivity of launch window to the inclination of the target plane.

This parallel azimuth launch technique is relatively easy to implement in the launch vehicle's guidance. No additional performance is required from the launch vehicle. The payload capability of the launch vehicle will have negligilble variations due to the variation of the earth component as the azimuth changes with launch time.

A second variable-azimuth launch technique makes use of launch-vehicle guidance in yaw during the latter phases of powered flight to minimize or steer out plane errors. Guidance is accomplished by varying the launch azimuth of the spacecraft so that the azimuth is an optimum angle directed toward the target's plane. This will reduce the out-of-plane distance prior to initiation of yaw guidance. This technique requires considerable additional performance from the launch vehicle.

For the Gemini program, appreciable performance margins may not be available; therefore, the Gemini parameters are being selected on the basis of a parallel-launch variable-azimuth technique. However, provision is being made for the variable-azimuth yaw steering technique so that it could be used should a sufficient performance margin be available. The variable-azimuth launch techniques will provide biases to offset relative nodal regression effects of the two orbits so that a minimum out-of-plane error is provided at the start of the terminal phase rather than at insertion of the spacecraft in its initial plane.

An out-of-plane correction capability during the terminal maneuver of 0.53° will be provided in the Gemini spacecraft. The Gemini plane window for which the parallel launch technique will be used is shown on figure 7. The total length of this window is 135 minutes. This window is much larger than the Gemini phase window of 43 minutes, based on a 2-day mission time; therefore, the optimum launch time or time "zero" will be with respect to phase. Figure 8 shows the launch times for each possible launch day measured from the time of target lift-off. The beginning of each window represents the first permissible phase condition within the plane window. In most cases this phase position will be the optimum or zero condition. These numbers are based on a mission time of 2 days.

It is anticipated that these launch times will, at least ultimately, be adequate for the actual launch requirements of the spacecraft and launch vehicle. Since, as stated previously, a 1-day rendezvous mission is desired, these launch windows would be reduced to panes of approximately 20 minutes in duration.

Based on the expected maximum out-of-plane errors in the terminal phase of the mission and the energy level between the two initial orbits, sufficient fuel for a velocity capability of 700 ft/sec is provided in the spacecraft for conducting the rendezvous mission.

In the terminal phase of the mission, Gemini will develop terminal maneuver techniques in which an orbital mechanics guidance method and an optical guidance method are used. An interferometer-type radar with both angular and range measuring capability, together with the on-board computer, will provide the hardware system necessary to compute an optimum orbital mechanics solution. The on-board computer will be programed with a set of equations which describe the motion of the spacecraft with respect to a rotating coordinate frame centered in the target vehicle. These equations will be "modified Clohessy-Wiltshire[4]" linearized equations of motion. (The modification of these equations was performed by Edgar C. Lineberry of Flight Operations Division, NASA Manned Spacecraft Center.)

The spacecraft will acquire the Agena target vehicle with its radar at a range of approximately 250 nautical miles. Upon acquisition, the radar will provide range, range-rate, and angular-displacement information. Range and range-rate information will be displayed to the astronauts while range and angular-displacement data are introduced into the computer. With these raw data, the computer will calculate the relative velocity components along the three axes. From these velocity components, the relative motion equations will be used to determine the velocity increment and direction required in real time to establish the spacecraft on the optimum intercept course. This information will be displayed to the astronauts who will monitor the changing velocity and position requirements. After the velocity requirement tends to level off near a minimum value, the astronaut will select the time to orient the spacecraft to the proper attitude and apply the proper velocity impulse. When the range is reduced to approximately 2 miles, the astronaut will apply a braking impulse to reduce the closing rates for final docking operations. Between the initial impulse and final impulse, small intermediate corrections will be computed and applied. The number of intermediate corrections will depend on the maneuver completion time after the initial impulse is applied. Depending on the accuracy of the midcourse catchup maneuver, the time to complete the maneuver may correspond to as little as 30° of orbit travel and to as much as 330° of travel. Figure 9 shows an example of velocity requirements at different spacecraft positions from perigee using an orbital mechanics terminal maneuvering technique. These values of velocity were derived using a specific set of initial conditions. The optimum amount of orbital travel to complete the maneuver after first impulse is also shown on the figure. The plot of velocity represents the minimum amount of velocity that would be required to conduct the maneuver if it were initiated at any point over the final two orbits for the given set of starting conditions. The velocity requirement includes the initial and braking impulses. It might be noted that the catchup rate over the final two orbits for the conditions shown in figure 9 was 1.5° per revolution. For faster rates up to 5.5° per revolution and/or larger out-of-plane errors up to 0.53°, the required velocity would be greater, but the trend of the curve would be similar. Within the out-of-plane tolerances established by the launch technique, it will be possible to initiate the terminal maneuver, if required, at ranges as far away as 250 nautical miles.

The optical guidance technique, which will be used for the Gemini missions, will be basically simple from an operational aspect. The range and range-rate information from the Gemini radar will be displayed to the astronaut for establishing the initiation time and braking schedule. The astronauts will observe the relative motion between the spacecraft and the target vehicle with respect to a star background. The Agena vehicle will be equipped with a flashing light so that it can be detected against the star field. When within proper range of the target vehicle, an astronaut, by using his on-board propulsion system, will thrust normal to the angular motion of the target vehicle. The astronaut will continue to thrust until he observes that the relative motion has been eliminated. At this instant, the spacecraft is on a constant line-of-sight approach to the target vehicle. A constant line-of-sight approach to the target will be reestablished periodically during the maneuver when the relative motion is again noticeable. By monitoring the range and range rate, the astronaut establishes the proper braking schedule similar to that required for the orbital mechanics technique. The maneuver time will be to the order of 20 minutes and the initiation range will be approximately 20 nautical miles. Since the initiation range for this technique is much smaller than that required for the mechanics technique, necessary midcourse adjustments will be oriented to provide minimum miss distances of less than 20 miles. This latter procedure makes it possible to use either technique; thus, redundancy is added to the terminal phase. Figure 10 shows the velocity requirements for conducting the optical guidance maneuver for a specific out-of-plane condition with different ranges and closing rates. These results were obtained from a simulation study and represent the most severe initial starting conditions with respect to the out-of-plane velocity component.

Investigation has shown that the velocity variation with initial range is primarily a result of the variation in the magnitude and direction of the velocity vector as the spacecraft approaches the target on the catchup trajectory. In general, a catchup trajectory can be established such that as range decreases, the direction of the velocity vector approaches the direction of the line of sight between the target and the spacecraft. Therefore, the corrective velocity required to establish a flight path along the line of sight is reduced.

The optical system which uses an optical method for providing ranging information to the astronauts will be developed and exercised during the Gemini program. This technique will serve as backup in the event of a radar failure.

DOCKING PHASE

Once the two vehicles are within a quarter of a mile of each other and the relative velocity has been reduced to approximately 8 to 10 ft/sec, the docking maneuver will begin. Windows in the spacecraft make it possible for the astronauts to have adequate visibility so that they can perform the docking operation with manual control. A docking collar on the Agena vehicle which engages with the small end of the spacecraft is being designed to absorb shock loads up to $1\frac{1}{2}$ ft/sec. Since the relative velocity at contact has been shown by simulation to be to the order of $\frac{1}{2}$ ft/sec, there is an ample margin of safety.

In order to provide the design concept and provide the astronauts with docking experience, two mechanical docking simulators are now being built. One simulator is being built at the Langley Research Center and the other at McDonnell Aircraft Corporation. Both simulators should be completed in 1963. The Langley simulator will serve as a research simulator and will provide general information in rendezvous docking. The McDonnell simulator will utilize Gemini hardware and will be used for hardware qualification and training.

THE "ACTIVE" AGENA

As previously stated, Gemini will be the initial "work-horse" in developing general rendezvous techniques in each phase of the mission. In order to determine the actual accuracies and dispersions in developing rendezvous technology in all phases of a mission, it is necessary that the missions be completed. For the initial Gemini rendezvous missions, additional capability is considered necessary until adequate real-time knowledge of the accuracies and techniques is established.

In order to provide this additional capability, the Agena vehicle has been selected as the target vehicle for the Gemini program. The Agena is a space-stabilized vehicle and can be maneuvered in orbit. The Agena not only can aid the spacecraft in conducting the rendezvous mission, but it also provides the power source for performing postdocking maneuvers, which is another objective of Gemini.

The Agena provides the Gemini program with a flexible rendezvous technique. Figure 11 shows this flexible technique. The figure is a flow diagram similar to the operational diagram showing the possible real-time steps between the launch phase and the docking phase.

The Agena provides the additional capability in the launch and midcourse phases for directing the two vehicles to the terminal phase of the mission. This capability can be seen by the additional blocks in figure 11. Additional plane- and phase-correction capability provides additional launch-time capability. The Agena vehicle after insertion into its 160-nautical-mile circular orbit will have a velocity capability of approximately 5,400 ft/sec which can be used for midcourse corrections.

For a 1-day mission completion time, the Agena can provide a catchup rate that will allow a full 360° phase difference if required. This capability to correct all phase errors greater than 70° for a 1-day mission removes all restrictions with respect to a phase window. To perform this maneuver, the Agena would receive from a ground command site the required information for storage in its programer. At the proper point in its orbit, the main Agena engine will be ignited and an elliptical orbit whose apogee might extend out as far as 550 nautical miles will be achieved. The new Agena ellipse will increase the catchup rate so that catchup will occur on a selected orbit. The size of the ellipse will vary with the magnitude of the phase difference. The Agena will recircularize its orbit by applying a retroimpulse at its 160-nautical-mile perigee when the phase difference has been reduced to between the values of 5° and 11°. This impulse provides two spacecraft catchup orbits for completing the midcourse maneuver. Final midcourse adjustments and closing conditions will be identical to those required if the Agena had not maneuvered. The maximum Agena ellipse and the recircularization maneuver will require approximately 1,300 ft/sec from the Agena vehicle with two main engine ignitions.

Depending on the phase position relative to the plane position at spacecraft launch, the amount of Agena velocity available for correcting excessive plane errors will vary between approximately 5,400 ft/sec to 4,100 ft/sec. Since the relationship between phase and plane can be calculated, accurate plane windows have been determined. A velocity of 5,400 ft/sec corresponds to a 12° out-of-plane correction, and 4,100 ft/sec corresponds to a 9.4° correction.

When the spacecraft is launched and the angle formed between the Agena and spacecraft planes exceeds 0.53°, the Agena will make a plane maneuver. From tracking data, the ground computer complex will determine the two positions where the planes intersect, the required velocity and direction, and the time of engine ignition. The first convenient command site will transmit this information to the Agena where it will be stored in the Agena programer until time to begin the maneuver. There will be portions of the launch window in which the Agena will be required to conduct both a plane and a phase maneuver. These maneuvers could be initiated simultaneously; however, from accuracy considerations, the present plan is to conduct them separately.

The Agena plane-correction capability will increase the plane window appreciably, but not as much as might be expected for two reasons. The first is that the optimum or "zero" time of launch will still be at the first optimum phasing condition after the plane position is within 0.53° of the launch site. This allows the spacecraft to conduct the mission without Agena maneuvers. The window, therefore, will only be opened in one direction. The second reason is that the parallel-launch variable-azimuth technique cannot be employed for out-of-plane conditions as large as 9.4° because the spacecraft launch azimuth is becoming more southerly with time, and a range safety limit, as well as a landing-area restraint, will be imposed. A launch azimuth of 105° has been established as the southerly bound for the variable-azimuth launch technique. When the azimuth reaches 105°, the out-of-plane error using the parallel-launch procedure will be approximately 3.2°. A constant launch azimuth at 105° will be used until the plane angle, which will be formed between the two planes, reaches the limit of the Agena capability. This is noted on the flow diagram shown in figure 11. The launch window, which includes the Agena capability, is shown in figure 12. The launch window denotes the areas in which spacecraft maneuvering only is required, areas in which only Agena phase maneuvering is required, areas in which only Agena plane maneuvering is required, and areas in which both Agena plane and phase maneuvers are necessary.

As the program progresses and experience is gained, a better knowledge of the accuracies and dispersions associated with the techniques in the different phases of the mission will be acquired. It is then anticipated that the increased capability provided by the Agena will not be required and the rendezvous technique will become the operational technique shown in figure 3.

POSTRENDEZVOUS CAPABILITY

At the completion of an operational rendezvous exercise, the Agena when docked with the Gemini spacecraft will have a velocity capability of approximately 2,340 ft/sec for conducting postrendezvous maneuvers. After docking, the astronauts will be able to control the Agena's attitude, monitor Agena systems, and ignite the Agena engine.

RENDEZVOUS PROGRAM SUMMARY

The Gemini rendezvous approach is divided into two phases. Phase 1 of the rendezvous program, in which a radar transponder package will act as a target, will be carried out in conjunction with the other long-duration requirements. This phase will be a preliminary terminal guidance study. Phase 2 will begin when the Agena is introduced into the flight schedule as the target vehicle. The four facets of Phase 2 are: the Agena as active, the Agena as passive, operational rendezvous, and Gemini-Agena with systems operational. The active Agena portion of Phase 2 refers to the utilization of the Agena propulsion system in increasing the confidence level of completing rendezvous and in developing rendezvous techniques. The passive Agena portion of Phase 2 refers to the nonpropulsive requirement on the Agena effecting rendezvous as the operational technique is developed. Operational rendezvous refers to the development of the technique so that a high confidence level is anticipated for future flights. The Gemini-Agena with systems operational refers to postrendezvous maneuvers in the docked configuration.

APPENDIX

DEFINITIONS

Launch window is the time span over which a launch may be performed and still meet the timing conditions for rendezvous.

Operational launch window is the launch window as defined and limited by the practical hardware constraints of the launch system.

Phase (launch) window is the launch window as determined by the ability of the rendezvousing vehicles to catch up with each other when they are separated by a phase difference or geocentric angular displacement.

Plane (launch) window is the launch window as determined by the ability of the rendezvousing vehicles' maneuverability to compensate for angular differences between the orbital planes of the two vehicles.

Midcourse phase is the portion of a rendezvous flight from injection into orbit up to the point where contact is made between the two vehicles so that command of the subsequent closing operations is on board the rendezvousing vehicles.

Terminal phase is that portion of a rendezvous flight during which the astronauts in the spacecraft determine and execute the proper maneuver to establish the precise intercept course between the target and spacecraft. This maneuver will bring the two vehicles close enough together in both position and velocity to permit final docking operations.

Docking is the final phase of a rendezvous flight in which the two vehicles are brought into bodily contact and latched solidly together.

Inclination of orbital plane is the angle between the equatorial plane and the plane of the satellite vehicle which passes through the center of the earth.

Launch azimuth is the initial heading of the satellite vehicle from true North. The launch azimuth is defined by the position of the desired orbital plane with respect to the position of the launch site.

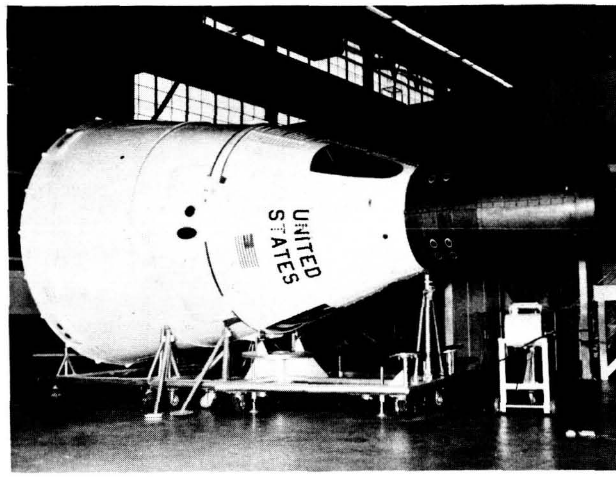

Figure 1.- Mock-up of Gemini spacecraft.

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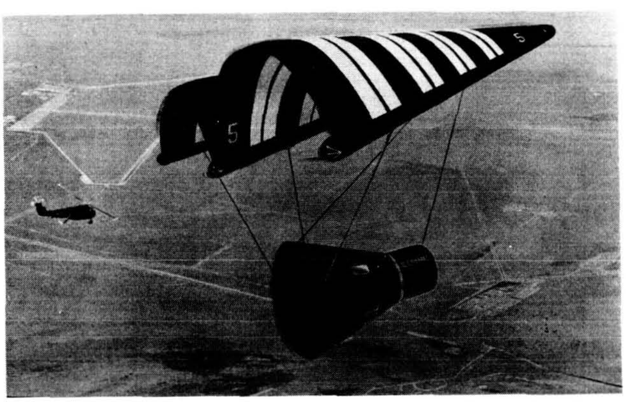

Figure 2.- Paraglider deployment.

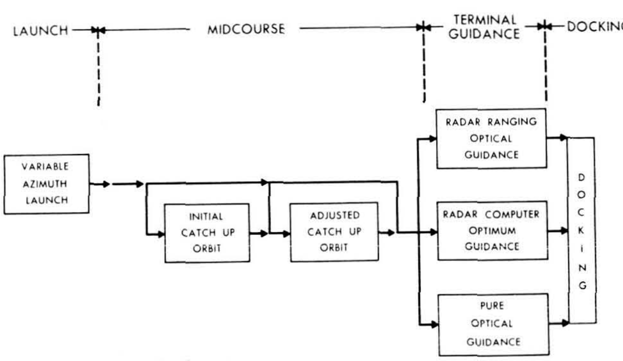

Figure 3.- Operational rendezvous technique.

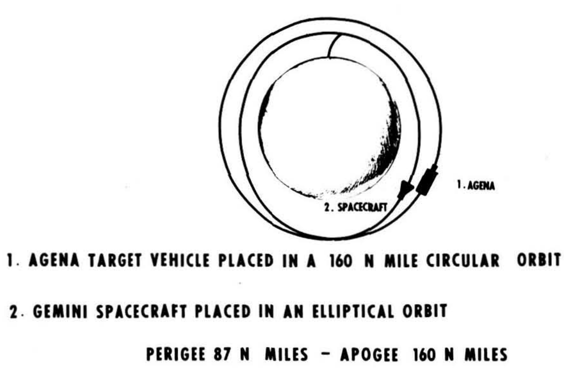

Figure 4.- Orbits selected for Gemini rendezvous program

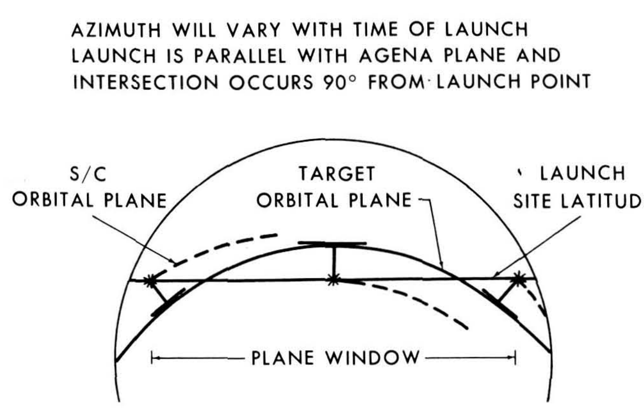

Figure 5.- Variable azimuth launch.

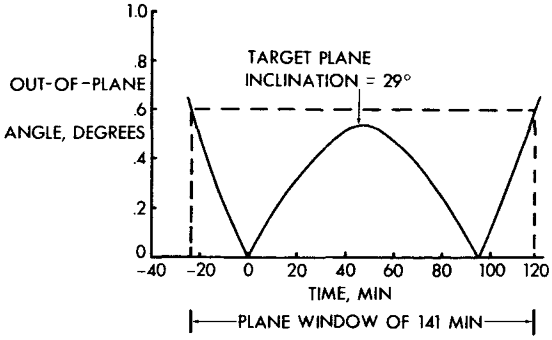

Figure 6.- Plane launch window variation with target plane inclination.

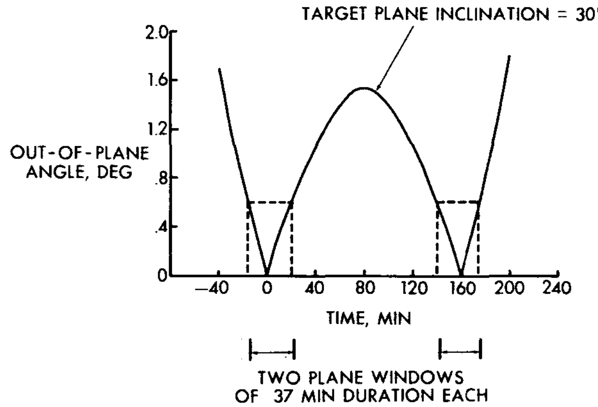

Figure 6.- Plane launch window variation with target plane inclination - Continued.

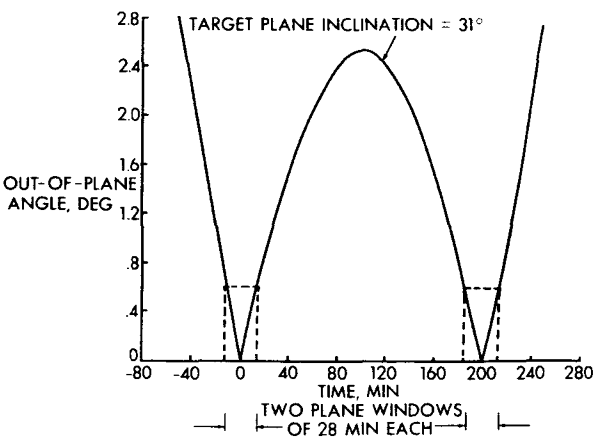

Figure 6.- Plane launch window variation with target plane inclination - Concluded.

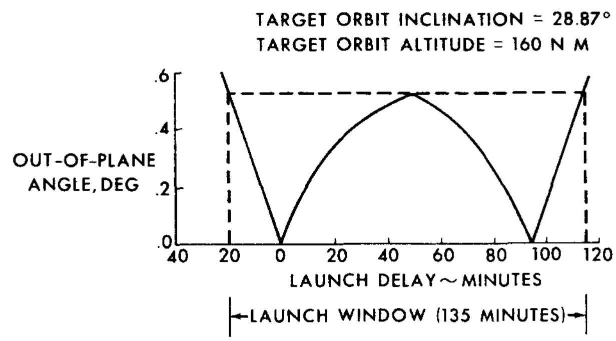

Figure 7.- Gemini plane window.

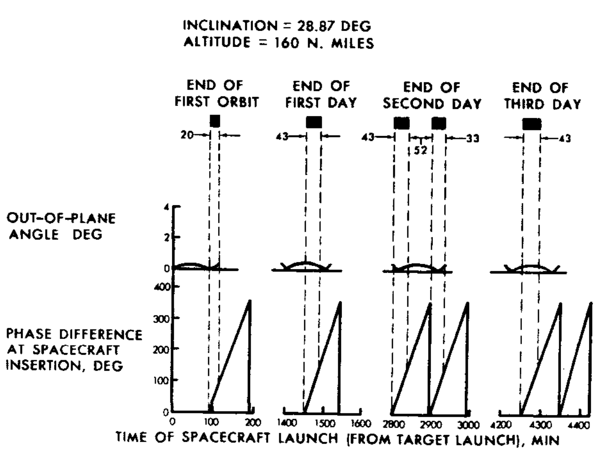

Figure 8.- Gemini launch window.

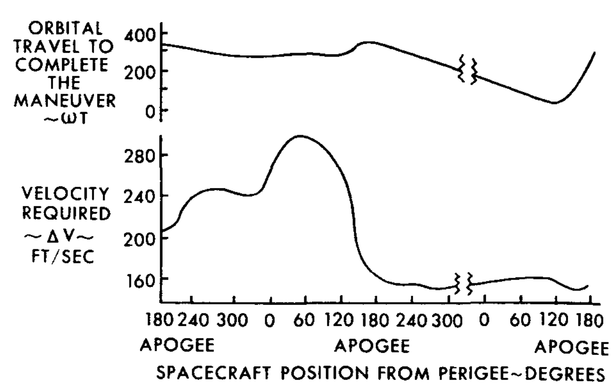

Figure 9.- Orbital mechanics terminal maneuver techniques.

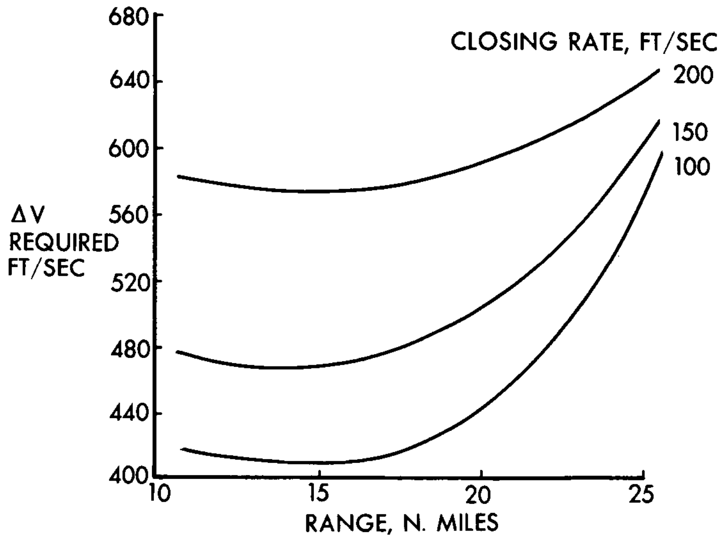

Figure 10.- Optical guidance terminal maneuver technique.

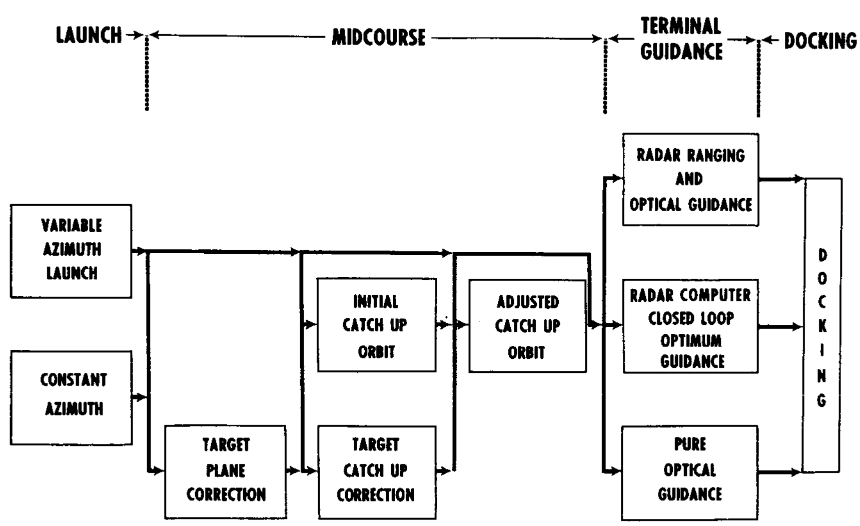

Figure 11.- Flexible rendezvous technique.

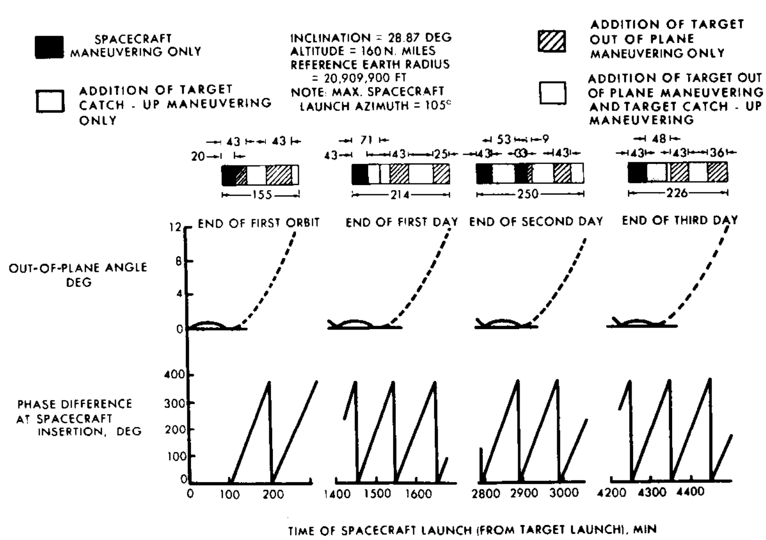

Figure 12.- Gemini increased launch window using "active" Agena.

THE TITAN III STANDARDIZED SPACE LAUNCH SYSTEM

0225458

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Titan III Program Director
Space Systems Division
Air Force Systems Command
Los Angeles, Calif. Space Systems Div.

Introduction

Titan III is the popular name for Program 624A Standard Space Launch System now under development by the Space Systems Division of the Air Force Systems Command for the Department of Defense.

At the conclusion of the 45-month development program, which includes 17 launches from the Atlantic Missile Range, it will take its place as a versatile and powerful addition to the National Launch Vehicle Program.

Titan III, when it is operational, will be the product of a new Department of Defense management concept and the first so-called "billion dollar" program where the contractor effort is conducted almost exclusively under cost-plus-incentive-fee type contracts.

Standardization -- one of the keynotes to the Titan III philosophy -- will give it greater mission flexibility than any other launch vehicle or system now under development. Payload capability will range from 5,000 to 25,000 pounds and mission capability will include:

--low altitude elliptical orbit by direct injection,

--low altitude circular orbit,

--low altitude circular orbit with Hohmann transfer to another orbit,

--synchronous orbit, and

--deep space trajectory to escape.

The ITL facility -- integrate-transfer-launch -- will give Titan III quick responsiveness, a quality required of any military system. Minimum pad time is insured through off-pad assembly, payload integration and checkout leading to high launch rates from fewer pads with resulting economies.

Repeatability -- or if you will, reliability -- is inherent in the conservative design approach which makes maximum use of state-of-the-art and does not require any technical breakthroughs.

The Titan III will be man-rated and as such will have a capability to support the manned as well as the unmanned systems now indicated for the near future.

In reference to the Titan III Standard Space Launch System, you will note the emphasis always is placed on SYSTEM. Although this system features a versatile and powerful new booster in two separate configurations, the boosters alone would add little to this nation's capability in space, particularly in support of military requirements.

Associated aerospace ground equipment developed to complement the booster and the unique ITL facility at Cape Canaveral are integral parts of the Titan III.

Together these three vital elements comprise a new space launch capability -- one conceived and developed "from the ground up" specifically to meet those space launch requirements of the next decade which cannot be met by any other space launch system now under development.

Design Approach

The 624A Space Launching System is planned as a national standardized launch vehicle adaptable to a wide range of applications. It is being designed for use in both military and non-military missions in the period from 1965 through 1975. Development objectives have been versatility, reliability, economy, high launch rates, and early availability. The philosophical and technical approaches to these objectives have given strong consideration to the building block concept, standardization, conservatism and simplicity of design, the capability of industry to produce, and cost effectiveness.

A conservative approach to design has been adopted uniformly throughout the program by maintaining a second generation technology and resisting adoption of second-and-a-half or third generation techniques. This philosophy has led to maximum utilization of the best current state of technology while avoiding dependence on not-yet achieved technical advancements or new research. Emphasis has been placed on functional integrity of equipment rather than peak performance. Conservative ratings and design margins have been used in those cases where new equipment is being specified.

The entire 624A program has been planned to derive the maximum benefit from the accumulation of knowledge and experience resulting from the ballistic missile and space programs. Every effort has been made to make this a thoroughly up-to-date system using the best available demonstrated and proven techniques. The System specifically has not been made dependent upon new major advancements in any area. The employment of such an approach will result in providing the United States with a space launching system which reflects a carefully planned balance between operating versatility, performance, cost, reliability, and utilization of national resources.

Management Approach

To set the stage for a discussion of the Titan III management approach one must refer to the remarks of Secretary of Defense Robert S. McNamara before the House Armed Services Committee.

The problems outlined and the solutions advanced have a direct relation to Titan III because the 624A program was selected as the guinea pig to prove Mr. McNamara's point.

He told the Congress:

"Research and development expenditures, whether measured in budget terms or in program terms, have been mounting steadily over the years, but too much of this effort is not producing useful results. What we want are weapons and equipment that the fighting man can use. We are not interested in supporting the intellectually challenging, but militarily useless, engineering 'tour de force'. If we are to make optimum use of our available scientific and engineering manpower resources, we must plan our program carefully and concentrate these resources where they will make the greatest contribution to our military posture.

"Poor planning, unrealistic schedules, unnecessary design changes and enormous cost increases over original estimates have continuously disrupted the efficient operation of our research and development program. Most of these difficulties have resulted from inadequate prior planning and unwarranted haste in undertaking large-scale development, and even production, before we have clearly defined what is wanted and before we have clearly determined that a suitable technological basis has been developed on which to build the system. We have often paid too little attention to how a proposed weapon system would be used and what it would cost, and finally, whether the contribution the development could make to our forces would be worth the cost.

"Accordingly, we are now following the practice of inaugurating large system development projects only after the completion of what we call a 'program definition' phase. To the greatest extent possible, we want to do our thinking and planning before we start 'bending metal'. Pencils and paper, and even the feasibility testing of 'pacing' components, are a lot cheaper than the termination of programs. By a more thorough and complete study and assessment of the facets of each new development -- prior to major commitments -- we can reduce the number of expensive projects which might otherwise later have to be reoriented, stretched out or terminated."

Mr. McNamara's philosophy has been followed to the letter in the Titan III Program. A year of exhaustive effort went into Phase I. When Phase I was completed, system definition and the program management package was forwarded to DOD. A strong management organization had been structured with close control through PERT-Time cost techniques established to insure attainment of all objectives in accordance with program plans.

Within the Space Systems Division, the Titan III System Program Office (SPO) consisting of the Air Force and Aerospace management team was well established, providing an integrated partnership bearing full management responsibility for the total standard launch vehicle system.

The Booster Vehicle

Since one of the time-consuming factors plaguing our space efforts to date is the need to virtually hand-make each booster to fit some particular payload and mission profile, one of the over-riding considerations in the design of the Titan III was maximum standardization to improve utilization and decrease long-term costs by providing a basic vehicle to which many payload packages can be affixed with no modifications to the booster other than the payload-to-booster fairing. The Titan III system also includes a standard payload shroud that will accommodate any payload up to 10 feet in diameter.

Further simplification and versatility comes from the "building block" approach used in the design of Titan III boosters. The basic core of the Titan III vehicle is a structurally-modified Titan II ICBM -- a two-stage vehicle powered by storable propellant liquid fuel engines. Thrust of the engines will remain the same as in the ICBM version for the forseeable future -- 430,000 pounds for the first stage and 100,000 pounds for the second.

To this core a new upper stage or transtage is added. The transtage engine will burn the same fuel as the core. An integral part of the transstage structure will be a control module housing all flight control and guidance components for all stages of the vehicle. Transtage thrust will be approximately 16,000 pounds and the engine will have a multiple restart capability.

The configuration resulting from combining the modified Titan II with the transtage and control module will be designated Titan III A. When two, five-segment, 120-inch diameter, solid rocket engines are added to the "A" configuration the vehicle will be designated the Titan III C.

Lift-off thrust of the two solid propellant first stage booster engines of the Titan III C will be in excess of 2,000,000 pounds. Secondary liquid injection thrust vector control will be used in the solid booster stage. Thrust vector control for all other stages will be performed by conventional hydraulic engine gimbaling.

The Integrate-Transfer-Launch Facility

To fully understand the significance of the ITL facility now under construction at Cape Canaveral, it is important to refer to a statement by the Honorable Brockway McMillan, Assistant Secretary of the Air Force for Research and Development.

Mr. McMillan began his statement with this question:

"Can a launch vehicle which must stand on the pad for six weeks of pre-launch checkout support a military capability for anything?"

He went on to answer his own question as follows:

"I think not. The essence of any military capability, be it in the Armed Forces Police or in the strategic retaliatory force, is readiness, responsiveness to command, and adaptability to the changing needs of policy or the fickle fortunes of war.

"To me this means that military vehicles must be simple, reliable, dependable, and flexible.

"If space vehicles are ever to support military missions in the same sense that air vehicles, ground vehicles, and marine vehicles now do, they must be capable of flexibility during the mission. They must be capable of responding to circumstances which were not known at the time the mission originated, by maneuver, by change of attitude, by shift of orbit, and by re-entry at a time and place determined by a military commander, not by an astrologer."

Today, when we see or hear of an astronaut being launched into orbit, or of a space probe sped on its way to wrest from the universe some of its secrets, we can be certain that many weeks were spent painstakingly checking each and every detail of the vehicle used to boost the payload - be it man or machine - into its path. These myriad checks are necessary to insure that every part of the complex machines used to probe space do and will function properly; anything less than this means the loss and waste of valuable scientific equipment, or more regrettable, a human life. To ensure the success of the mission and the safety of passengers each component of the booster and its payload is not only thoroughly checked prior to erecting the vehicle on its launching pad, but checked again after the components have been mated to form the complete vehicle. Understandably, the time needed to assemble the booster and its payload on the launching pad and then check it out completely runs into weeks, during which time that particular launching pad cannot be used for any other purpose. In the event of a change of mission or a major malfunction requiring the replacement of a payload or booster stage, the pre-launch checkout must begin anew, and more weeks are consumed. Thus - the time factor - is the one irreplacable quantity.

Obviously, if we continue to tie up launching facilities for weeks at a time for each payload, it will not be long before our present facilities are saturated and more must be constructed. Simultaneously, the trend toward larger, more exotically fueled boosters has increased safe-separation distances in turn demanding more real estate per launch site and complicating the support problem. Since there are few areas in this country which fulfill the requirements for launching sites, it is equally obvious that there is a limit to the number of launch complexes that can be built before we run out of available real estate. If we are not to be limited by this factor in the near future, some way must be found to better use present and future facilities; the most obvious method being to increase launch rates and to provide for greater flexibility in the event of mission changes or pre-launch aborts. One of the solutions arising from Air Force research is the Titan III, the first Department of Defense Standardized Space Launch System.

Although this state-of-the-art space booster makes maximum use of the proven components and techniques, it must still be put through a grueling series of pre-launch checks before it is ready to launch its payload. These checks are the major stumbling blocks to higher launch rates from present facilities; yet, eliminating them runs the risk of losing a valuable payload due to some malfunction during or after launch. Until out technology produces a space vehicle so foolproof that extensive pre-launch checking is unnecessary, we must live with out present procedures, and consider other means of improving our launching facilities.

A system to increase launch rates came from an Air Force sponsored study which sought new means of speeding up the pre-launch checkout of space vehicles in general. This solution envisioned off-pad assembly of the entire booster/payload, complete checkout in a consolidated preparation facility remote from the launching site proper, transportation to the launching site in a "ready" configuration, and occupancy of the launching pad only for that time necessary to fuel and launch the vehicle. This approach allows simultaneous preparation of several complete systems, the reduction of "on-pad" time to a minimum, the ability to react rapidly to mission changes with <u>in toto</u> replacement of one vehicle by another which is ready to be fueled and fired. It also makes more effective use of support facilities through single location preparation activities, and the resulting higher launch rate. This is, in essence, the Integrate-Transfer-Launch concept.

The Engineering Challenge

Although the philosophy under which this system is being developed entails the application of the present state-of-the-art, there are some areas which must be recognized as potential risks. These include:

In the solid motors, the area with the greatest question marks is the liquid injection thrust vector control. The function of liquid injection TVC has been used on the Minuteman and in its most advanced configuration on the Polaris second stage. This has been successfully flight demonstrated. The significant deviation in applying this on the Titan III system lies in full directional control in all planes through a single nozzle. Also, two motors, operating independently, but as a single stage, offer the possibility of higher vector angle requirements than previously demonstrated. To minimize this risk we have already demonstrated the feasibility of the system in our 100" motor program; we have programmed full scale demonstrations early in the development phase; and we have planned a very stringent cold flow program.

The second area of concern regarding the use of the solid motor stage is at separation. This is being studied analytically in great detail and an early wind tunnel test program has been initiated. Here again the differential in performance between the two motors comes under close scrutiny and all means to assure positive control will be implemented.

Another advancement in technology which is being applied in this system is the extended burning time of the solid motors. This is new, almost double that used in the Minuteman first stage, however, we have shown the feasibility of this in our subscale static test work and will demonstrate materials life and component configuration during the early test firings.

One area of concern regarding the development of the pressure fed engines for the transtage is the effect of the high combustion temperature on the ablative material used in the fabrication of the thrust chambers. Apollo subscale tests conducted during December 62 and January 1963 in this area of concern were very encouraging. Ablative Thrust Chambers withstood the high temperature affects for a considerably longer burning time (50%) than that required for the transtage engines. Early last month, a successful firing for a duration in excess of the transtage burning time was conducted at the Contractor's facility. This test is viewed as a significant achievement in the development program since loss in chamber pressure was very slight and the event occurred approximately forty-five days ahead of schedule. Admittedly, the test was not conducted with a fully developed injector but the injector efficiency was sufficiently high to merit the "significant achievement" classification.

Another area of concern is the use of a columbium-titanium nozzle extension and the area ratio at which this extension can be fastened to the ablative thrust chamber. A materials testing program is being conducted by the engine manufacturer involved tensile tests on both flat and cylindrical specimens. To date, we have seen no indication of a serious problem in this area. Nevertheless, the explosive forming dies will be capable of forming an all columbium nozzle extension.

The skirt is fastened to the thrust chamber at an area ratio of 6:1. Previous experience on a full titanium skirt was with the Ablestar engine which had the skirt fastened at an area ratio of 20:1 and had experienced no difficulty. However, the subscale Apollo test program mentioned previously proved conclusively that titanium could not be used at the 6:1 area ratio. We believe that columbium can be used at the 6:1 ratio but may require a coating to eliminate oxidation. In any event, the thrust chamber wrapping mandrel is being fabricated to a much greater area ratio than 6:1 so that should difficulties arise in this area the ablative material can be extended out to whatever area ratio is required in order to effectively fasten the nozzle extension.

The major, but by no means the only cause for concern in the development of the Titan III guidance system is in modifying the Titan II guidance system to operate in the predicted Titan III environment. The most likely candidate for trouble is operational shock. It has been determined by analysis and simulation that a limiting feature of the inertial measurement unit when subjected to the shock impulse generated from explosive bolt ignition during staging may be in the accelerometer servo loop. In particular the 25 PIG gyro float being limited in accelerometer freedom might go to its mechanical stop. During the time that this situation persists, the unit cannot function as an accelerometer and optisyn counts (the measure of incremental velocity) are lost. Limited testing on a Titan II system has been performed by AC Spark Plug; and test results confirmed the analysis. The requirement for a hardware change and the extent of such a change are presently undergoing careful investigation. It must be emphasized that considerable technical risk is always present in making a major change to a device as complex as the 25 PIG.

Another potential problem area associated with environment is the fact that the system must be able to live in a hard vacuum for $6\frac{1}{2}$ hours instead of a few minutes. The development of new equipment such as the Environmental Control Unit, the modification to existing equipment and the effect of this environment on unmodified equipment may add up to possible difficulties.

Why Titan III?

Peaceful and military uses for space are not separate and distinct. They are, in fact, a single requirement and are indivisible. This nation has no aggressive designs on anyone or anything, therefore any military use we may make of space is for the purpose of keeping the peace.

This was well stated by Dr. Edward Welch, Executive Secretary of the National Aeronautics and Space Council, when he said:

"We have space missions to help keep the peace and space missions to increase our ability to live well in peace ... national security missions are a major portion of our national space program, at least as important as any other phase in space. If we do not take adequate care of our national defense, we will not have a chance to do any of the other things in space -- at least as free men."

Our policy of peaceful use of outer space has never been intended or interpreted to deny the military the use of space for peaceful or for stabilizing purposes.

This point has been very clearly stated by Dr. Harold Brown, Director, Defense Research & Engineering. In his recent testimony before the Senate Committee on Aeronautics and Space Science, Dr. Brown said:

"We must, therefore, engage in a broad program covering basic building blocks which will develop technological capabilities to meet many possible contingencies. In this way, we will provide necessary insurance against military surprise in space by advancing our knowledge on a systematic basis so as to permit the shortest possible time lag in undertaking full scale development programs as specific needs are identified."

What are the peaceful and stabilizing activities in space that are of military importance? Vice President Johnson enumerated some of them in a public address. He mentioned, early warning of ballistic missile attack, various kinds of surveillance and reconnaissance, communications of a secure and invulnerable kind, and navigation. Roswell L. Gilpatric, Deputy Secretary of Defense, noted recently the list also includes defensive missions and missions to inspect and verify that unidentified space vehicles are in fact peaceful. If they are proven hostile, they will be neutralized before they can do harm to mankind.

Titan III is being developed to support just such systems referred to by the Vice President and Mr. Gilpatric.

Prior to Department of Defense approval of the Titan III system, the whole picture of U. S. space launch capability was examined at length by a joint planning group made up of representatives of both the National Aeronautics and Space Administration and the Department of Defense.

Following the policy decision of President John F. Kennedy on May 25, 1961 establishing this nation's expanded space program, the Large Launch Vehicle Planning Group was established by agreement between Mr. McNamara and the NASA Administrator Mr. Webb. The LLVPG brought together the best knowledge in DOD and NASA as well as from outside agencies and extended an existing agreement on booster development to include the actual planning for space boosters of the future.

The LLVPG scrutinized each and every booster available as well as those approved for development at that time. They included the NASA-Air Force Scout; the Air Force Thor, Thor Agena and Thor Ablestar; the NASA Thor Delta; the Air Force Atlas and Atlas Agena combination; the Air Force-NASA Titan II Gemini booster and the NASA Saturn I and II large launch vehicle.

Subsequent recommendations of the LLVPG led to NASA's work on the advanced SATURN, the large 1,000,000 pound thrust hydrogen engine, and also to DOD's work on the Titan III proposal. It had been previously proposed to make a space launch vehicle from the Titan II ICBM by modifying Titan II and putting on an upper stage, but detailed studies by the LLVPG recommended against this approach in favor of the Titan III approach -- a standard launch vehicle system based on a structurally modified Titan II missile with strap-on solids. The importance of such decisions becomes evident when one realizes that about one half of funds expended for space projects are for launch vehicles.

It is established that not only the Air Force but DOD, NASA and independent technical authorities recognized the clear cut need for a space launch system which would become a part of the National Launch Vehicle Program, and at the same time and more importantly, would meet military space launch requirements.

What are some military requirements for Titan III?

First, the matter of rendezvous and inspection.

Rendezvous in space is not an easy task and there are three aspects of rendezvous -- and the attendant functions of inspection and transfer -- which complicate the problem for the military planner.

Simple rendezvous in space and the transfer of fuels or liquid food cannot satisfy the military requirements for supporting a military mission. Logistically, there is a need to transfer numbers of men and quantities of materials, in bulk. As another point of difference, it is one thing to rendezvous with a friendly satellite in a controlled environment and pre-determined orbit; it is quite another thing to rendezvous with an uncooperative vehicle which might be equipped to take evasive action or be booby-trapped. Finally in terms of inspection, there is little advantage to coming close enough to a foreign vehicle in space to be able to inspect it, unless we are also prepared to capture it, or otherwise negate it.

There may be no importance to such inspections at all, considering how easily the enemy could launch harmless decoys or dangerous booby traps while concealing the contents of a satellite. To turn the problem around, we would want to be able to counter such an effort on the part of an enemy satellite.

Second, consider the practicality of the manned orbital development station, popularly called MODS.

There are a variety of purely scientific reasons for putting a manned laboratory into space. And there are equally valid and pressing reasons for justifying a military-oriented "space station." For one thing, military experience can contribute to the success and the effectiveness of MODS programs. Military officers have pioneered in the Mercury program, and military pilots have shared X-15 experience with civilians. Military man eventually must become functional in space. To do so, he must acquire the kind of experience best afforded by an orbiting development station. In such a station, man can undergo the effects of the genuine space environment, conditions which cannot be entirely simulated on the ground. He can test and check-out equipment, determine the limits of human endurance under severe circumstances, and evaluate the life support that will become essential to future space operations.

Third, there is the communications satellite.

Last May the Secretary of Defense directed the development of two military communication satellite systems with the goal of reaching an operational capability as early as possible. At the same time, following the publicized success of Telstar, a program got underway to produce a commercial space communications system. Why won't a single program satisfy both requirements? The answers are basic and irrefutable.

Commercial communications systems require many channels, a considerable number of which are normally leased by the military for its routine administrative communications; the strictly military require relatively few channels. But these few channels must be global and they must be reliable even in the gravest of circumstances. Further, the channels must be jam-free, and any command control function must be secure from enemy interference or operation.

Not only does a military system require a few jam-proof channels, but the satellites must function with simple ground stations. Eighty-five foot antennas and receivers soaked in liquid helium are acceptable for commercial installations, but they are hardly suitable for installation in a trailer or for landing in the Congo by airlift.

Another requirement of military communications satellites is the desirability of providing multiple ground stations or of hardening key stations, both in the interests of survivability. Neither factor is required for commercial purposes.

A military system must function 24 hours a day, with built-in redundancy to guard against chance or induced failure. In this respect, we must remember that Telstar can operate only a small percentage of the time, and that many silent orbits have to follow each transmission in order to allow for the recharging of batteries.

Communications may well be the life-line of survival in any future conflict. As a "last ditch" system, land-line or micro-wave techniques leave much to be desired, and any conflict between military and commercial utilization would be intolerable in a critical situation.

Establishing, maintaining the operating military space systems to perform the missions outlined here requires a space launch system with the exact dimensions designed into Titan III --

--almost immediate reaction time,

--capability to economically support frequent and multiple launch requirements

--great mission and payload flexibility,

--high degree of repeatability, and

--maintainability of a high order.

Taking into consideration the management and technical approaches being taken in the development of the Titan III Standard Space Launch System, the radically new contractural approach which should stimulate maximum performance from the industrial family, the appraisal of the technical risks and the Air Force determination to maintain a conservative state-of-the-art program from approval through initial operational capability permits great confidence that Titan III will join the national launch vehicle program on schedule in 1965 and will perform the missions it is designed to perform.

FUTURE OF LARGE LAUNCH VEHICLES

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Introduction

Launch vehicles have been one of the most important, if not the pacing item in determining the progress of our National Space Flight Program. Unquestionably, the capability and the availability of our launch vehicles will, for some years to come, continue to be key factors and will have a major effect on our progress.

Our large launch vehicles currently under development or being studied for future development represent an ambitious growth in capabilities to support manned exploration of space. SATURN I and SATURN I-B, planned to be operational in 1965 or 1966, will be significant milestones in the art of large boosters. The next increase in booster capability will be the SATURN V, which is planned for operational status in 1967. SATURN V is the largest launch vehicle the free world has under development at the present time. It is sized to meet the launch requirements to accomplish the APOLLO mission of a manned lunar landing.

Following SATURN V, the next large launch vehicle planned for development is the NOVA. The original NOVA concept was developed in 1959-1961 with the goal of providing a direct flight capability for a manned lunar landing. NOVA was to be essentially a back-up for SATURN V and, therefore, of the same or present state-of-the-art technology. The selection of the Lunar Orbit Rendezvous (LOR) mode of operation in conjunction with SATURN V now provides for the APOLLO mission capability, thus eliminating the necessity for an early (1967-1969) NOVA vehicle for this particular mission. Therefore, the primary requirements for NOVA were shifted from early manned lunar landing to more ambitious and later missions. The original concepts became both obsolete and insufficient. Moreover, it became more and more apparent that resources during the next few years would not permit the development of another large launch vehicle at a rapid pace with an early availability goal. This situation necessitated a re-direction of the NOVA program during 1962; and a study effort was initiated to determine what the NOVA should be, how it could be employed, and when it should be developed.

Potential Missions

Potential missions which could utilize a NOVA class vehicle include:

a. Manned planetary exploration
b. Large lunar base
c. Large orbital or global cargo missions.

One or more of these requirements must materialize before development of a NOVA launch vehicle can be justified. Of all of these potential missions, the one of manned planetary exploration appears to be the most ambitious and probably will require the largest weight lifting capability. Therefore, the accomplishment of this mission is one of the major considerations in determining NOVA.

NOVA Study Program

Since NOVA will be the next large launch vehicle after the SATURN V, it must provide a "Big Step" in vehicle technology and payload capability. Therefore, NOVA must carry the largest payloads, with the greatest reliability and at the lowest possible cost consistent with the projected state-of-the-art technology. To determine the "Ideal NOVA" NASA is conducting extensive studies both in-house and by contracts with industry. NASA is also benefiting from efforts by other government agencies, as well as company-sponsored efforts. The Marshall Space Flight Center is the technical director and coordinator of the NOVA launch vehicle program. The major contracts with industry are for studies by General Dynamics/Astronautics and Martin-Marietta/Baltimore to determine vehicle configurations, define research technology developments required to support various vehicle concepts, establish basic model specifications, determine facility requirements, and define development and operating plans. Contracts are also in effect with Douglas Aircraft, General Dynamics/Astronautics, and Space Technology Laboratories to study Advanced NOVA concepts. Boeing Aircraft has been studying a solid boosted NOVA. Martin-Marietta/Denver is studying the facilities required to launch the various vehicles conceived by the various contractors. These are the major or prime NOVA contractors. Additional inputs too numerous to cover here are being made by various contractors, both from government contract and company-sponsored efforts.

Growth in Payloads

Figure 1 shows the growth in payloads which these future large launch vehicles will provide. Plotted on this graph are single-launch weight lifting capabilities to earth orbit. SATURN I will be able to lift 10-ton payloads to orbit by 1965. A year later SATURN I-B will increase this capability to almost 15 tons. You will note the increase provided by SATURN V. This vehicle will single launch 120-ton payloads to earth orbit. The next step is to a NOVA class vehicle. A single NOVA will be able to place a payload of 500 tons (or one million pounds) to earth orbit, and this could be accomplished before the mid-1970's. This would be four times the SATURN V capability, or an increase by more than two orders of magnitude over that of today's launch vehicles.

Direct Operating Cost Trends

As payload capabilities increase, the vehicles increase in size, require more thrust, and become more expensive. At the same time, however, they accomplish missions which their predecessors couldn't do and, of equal importance, provide us with a more economical transportation system. Figure 2 illustrates this this trend by considering costs on a basis of dollars per pound to orbit.

It is expected that the direct operating costs of the first SATURN I-B will approach a \$1,000 per pound into orbit. The cost curve drops quite rapidly but tends to level out around \$400 per pound. The SATURN V will orbit payloads initially for about \$250 per pound, with subsequent flights approaching a minimum level of \$150. NOVA will have an even flatter curve. An early state-of-the-art vehicle could orbit payloads for \$150 per pound, with eventual reduction to less than \$100. A more advanced state-of-the-art NOVA could conceivably place one-million-pound payloads into orbit for even less.

Vehicle Configurations

The vehicle studies fell into three categories or classes. Class I represent present or SATURN V state-of-the-art which could be developed and become operational in the early 1970's. Class II, while still considered conventional, are advanced somewhat beyond the present state-of-the-art. They would require developments, such as new propulsion systems, and could be operational in the mid-1970's. Class III are beyond what can be considered conventional. They require technology advancements, such as altitude compensating engines, and will utilize recovery and reusability.

Figure 3 shows some typical configurations of Class I and Class II vehicles. The two vehicles on the left represent current technology. The first is a two-stage, all-liquid vehicle, with LO_2/RP propellant first stage using 18 F-1 engines (uprated to 1.8 million pounds of thrust each) and having an engine-out capability. "Engine out" means that should one engine fail, the remaining engines will provide sufficient performance to accomplish the intended mission. The second stage has three M-1 engines (1.5 million pounds thrust each) and uses LO_2/LH_2 propellant. This vehicle is 450 feet high and is some 80 feet in diameter, has a take-off weight of 25 million pounds, and develops 32 million pounds of thrust.

The second configuration is typical of vehicles using a solid propellant first stage. This one uses four 300-inch motors. The second stage uses liquid (LO_2/LH_2) propellant with M-1 engines. This vehicle is over 500 feet in height, 60 feet in diameter, weighs 34 million pounds at launch, and produces about 50 million pounds of thrust. Solid vehicles are both taller and heavier than liquid ones.

The three configurations on the right of Figure 3 are typical of the Class II vehicles. All three require advancement of propulsion technology. All would use LO_2/LH_2 propellant. The first two would require development of a plug nozzle. Due to the quantity of modules, engine-out capability would be required, as well as vector control. The two-stage vehicle is 385 feet in height, 71 feet in diameter, launch weight 14.5 million pounds, with 18 million pounds of thrust developed at lift-off. The single stage vehicle is 377 feet in height, 81 feet in diameter, launch weight 24 million pounds, and take-off thrust of 30 million pounds.

The last configuration shown on Figure 3 is a stage-and-a-half vehicle having five 6-million-pound-thrust engines. Four of the engines would be staged at the end of the boost phase, with the sustainer engine operating to orbit. This vehicle would be 406 feet in height, 90 feet in diameter, 24 million pounds in weight, and develop 30 million pounds of thrust.

Advanced NOVA Concepts

The Class III vehicles are the furtherest advanced vehicles studied. These vehicles are somewhat unconventional in approach, as well as using new techniques in propulsion and recovery.

Figure 4 is a configuration studied by General Dynamics/Astronautics. This vehicle has an unconventional shaped first stage which is recovered by use of parachutes and retrorockets fired just before water impact. The first stage is 140 feet in diameter, with the thrust structure going between the LH_2 and LO_2 tanks. The vehicle shown has a chemical (LO_2/LH_2) second stage, however, this would lend itself very well to a nuclear second stage when it becomes available.

Martin/Baltimore also studied the configuration shown in Figure 5. They named it RENOVA. It is reusable, and features such things as air augmentation. Upon leaving the atmosphere the air scoops close, the ring which limited the expansion ratio is jettisoned, and the vehicle goes on to orbit. In orbit the payload is separated, the vehicle rotates 180 degrees, fires a retro, and reenters the atmosphere. Parachutes are deployed, and retrorockets are fired just prior to water entry. In the same manner as the General Dynamics vehicle, the vehicle floats with its engines out of the water. It is towed to shore, refurbished, and reused.

Douglas Aircraft studied an interesting concept called ROMBUS. Figure 7 shows it on its launching pad, which is a water filled basin. The vehicle has eight LH_2 tanks attached to the outside. As the propellant is used from the tanks, they are staged as shown in Figure 8. Parachutes slow the descent of the tanks, and they are recovered from the ocean and refurbished for reuse. Upon reaching orbit, the payload separates from the vehicle in much the same manner as for the RENOVA. Retrorockets give the vehicle the proper trajectory for entering the atmosphere and land recovery.

NOVA Launch Facilities

Due to the tremendous size of the vehicle required to lift one-million pound payloads, it was necessary to support the vehicle study contractors with a launch facilities study. A contract was issued to Martin-Denver by the Launch Operations Center, Cape Canaveral, to evaluate the various vehicle configurations in terms of launch facility requirements and feasibility. Factors such as vehicle concept, launch rates, assembly and checkout methods and procedures, acoustic limits and safety limits were among the considerations. Figure 9 is a typical view of a stage and its transporters in the stage checkout building. After checkout, the stages are removed from the checkout building by the transporters and taken to the launch area.

Figure 10 shows a concept of a combination assembly and launch building. The design includes removable flame ducts and deflector, and provides for internal erection and assembly of the vehicle. The umbilical arms swing into protected areas in the walls of the building at lift-off. The roof provides weather protection during erection, assembly, and checkout, but is opened at the time of launch. The inside of the building is provided with an acoustic liner that is also designed to withstand (with a minimum of refurbishment) the temperatures expected during lift-off. The launch building will stand about 650 feet high, with an outside diameter of about 350 feet. In some cases, the number of pads required would exceed the quantity that could be placed in the NOVA launch pad area without violating separation distance criteria. In these cases, it would be possible to harden the structure of the launch building so the pads could be placed closer together.

Conclusions

Results, to date, of the large launch vehicle studies lead us to several conclusions concerning the future vehicles which will follow or supersede SATURN V. These conclusions include the following:

1. A new large launch vehicle in the 500-ton orbital payload class will be justified only if one or more of the following mission requirements materialize:

   a. Manned planetary exploration
   b. Large lunar base
   c. Large orbital or global cargo missions.

2. Take-off weights of a 500-ton orbital payload class vehicle range from 15 to 40 pounds. Two-stage expendable liquid vehicles are the lightest, and solid first stage vehicles are the heaviest.

3. Development time for a vehicle in this class will range from seven to nine years.

4. Development costs for such a vehicle will be from 3.5 to 6 billion dollars.

5. Reusability is most desirable and will result in transportation efficiencies. Although expendable vehicles have less cost effectiveness, they still make manned planetary flights an attractive proposition.

6. The vehicle concepts considered must be compatible with the requirements for low density payloads of 2 to 3 pounds per cubic foot and diameters of not less than 60 feet.

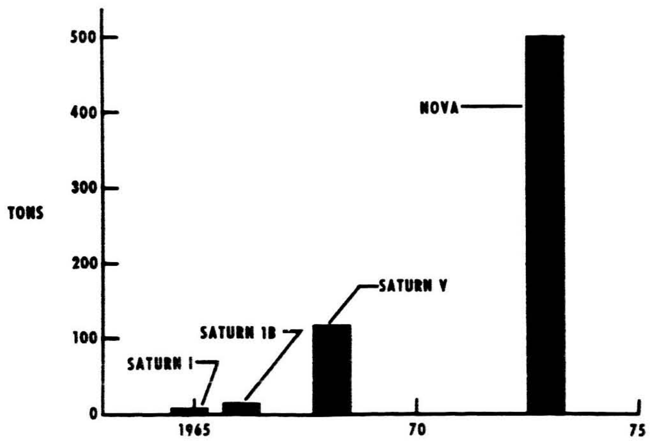
7. Although a reusable single-stage vehicle appears desirable, development risk makes it marginal. One or more of the following features possibly could make them less marginal:

   a. Solid or liquid JATO's.
   b. Air augmentation.
   c. Staging of tanks and/or engines.

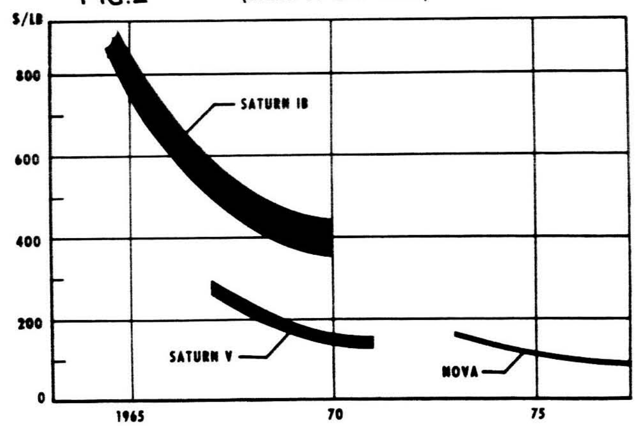
Acknowledgments

The author wishes to express his sincere gratitude to the Future Projects Office of the Marshall Space Flight Center, General Dynamics/Astronautics, Martin-Baltimore, Douglas Aircraft, and Martin-Denver for the visual aids used in this paper.

TREND OF PAYLOAD CAPABILITY OF LARGE LAUNCH VEHICLES
FIG. 1 (EARTH TO LOW ORBIT)



TREND OF DIRECT OPERATING COST FOR LARGE LAUNCH VEHICLES
FIG.2 (EARTH TO LOW ORBIT)



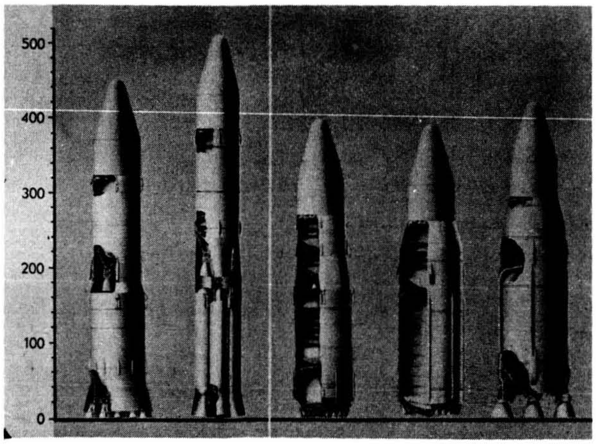


Figure 3 TYPICAL LARGE LAUNCH VEHICLES

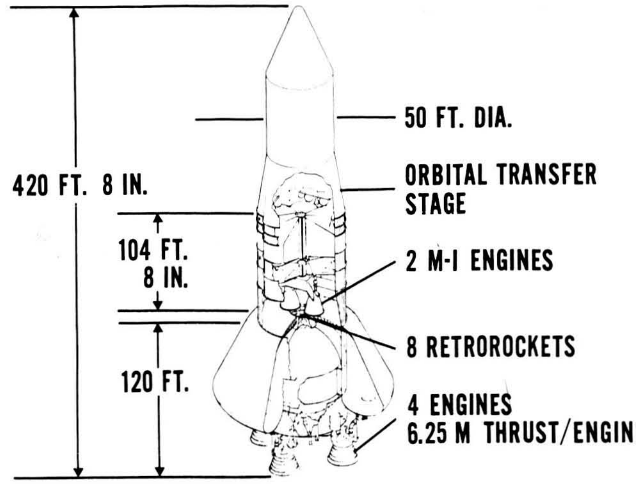


Figure 4 VEHICLE OPTIMIZED FOR FULL STAGE RECOVERY AND REUSE (General Dynamics/Astronautics)

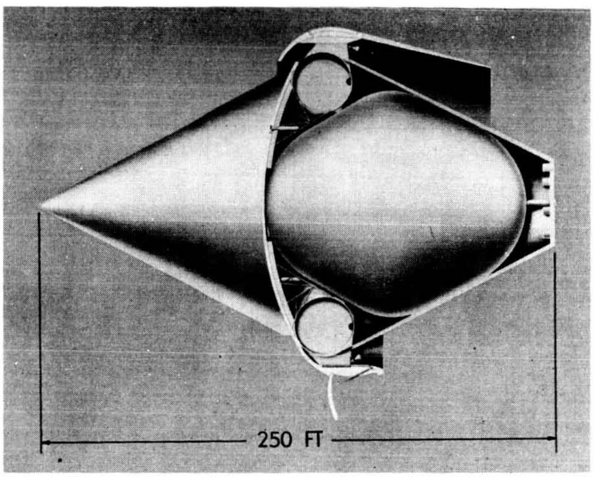


Figure 5 RENOVA - INBOARD PROFILE (Martin)

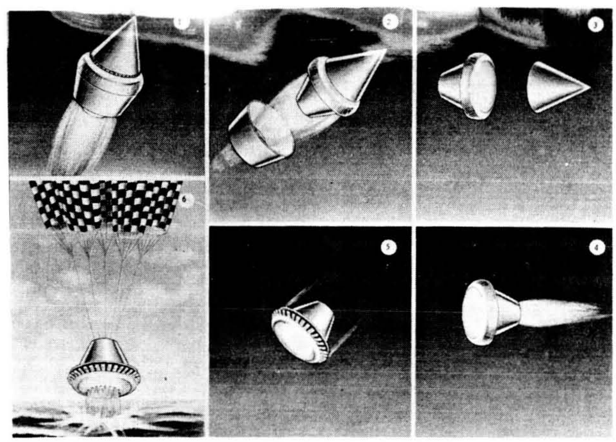


Figure 6 RENOVA - LAUNCH & RECOVERY (Martin)

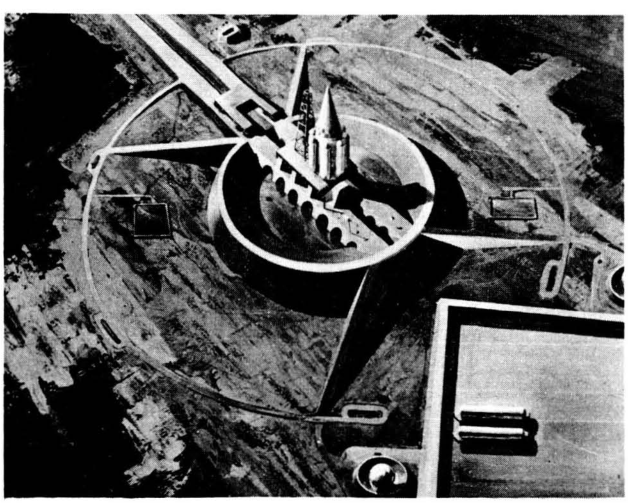


Figure 7 ROMBUS (Douglas)

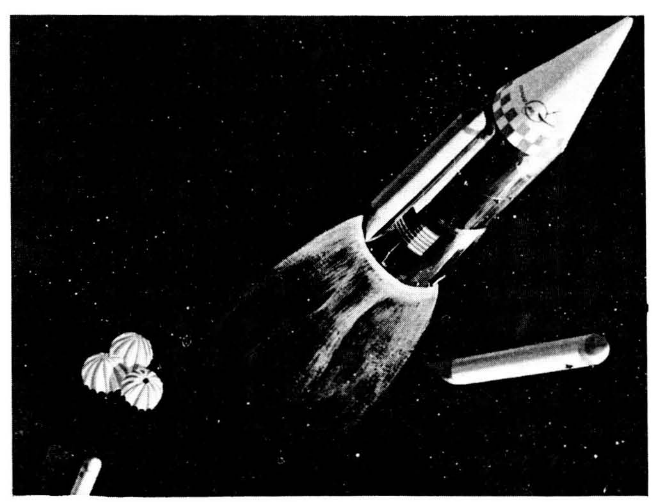


Figure 8 ROMBUS - TANK STAGING (Douglas)

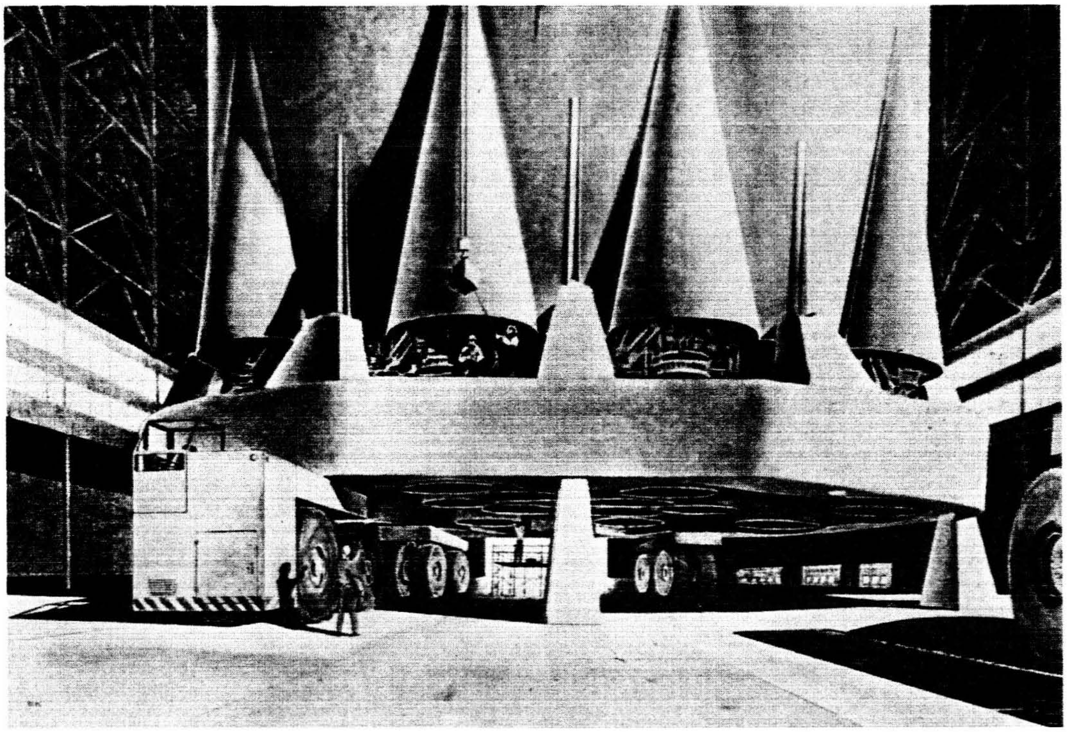


Figure 9 POSITIONING AT SCB (Martin)

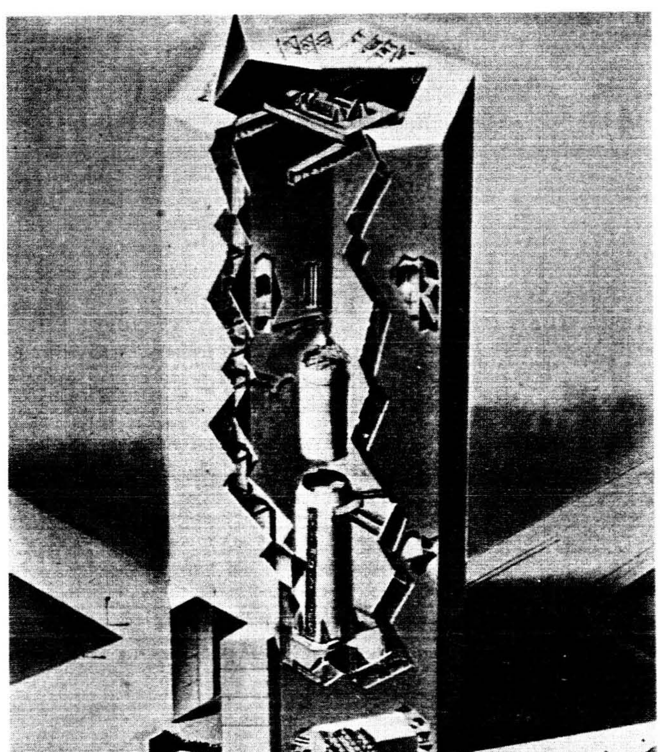


Figure 10 LAUNCH BUILDING (Martin)

MANUAL CONTROL OF LAUNCH VEHICLE SYSTEMS

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179G101

I. INTRODUCTION

When the launch vehicle of a manned space system is ignited and the hold-down clamps are pulled, tens of millions of dollars in equipment and ground support labor will already have been spent whether the primary mission is accomplished or not. A great variety of malfunctions can result in aborting the primary mission and yet allow the accomplishment of one or more alternate missions. The complete Apollo spacecraft, with its Command Module, Service Module, and Lunar Excursion Module, will be a very high capability craft which could be used for several advantageous missions other than the manned lunar landing. As an example, let us assume that during the boosted phase of flight prior to reaching the desired injection, one of the stages operates with less than the allowable minimum thrust. The lunar landing cannot be made. Depending upon the degree of the malfunction here are some possibilities of what the pilot might do:

1. Make a fly-by of the moon.
2. Eject LEM to reduce weight and proceed to orbit the moon.
3. Inject the Command Module, Service Module, and the Lunar Excursion Module into a circular earth orbit at some pre-determined altitude and:
    a. Remain in this orbit for several days using the C/M as a space station.
    b. Deposit LEM for possible future use.
    c. Separate LEM with crew to make one or more long range rendezvous to further demonstrate system capability for subsequent lunar attempts.
    d. Use the Service Module to inject the C/M and LEM into the required orbit for effecting a rendezvous with some space station which may be in service at that time.

Certainly there are more types of malfunctions and more possibilities of what may be done to salvage as much from the mission as possible. Also, those malfunctions which would not allow the completion of any practical secondary mission can result in a variety of circumstances requiring a number of abort procedures if an optimum recovery is to be achieved in each case. It is unlikely that a pre-programmed automatic guidance system can handle all of the practical possibilities, whereas a manual system quite possibly can.

In deciding whether or not to provide manual guidance capability, the first question to arise is, what is the probability of an abortive malfunction? We need only admit that there is at least some probability. Then, what is the burden on the primary mission if we provide the manual control and guidance capability necessary to accomplish the secondary missions? The answer to this question is that we don't know until we have determined man's ability, established what he requires, and developed the manual techniques necessary to accomplish an exacting type of human activity never before attempted. Simulation with man-in-the-loop is the only means available to us to arrive at an answer we can rely on with any real confidence.

Chance Vought has conducted several man-in-the-loop studies using the Manned Aerospace Flight Simulator facility. The following paragraphs briefly describe this simulator and discuss two studies of manual control of launch vehicles.

II. SIMULATOR DESCRIPTION

The Manned Aerospace Flight Simulator was designed as a general purpose research facility for man-in-the-loop studies. A digital-analog computer performs a real-time solution of all of the general equations, vehicle dynamics, and the control and guidance equations in a full six-degrees-of-freedom. The simulator is shown in Figures 1 and 2. The gondola contains a single seat cockpit with a fully mechanized instrument panel and cockpit controls. The gondola is suspended on a hydraulically powered gimbal system to provide angular and translatory motions. This system is in turn mounted on a pitching base which positions the longitudinal axis of the gondola from full nose-up to full nose-down which, for example, can be programmed according to the computed longitudinal load factor, thus simulating axial acceleration forces from 0 to $\pm$ 1 g. The cockpit instrument panel, shown in Figure 3, is a general purpose research panel which is modified as required to present the necessary display parameters for each simulation and can be readily interchanged with any other panel.

The simulator is surrounded by a spherical screen and a gimbal mounted apparatus projects a scene of the earth (or moon) in accordance with the attitude changes of the simulated vehicle. A high-fidelity speaker system provides characteristic noises of rocket engines, etc. Vibrations are programmed through the moving base and through the cockpit seat which is separately suspended and powered. A master control station, as shown in Figure 4, coordinates the simulated flights and provides repeater instruments, x-y plotters, and pen recorders.

III. EARTH ESCAPE LAUNCH STUDY

The first of our studies to be discussed is a simulation of a two-stage launch vehicle and a manned lifting ballistic capsule. The basic equations of translational motion, velocities, altitude, position, dynamic pressure, etc., were computed in six-degrees-of-freedom. All position calculations were referenced to a fixed inertial plane and a two-body gravitational system (earth and vehicle) was assumed to exist. A standard 1959 ARDC atmosphere was used and perturbations to the velocities resulted from the aerodynamic and thrust loads on the vehicle. Geographic headings and positions relative to the reference plane and to a surface target on the rotating earth were resolved in the digital computer.

The simulated vehicle was assumed to be a rigid body. The two-stage booster system contained five engines in the first stage and two in the second. All engines were handled independently in the computer, i.e., the thrusts, fuel flows, and nozzle deflections were computed separately for each engine.

Moments of inertia and center-of-gravity position were computed as functions of vehicle mass which varied with fuel flow, time and staging. The second stage was equipped with four fixed direction, low constant thrust engines for velocity control and a single engine was on the capsule for separation from the second stage. Deflection of the booster nozzles and reaction controls on the second stage and capsule provided attitude control.

The pilot manually lighted and terminated the booster engines as required. Separation of the first stage was automatic with first stage thrust termination. Second stage separation and engaging the capsule's reaction controls were manual functions. A three-axis side arm controller controlled attitude through a rate-command system with a rate damping stability augmentation system.

The cockpit displays included the following:

a. Pitch attitude

b. Roll attitude

c. Sideslip

d. Horizontal velocity and burnout command

e. Vertical velocity and burnout command

f. Altitude

g. Angle-of-attack

h. Normal acceleration

i. Longitudinal acceleration

j. Vehicle heading

k. Present course

l. Programmed course

m. Differential heading of present course and programmed course.

n. Offset distance of vehicle from the reference inertial plane.

o. Altitude vs. velocity situation of the vehicle relative to a graphic presentation of the nominal trajectory.

Figure 5 shows the nominal altitude vs. velocity trajectory and some typical trajectories achieved by manual control. The flights were launched from a rotating earth 15 miles off the inertial plane intended for burnout. Therefore, the inertial trajectory of the launch vehicle had to be turned into this reference inertial plane prior to burnout and while simultaneously flying the launch trajectory. The degree of success in meeting the requirements for burnout is indicated in Table I. The required accuracies indicated are those deemed suitable for a direct launch into an earth to moon trajectory.

The results of this study are by no means conclusive but they give an indication that the human operator, when given adequate display information, can control the trajectory of a launch vehicle with the required precision.

During the study, equipment malfunctions and errors made by the pilot while he was still learning the procedure resulted in random situations that would not allow completion of the mission. In these cases the pilot executed various abort procedures at his own discretion. Figure 6 illustrates one case in which the pilot aborted at an altitude of approximately 700,000 feet with a relative velocity of 17,500 fps and a sink rate of some 1,000 fps. He terminated thrust of the second stage booster and allowed the vehicle to descend to an altitude of approximately 500,000 ft. at which time he oriented the vehicle to a pitch angle of plus 75° and re-lighted the booster. This checked the descent and accelerated the vehicle toward circular satellite speed. He manipulated the pitch angle as required to bring the vertical velocity to zero upon reaching an altitude of 400,000 ft. Holding altitude constant, he continued thrust and coasted in near circular orbit until given the command to de-orbit. With the vehicle rotated 180° in yaw, he re-lighted the second stage booster until the velocity was reduced by 500 fps. As the vehicle descended toward the sensible atmosphere, he separated the command module from the booster and set an L/D max glide attitude. The capsule descended to impact the earth at 200 fps since no drag chute or landing chute was deployed in this simulation.

The variety of abort situations encountered demonstrated how a pilot's judgement and flexibility can be utilized to:

1. Adjust trajectory errors.

2. Abort the primary mission and select and carry out an alternate mission.

3. Delay thrust termination and alter the trajectory to make the safest capsule escape and/or reach an optimum recovery area.

TABLE I

TABULATION OF RESULTS

Burnout Parameter	Required Accuracy	Results	Comments
Horizontal velocity 34,300 ft/sec.	± 13 ft./sec.	Predominantly negative but usually very close.	More sensitive velocity indicator would provide required accuracy.
Vertical velocity 2,460 ft/sec.	± 25 ft./sec.	Predominantly positive - occasional miss	Readability of display not quite satisfactory.
Altitude 629,000 ft.	± 16,000 ft.	Predominantly negative but usually very close.	Display used was designed for qualitative information and had to be used as a quantitative display.
Lateral displacement 0 N. Miles	± 5 N. miles	Always positive	Display provided high accuracy and readability.
Lateral velocity 0 ft./sec.	± 13 ft./sec.	Predominantly positive - occasional miss.	Required accuracy same order of magnitude as display readability.
Range 1972 N. Miles	± 4 N. miles		No attempt was made to control range in this program.

A great amount of work is yet to be done to develop display concepts and flight techniques before we can know what the human operator's capabilities and requirements really are.

IV. LUNAR LAUNCH AND RENDEZVOUS

Another of our studies regarding pilot control of launch vehicle systems was a simulation of a lunar landing craft. The objective of this study was to obtain data regarding a pilot's ability to control the launch, transfer, and terminal phases of the lunar lander mission and to determine how much assistance he will require from cockpit displays, on-board computers, autopilots, etc.

The simulated flights began with the lunar lander on the lunar surface and ended when it was within a few thousand feet of the mother ship. The final closure and docking maneuver was not included in this experiment. The lunar surface position of the lunar lander at the time of launch was at various distances, up to 60 miles, away from the orbital plane of the mother ship, thus requiring a plane change maneuver.

The mechanics of the problem are shown in Figure 7 and Table II. The mother ship was in a circular orbit at an altitude of 59 N. miles. The lunar lander was launched when the line-of-sight elevation to target was 86.4°. After lift-off the lunar lander was yawed to align the pitch plane parallel to the orbital plane of the mother ship. The lunar lander was then pitched as required to maintain an efficient trajectory to an altitude of 50,000 feet and level flight speed of 5,196 fps. The pilot then reduced thrust to 1/3 of full thrust and continued at constant altitude until a Hohmann injection speed of 5,555 fps was obtained. This put the lunar lander on a transfer orbit having an apolune of 59 miles. During the transfer the pilot made mid-course corrections as required to arrive at the terminal window, which we defined as a sphere around the mother ship having a radius of 15,000 feet.

The boost trajectory is shown in Figures 8 and 9. The altitude vs. range plot shows that the nominal trajectory climbs to 50,000 feet and is held at this altitude while accelerating to the required injection speed. The out-of-plane range vs. in-plane range plot shows that the launch is always made parallel to the orbital plane of the mother ship.

The horizontal and vertical velocity schedules required to govern the nominal trajectory are shown in Figures 10 and 11. Circular orbital speed is 5484 fps and the Hohmann injection speed for an apolune of 59 N. miles is 5555 fps. The vertical velocity increases from zero to 300 fps and, of course, back to zero on reaching burnout altitude.

By launching parallel to the orbital plane of the mother ship, the lunar lander's orbit crossed the mother ship's orbit 90° after injection. On approaching this plane intersection, thrust was applied in the local horizontal perpendicular to the lunar lander's orbital plane as required to make the turn into the plane of the mother ship.

The cockpit instrument panel included the following mechanized displays:

1. Three axis attitude ball
2. Pitch command bar
3. Present course heading (relative to target plane)

TARGET ALTITUDE = 358,720 FT. (59 N. MI.)
TARGET VELOCITY = 5,342 FPS (ANOMALY RATE = 0.0505022 DEG/SEC)

FLIGHT CONDITION	TIME	VEHICLE			TARGET				
		HORIZ. VEL.	VERT. VEL.	ALTITUDE	POSITION ANGLE REL. TO VEHICLE	ELEVATION ANGLE REL. TO VEHICLE XY PLANE & CORRESPONDING VEHICLE ATTITUDE			
						ELEVATION DEG.	ROLL DEG.	PITCH DEG.	HEADING DEG.
	SEC	FPS	FPS	FT.	DEG.				
BOOST INITIATION	0	0	0	0	-0.212[1] -0.302[2]	86.4[1] 84.9[2]	0	0	180
LEVEL OFF AT INSERTION ALT.	261.8	5,196	0	50,000	6.968[1] 6.878[2]	73.6[1] 71.1[2]	0	86.5[1] 88.8[2]	180
CIRCULAR SPEED	293[1] 272.2[2]	5,484	0	50,000	6.900[1] 6.835[2]	70.1[1] 69.8[2]	0	90	180/0
BURN OUT	300.6[1] 274.8[2]	5,555	0	50,000	6.832	69.0[1] 69.5[2]	0	90.8[1] 90.3[2]	0
LINE OF SIGHT DIST. REDUCES TO 70,000 FT.	2,663	5,330	114	293,300	-0.233	69.5	0	0	180
						0	0	69.5	180
TRANSFER ORBIT APOLUNE	3,729.5	5,272	0	358,720	0	0	0	0	180

NOTES: 1 - FULL THRUST UP TO LEVEL-OFF, THEN 1/3 THRUST TO BURNOUT
2 - FULL THRUST FOR ENTIRE BOOST

TABLE II NOMINAL DATA - BOOST & RENDEZVOUS

4. Vertical velocity with command bug
5. Horizontal velocity
6. Altitude
7. Normal acceleration
8. Body axis angular rates
9. CRT display - altitude vs. horizontal velocity
10. TV display - guidance modes

The command display indicated to the pilot the correct pitch attitude to maintain at all times and the required vertical velocity. The pitch command bar showed pitch error determined from combined deviations in pitch and/or vertical velocity from nominal.

The guidance modes on the TV display were selected by the pilot. The boost guidance mode displayed:

a. Differential anomaly angle
b. Differential altitude
c. Lateral displacement from orbital plane of the mother ship.
d. Differential angle between orbital planes
e. Predicted position of the mother ship at lunar lander apolune
f. Predicted position of lunar lander at apolune

The mid-course guidance mode displayed the above parameters with increased scale resolution. The third TV display mode simulated a line-of-sight radar presentation of line-of-sight elevation angle, bearing angle, and range. A reference star was also pictured on the screen to represent an actual star relative to the mother ship for use in governing the line-of-sight during the terminal phase of the rendezvous.

During the experiment four different modes of operation were included:

a. Automatic Mode - Completely automatic except that the booster was terminated by the pilot using a burnout command display.

b. Manual-Command Display - The pilot in full control using command, predictive, and status displays.

c. Manual-Predictive Display - The pilot in full control using predictive and status displays.

d. Manual-Status Display - The pilot in full control using only status displays.

The results of some 136 flights in the four modes indicated that the fuel required to reach the terminal rendezvous window 15,000 feet from the mother ship, in a suitable situation for the final closure, was less than 4 percent in excess of the theoretical minimum. The concensus of opinion is that minor improvements in displays and technique can reliably reduce this to about 2 percent. It further indicated that the difference in the efficiency of the four modes of operation is only approximately 1 percent of the fuel.

It is important to note that the least of the display modes consisted of only the following parameters:

a. Attitude

b. Altitude

c. Vertical Velocity

d. Horizontal velocity

e. Course heading relative to target plane

f. Line-of-sight bearing

g. Line-of-sight elevation

h. Line-of-sight range

Figures 8 and 9 show a typical case of the pilot's ability to control the boost trajectory using only display parameters a through e above and Figures 10 and 11 show how the velocity programs as a function of altitude were maintained.

V. CONCLUDING REMARKS

These two simulator studies have provided evidence that launch vehicle guidance is within the capabilities of the pilot, that display requirements are considerably less complex than many technically qualified personnel believe them to be, and that the pilot can serve as a very flexible and reliable guidance and control component. Modern rocketry over the past several decades has necessarily developed without man-in-the-loop and has forced the development of sophisticated automatic guidance systems capable of such feats as that of Mariner II. Manned space systems are now a reality and there has been relatively little work toward developing manual guidance and control systems. The pilot has been by-passed in this function and the tendency more often than not seems to be to leave it that way.

This should not be the case. Manned space vehicles must develop as multi-mission capability systems in which optimum utilization of the human operator as a guidance and control component will put automatic systems into their proper place as auto-pilots to be used at pilot option.

References

1. "Preliminary Investigation of Manual Control of the Apollo Booster System Using the Manned Space Flight Simulator," Chance Vought Corporation Report No. AST/E1R-00.15, November 1961.

2. "Lunar Boost and Rendezvous Simulator Study," Chance Vought Corporation Report No. 00.133, December 1962.

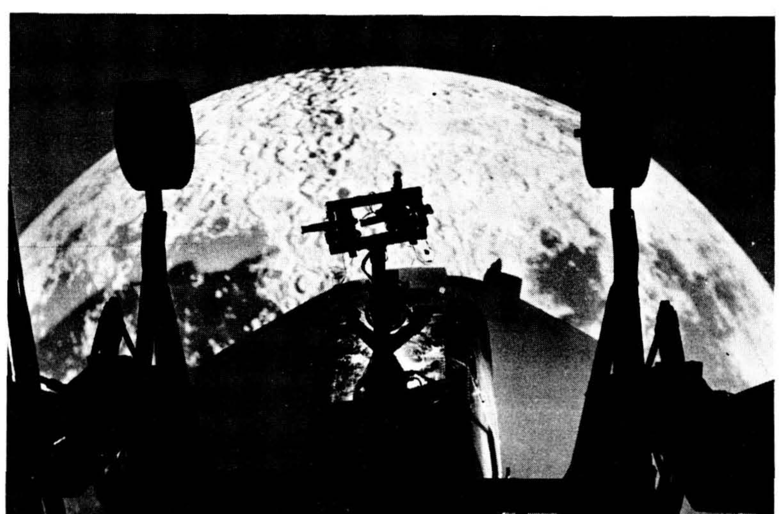

FIGURE 1
MANNED AEROSPACE FLIGHT SIMULATOR

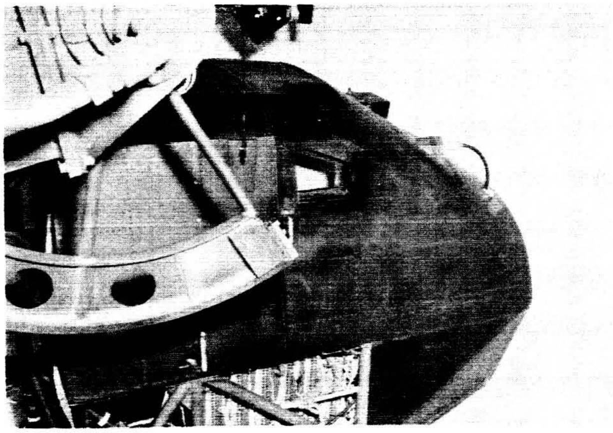

FIGURE 2
SIMULATOR GONDOLA

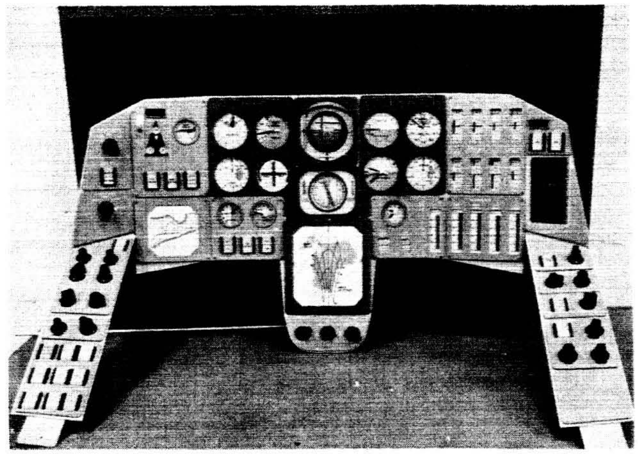

FIGURE 3
SIMULATOR INSTRUMENT PANEL

FIGURE 4
MASTER CONTROL STATION

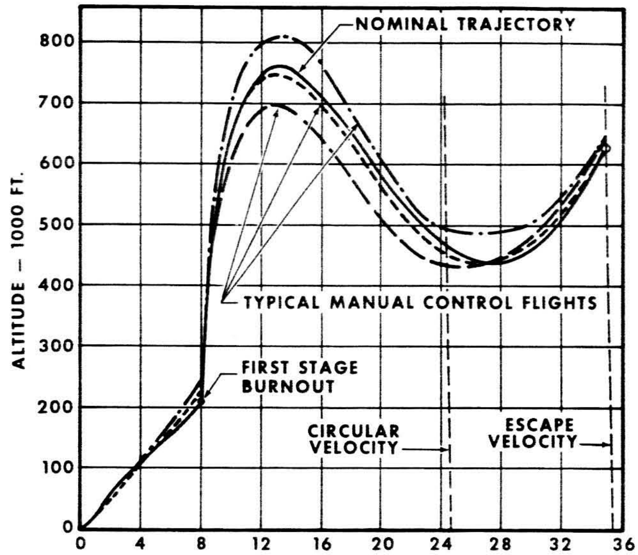

FIGURE 5
EARTH ESCAPE BOOST TRAJECTORIES

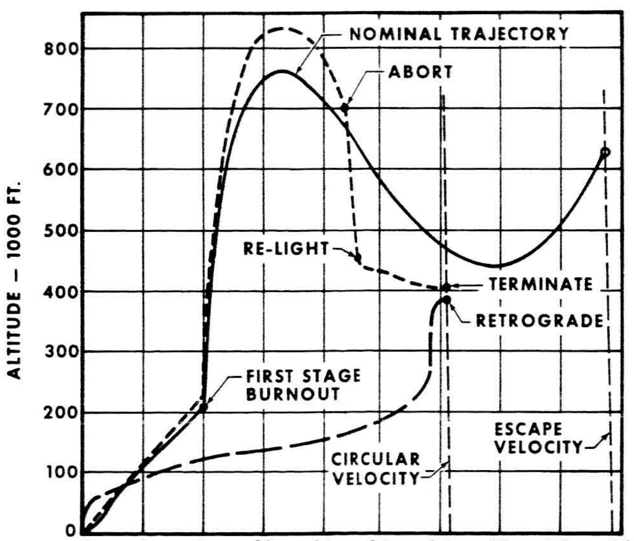

FIGURE 6
BOOST ABORT TRAJECTORY

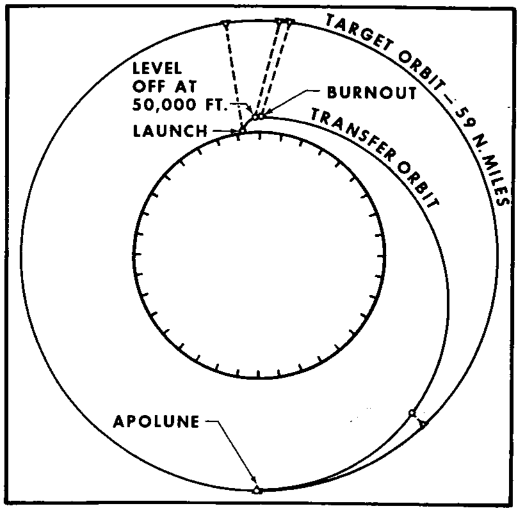

FIGURE 7
LUNAR BOOST AND RENDEZVOUS TRAJECTORY

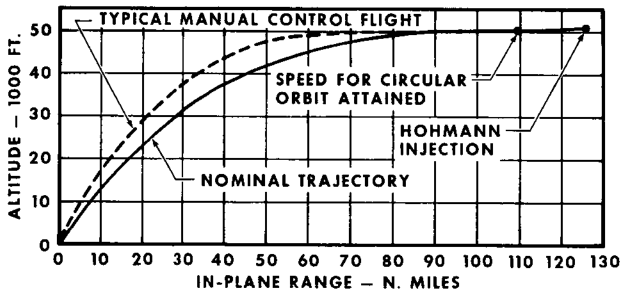

FIGURE 8
LUNAR BOOST TRAJECTORY

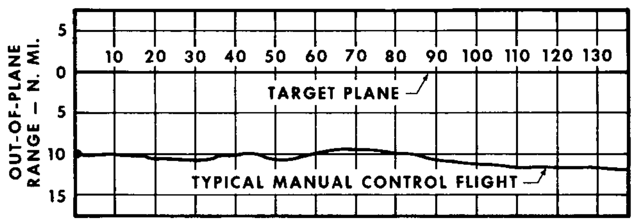

FIGURE 9
LUNAR BOOST PLANE

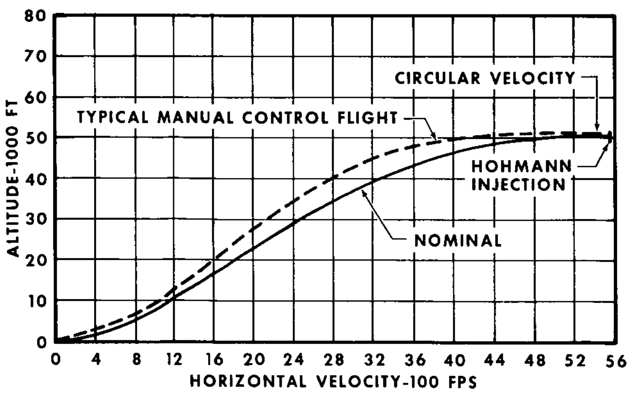

FIGURE 10
HORIZONTAL VELOCITY PROFILE

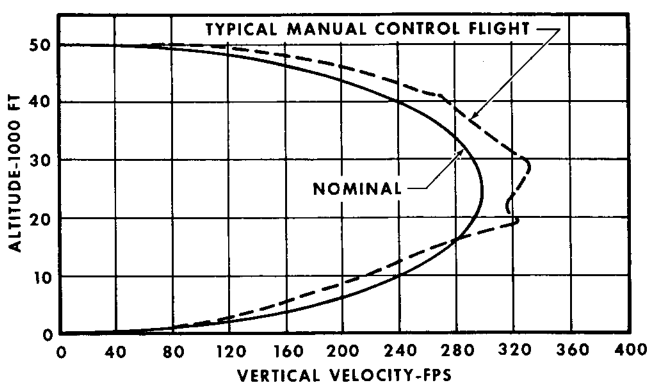

FIGURE 11
VERTICAL VELOCITY PROFILE

double code

LAUNCH VEHICLE COST ANALYSIS AND SYSTEM EVALUATION

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Abstract 23210

Space Technology Laboratories has been engaged in design-cost studies for the Future Projects Office, Marshall Space Flight Center for the past two years. These studies were performed under contract number NAS 8-2599. This paper is a brief summary of the Phase III report. The report is unclassified and available to industry.

The purpose of these studies was to develop a methodology and data for performing cost analyses and system evaluation for different launch vehicle designs in the Saturn and Nova classes. It was desired to have a consistent and usable technique for estimating the total cost and cost-effectiveness of large launch vehicle programs.

This paper presents in summary form, a technique for developing cost category models. Design values for the C-1 launch vehicle were used and estimated costs for this system are listed. In addition, some system evaluation techniques are illustrated.

AUTHOR

LIST OF CHARTS

CHART NO.	TITLE
1.	OBJECTIVES
2.	COST CATEGORIES
3.	COST VARIABLES
4.	INDIRECT COSTS, C_R AND C_G
5.	INDIRECT COSTS, C_F AND C_D
6.	DIRECT COSTS, C_V
7.	DIRECT COSTS, C_P, C_T, AND C_L
8.	TOTAL COST ANALYSIS
9.	MISSION ANALYSIS
10.	TRADE-OFF STUDIES
11.	MATHEMATICAL PROGRAMMING

Chart 1 lists the four major objectives of the study.

The first objective was to define the mission, performance-design, and operational variables in a way that would be applicable for a wide variety of missions, designs, and operations.

The mission was defined as payload (pounds) placed in a standardized orbit, put into escape velocity, or cargo delivered to the moon. Performance-design variables refer to the usual equations for payload fraction, mass ratio, structure ratio, etc. A common variable of these and the cost equations is weight. The operational variables refer to the launch base parameters i.e. launch rate, number of pads, required number of operational launches, etc.

The second objective was to develop a technique for estimating the cost of launch vehicles. This consisted of first defining the cost categories in detail. The mathematical form of cost equations for each cost category was then assumed and cost factors or cost estimating relationships were then developed. The relationships were determined by extrapolation from historical data, trade publications and contractor studies.

The third objective was to integrate the mission, performance-design and operational variables in a cost-effectiveness model so that perturbations of these variables and trade-offs could be studied.

Finally, the cost and cost-effective models were analyzed to determine efficient system specifications and operating characteristics.

OBJECTIVES

1. DEFINE MISSION, PERFORMANCE-DESIGN AND OPERATIONAL VARIABLES.
2. DEFINE COST CATEGORIES, ESTIMATE FACTORS, STRUCTURE COST EQUATIONS.
3. INTEGRATE SYSTEM VARIABLES IN A COST-EFFECTIVE MODEL.
4. ANALYZE MODEL TO DETERMINE EFFICIENT SYSTEM SPECIFICATIONS AND OPERATING CHARACTERISTICS.

Chart 1

Chart 2 lists the indirect and direct cost categories as defined by the Future Projects Office.

The indirect costs can be considered fixed in nature. The direct costs vary with the number of launches. Another name for direct costs are specific direct operating costs (SDOC).

COST CATEGORIES

INDIRECT COSTS, C_I	DIRECT COSTS, C_O
RANGE AND GENERAL OVERHEAD, C_R	PRODUCTION, C_V
GROUND SUPPORT EQUIPMENT, C_G	PROPELLANT, C_P
LAUNCH FACILITY, C_F	TRANSPORTATION, C_T
DEVELOPMENT, ENGINEERING, TEST, C_D	PERSONNEL, C_L

Chart 2

Chart 3 lists the variables which account for different cost estimating relationships or cost standards.

The first set of variables, the vehicle class, indicates the size and complexity of the launch vehicle. Cost estimating relationships and cost equations are unique for each class. They must be modified if it is desired to apply them to another class. An example of modification is the use of scaling down factors for dollars per pound of structure.

The second set of variables, the design variables, affect the development, production and propellant cost categories. Stage designation (first, second, third), propellant type (LOX-RP, LOX-LH, etc.), and the subsystem weights determine some of the parameter values in the cost equations.

The third set of variables, the operational variables, affect the range and general overhead, ground support equipment, launch facilities, transportation, and personnel cost categories. In general, these variables pertain to the launch base operations. The cost models for these categories are structured as a function of program years, number of operational pads, number of launches, launch rate per year, time on pad, and countdown time.

The fourth set of variables can be termed a complexity-judgment factor which perturbs the cost standard or cost estimating relationship. Complexity is determined by the materials used, density factors, structure ratios, and manufacturing techniques. The judgment or "mental set" of the analyst is also a modifying factor in the development of the parameters and the cost of equations.

COST VARIABLES

1. VEHICLE CLASS
   - A. THOR, ATLAS, TITAN
   - B. SATURN C-1, C-5
   - C. NOVA
2. DESIGN VARIABLES
   - A. STAGES
   - B. PROPELLANT TYPE
   - C. SUBSYSTEM WEIGHTS: STRUCTURE; PROPULSION; GUIDANCE AND CONTROLS
3. OPERATIONAL VARIABLES

   PROGRAM YEARS, PADS, OPERATIONAL LAUNCHES, LAUNCH RATE, TIME ON PAD
4. COMPLEXITY-JUDGEMENT FACTOR

Chart 3

Chart 4 lists the equations that were developed for the range and general overhead; and ground support equipment cost categories.

C_R Cost Category - Equation 1 states C_R as the sum of range operations and launch base operations. The range operations have been expressed as a function of countdown hours cost per launch. The launch base operations have been computed by prorating yearly overhead costs to the number of pads. Equation 2 results from substituting probable values for the C-1 launch vehicle in equation 1. The symbols are:

C_R (C-1) = range and general overhead costs, $10^6\$$

K_1 = range operations cost per countdown hour per launch, $10^6\$$; = 0.025, assumed value for C-1 system

T_c = average number of countdown hours per launch; = 10, assumed value for C-1 system

N = number of launches

K_2 = launch base operations cost per pad per complex per year, $10^6\$$; = 4, assumed value for C-1 system

B = number of launch complexes

P = number of launch pads per complex; = 2 for C-1 system

Y = number of operational years in the program

C_G Cost Category - Equation 3 states C_G as the sum of its development costs, complex associated equipment, pad associated equipment and its maintenance cost. Equation 4 results from substituting probables values for the C-1 launch vehicle in equation 3. The symbols are:

C_G (C-1) = ground support equipment costs, $10^6\$$

k_G = development costs, $10^6\$$. A probable value for this term could be 20% of the cost of the ground support equipment.

c = cost of launch control ground support equipment per complex, $10^6\$$; = 10, assumed value for C-1 system

N = cost of checkout and related ground support equipment per pad per complex, $10^6\$$; = 4, assumed value for C-1 system

M = number of pads serviced by one set of launch control ground support equipment; assume 2.

n = yearly maintenance cost factor, percent of ground support equipment hardware cost; assume 8% per year

INDIRECT COSTS

RANGE AND GENERAL OVERHEAD, C_R

$$C_R = K_1 T_C N + K_2 BPY \tag{1}$$

$$C_R(C\text{-}1) = (.025)(10)N + 4B(2)Y \tag{2}$$

GROUND SUPPORT EQUIPMENT, C_G

$$C_G = k_G + cB + \frac{rPB}{M} + nY\left(cB + \frac{rPB}{M}\right) \tag{3}$$

$$C_G(C\text{-}1) = k_G + 14B(1 + .08Y) \tag{4}$$

Chart 4

Chart 5 lists the equations that were developed for the launch facility and development cost categories.

C_F Cost Category - Equation 1 states C_F as the sum of its development cost, a constant term, complex associated facilities, pad associated facilities and its maintenance cost. Equation 2 results from substituting probable values for the C-1 launch vehicle in equation 1. The symbols are:

C_F (C-1) = launch facility costs, $10^6\$$

k_F = development costs, $10^6\$$. A probable value for this term could be 15% of the cost of the launch facilities

X = launch base expansion requirements for facility equipment, $10^6\$$; = 5, assumed value for C-1 system

b = cost of complex related facility cost, $10^6\$$; = 20, assumed value for C-1 system

p = cost of pad related facility cost, $10^6\$$; = 15 per pad per complex, assumed value for C-1 system

m = yearly maintenance cost factor, percent of launch facility cost; assume 5% per year

C_D Cost Category - Summary development cost category items and their probable cost estimating relationships are noted. Item IB, Engineering is most difficult to estimate and most critical because it is the item having the highest amount. Its value can range from \$2,000 to \$4,000 per pound of structure depending on the design configuration. The equation, $7.0\ W_S^{0.33}$ was developed from contractor cost estimates. Miscellaneous items include instrumentation, handbooks, and documentation, all denoted by the symbol k; and tooling. Tooling costs have been divided into basic and sustaining. They are most difficult to estimate and are unique for each design. The cost of item ID, Test Items, consists of a factor, 1.08, for spares; the first unit vehicle cost, a; and the number of development test items, N_D. Contractor test facilities, item IIA, was not estimated. Test operations, item IIB, consist of launch base test operations and/or contractor test operations. The symbol c, can be the unit test operations cost. Capital expenses consist of contractor engineering and production facilities and are unique for each contract. Sustaining effort consists of the cost of design changes and related production changes. An arbitrary percentage factor, 5% to 25% of C_D and 5% to 10% of C_V, can be used.

Equation 3 states that C_D is a function of a constant term, K; the structure weight, W_S; the first unit production cost, a; the number of development test units, N_D; and the total vehicle production cost, C_V.

Equation 4 results from substituting probable values for the C-1 system. The symbols are:

C_D (C-1) = total development cost for the C-1 system, $10^6\$$

342 = constant value for capital costs, $10^6\$$

W_S = total structure weights of launch vehicle

a_f = first unit launch vehicle production cost of structure only, $10^6\$$

N = number of operational launches

a_{s_i} = first unit cost of i the stage, $10^6\$$

N_{s_i} = number of ith stages that would be tested separately from a complete launch vehicle

a = first unit production cost of the launch vehicle, $10^6\$$

N_D = number of development vehicles

8.5 = unit cost, $10^6\$$, of a test launch

C^1_V = total operational launch vehicle production cost less engines, $10^6\$$

INDIRECT COSTS

LAUNCH FACILITY, C_F

$$C_F = k_F + X + bB + pPB + mY(bB + pPB) \quad (1)$$

$$C_F(C\text{-}1) = k_F + 5 + 50B(1 + .05Y) \quad (2)$$

DEVELOPMENT C_D

<u>ITEM</u>

I.	DEVELOPMENT	
	A. SYSTEMS MANAGEMENT	6% OF (I + II)
	B. ENGINEERING	2,000 TO 4,000 \$/# OR $7.0\ W_S^{.33}$
	C. MISCELLANEOUS	k + BT + ST
	D. TEST ITEMS	$1.08\ aN_D$
II.	TEST	
	A. FACILITIES	---
	B. OPERATIONS	cN_D
III.	CAPITAL EXPENSES	---
IV.	SUSTAINING EFFORT	$\%\ C_D + \%\ C_V$

$$C_D = f(K, W_S, a, N_D, C_V) \quad (3)$$

$$C_D\ (C\text{-}1) = 342 + 7.77\ W_S^{.33} + 16.7\ a_f^{.5} + .0333\ a_f N^{1.1} + 1.15 \sum a_{s_i} N_{s_i} + 1.15\ aN_D + 8.5\ N_D + .05 C_V^1 \quad (4)$$

Chart 5

Chart 6 lists the equations that were developed for the launch vehicle production cost category.

Equation 1 states the functional dependence of cost on weight, engine type, quantity of launches and learning curve.

Equation 2 is a simplified version of equation 3. In equation 3, costs are estimated in detail by stage, engine, guidance subsystem or structure subsystem. The dollars per pound parameter for structure for upper stages may be increased by from 5% to 10% to account for increased manufacturing complexity and a like amount for a propellant combination other than LOX-RP.

The symbols are:

C_V = Launch vehicle production cost, millions of dollars

f = Functional relationship

W = Weight, pounds

E = Number and types of engine

N = Number of operational launch vehicles

$\left.\begin{matrix}\alpha\\ \beta\\ \gamma\end{matrix}\right\}$ = Learning curve exponents

a_e = First unit engine cost, dollars

n_e = Number of a particular type of engine per vehicle

c_S = Dollars per pound of structure, first unit, dollars per pound

a_G = First unit guidance cost, dollars

W_S = Structural weight, pound

k_5 = Constant to account for spares, assume 8%

j = Refers to type of engine

i = Refers to stage

a = First unit vehicle production cost, dollars

W_E = Total empty weight, pounds

c_E = Dollars per pound of total empty weight, dollars per pound, assume 265

0.848 = 90% learning curve exponent

DIRECT COSTS

PRODUCTION, C_V

$$C_V = f(W, E, N, \alpha) \quad (1)$$

W = f(VEHICLE CLASS, STAGE, PROPELLANT MIX, PERFORMANCE-DESIGN EQUATIONS, PRODUCTION TECHNIQUE, MATERIAL)

$$C_V = k_5 a N^{1+\alpha} \approx 1.08\,(265\,W_E)\,N^{.848} \quad (2)$$

$$C_V = k_5 \left[c_S W_S N^{1+\alpha} + a_G N^{1+\gamma} + \sum_i \left\{ a_e (n_e N)^{1+\beta} \right\}_i \right]_i \quad (3)$$

Chart 6

Chart 7 lists the equations that were developed for the propellant, transportation and personnel cost categories.

C_P Cost Category - Equation 1 states the functional relationship for C_P and equation 2 notes the results after substituting values for the C-1 launch vehicle. The symbols are:

C_P (C-1) = total propellant costs, 10^6\$

c_p = unit cost of propellant, dollars per pound

u = utilization factor to account for boil off and transfer losses

W_P = weight of propellant (oxidizer and fuel) pounds

N = number of operational vehicles launched

C_T Cost Category - Equation 3 states C_T (barge transportation of stages from manufacturing facility to launch base) as the sum of the transporter investment and operating costs. Equations 4 and 5 result from substituting appropriate values for the C-1 system. The symbols are:

C_T (C-1) = total transportation costs, 10^6\$

a_T = unit transporter cost, 10^6\$ per transporter

N_T = number of transporters required for the program

b_T = annual operating cost of transporters, 10^6\$ per transporter per year

Y = number of program operational years

L = launch rate per year for the system

DIRECT COSTS

PROPELLANT, C_P

$$C_P = f(c_p, u, W_p, N) \quad (1)$$

$$C_P\,(C\text{-}1) = .04N \quad (2)$$

TRANSPORTATION, C_T

$$C_T = f(A_T, B_T) = (a_T N_T + b_T N_T Y) \quad (3)$$

$$C_T\,(C\text{-}1) = 7.6\,N_T \quad (4)$$

$$N_T\,(C\text{-}1) = .155L \quad (5)$$

PERSONNEL, C_L

$$C_L = k_6 M_L Y \quad (6)$$

$$k_6 = .022 \times 10^6 \quad (7)$$

$$M_L = f(B, P, S, E, L) \quad (8)$$

$$M_L\,(C\text{-}1) = 115B + 43L \quad (9)$$

$$C_L\,(C\text{-}1) = 2.53\,BY + 1.0N \quad (10)$$

Chart 7

C_L Cost Category - Equation 6 states that launch base personnel costs are the product of a yearly personnel cost factor, the number of personnel required to launch the vehicle and the program years. Equation 7 includes factors for supervision, fatique, vacations, etc. Equation 8 illustrates the many factors considered to determine personnel requirements. Equation 9 results from substituting C-1 values in equation 8 after it was structured as a linear sum of the variables. Equation 10 results from substituting equations 7 and 9 in equation 6. The symbols are:

C_L (C-1) = total launch base personnel cost, 10^6\$

k_6 = factor - 10^6\$ per personnel per year

M_L = total number of launch base direct personnel

B = number of launch base complexes

P = pads per complex

S = number of stages per vehicle

E = number of engines per vehicle

Chart 8 lists the cost categories, their amounts and percentages of the total amount. These amounts were calculated by substituting the input data (120 launches, 10 years, 1 complex, 2 pads, etc.) for the C-1 launch vehicle in the cost category models.

It is of interest to note that the development and production cost categories account for about 90% of the total cost. This is usually the case for launch quantities above 50. It also indicates the areas for greatest cost estimating effort.

TOTAL COST ANALYSIS, 120 LAUNCHES IN 10 YEARS, 1 COMPLEX

CATEGORY	AMOUNT (10^6 \$)	%
INDIRECT COSTS	(1,621)	(45)
RANGE AND GENERAL OVERHEAD	110	3
GROUND SUPPORT EQUIPMENT	34	1
LAUNCH FACILITY	88	3
DEVELOPMENT	1,389	38
DIRECT COSTS	(1,984)	(55)
PRODUCTION	1,832	51
PROPELLANT	5	-
TRANSPORTATION	8	-
PERSONNEL	139	4
TOTAL	3,605	100

Chart 8

MISSION ANALYSIS

$$S = 19{,}000\, N_1 + 220{,}000\, N_2 \qquad (1)$$

$$B = 33\, N_1^{.848} + 80\, N_2^{.848} + 3\, N_1 + 6\, N_2 \qquad (2)$$

$$E = \frac{B}{S} \qquad (3)$$

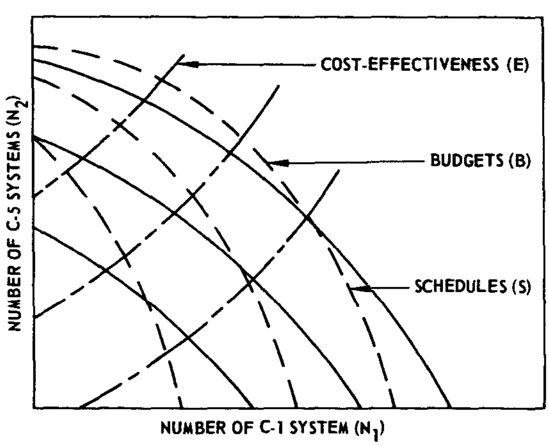


Chart 9

Chart 9 illustrates how integrated mission analysis (schedule, budget and cost-effectiveness) can be performed for two launch vehicle systems using some of the cost category models.

Equation 1 states the schedule or payload requirements for the C-1 and C-5 launch vehicles in an orbital mission. Equation 2 states the budget constraint for the two systems. Although the equation denotes budget in terms of direct operating cost, total cost or production cost equations may be used instead. Equation 3 is the cost-effectiveness ratio, dollars per pound.

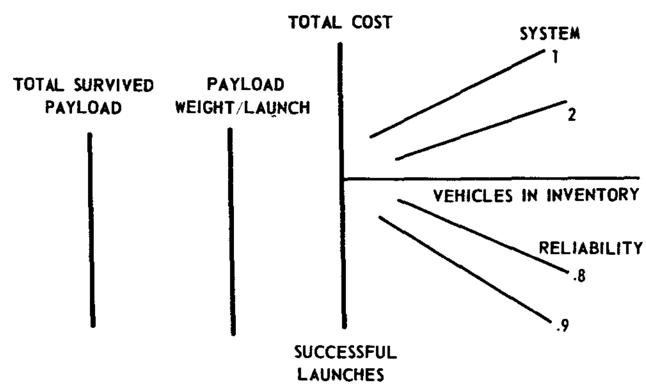
The advantage of the chart is that it quickly shows the interrelationship of the five variables. In addition constraint (for example, 20 C-1 launch vehicles must be programmed) lines can be added.

Chart 10 illustrates two types of tradeoff studies that can be performed using the cost category models.

Reliability Tradeoff - This chart can be used to examine the effects of reliability upon total cost. Assume the mission (total survived payload in orbit or escape) is fixed. The number of successful launches required by each system can then be determined. Applicable reliability (ratio of successes to trials) factors may then be selected and the total cost determined.

TRADE-OFF STUDIES

RELIABILITY



ISO-COST TECHNIQUE

LET TC (SYSTEM 1) = TC (SYSTEM 2)

AND R_1 = CONSTANT

OBJECTIVE: TRADE DEVELOPMENT AND PRODUCTION COSTS FOR IMPROVED (HIGHER) RELIABILITY OF SYSTEM 2.

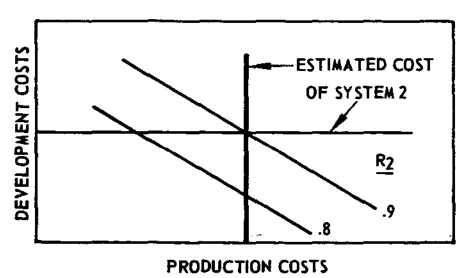


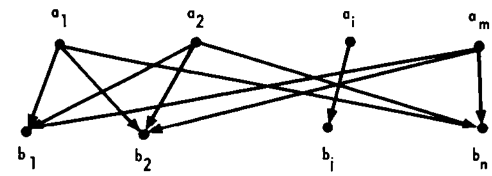
Chart 10

Iso-Cost Technique - This chart can be used to trade dollars from one category to another while maintaining total cost fixed. Assume that two systems must have equal cost but have different reliabilities. The chart illustrates how cost can be allocated to either or both categories for various reliability levels of the second system.

Chart 11 illustrates how the transportation or "traveling salesman" problem is similar in concept and solution to an interplanetary transportation scheme. The technique is particularly amenable to computer solutions. The result would be to recommend specific launch vehicles (and systems) that should be developed at specific time periods to obtain overall minimum cost-effectiveness for a given schedule i. e. mission.

Consider m earth launched systems or launch vehicles which supply n locations (orbital, lunar and planetary missions). The systems produce schedules at levels $a_1, a_2, \ldots a_m$ and the demands at the locations for these schedules are $b_1, b_2, \ldots b_n$. The unit cost (dollars per pound of payload) of transporting system i to location j is denoted by c_{ij}; and the amount of pounds of payload is denoted by X_{ij}. The transportation pattern which minimizes total cost can be determined by solving the equations.

MATHEMATICAL PROGRAMMING



$$\sum_{j=1}^{n} X_{ij} = a_i \qquad i = 1, 2, \ldots m$$

$$\sum_{i=1}^{m} X_{ij} = b_j \qquad j = 1, 2, \ldots n$$

$$\sum_{i=1}^{m} \sum_{j=1}^{n} c_{ij} X_{ij} = \min$$

Chart 11

SELF-SEALING SPACECRAFT STRUCTURES IN THE METEOROID ENVIRONMENT

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ABSTRACT

The likelihood of meteoroid encounters poses various problems to the spacecraft designer, one of them being the loss of fluids vital to mission completion. It is demonstrated that, for the more ambitious projects, a structure which provides penetration resistance only can impose weight penalties of increasing seriousness as mission times rise. Structural composites utilizing elastomeric materials reliably demonstrate in the laboratory a capability of self-sealing compatible with system requirements. In an extensive program at Northrop, and in support of a NASA-sponsored study, a variety of structural configurations successfully sealed following penetrations at speeds from 8,000 to 20,000 feet per second. The sealants are evaluated by excitation on an electrodynamic shaker, and it is shown that the more successful sealants exhibit high damping and energy dissipation when compared with the less successful materials. In a unique application of the theory of viscoelasticity, models of material response are proposed, and a qualitative description of successful operation is advanced. Material property data plus the results of hypervelocity perforations of elastomers and complete panels are presented in support of the derived conclusions.

A survey of suggested self-sealing configurations is presented. Recommendations for the practical applications of these systems are given.

1.0 INTRODUCTION

With the advent of space excursions of longer duration, the meteoroid environment must receive careful attention. No experimental data for significantly lengthy durations are as yet available to confirm the existing predictions. It is important, therefore, to assess the need for self-sealing applications and to survey the utility of these systems. As armor plate and bumper approaches have received a great amount of attention, the added or alternative benefits of self-sealing have remained hitherto unexplored. While one may recognize that self-repair concepts are not a panacea in that they do not provide for defeating penetration, it is expedient that their utility be studied. Moreover, a need to explore various sealing concepts is mandatory in view of the variety of anticipated environmental considerations and possible areas of application.

2.0 ANALYSIS OF THE REQUIREMENT FOR SELF-SEALING

2.1 The Meteoroid Environment

The presence of extra-terrestial material has been observed by man for many centuries in the manifestation of meteorites, and the threat to space travel has been discussed at great length. However, it has only been in the past decade that the hazard has been subject to quantitative scrutiny. The science of meteoroid observation has many uncertainties, and most complete evaluation will undoubtedly be gained by long-term detection satellites to realistically assess the degree of danger.

In the meantime, statistical analyses of various earth-bound observational approaches will proceed. Photographic methods seem applicable to meteor magnitudes of 6. Radio techniques extend the detection range to about magnitude 13. Some data further into the micrometeoric range have been collected by orbiting vehicles exposing relatively small areas o for limited durations.

An accurate appraisal of the environment hinges on the relation between an observed luminosity and the particle mass as it ablates in its fiery earth atmosphere entry. Some of the most reliable information of this nature was obtained in an experiment coded Trailblazer I, developed by NASA, and reported by Whipple (Ref. 1). Correlation with frequency of occurrence and meteor mass is then presented on flux versus mass charts, where the mass is presented as a threshold value. It is interesting to note the historical changes in the flux information as tabulated by various investigators. In Figure 1, it can be seen that the most current results present a somewhat more optimistic picture. The data of Whipple, presented as the 1963A line, indicates a decrease in frequency over his earlier (1957) estimate beginning from magnitude 23 and showing better than one decade decrease for magnitude 5 meteors. His conclusion tends toward the 1956 results of Watson[2]. Indeed, an alteration in the mass-magnitude relation reflects this change.

Meteoroid size can only be determined from mass and density relations, both of which are speculative. However, there is general agreement among investigators that the prime hazard is due to the micro-sized debris, or micrometeorites.

The flux data can be converted into an index of the penetration hazard. This is done by converting the meteor mass to an equivalent wall thickness penetrated by it, assuming values for the meteor parameters of velocity and density, and a pertinent penetration equation. The reciprocal flux plotted versus the threshold wall thickness defines an average time to penetration of a unit area for any selected group material. The assumed values of the attendant parameters are:

(a) Average meteoroid density, $\rho = 0.44$ gm/cm^3

(b) Average meteoroid velocity, $V = 22$ km/sec

(c) Herrmann - Jones thick target penetration formula (Ref. 3):

$$P = 0.6\left(\frac{6}{\pi}\right)^{\frac{1}{3}} \frac{\rho^{1/3}}{\rho_t^{2/3}} \ln\left[1 + \left(\frac{\rho}{\rho_t}\right)^{2/3} \frac{\rho_t V^2}{4H}\right] m^{1/3}$$

with p = thick-target depth of penetration
ρ_t = target density
H = target Brinnel hardness
m = meteoroid mass

(d) Thick target penetration depth = 2/3 thin sheet penetration depth

The relatively low meteoroid density is proposed in Reference 1, and is consistent with the generally accepted notion of the cometary origin of over 90 percent of the debris (Ref. 4). Whipple also accepts the listed penetration relation as applicable for low density particles.

The average-time-to-penetration data is shown in Figure 2 for the two sets of Whipple data, viz., that of 1957 and 1963 estimates. It can be seen that for a 0.1 cm skin thickness (about 0.040-inch) the penetration time has been increased by a factor of over 3,000. This result accrued with an intended pessimism of one order of magnitude in the 1957 data. The changes in the data over six years of effort arise from the alterations in the flux information and the penetration relation and criterion.

Future changes in the penetration equations are not expected to be drastic, although laboratory projection techniques for realistic speeds and particle masses still require development. Uncertainties in the density estimates could reduce the average time to penetration by one order of magnitude below the "best estimate." According to Figure 3, a vehicle of 1,000 ft^2 exposed surface (about 90 $meters^2$) with an aluminum skin thickness of 40 mils designed to a realistic zero penetration probability could be penetrated in less than one month. The problem increases linearly with larger vehicle size.

The need for meteoroid protection for extended missions is obvious, as is the existence of a variety of uncertainties in its implementation for the shorter ones. Long-time orbiting detectors will provide better data for a realistic assessment of probabilistic vehicle design, but the time to implement these efforts must not inhibit current or future development. The biomedical results of a penetration-induced decompression have been detailed elsewhere, and will not be discussed here. However, it has been reliably stated that incapacitation of an exposed biospecimen could result in seconds.[5] Self-sealing systems, with nearly instantaneous response and no extraneous crew monitoring, are logically suggested. Hence, for unmanned compartments, for inaccessible areas of the vehicle, and even as a complement to armor plate, self-sealing structures "design around" the environmental problem and present a greater degree of crew comfort and safety.

2.2 System Comparative Evaluation

One method of observing the utility of self-sealing is to compare the expected weights of various systems for different mission times. Such an analysis has been made for the following systems employing aluminum skins:

(a) An air replenishment system with provisions for maintaining a 14.7 psi air atmosphere with no provisions for repair.

(b) An armor plate structure with a zero penetration probability P(o), of 0.99.

(c) A self-sealing structural composite.

The Whipple 1963A "best estimate" flux data was assumed. The air replenishment system is assumed to have the capacity to replace air at a rate necessary to maintain the specified pressure to offset the predicted loss from hole production flux, with an equipment weight penalty of 40 percent of the lost mass[6]. Ideal nozzle flow was assumed in calculating the mass loss. The armor plate concept is applied to a vehicle of 1,000 ft^2 of exposed surface area. The penetration relationship selected for ease of computation is that of Rodriquez[7], which is a generalized form of that of Kornhauser[8].

The self-sealing systems employ an elastomeric sealant confined in a honeycomb core sandwich, and will be described in detail later. Successful laboratory specimens of this type have been fabricated with a unit weight of 1.7 lb/ft^2 (not including the metallic face). This current optimum has been used for the weight analysis.

An average meteoroid velocity and density of 22 km/sec and 0.44 gm/cm^3, respectively, taken from Whipple, are assumed. The complete analysis is shown in Figure 3. The air replenishment concept becomes noncompetitive at mission times beyond one month. For times greater than two weeks, the 0.100-inch system is actually lighter than its companion 0.020-inch skin system since only the less frequent, more massive, perforations are experienced. A comparison of armor plate and self-sealing yields interesting results. For the range of sealant constructions between 0.020-inch and 0.100-inch aluminum face thicknesses, a weight advantage over armor plate is predicted for mission times beyond two days to about two weeks. Moreover, the 0.020-inch air replenishment concept would still be lighter up to about two weeks.

It becomes apparent that the weight tradeoff point for self-sealing configuration occurs for mission times beyond two weeks. Hence, for near-earth performance, vehicles could utilize these structures to a weight advantage assuming that the particle flux rate is not significantly altered. As discussed earlier, the flux data are considered to be reliable to the extent that any future revisions should not produce drastic changes. Some further optimizations in the self-sealing geometries can be expected, but their effect should not be drastic either. Hence, the weight picture of armor versus self-sealing as presented here is considered quite realistic.

The "bumper" concept has not been analyzed here because of the scarcity of data at this time. It can be reasonably expected that considerable weight saving can be effected if one were to apply the laboratory optimums.[9] Indeed, factors of 50 percent and higher have been reported. However, it is interesting to note that these optimums occur at geometries where a considerable percentage of the total penetrated depth (bumper plus witness plate) is in the rear wall. This situation may be intolerable for cryogenic tankage where the residual energy of penetration (past the bumper) may be severe enough to initiate shock effects in the confined fluid. Applying this further to manned capsules, one need be concerned about structural wall crack formation with or without a bumper, since a penetration-resistant structure may not necessarily be leak-proof. However, nonoptimum bumpers may be employed. For this reason, a factor of 30 percent weight savings may be a more realistic assessment of bumper efficiency. With this factor applied directly to the armor plate curve of Figure 3, the weight tradeoff for bumper versus self-sealing occurs between ten days and three months for self-sealing configurations using aluminum skins of 0.020-inch to 0.100-inch thickness for the structural requirement.

3.0 SEALANT MATERIAL REQUIREMENTS

Depending on the area of application, self-sealing materials must satisfy specific requirements. Many of these demands are obvious and are listed as follows:

- Minimum permanent deformation following hypervelocity perforation coupled with good recovery characteristics.
- Sufficient strength and crack propagation resistance under extremely rapid loading to localize damage and material removal.
- High internal loss characteristics to facilitate energy dissipation.
- Resistance to degradation by the induced environment.
- Chemical properties compatible with fire resistance, permeability, and minimal odor and toxicity of the degradation products.

Bjork[10] describes the process of hypervelocity penetration as a fluid phenomenon, neglecting the inherent strength of the material, and considering it equivalent to the propagation of a pressure wave in the megabar range. Eichelberger and Gehring[11] have further postulated that the hydrodynamic analogy yields accurate predictions for the initial stages of crater formation, and that descriptions of the final stages must account for the mechanical after-flow of the impacted material. Charters[12], in describing crater formation, refers to a rebound or recovery of 15 percent of the maximum formed crater volume in the final stage of penetration.

These descriptions were made from observations of impacts on thick metallic targets. However, many aspects of the penetration mechanism apply equally well to the elastomers. It is this inherent ability for recovery from as much as 100 percent strain which renders the elastomers applicable to self-sealant usage. Figures 4 and 5 show the entry and exit face, respectively, of a penetrated panel. The remarkable degree of recovery is evident in both photos of this penetration which occurred at approximately 7,000 feet per second. The tendency of elastomers to exhibit viscous responses further serves to enhance their application in view of the attendant energy dissipation to be expected. The notion of a sealing mechanism utilizing a flow process is a feasible (as well as aesthetic) possibility and will be described later.

These salient requirements, along with compatibility with the structural function of the shell, made the elastomers a logical class of materials for this purpose. The use of the elastomers for static seals is well known and prevalent. The further possibility of tailoring the material to specific requirements is available to a high degree in view of the presence of this effort in polymer research.

4.0 SURVEY OF SELF-SEALING CONCEPTS

All of the concepts discussed here were initially subjected to perforations at 7,000 to 8,000 fps. Lead, steel, and glass projectiles of 1/8-inch diameter were used. The particle accelerator range was evacuated to approximately 200 microns for each shot. The exit side of the specimens were exposed to ambient conditions, so that sealing was being observed across essentially a 14.7 psi pressure differential. Immediately after firing, sealing was qualitatively checked by ear and with the assistance of a stethoscope placed onto the exit side near the perforation. Following this, leakage rates from a known volume container over a spectrum of pressure differentials were measured using laboratory-type flowmeters. Further testing of the more promising configurations was then conducted at hypervelocity facilities at velocities in excess of 20,000 fps.

4.1 The Honeycomb-Core Sealant Concept

The basic configuration is shown in Figure 6. It consists of a metallic face sheet and a reinforced neoprene backstrip confining a phenolic fiberglass honeycomb core. The core is filled with the sealant and proper surface treatments are used to insure good sealant-to-core and -face bonding.

A rubber backup strip, with a good bond to the core-sealant, seems to be a necessity. Early experiments which included shots into panels with metallic rear faces exhibited severe damage on the pellet exit side (see Figure 5). In most cases, sealant material local to this area was also observed to be severely damaged and hole closure was not achieved. When neoprene was substituted, immediate improvement was observed in the form of decreased deformation of the backup sheet and adjacent sealant. Since the area bounded by the backup strip represents a free surface with expected shock wave reinforcement due to reflection, it is apparent that a dissipative material is required. The resiliency of the neoprene further permits recovery from radial deformations due to the passage of the particle, and enhances hole closure.

The function of the honeycomb core is to contain the sealant and provide damage confinement through the addition of bonding surfaces in the sealant volume. However, aluminum honeycomb geometries exhibited massive damage. Radial collapse of the cell walls as much as twelve times the projectile diameter, with attendant sealant damage, was observed. Again, the need for a more dissipative material was apparent. The use of phenolic fiberglass core, with attention to maintaining good core-to-sealant bonding, was seen to greatly alleviate the damage. Subsequent firings resulted in little or no core damage and localization was achieved.

Sealant materials used in the initial work were commercially available silicones, polysulfides, and polyurethanes. The silicones were observed to exhibit minute radial cracking local to the particle path and did not seal with the reliability of the latter materials. The polysulfides and polyurethanes used successfully can best be described as less viscous and, hence, more amenable to creep or flow characteristics which are obvious from static handling of the materials. It was these observations which suggested a method of dynamic material testing and evaluation, to be described later in this paper.

A highly successful configuration consists of a 0.020-inch aluminum face, 3/16-inch thick core, with 3/16 or 1/4-inch cells filled with a polysulfide elastomer, and a 1/16-inch fiber-reinforced neoprene backup sheet. The self-sealing weight contribution is 1.7 pounds per square foot, and the over-all weight is approximately 2.0 pounds per square foot. Best results are observed when the bonding agents used are the same material as that of the sealant. Since all tests were conducted with 1/8-inch diameter projectiles, it is conceivable that some further weight reduction may be achieved with more realistic (vis., smaller) projectile sizes.

To further observe isolated sealant response to dynamic puncturing, various configurations were penetrated. These consisted of shots into 1/4-inch and 1/2-inch discs of sealant material in variously supported geometries. Shots were made into various permutations with and without face sheets and backup sheets. Where possible, a circular grid was molded onto the surface of the sealant disc, and both pre- and post shot spacings of the grid lines were made. In some cases, residual radial surface strains adjacent to the entry crater were observed to be tensile. For the majority of cases, residual strains were found to be undetectable or below 3 percent, indicating that the specimen surface was relatively undamaged. Upon dissection of the specimens, inspection of the damage at the particle path revealed local tearing in a direction opposite to the direction of the projectile. Delamination in the radial direction extending out from the particle path was indicative of the presence of localized shearing through the specimen thickness. The deposition of material toward the hole in both instances just discussed is considered desirable, in that both surface and subsurface inward displacement of the sealant minimize hole size and, hence, leakage rates.

Figure 7 shows a typical surface damage in what is considered a brittle silicone rubber. Figure 8 shows the identical situation for a highly successful polysulfide sealant. The cross-sectional view of a perforation in a 1/4-inch silicone rubber is shown in Figure 9. The displacement of the material back along the pellet direction cannot be explained in terms of wave theory. One simple explanation is that the short-duration heat pulse upon puncturing melts the surrounding material, and the subsequent air flow across the slab initially forces material back along the particle path. In any event, this displacement of the material back into the hole assists the sealing process.

An example of successful self-sealing following a 20,000 fps penetration in the mechanical configuration is shown in Figure 10. The backup sheet has been pulled back to show the localized interior damage. In every instance shown, the sealant material is seen to be deposited in the entrant hole, and the exit damage is quite remarkably localized.

Residual leakage rates for the successful panels were measured by observing the pressure drop from a known volume container capped with the penetrated specimens. Leakage rates as low as 2.0 lb/yr to zero have been observed. Average leakage rates have been found to be on the order of 1 to 2 lbs/day, which is an improvement of better than 99 percent over the observed rates through the penetrated aluminum face sheet hole. In many instances, the specimens showed a detectable leakage rate across a 14.7 psi pressure differential, but exhibited almost complete sealing at 4 to 5 psi internal pressures.

4.2 Elastomeric Sphere Concept

The concept just described relies on both the energy dissipation and the recovery of the sealant material for successful operation. For extremely localized damage, the recovery of the material in the domain of infinitesimal deformations is utilized to effect hole closure. It is conceivable that under certain environmental conditions and with massive face sheet and sealant damage or tear-out, macro-motion of the sealant into the perforated zone will be required. For this reason, the elastomeric sphere concept was investigated. Here, the conventional sealant is replaced by discreet elastomeric spheres to a predetermined packing density (see Figure 11).

Upon penetration, the pressure differential between the inside cabin and the vacuum of space forces the balls toward the entrant hole and effects the sealing. Ball size, packing density, and material will control the mobility of the spheres and the sealability. This concept exhibited residual leakage rates that are comparable with the mechanical system described earlier, and is currently under further study.

4.3 Other Systems

A number of alternate concepts developed to fit a variety of requirements have been successfully tested. They may be classed as mechanical or chemical depending on the mode of activating the sealing process.

One highly successful approach is to prestress the sealant in compression. This enhances the sealability of materials which have desirable thermal or vacuum properties, but which exhibit brittle-type fractures or low shear strength characteristics under dynamic conditions when simply confined. The relatively low modulus characteristics of the elastomers facilitate the buildup of moderate internal stresses with appreciable strain recovery in the penetration hole. The prestressing can be accomplished in a variety of simple and ingenious ways.

One prestressing technique uses an elastomer and foaming reagent which causes an unconfined volume expansion of 200 percent to 300 percent upon curing. The structural panel is filled with the uncured sealant compound to a volume fraction which will produce a desired prestress level. Face plate bonding is accomplished with a material whose softening point is above the cure temperature of the sealant. The sealant is then cured at the required heat. This technique has shown successful sealing characteristics with a high degree of reliability.

Chemical concepts rely on the dynamic action of the penetrating particle to initiate a reaction which closes the hole. In one concept, shown in Figure 12, an uncured polymer is separated from the catalyst by a thin, nonreactive membrane. Upon complete perforation, the pressure differential across the panel forces a mixture of polymer and catalyst through the hole. Very fast curing mixtures have been used with complete and repeatable sealing action. In another method, small bags of catalyst are interspersed in the sealant void (see Figure 13). This seems to localize the curing action to the area of the penetration. Bag size is an important factor in this method. For very uniform distribution of the catalyst in the uncured elastomer, microencapsulation techniques can be adopted.

In all of the chemical concepts where the sealant materials are initially fluid, careful attention must be given to the rheological or flow properties of the polymer. The viscosity of the material must be such as to permit an initial gradual flow through the hole without excessive loss. Cure rate must obviously be rapid enough to "set up" the material in the hole. Environmental stability must be carefully considered against mission time, as degradation can severely alter flow rates and, hence, sealability.

5.0 SEALANT MATERIAL EVALUATION

Observations of a variety of sealing results following perforation indicate the importance of the proper selection of the sealant material. Indeed, material differences are apparent from static handling of the specimen material. The highly elastic or rubbery materials tested exhibit a tendency toward crack formation and excessive volume removal or tear-out. When a less viscous material was substituted, immediate improvement was noted. This improvement is characterized by a remarkable degree of damage localization in the form of a minute particle path with little or no surface damage. Unfortunately, such terms as "less viscous" or "more elastic" are purely relative and do not assist in a quantitative description of material response. One method of assessing the use of a solid elastomer may be derived from a consideration of the theoretical response of viscoelastic materials.

5.1 Theoretical Material Response

Ideally, viscoelastic materials are assumed to consist of elements whose over-all behavior can be described as a combination of viscous and elastic responses. Ideal springs and Newtonian viscosity characterize linear behavior and permit the use of the principle of superposition.

Some classical bodies are shown in Figure 14 along with their stress-strain equations. Most real elastomers depart from this ideal behavior, exhibiting a spectrum of retardation times which makes difficult a complete description of response over a broad range of loading rates or frequencies. However, responses over a selected frequency bandwidth, specified by design criteria, can be accomplished. A response fit over a decade of frequency is considered adequate.

The most general characterization of a material with a time response in shear can be written as (Ref. 13):

$$P\{S_{ij}\} = 2Q\{E_{ij}\}$$

where S_{ij} and E_{ij} are the deviatoric stress and strain tensors, respectively, and P and Q are differential operators whose coefficients define material properties. A Laplace or Fourier mathematical transform may be performed which alters the time-dependency to one of frequency. The notion of modulus is retained in the form

$$\frac{\bar{Q}}{\bar{P}} = E' + iE''$$

denoting the complex modulus, with barred quantities denoting transformed properties. Since the value is complex, it denotes in a vector representation in-phase and out-of-phase components. The in-phase component is referred to as the elastic or storage modulus; the out-of-phase component is called the loss modulus. For low to moderate damping ($E''/E' < 0.2$), it can be shown that the ratio of E''/E', or loss factor, is a measure of the percent energy dissipation of the material (Ref. 13).

Sinusoidal excitation may be performed by a variety of means which include the vibrating reed or rocking beam. The intent at this point is to simulate the loading rate of hypervelocity perforation. This implies that excitation be performed in the kilocycle range. However, power limitations severely hamper the generation of detectable mechanical displacements at those extremely high frequencies. For this reason, correlation with theoretical models of material response was attempted at moderate frequencies from 100 to 2,000 cps in an attempt to describe successful self-sealing operation of solid elastomers at impact speeds of 5,000 to 7,000 feet per second.

5.2 Results of Dynamic Excitation

The oscillatory test method employs a three-inch diameter, 1/4-inch thick elastomer specimen, with a mass bonded to its upper surface. The elastomer-mass combination is bonded to a rigid mounting plate, which is in turn attached to the head of a conventional electrodynamic shaker. Bonding agents used were the same material as that of the specimen to insure homogeneity. The test geometry is shown in Figure 15. Input and output accelerometers, along with the associated phase angle were recorded for fixed output "g" levels to maintain constant peak stress on the sealant over the frequency range tested.

Considerable in-plane restraint was evident on the specimen, resulting in an apparent stiffening of the material. However, it was felt that this geometry most nearly simulated the confined sealant in a panel configuration. Two silicone rubbers, which exhibited poor to inconsistent self-sealing, and a polysulfide and a polyurethane elastomer with superior performance, were the four materials selected for initial evaluation.

The results of the oscillatory tests are shown in Figures 16 and 17. The in-phase or elastic modulus results indicated that the two successful self-sealants exhibited higher in-phase or recovery moduli. From Figure 17, it is seen that the energy dissipation characteristics of the polysulfide and polyurethane materials were superior at the higher frequencies. It is concluded that the extremely high damping capability of the latter two materials at the higher frequencies renders them more suitable for this application.

It is to be noted that the in-phase or elastic moduli values of Figure 16 are considerably higher than Young's modulus information reported in the literature. Owing to the test geometry which imposes radial restraint on the specimen due to the bonded surfaces and the short specimen height, the results are more closely in agreement with bulk modulus. For a purely elastic, isotropic material with complete lateral restraint, normal deformations are volumetric, and the associated stiffness is the bulk modulus, K, where

$$K = \frac{E}{3(1-2\nu)}$$

and ν is Poisson's ratio. An incompressible material (ν=0.5) has an infinite bulk modulus. Furthermore, pure volumetric deformations are minimized with increasing bulk stiffness. Obviously, complete recovery ameliorates the self-sealing situation by effecting hole closure following perforation. Naturally, maximum closure, and hence, minimum leakage would result with minimum material loss or tear-out. Consequently, high internal dissipation in shear from a controlled viscosity is consistent with minimizing material loss. A sealant material rigidly confined, having a relatively high bulk modulus and internal dissipation is adjudged superior.

Another method of material evaluation used is the standard Lupke resiliency test. In this test, a specimen of the candidate material is impacted by a pendulum and the degree of successive rebounds is recorded. Although this test provides relative information only, it has been highly useful for initial screening of materials for energy absorption capability.

6.0 CONCLUSIONS AND RECOMMENDATIONS

The need for self-sealing structural systems has been described in view of the many uncertainties in the meteoroid environment assessment. Moreover, these uncertainties are expected to be present for a number of years to come. Practical self-sealing composites have been developed in the laboratory which provide a greater degree of safety with the added feature of self-repair to minimize crew responsibility. Hole closure can be accomplished either mechanically by utilizing the inherent ability of the elastomers to recover deformations, or chemically by employing a mechanism actuated by the penetrating particle. Weight considerations appear attractive for the longer missions, where the term "longer" applies by present calculations to missions of a duration beyond two to four weeks. The need is even more urgent if one considers that penetration-resistant structures may not necessarily be completely leakproof following a meteoroidal collision.

Material behavior under the space environment must be carefully assessed, and stable materials are mandatory. Both mechanical and chemical methods have been shown to be practical when careful attention is given to material bulk and compressibility properties, and that of flow. The combined attention of the structural designer and the chemist has been a necessity in the development of the current concepts.

A qualitative description of the mechanical self-sealing mechanism has been forwarded based on the measurement of aggregate material dynamic properties. A complete theoretical analysis would be beneficial, but is beset with the vagaries also associated with the thick-target hypervelocity penetration problem.

As self-sealing structures are not specifically intended to defeat the incoming meteoroid, their area of application may seem to be restricted. However, as a complementary system for either armor plate or bumper constructions, a significant increase in safety can be expected. Use of elastomers as a back-spall preventative appears possible, but requires further study.

Greater usage can be expected in areas of the vehicle having a minimum of personnel activity (e.g., passageways, airlocks, etc....). Tankage protection is another area where the dissipative capabilities of elastomers may prove feasible, with uniquely designed self-sealing systems employing tailored materials.

ACKNOWLEDGMENT

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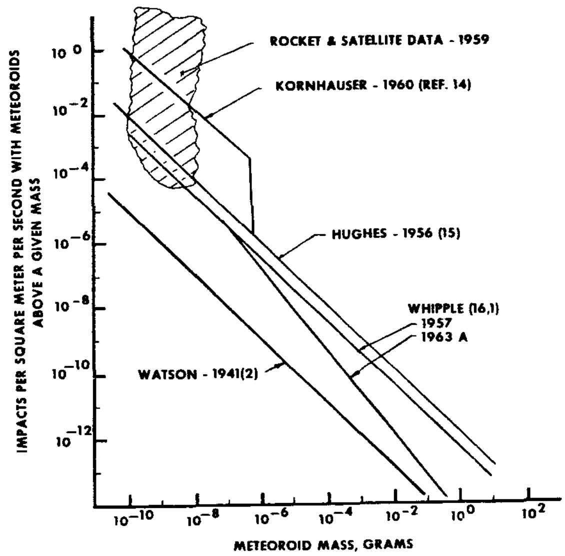

FIGURE 1 METEOROID FLUX ESTIMATES

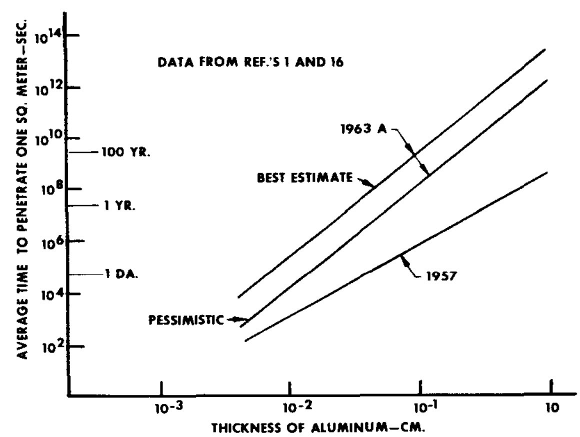

FIGURE 2 WHIPPLE ESTIMATES OF THIN-SKIN PERFORATION

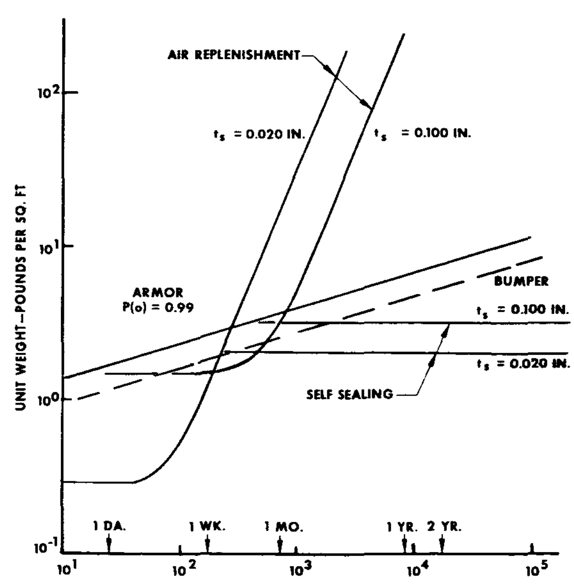

FIGURE 3 SYSTEM WEIGHT COMPARISON

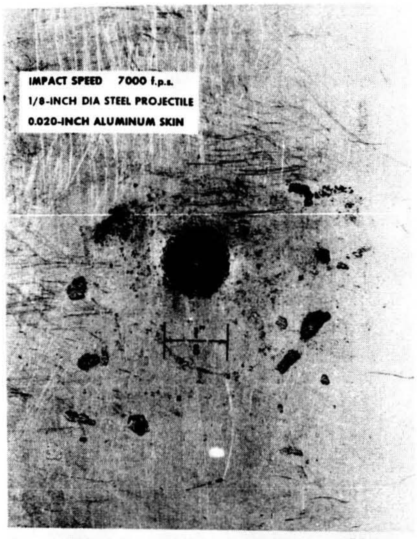

FIGURE 4 ENTRY DAMAGE TO SELF-SEALING PANEL

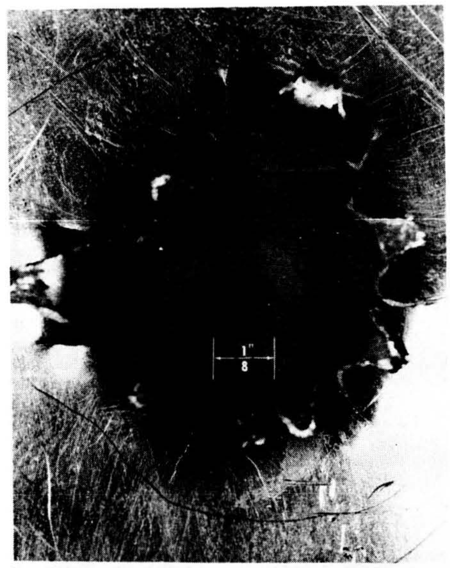

FIGURE 5 EXIT DAMAGE TO ALUMINUM REAR WALL

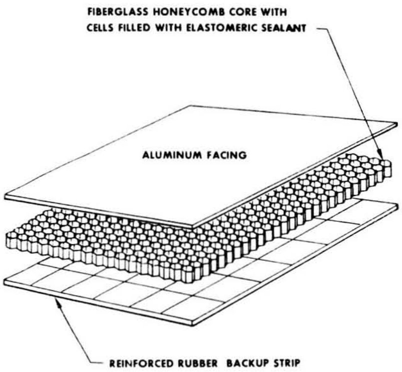

FIGURE 6 BASIC HONEYCOMB CORE
SELF-SEALING CONCEPT

FIGURE 7 ENTRY DAMAGE TO SILICONE RUBBER SPECIMEN

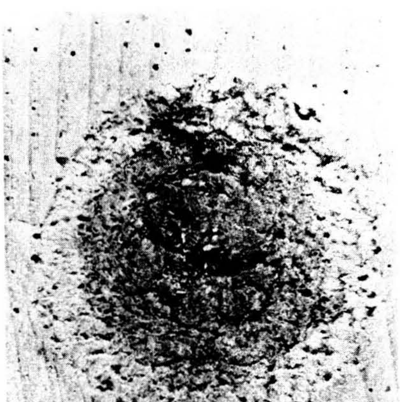

FIGURE 8 ENTRY DAMAGE TO POLYSULFIDE
ELASTOMER SPECIMEN

FIGURE 9 PERFORATION DAMAGE TO
½-INCH THICK SILICONE ELASTOMER

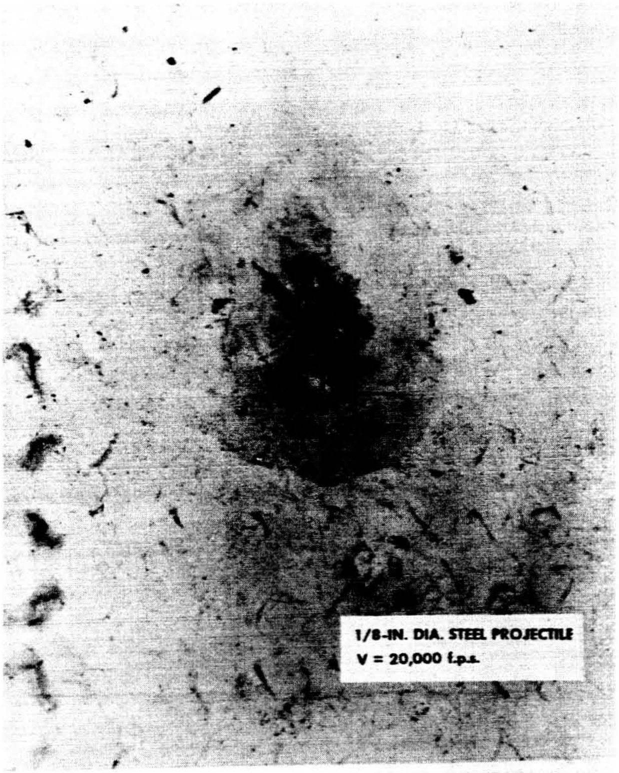

FIGURE 10 SEALANT EXIT DAMAGE IN HONEYCOMB-CORE PANEL

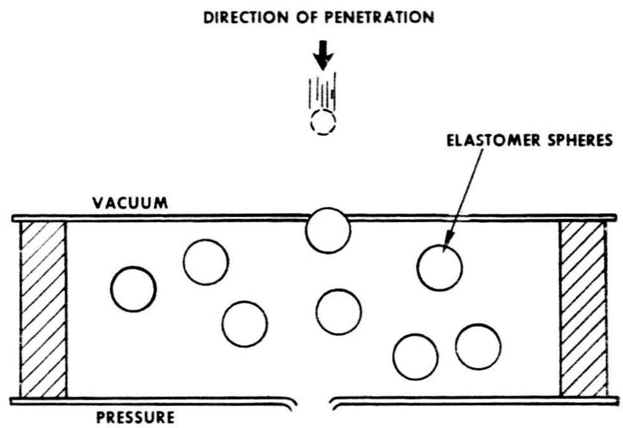

FIGURE 11 ELASTOMERIC SPHERES SELF-SEALING CONCEPT

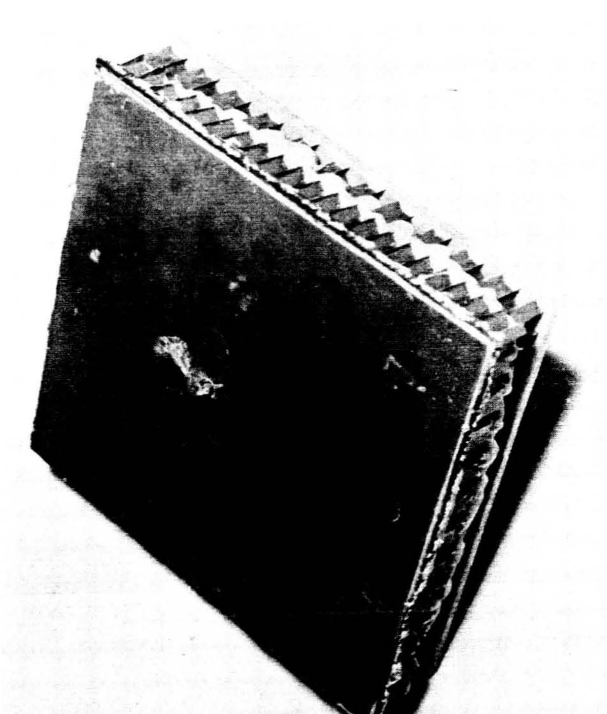

FIGURE 12 CHEMICAL MEMBRANE CONCEPT

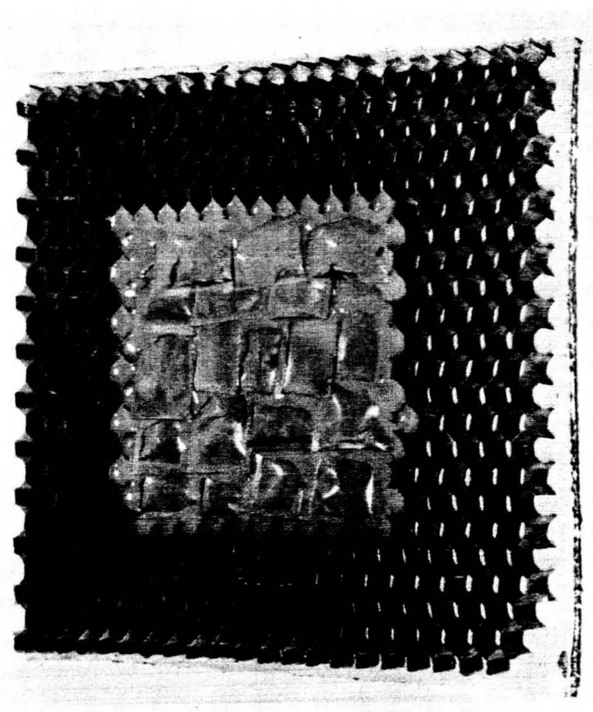

FIGURE 13 CHEMICAL BAG CONCEPT

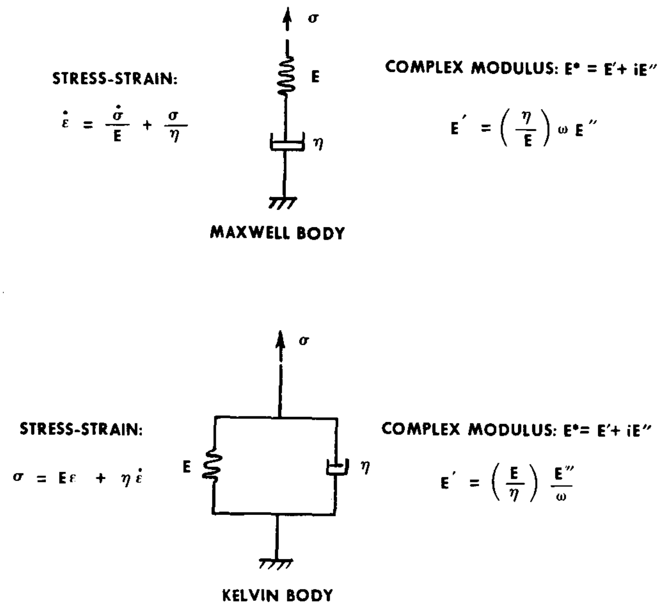

FIGURE 14 CLASSICAL VISCOELASTIC BODIES

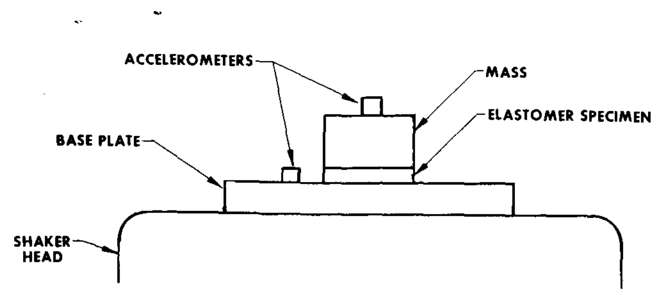

FIGURE 15 SHAKER TEST SETUP

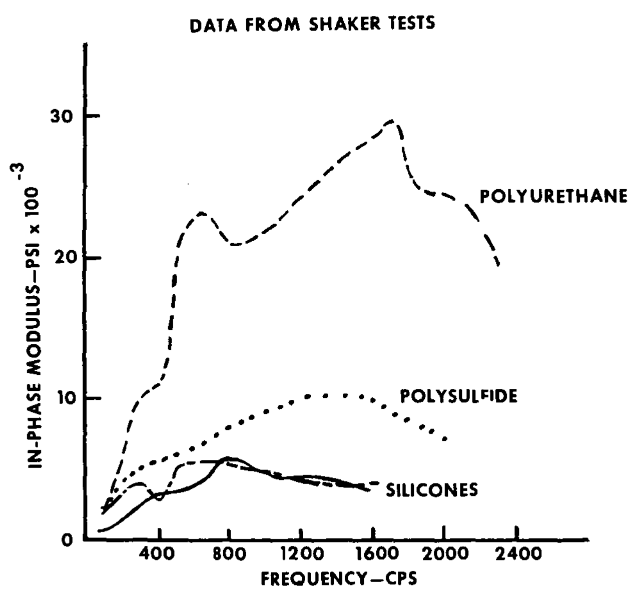

FIGURE 16 IN-PHASE (ELASTIC) MODULUS VS. FREQUENCY

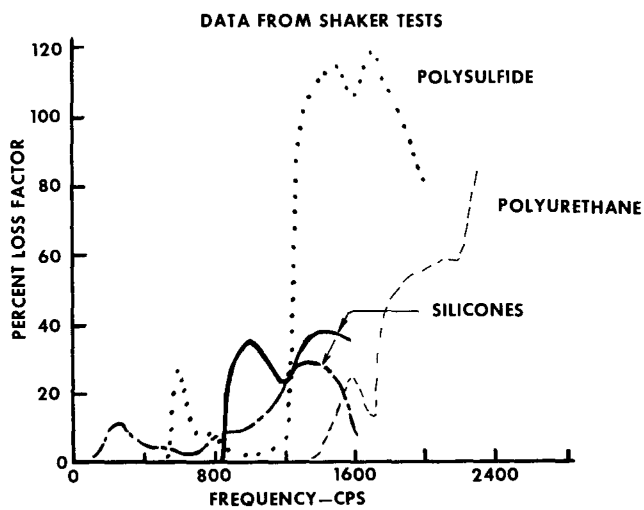

FIGURE 17 LOSS FACTOR VS. FREQUENCY

NEW FABRICATION TECHNIQUES FOR SPACE STRUCTURES*

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Introduction

The extension of man's activity beyond the Earth's atmosphere is barely into the pioneering stage. The astronauts who have penetrated space are watched apprehensively by millions of viewers on television, and every little mishap almost becomes a national emergency. As such flights become more common, public interest will wane until the day comes when the launching of an astronaut will create no more interest than the passing overhead of Echo I. However, before this situation arises, a great deal of costly work must be done in all areas of spacecraft design, not the least of which is in the field of space structures.

The demand for lightweight, reliable, compact vehicles will result in design concepts which utilize foil-gage, high-temperature materials, and in some cases, expandable structures. Some of the fabrication techniques which are required to build these structures is the subject of this paper.

Lightweight Structures

Nearly every space vehicle to date has been weight limited because of booster payload capability, and this restriction will undoubtedly continue to exist. The weight required for structure subtracts directly from the weight available for the instrumentation, equipment, and supplies required for the mission. Although booster sizes have increased, the missions have become more ambitious and the program manager continues to be faced with making the decision of "what to leave off". The result is that highly efficient, lightweight structures are very important.

To illustrate the point, Table 1 shows a typical group weight summary for an orbital carrier vehicle expressed in terms of percent reentry weight.

Table 1
WEIGHT BREAKDOWN FOR TYPICAL ORBITAL CARRIER VEHICLE

	Typical Percentage	Goal Percentage	Percent Difference
Structure	46	40	-13
Equipment	32	32	0
Propulsion	10	10	0
Payload	12	18	+50

The structure comprises almost half of the weight and any reduction in this item permits a like increase in another subsystem to improve the mission utility. In the example cited here, a 13 percent decrease in structural weight increased the payload capability by 50 percent. This weight margin could be used to increase life support and power supply fluids to extend the mission life, impulse propellants could be added for additional maneuvering capability, or additional scientific equipment could be carried. It is conceivable that with this increased capability one launch might be able to do the job of two. Both present and projected high costs of placing a pound of payload in orbit permit the structural designer to consider materials and types of construction previously considered economically unfeasible.

The need for highly efficient structures is not restricted solely to the spacecraft. Improved structures throughout the booster also improve mission capability by increasing the booster payload capability. In a multi-stage booster system the final stage structure is almost as important as the spacecraft, since the inert weight of this stage also reaches orbital velocity. On lower stages the effectivity becomes progressively lower since the inert weights of these stages do not reach as high a velocity; however, even in these, structural weight affects the payload capability. Figure 1 is a typical plot of stage structural weight fraction vs stage gross weight for booster upper stages. (Structural weight as used here includes propulsion, unusable fluids and other inert weight items as well as the actual structure.) Figure 2 shows the stage structural weight fraction vs limiting ideal velocity obtainable by the stage regardless of its size. While the problem of optimizing a booster for a particular mission will not be considered here, Figure 2 does indicate the degree of performance change with stage structural weight fraction.

Although structural improvements, both in the space vehicle and booster, must be weighed against cost, reliability, and many other factors, the potential improvement to space systems offered by highly efficient lightweight structures is worth a great deal of effort to effect such improvements. A substantial part of this effort must be expended in the area of fabrication techniques for exotic materials and structural concepts.

Fabrication Techniques

As in most technologies, the problem of fabricating structures is one of evolution. Very seldom does a sudden break through occur which enables the manufacturer to build something today which he could not do yesterday. Even the relatively new technique of "explosive forming" has been under development for several years, and in fact, is still under development.

In a field as broad as this one, it is necessary to restrict the subject matter. Consequently, we will discuss some of the work which has been done at the Lockheed-California Company on the fabrication of structures from foil gage (.003 to .020) Rene 41; the fabrication of honeycomb sandwich from foil gage refractory materials (.001-inch) work being done at the Lockheed Missiles and Space Company; and a look at some of the techniques being used and proposed for application to expandable structures (Lockheed-California Company under contract to ASD).

Foil Gage Structures. Because of the weight problem previously discussed, there is a great interest in the development of methods for utilizing foil-gage materials, particularly the high-temperature materials, in the primary structure of aerospace vehicles. This interest is due to the fact that all high-temperature metals, i.e., super alloys and refractory metals, have densities in the neighborhood of .3 lb/cu in. or more; therefore, in order to reduce the weight it is necessary to use these materials in very thin gages. Old ideas of "minimum gages" have to be discarded because we must not be bound by such restrictions. The minimum gage for any application is that one which is required to do the job, and the manufacturing people must learn how to handle, bend, join, and otherwise process the material.

As a result of this philosophy, the Lockheed-California Company began to explore the problems which might be encountered when fabricating actual structures from foil gage materials. We were aware of the work which had been done by North American on their "Spacemetal", Ryan's "Miniweight", and the Budd integrated construction. Welded honeycomb sandwich panels as developed by John J. Foster were also of considerable interest. However, these techniques had all used stainless steel and were therefore temperature limited to values below those of interest for reentry vehicles. To our knowledge no one had done any real work using foil gage Rene 41 which appeared to be the best all around material for the applications we had in mind. We were also interested in building something large enough to have real meaning. It has been our experience that small panels, while handy to carry around, do not really represent the problems encountered in fabricating an actual structure.

Consequently, a specimen representing a portion of a winged reentry vehicle, Figure 3, was chosen as typical of structures likely to be designed in the future. Figure 4 shows this specimen which is 54 in. wide, 75 in. long, and 43 in. high. It consisted of a series of trusses in two directions covered on one side by a skin-corrugation surface. The gages of material varied from .003 in. for the corrugations to .020 in. for some of the beam members. The weight of the completed specimen was 2-3/4 lb/sq ft of platform area. The addition of a second surface would have increased this weight to about 3-1/4 lb/sq ft. The manufacturing research organization at CALAC courageously set out to make the specimen. Problems encountered were as follows:

1. Handling
2. Shearing
3. Forming corrugations and struts
4. Trimming - deburring
5. Fusion welding
6. Spotwelding
7. Heat treatment

Fabrication Approach. Having had virtually no experience with the foil gage Rene 41, the problem was approached cautiously by making a few preliminary tests. As a result it was decided that, for the most part, standard aircraft machine tools and methods could be used. Subsequent experience proved that, with slight modification, these methods were indeed satisfactory. The primary machine tools used were power shear, power brake, Guerin rubber hydropress and roll former. Due to the limited quantities required, hand check and straighten operations were often employed for final finish rather than adding additional tooling. All assemblies were joined by welding. Fusion welding was used for flat skin joints, roll seam welding for corrugation to skin and beam subassemblies, and spotwelding for all other joints.

Handling. Handling of foil gage materials was the most difficult problem encountered. The relatively soft solution-treated Rene 41 was very susceptable to "dings", "oilcans", edge crushing, and all of the other ills of a fragile material. If one item in foil gage sheet metal fabrication should be stressed, it is education. Figure 5 shows a strut and bulkhead fabricated from .004 in. thick material. From raw stock inspection through transportation, shearing, forming, welding and heat treatment, employees must be educated to the fact that seemingly inconsequential pressures can produce damaged parts. One of our introductions to this fact occurred during the first shearing operation on .003-in. gage material. A small door was left open to take advantage of any breeze on a warm day. The foot wide, nine feet long sheared sheets were placed on a workbench about 30 feet from the door. A breeze picked up this material and lodged it against adjacent equipment. The ensuing wrinkles caused the material to be scrapped.

At this point it should be stressed that the final structure as aged hardened was much more resistant to damage than was the soft material. In fact, it was surprisingly sturdy, and with reasonable care would be entirely satisfactory as a space vehicle structure.

Generally, mill packaging was satisfactory. Coils were wound on overlength cores and securely taped in place to prevent edge crushing. The coils were then fastened securely in sturdy wooden boxes. In the instances where the coils were not on overlength cores, edge crushing did occur.

Good housekeeping of all work areas was a necessity to prevent part damage. Felt-topped work benches minimized damage from particle and chip indentations. Tooling used to fabricate parts must be cleaned of dust accumulated on oil films before use. Seemingly small nicks in tooling appeared as large dents in finished parts. In order to minimize damage during transportation, shipping containers were provided. Each part was separately packed to retain shape and prevent dents. Interleaving parts in craft paper inside a cardboard carton did a very acceptable job of protection.

One problem that was not overcome was the inertia of the assembly tools. In most cases the assembly fixtures and welding guns were hundreds of times heavier than the parts. Under these conditions operator "feel" is virtually nonexistent. Tools could bump into and halfway through a detail part before any tool reaction was noticed. This was especially true during final assembly operations in restricted spaces.

Shearing. Shearing of foil gages requires that cutting equipment be reserved for thin gages only. A general purpose power shear that was used on ordinary materials did a good bending job on foil gages, but did not cut. This same shear, with blades ground sharp and set at .000 in. to .001 in. clearance was satisfactory. Heavier gages may be sheared successfully on standard equipment used for stainless steels. Hold-down clamps for thin gages must be firm, cleaned of all dirt to prevent marking, and rounded on the corners. Beveled micarta clamp faces with heavy back-up springs were satisfactory. A substitute method of shearing was used for gages up to .006 in. when a special shear was not available. The thin Rene 41 was packed between sheets of .020 in. thick 7075-T6 aluminum and sheared as a sandwich.

Forming. The tooling used to form the foil gages had to be held to unusually close aircraft tolerances. In some cases, such as in roll forming, sets of tools were held to a total tolerance of .0005 in. Where complete part enclosure was not required, normal tool tolerances were held and produced satisfactory parts. In order to save Rene 41 material costs, 301 stainless steel sheet was used for tool try to check clearances and overall operation of the tools.

Springback was somewhat variable between heats. The solution treated material average 1 percent and the 30 percent cold reduced material averaged 11 percent springback. Test sections of tools were made to verify springback prior to final construction of rubber press tools. To obtain and set the bend radii required in the beads on the beam and rib members, a matching cover plate was used in the rubber press operation. Figure 6 shows the die used to form the truss members.

Interchangeable inserts allowed use of the same basic tool for all members. The hat shape for these members was obtained with a leaf brake operation. The reverse flange on the hat brim was given a final sizing operation by passing the sections through one set of forming rolls. Figure 7 is an assemblage of parts fabricated for this specimen.

The skin backup corrugations were made of .003-in. gage material. Because of the foil gage, small cross section, and relatively long parts, progressive roll forming was used. In order to save on tool development costs, the usual approach of developing rolls to form a complete panel width in one pass was not used. Figure 8 shows the set of five rolls which were developed to form one half corrugation in one pass. A powerbrake, preformed, straight bead was used to guide the rolls. The second pass was made with the sheet turned end for end and upside down so that any tendency toward misalignment would be balanced. Through this unorthodox approach corrugations were made at low cost to tolerances of plus or minus .002 in. and 1/2 deg in the full sheet width of twelve inches.

As a result of this and more recent experiences, progressive, multiple spindle roll forming was one solution to fabrication of foil gage materials to close tolerances. Even for small prototype quantities the overall cost was very competitive.

Trimming - Deburring. Economical cutoff methods for long lengths of foil gage material formed into complex shapes were limited. No true production method was developed for parts of this type. Shearing with formed dies was not feasible because of low quantity. Even with good support, friction sawing produced a ragged, torn edge. The method finally adopted was a 14-inch diameter abrasive cutoff wheel operating at 3,400 rpm in a table saw. A movable table-top plate was fixed to a guide bar in the normal cutoff groove in the saw table. This new table was jig bored for pins at the various cutoff angles required. Plastic fixtures were cast to the exterior shape of the parts to be cut and pinned to the movable table. An interior plastic plug was inserted inside the part. (Figure 9) With the whole assembly clamped firmly in place, cutoff operations produced a smooth precision joint. For the 90 degree cuts on the corrugations, each panel was cast solid, both top and bottom, with Kerr tool stone. This rigid fixturing produced a smooth controlled cut. After all foil gage cutoff operations, the residual burr was removed with a rotary file operating in a die grinder at 100,000 rpm. Heavier gages were deburred on a belt sander with good results.

Fusion Welding. The only fusion welding involved in the fabrication of the test panel was for joining the skin segments. As the .004-in. thick material was only available in 12-inch widths, five sheets were required for the full skin width. The solution heat treated material was prepared for welding with a standing flange .012 in. to .015 in. high. The butt Heliarc welds were produced without the addition of weld metal. A smooth copper backup bar was used with no groove or backup gas. The sensing head was locked out, and the arc length was fixed with a precision slide.

Spotwelding. Resistance welding was the basic joining method employed. The initial problem encountered was accessability during assembly. Figure 10 illustrates the complexity of some of the joints. A careful appraisal and detail plan of subassemblies was required to maintain accessability through final assembly.

Expulsion of metal from the spotwelds was also encountered. Although this problem was not completely eliminated, the welding parameters were controlled sufficiently to obtain satisfactory spotweld shear strengths.

The .003-inch corrugations were joined to the .004-inch skin as a subassembly. The corrugated panels were tack welded to the skin from the center of the panel outwards. Micarta fixtures maintained proper pitch spacing. The corrugations were then roll welded to the skin. Copper mandrels were inserted into the corrugations on each side of the weld to prevent crushing (Figure 11). Guide wheels were mounted on the welding machine frame ahead of and behind the welding wheel to maintain alignment. The small bead on the corrugation was used as a track for the guide wheels. All of these welds were made with a low inertia, 100 kva, Stryco rollspot seam welder. The corrugation doublers were then roll welded to the tops of the corrugations using insert mandrels. Beam and rib cap subassemblies were then joined to the doublers. Figure 12 shows the completed surface subassembly.

As might be expected for the gages being welded, the major problem was the prevention of wrinkles in the skin, especially near the edges. For the one specimen being fabricated within the time and budget allocation, this problem was not completely solved.

All beam, rib, and intercostal members were subassembled by first tackwelding the halves together and then roll welding (Figure 13). Gussets joining the vertical and diagonal members were attached with a rocker arm welder using copper insert mandrels and aluminum holding fixtures.

Final assembly welding was accomplished with 100 kva portable guns of scissor and 'C' type (Figure 14). Lightweight, compact size, and versatility are the prime requisites for portable equipment. Equipment not meeting these requirements can produce excessive mechanical damage to detail parts.

Heat Treatment. All of the sheet material was purchased from the mill in the solution-treated condition. The heavier gage materials, .010 in. and .020 in. thick, were cold reduced an additional 30 percent at the mill after solution treatment. By purchasing the material in this condition, satisfactory formability was maintained and in-house processing minimized. No re-annealing of the material was required during any of the fabrication operations.

The final aging treatment required 1400°F for 16 hours. The completed box truss section was aged as a unit after all welding and assembly. Although Rene 41 retains good strength at this temperature, adequate support was provided so that part sag would not cause permanent distortion. Care was exercised to minimize the heat sink effect of any heavy fixturing. Thermal gradients across a large part with the attendant differential in thermal expansion can cause permanent wrinkling.

Conclusions. The primary problem encountered in fabrication of foil gage structures was not the inherent formability, but rather in handling and education of employees. Parts take on the aura of a fruit stand ("Don't squeeze the tomatoes") and must be handled, packaged, and transported with special care. Shearing and forming can be accomplished with conventional techniques. Closer fits and tighter tolerances are required but only in proportion to the reduction in gage. Spotwelding requires special techniques to produce optimum strength joints without metal expulsion. Taking the broad view, an average aerospace company can fabricate lightweight foil gage structures of the type described here with minimum development and minor equipment modification. Another view of the completed specimen is shown in Figure 15.

Refractory Metal Honeycomb Structures

The previous section described structures and fabrication techniques for the superalloy materials, specifically Rene 41. These materials are good for temperatures ranging from 1500°F to 2000°F depending on the material, application, and length of exposure. Above this temperature range, and indeed beginning at about 1750°F (Figure 16), the refractory metals provide a more efficient structure material.

All these materials, however, suffer from an affinity for oxygen at these temperatures which renders them useless unless properly protected from the ambient oxygen environment or unless used in a vacuum (Figure 17). Means of protecting by coatings have been developed during the past few years which are effective on relatively thick sheets and massive parts. However, foil gage materials present a more difficult problem since the usual coatings diffuse into the surface of the material, and if the sheet is very thin, diffusion may penetrate through the complete thickness, if not initially, then perhaps with continued exposure to high temperatures during use. It is also pointed out that thick coatings on thin sheets partially defeats the purpose of using the thin sheet for achieving light weight.

The previous discussion also described structures which are "open face" types. Many studies have been made which show that for certain applications more efficient structures can be made by using "double face" sandwich structures (Ref. 1 to 3). This is particularly true where the structure is very lightly loaded. For these applications, where gages of material must be very small, the natural support to the faces afforded by a honeycomb core can be exploited to provide smooth surfaces and rigid panels.

With these characteristics in mind the Lockheed Missiles and Space Company, under contract to Aeronautical Systems Divisio", is developing a sandwich panel 18 in. by 18 in. in size consisting of .001-inch foil Molybdenum alloy faces. Due to material limitations, the present core consists of a 1/8-inch cell honeycomb of .001-inch gage Inconel 702. Recently molybdenum honeycomb core has become available and will be used for future panels.

The problems involved in producing such a panel are formidable. The very thin gage requires careful handling to prevent damage, diffusion of brazing alloys into the parent material can destroy the integrity of the joint, differential thermal expansions result in buckles and wrinkles in the face sheet, and protection of the completed panel against oxidation, are typical of some of these problems.

On the other hand the picture has a bright side. There is a good possibility that the panels can be assembled by diffusion bonding thus eliminating the braze material. The completed panels have a high stiffness-to-weight ratio, and the handling requirements during manufacture, assembly, transit, and ground handling are less demanding.

When starting a program of this sort, advantage must be taken of all the pertinent current technology. There is considerable technology in the fabrication of honeycomb sandwich from stainless steel. This technology came primarily from the B-58 and RS-70 programs. However, as applied to the present program, these have serious limitations, one of which is the temperature required for brazing. While temperatures for stainless steel brazing are below 2000°F, refractory materials require temperatures considerably above this. Both the tooling and the materials to be used in the composite itself are affected by this requirement.

Tooling Materials. The raw materials used in making tools were glasrock, micro-quartz, fiberpax, and mica-bonded aluminum. The glasrock-foamed silica (25 to 50 lb/cu ft.) has proven to be a satisfactory material for brazing temperatures of 2200°F to 2400°F, although some deterioration does occur at temperatures above 2000°F. The brazing surfaces for the initial small panels were ground to the required flatness and grooved to receive the heating elements with a hacksaw blade. The brazing tool for the full size panels is cast on a flat surface plate from a slurry. The holes for the heater wires are integrally cast by plastic tubes which are subsequently burned out. Figure 18 shows the assembled brazing fixture for one side with the glasrock tool, heater wires, and coated caul sheet.

A problem area not being investigated, but which is important as brazing or diffusion temperatures exceed 2500°F, is a suitable material for the brazing tool. The use of tungsten for cases where production quantities will justify the cost may provide the solution.

Brazing Environment. The original brazing was done in an argon atmosphere confined in a plastic bag (Figure 19). However, subsequent brazing in a vacuum, 5×10^{-4} torr, was so successful and required less preparation with fewer problems that all current work is being done in a vacuum furnace. The one used for this project is the work area of an electron beam welder.

Core Preparation. The surfacing of the core materials to the required flatness is a critical process. Previous experience has shown that for heavier gages, a variation in core thickness less than .002 in. must be maintained. It is presumed that for the foil gages, a tolerance of .001 in. is required. Hand lapping has been selected as the most practical process presently available. The core slice is glue-bonded to a ground surface plate with an adhesive that can be dissolved in a solvent. The lapping is done with a cast (meehanite) surface plate prepared with a cross grid of grooves. A 240-mesh alundum grit is used as the lapping agent.

Heater Elements. A number of possible materials for use as heater elements were investigated. These materials were narrowed down to nichrome and columbium foil ribbons. The placement of the elements in the fixture was found to be critical, since the thickness of material between the wire and the surface provided insulation and the wire temperature was a function of this distance. The use of thick caul sheets allowed these wires to be placed on the surface or in shallow grooves which alleviated this problem.

Brazing Alloys. The criteria for choosing a brazing material must take into account fillet shrinkage, diffusion into the parent material, oxidation resistance, ratio of melting point to operating temperature, wetting characteristics, etc. For this project the gold-palladium alloys were chosen initially, and from the various possible combinations a 50-percent gold - 50-percent palladium, .0005-inch thick foil was found to be satisfactory, except that diffusion into the parent materials has proven to be a serious problem. A better brazing material was subsequently found to be 70-percent silver - 30-percent palladium. However, it appears that diffusion bonding will provide the best solution.

Assembly Process. One of the more important requirements in the final assembly of the panels is that adequate support be provided to insure uniform temperature across the face and that the parts be held firmly in place with uniform contact. This resulted in finding that heavy caul sheets are necessary. Because of problems of thermal expansion and compatability the caul sheets were made of molybdenum.

The assembly process consists of placing the lower half of the brazing tool with the face up. Heater wires are inserted or laid on the surface and covered with electrical insulation. The caul sheet is placed on top and a stop-off of water-suspended MgO is applied. The composite panel is laid on top. The upper half of the fixture which has been assembled in the same way as the lower half is now clamped and placed on top of the panel. The entire assembly is loosely clamped to prevent slippage and placed in the brazing chamber where the heater wires are attached to the power lead (Figure 20.)

The resulting panel is shown in Figure 21 as it appears when the fixture is opened. As can be seen, perfection has not yet been achieved and the surface skin, which I would remind you is only one mil thick, is not perfectly smooth. The reasons for this roughness are attributed to the difference in the coefficient of expansion of the Inconel 702 core material and the molybdenum facings and the shrinkage of the braze alloy during solidification. Recent unreported work indicates that this effect can be reduced to the vanishing point.

Work is continuing on this project with the ultimate objective of developing a panel which will be capable of operation at temperatures above 2500°F. To achieve this will require core materials of refractory metals (these materials have recently become available in thin gage honeycomb); higher melting brazing alloys or more likely, the elimination of the braze material all together and dependence upon diffusion bonding; the development of fixture and heater element materials capable of resisting higher temperatures; and of particular importance, a means of protection against catastrophic oxidation in service.

Expandable Structures

Two types of metallic structural concepts designed for high-temperature usage have been briefly discussed. We will now turn to a completely different concept, i.e., expandable structures for space applications. In trying to discuss this subject we find ourselves in the thicket of diversity of opinion. Lockheed has collected nearly one thousand documents related, in one way or another, to expandable structures. These documents were collected as part of an ASD contract to survey and provide handbook information on expandable structures. We would like to take this opportunity to thank the many firms who generously cooperated with this effort by supplying us with reports and data.

It is obvious that with all this information available we must be narrowly selective in the presentation made in this paper. Consequently, we will discuss only those expandable structures which are folded for launch, then expanded to a greater volume after reaching orbit. The final structures so obtained will be assumed to be rigidized by chemical means rather than depending on air pressure. They are of a size and nature to be inhabitable by men; however, they are not designed for reentry. Great strides have been made in the past few years to provide the technology required to build a vehicle which meets these requirements, but there still remains much to be done before these concepts will replace the more conventional first-generation, mechanically unfolding concepts.

The advantages make the gain worth the task. Some of these advantages are:

1. As the boosters increase in efficiency, the required ratio of volume to weight of the payload will tend to decrease. Expandable structures can provide a favorable ratio.
2. Due to the unitized construction fewer joints are required with a resultant reduction in potential sources of leaks.
3. Since the primary loads in such a vehicle are tension due to the pressurization required, it is possible to exploit the high tensile strength properties of various filaments and fibers to provide a lightweight structure.
4. There is some evidence that such structures may be more resistant to meteoroid damage.

Design Concept

A major, if not the most important design consideration for a manned space station, is protection against meteoroid strikes. The amount of time of exposure and the vulnerable area is so large that strikes are inevitable, although more recent data have indicated less hazard than was previously thought to be the case. Two aspects of this problem must be considered. The first is that insofar as is practical, the structure should not allow meteroids to enter the station due to the hazard to personnel and equipment. For this purpose, double wall construction appears to provide the best solution. This means that a filler material or some method of maintaining the spacing between the walls is required. Various methods have been proposed including honeycomb cores, foamed in place plastics, and drop cords between the walls. The foam plastics will provide the best meteoroid protection. In order to be effective for meteoroid protection, the two walls should be two inches or more apart and the core density must be small to keep the core weight from being excessive.

Secondly, if the meteoroid has enough energy that it succeeds in penetrating the inner wall, a means of sealing off the hole is required. Self sealing methods under development may provide the solution, at least for the small holes most likely to be encountered. For the larger holes, which in any event will be a very rare occurrence, patches can be applied since leakage rates on even relatively large holes are slow enough to provide time to locate and repair. Much work remains to be done on meteoroid science - both in environment criteria and in effects on specific structures. Of equal importance, the type of work which has been done on the physiological effects of meteoroids which penetrate the walls of manned stations should be expanded in order to provide a basis for assessing the hazards (Ref. 9).

Most designs for expandable space stations incorporate the double wall principle in one form or other. Typical design concepts are shown in Figure 22. These designs are heavily biased to provide meteoroid protection. In addition, the sandwich construction provides an efficient structure, and the space between the faces can be used for insulation, self-sealing packets, ducts, etc.

It is apparent that with double walls, cores, insulation, etc., it is no longer possible to fold such a structure into a tight little package. Some indication of the effect of wall thickness on unfolding ratio can be obtained from Figure 23. For the type of mission being discussed here it is probable that high unfolding ratios are not required. This is the case if a certain amount of "hard" equipment is enclosed with the folded structure during launch.

Materials and Processes

A great many materials have been proposed for use in expandable structures. Many of these materials fall by the wayside when considered in the light of their resistance to the environment of space. Table 2 is illustrative of this point where we find only four materials which fall into this "excellent" class.

Foam-in-place plastics have been widely proposed as a method of rigidizing structures. Some of the materials which have been foamed in vacuum are polyurethanes and gellatins.

Table 2

RESISTANCE OF MATERIALS TO SPACE ENVIRONMENT

Materials	Vacuum (10^{-8} mm Hg)	Total Expected Radiation (Apprx. 2 yr)	Serviceable Temperature Limits
Films			
Teflon	G	F	-100° to +475°F
Mylar	G	G	-100° to +325°F
Saran	G	G	-100° to +275°F
H-Film (Polyimide)	E	E	-100° to +550°F
Fabrics			
Cotton	G	P	- 65° F to +200°F
Nylon	G	F	- 65°F to +250°F
Dacron	G	G	-320°F to +325°F
HT-1 (Nylon)	G	G	- 65°F to +450°F
Fiberglass	E	E	-320°F to +450°F
Plastics			
Epoxy	E	G	-320°F to +450°F
Phenolics	F	G	-320°F to +450°F
Polyester	G	F	-320°F to +350°F
Silicones	G	G	-320°F to +550°F
Elastomers			
Silicones	G	G	-200°F to +550°F
Butyl	F	P	- 65°F to +375°F
Chloroprene	G	G	- 65°F to +250°F
Polysulfide	P	P	- 65°F to +250°F
Polyurethane	G	E	- 65°F to +300°F
Viton	G	G	- 20°F to +450°F

P = Poor G = Good

F = Fair E = Excellent

NOTE: Comparisons are based on materials in their own category and not on category versus category.

Figure 24 shows a typical example of a foaming operation. It appears, however, that there is a long step from the laboratory samples which are small enough to be made in a bell jar or small vacuum chamber, and a full-size double wall cylinder ten feet or more in diameter which must automatically provide itself with the proper distribution of material to insure the structural integrity of the completed vessel in the zero-gravity environment. The solution to this problem may lie in the technique of applying layers of the basic materials to the inner or outer wall, then by means of solar heat setting off the reaction which would result in a foaming action. Once started the heat of reaction could sustain the process until complete foaming had been achieved.

Wall structures are generally considered to be made of Fiberglas and may be of filament wound or woven fabric. One very interesting concept which has been developed is an integrally woven sandwich structure. In the non-rigidized condition this material lies flat and can be folded, as rigidized it is a hard "truss core" type sandwich which provides excellent structural walls.

The key to successful expandable structures of the type discussed here is the technique for rigidization. It is obviously desirable for this process to be automatic, controlled, and rapid. The state of development is such that one cannot say with certainty that any one method is better than another. One way that holds considerable promise is to impregnate the wall material with a suitable resin material. It is then partially cured so it will not be sticky but is still pliable. After deployment in space, the material will then rigidize as the result of the space environment. Various types of materials will respond to different environmental conditions. Some are rigidized by ultraviolet rays, others by infrared or temperature, and still others evaporate. Some of the types of materials are listed in Table 3 which shows the process by which the rigidization is achieved.

Conclusion

It is recognized that much of the discussion on expandable structures has been in the nature of vague generalities. The research and development efforts in this field are just barely past the preliminary stage, and actually very few expandable structures have been flown. The work in this field is being pushed forward rapidly, and there are a number of important projects under way or contemplated which will provide information and confidence to incorporate these concepts into practical manned space vehicles. In the meantime it is perfectly feasible to consider the use of expandables in certain specialized applications of a secondary nature. It is anticipated that the information currently being collected and evaluated will provide the basis for such designs.

Summary

In this paper we have suggested some ways in which lightweight structures can be built from very thin gage superalloy and refractory metal materials and from non-metallic materials which are folded for launching, expanded in space, and rigidized.

The techniques employed for the superalloys, such as Rene 41, in gages down to .003 inches are well developed, and with a nominal amount of additional work for improvement, structures can be fabricated today from these materials. The fabrication of honeycomb panels from .001-inch thick molybdenum is currently in the process of being developed and there is every expectation of success for this effort in the near future. The practical application of expandable, rigidized structure for primary structural manned applications is further down the road, but the intensive efforts being exerted is resulting in rapid progress which will lead to designs which exploit the unique characteristics of these concepts.

Table 3

MATERIALS RIGIDIZED BY SPACE CONDITIONS

MATERIAL	VACUUM	CONDITIONS ULTRAVIOLET	CONDITIONS INFRA-RED
Epoxies		X	X
Polyvinyl Chloride	X		X
Acrylics	X	X	
Polyvinyl Formal	X		X
Polyesters		X	
Polyurethanes		X	X
Gelatin	X		X

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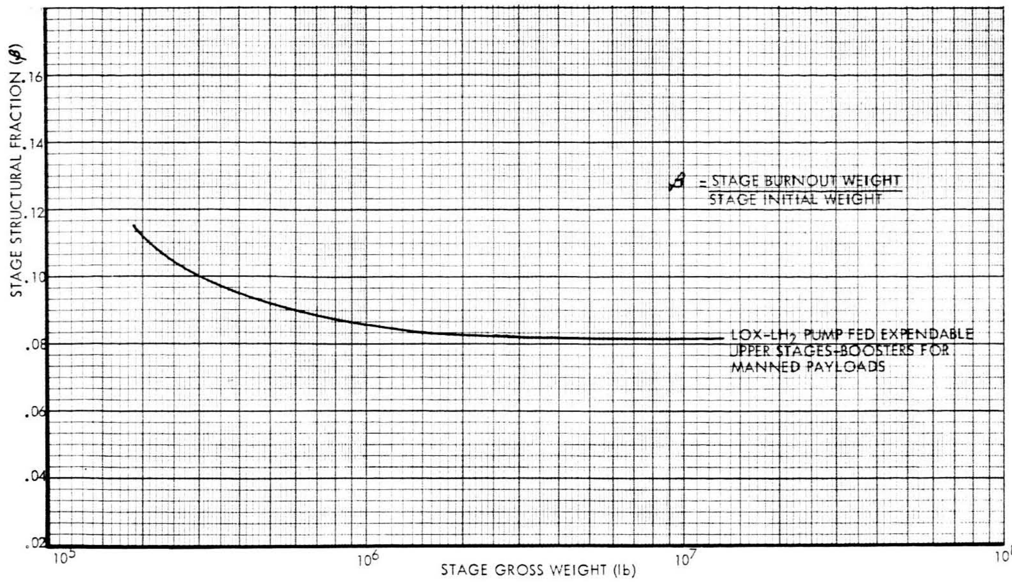

FIGURE 1. TYPICAL STAGE STRUCTURAL FRACTION VS STAGE GROSS WEIGHT

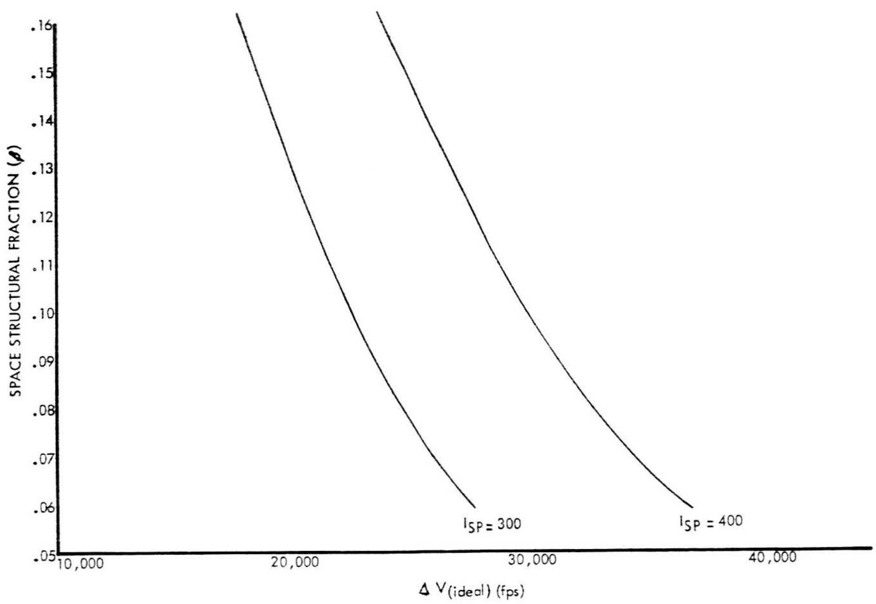

FIGURE 2. LIMIT OF ΔV_{IDEAL} VS STAGE STRUCTURAL FRACTION FOR A SINGLE BOOSTER STAGE

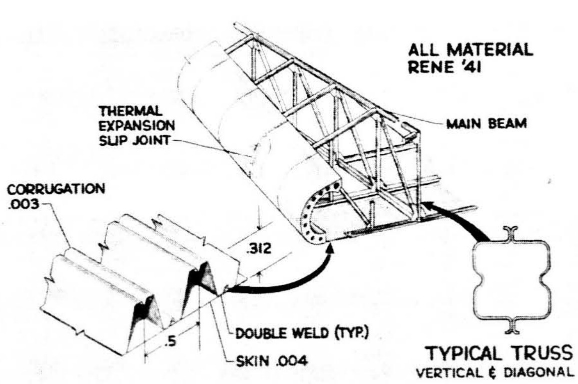

FIGURE 3. SPACE FERRY STRUCTURAL DESIGN

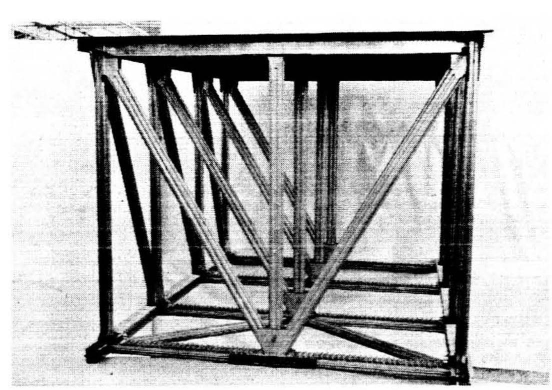

FIGURE 4. FOIL GAGE STRUCTURAL SPECIMEN

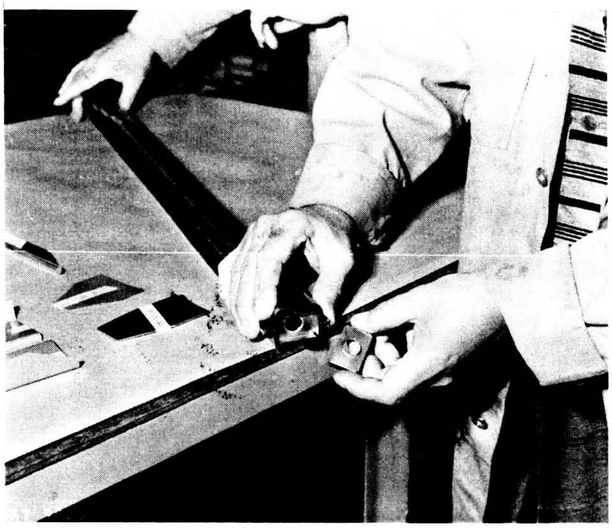

FIGURE 5. BEAM STRUT ASSEMBLY

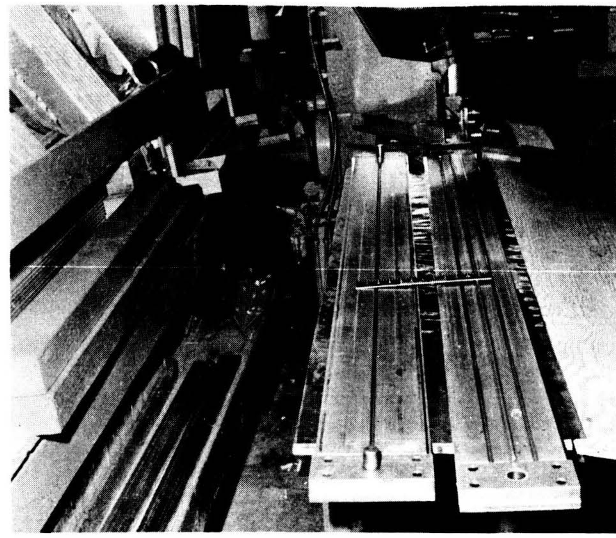

FIGURE 6. DIE FOR FORMING TRUSS MEMBERS

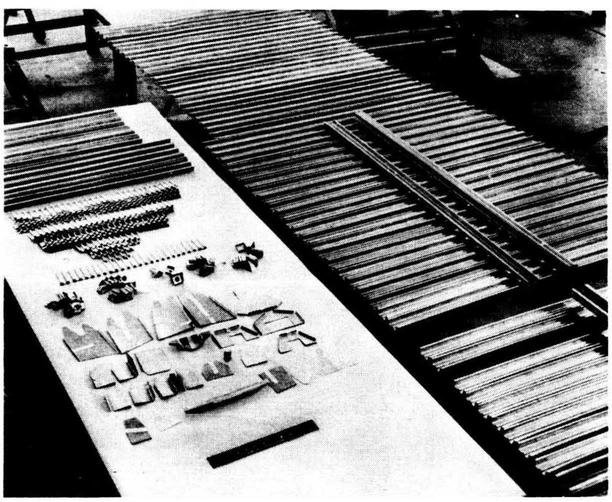

FIGURE 7. TEST SPECIMEN – DETAIL PARTS

FIGURE 8 ROLLS FOR FORMING CORRUGATIONS

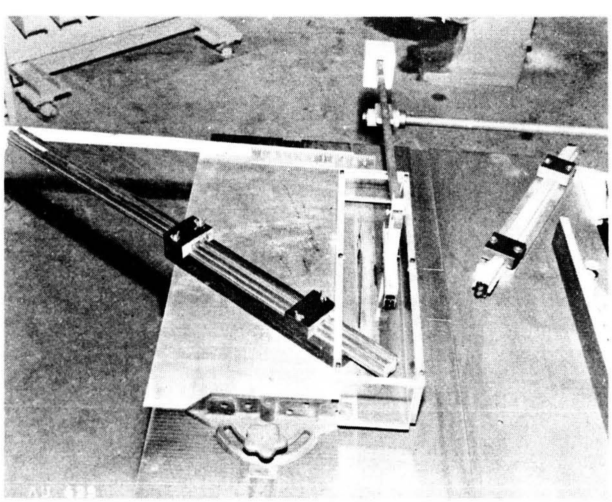

FIGURE 9. STRUT CUT OFF FIXTURE

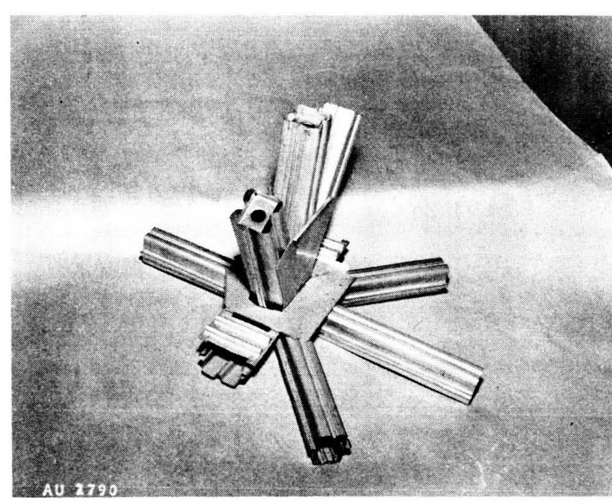

FIGURE 10. STRUCTURAL SPECIMEN – TYPICAL JOINT

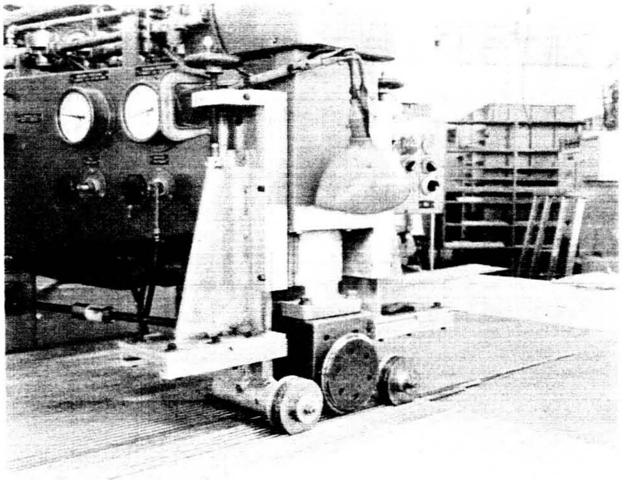

FIGURE 11. WELDING OF CORRUGATIONS TO FACE SHEET

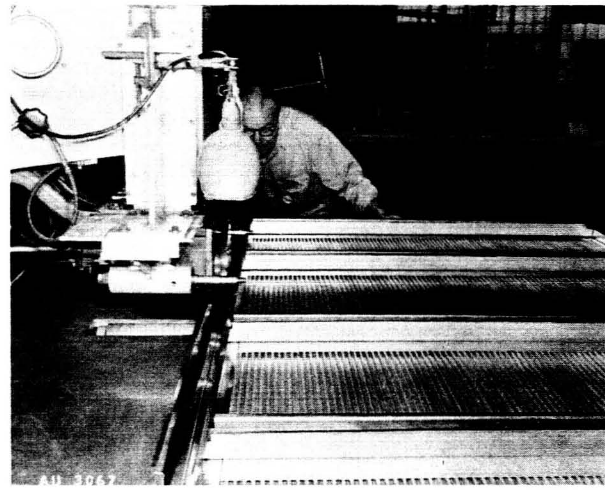

FIGURE 12. WELDING OF SURFACE ASSEMBLY

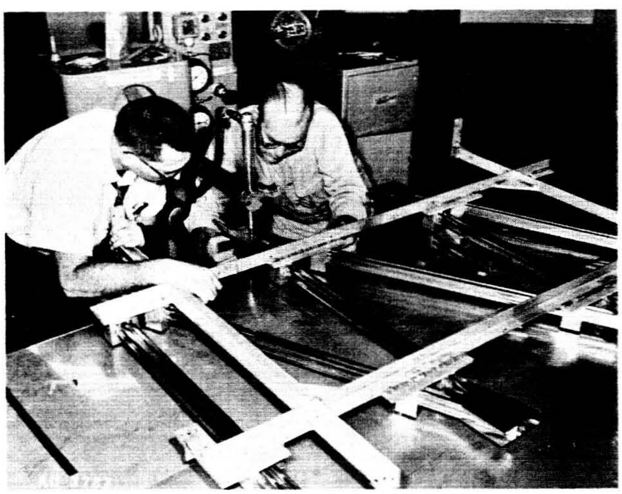

FIGURE 13. WELDING OF BEAM SUBASSEMBLY

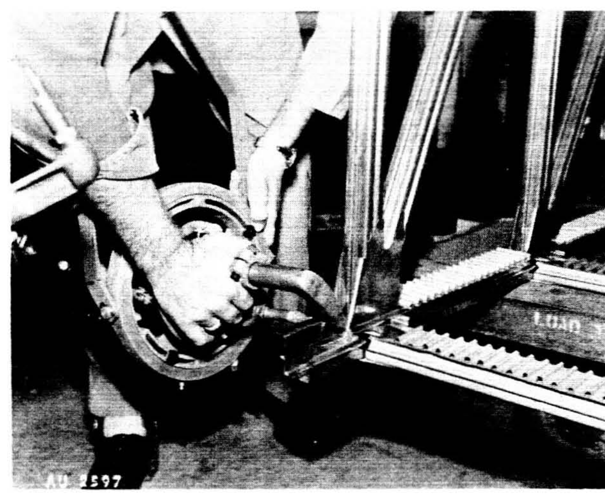

FIGURE 14. PORTABLE HAND WELDER

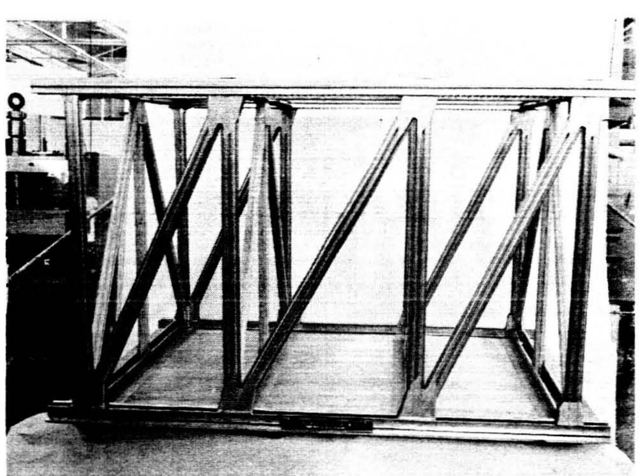

FIGURE 15. COMPLETED STRUCTURAL SPECIMEN

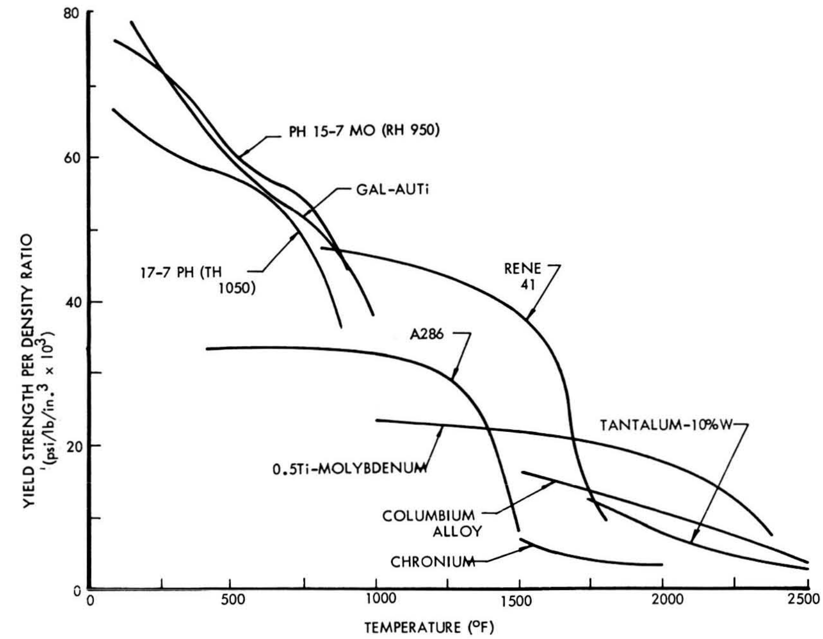

FIGURE 16.

YIELD-STRENGTH DENSITY RATIO OF VARIOUS MATERIALS VS TEMPERATURE

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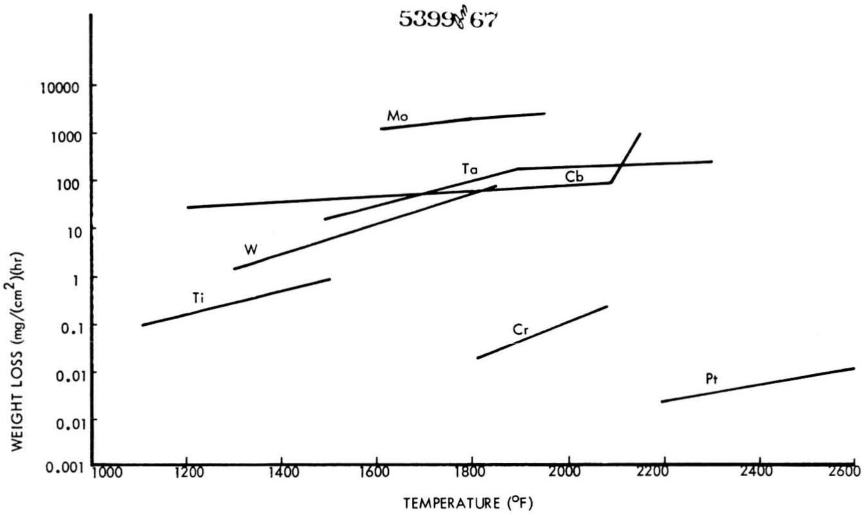


FIGURE 17.

OXIDATION RATES OF VARIOUS MATERIALS VS TEMPERATURE

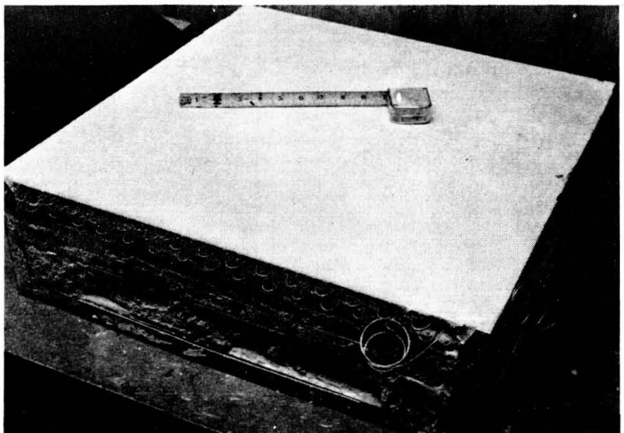


FIGURE 18. ONE HALF OF BRAZING FIXTURE

FIGURE 19. BRAZING FIXTURE **IN** PLASTIC BAG

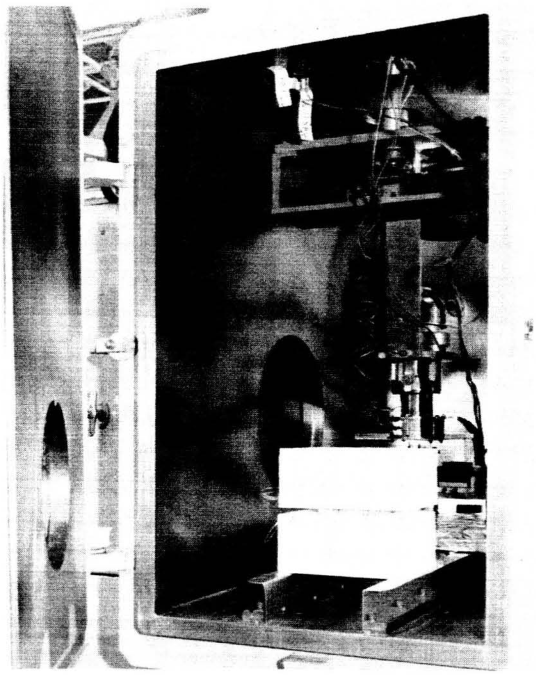


FIGURE 20. BRAZING ASSEMBLY

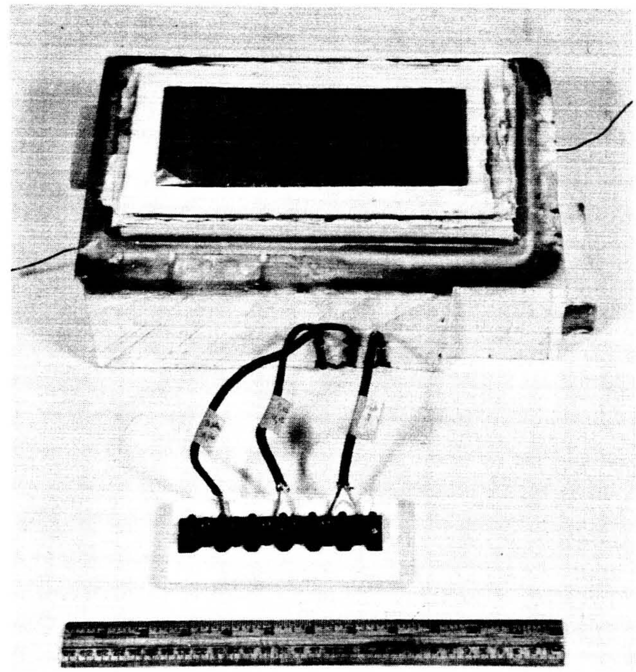
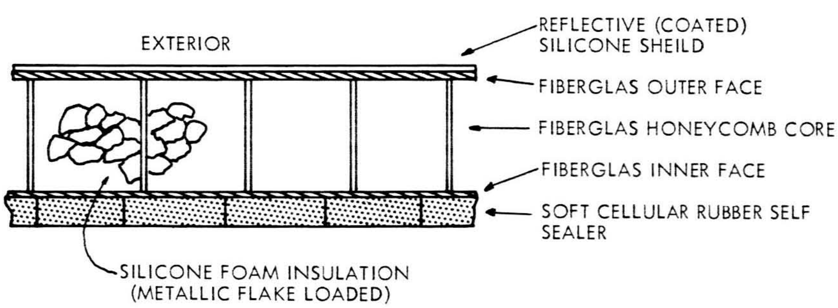
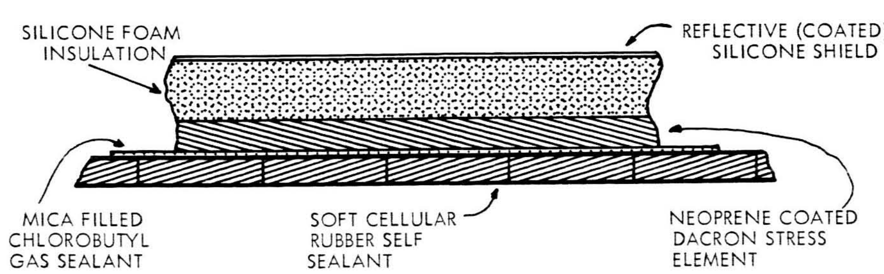


FIGURE 21. SMALL PANEL AFTER BRAZING





REF: U.S. RUBBER CO. R & D REPORT "INFLATABLE MODULES"
E. P. PERCARPIO, SEPT 20, 1962

FIGURE 22. EXAMPLES OF TYPICAL EXPANDABLE STRUCTURE – SPACE STATION WALL CONSTRUCTION

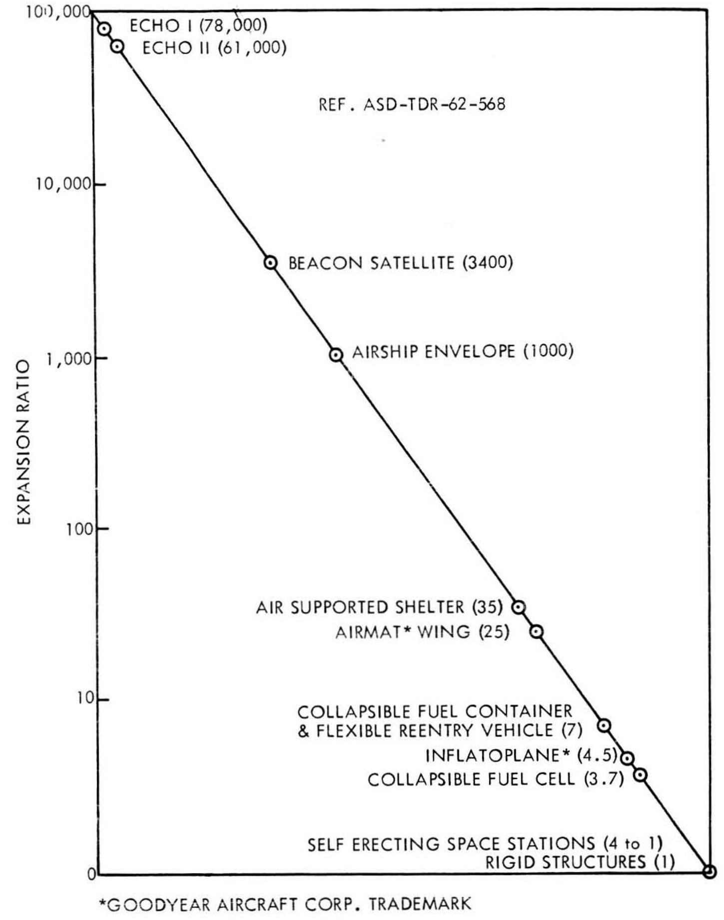
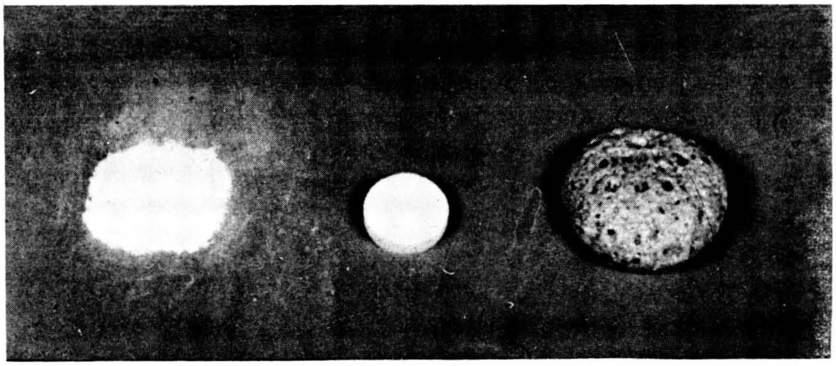


FIGURE 23.

EXPANSION RATIO FOR SEVERAL EXISTANT AND PROPOSED EXPANDABLE STRUCTURES



A B C

A. Powdered Material

B. Compression Molded Disc

C. Rigid Foam (obtained using vacuum and 250 °F)

FIGURE 24. SOLID DIISOCYNATE FOAMANT

ON BOARD PROPULSION DESIGN FOR MANNED SPACECRAFT

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INTRODUCTION

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Early vehicles, such as the Douglas D-558 Skyrocket series, the X-15 and the present Mercury, Gemini and Apollo systems illustrate the design changes in the evolution from conventional aircraft to realistic spacecraft designed to operate beyond the atmosphere. The rocket-powered aircraft, specifically the X-15, were designed to explore high speed phenomena and flight conditions associated with reentry into the earth's atmosphere.

Experience in prolonged space flight has been obtained from the Mercury spacecraft system. Gemini will provide an extended capability to maneuver in space for prolonged periods of time. It incorporates design changes, derived from Mercury experience, which shift the emphasis from simply achieving orbital flight to performing useful tasks in space. The Apollo design includes the capability of complete command on board the spacecraft. Beyond this, interplanetary vehicle designs are being studied, the recent EMPIRE study being one example. The trend in all of this is that of making fuller use of man's potential and capabilities as a functioning member in the spacecraft design.

This paper reviews the ways in which a propulsion system designed for a manned spacecraft differs from that of an unmanned vehicle. These differences seem small to the authors. They are certainly smaller than the differences between a nuclear and a chemical propulsion system. Yet, the success of a manned vehicle depends on patient attention to such small details. In light of the "what have we learned" theme of this session, this paper is limited to a discussion of chemical propulsion systems. Within this limitation, those general requirements and design considerations unique to manned propulsion systems are: engine requirements with regard to crew safety and reliability; requirements for pilot controls and displays; propellant management requirements; and the possibility of inflight maintenance and repair.

RELIABILITY AND SAFETY

The most striking difference between the design approach to manned and unmanned space vehicles is a persistent emphasis on reliability and crew safety. For an unmanned weapon system, reliability is one element in a cost effectiveness evaluation. But no dollar value can be placed on the life of a man. This emphasis on safety and reliability pervades all aspects of propulsion system design.

In missile propulsion design, safety and reliability are often antithetical. The ignition system, for example, may contain so many "safety" devices to prevent inadvertent firing, that the missile is less likely to work when it is supposed to. In spacecraft design, we refer to reliability as the probability that the mission objectives will be achieved, and to safety as the likelihood that the crew will be returned to earth unharmed. Safety, in this sense, becomes almost synonomous with reliability during many phases of a manned mission. Consider a vehicle in lunar orbit for example. For safe return of the crew, ignition of the propulsion system upon command is just as important as prevention of accidental ignition.

One approach toward reliability is that of redundancy. An example is the use of redundant engines in such a manner that if one engine fails, the remainder are sufficient to complete the mission. But this leads to a contradiction. With more engines, the flight control system and indeed the whole vehicle design becomes more complicated and hence can be expected to be less reliable. The greater the number of engines, the greater the possibility of human error during checkout. Moreover, not all engine failures are passive.

It should be noted that a well-developed, qualified engine rarely experiences a catastrophic in-flight failure. Failures do occur, and are corrected, during early engine development programs. But after that, the reliability of the engine matures to relatively high values. This argument can be supported by a survey of Thor flight test data. Over 50% of the flight malfunctions occurred during the first 20% of all Thor flight attempts. Of these malfunctions, most are attributable to systems not directly associated with the engine. As development continues, the record improves. In fact, the only Thor propulsion system failures in the last 100 flights can be traced to human errors during ground checkout, possibly caused by gross fatigue of the crews, rather than resulting from engine system design faults.

The inherent conflict between the "keep it simple" and the "make it redundant" approaches to reliability can be illustrated quantitatively by considering the question of engine malfunction detection. Use of engine malfunction detectors of some sort on a multiple engine system permits shutting down a bad engine. On the other hand, automatic monitoring and engine shut down devices must be held to a minimum, since each device added introduced additional complexity and therefore provided another source of failure.

As shown in Figures 1 and 2, the effects of monitor unreliability on the overall reliability of the system can be of considerable importance. A spare engine is helpful if the bad engine has damaged nothing but itself, and is successfully isolated. Its usefulness is lessened if a malfunction monitor occasionally shuts down good engines. From Figure 1, assuming an engine qualified to a mission reliability of 0.990, a perfectly monitored four-engine cluster would have a mission reliability of 0.9994 - a big improvement. But if the malfunction-monitor shuts down each good engine randomly, about once in every 20 missions, the reliability is down to about 0.98, only half as good as with a single engine.

Having questioned the reliability of an engine malfunction detection device, it is also reasonable to question the reliability of a man himself. Will all failure modes develop slowly or harmlessly enough to permit a man to catch them in time? It should be pointed out that this is done now on rocket static tests, during which test engineers frequently catch trouble in time to save the engine. A big difficulty is in assigning a number to the man's reliability. Is he 90% certain of controlling a bad engine, and 99% certain of not shutting down a good one? Of course, in the event of error, he might be able to restart the engine with no particular harm, a trait not shared by automatic sensors.

This discussion is not intended to prove that multiple engines are better than a single engine. That decision depends on the particular mission, configuration, etc., of the vehicle. The present opinion is, however, that, with good displays and controls, a well-trained flight engineer can justify the incorporation of a malfunction detection system.

CONTROLS AND DISPLAYS

What displays does the crew need? X-15 pilots have relied rather exclusively on throttle and chamber pressure readings during powered flight. It is felt that it would be unwise to conclude too much from this experience. With only one pilot, and a short powered phase, there is too much happening for him to devote much time to engine performance analysis using the variety of available gauges. The multi-engined airplane take-off maneuver, where a flight engineer or third pilot concentrates heavily on engine operation, might be a better model of the future spacecraft.

But there is one more element. For a long duration mission, the crew will want enough information to detect, locate, and analyze faulty components so that they can be repaired in flight.

Past experience suggests some guidelines for display design:

1. The Army-Navy Instrumentation Program, ANIP, has shown that simplified, integrated, displays yield significantly higher pilot efficiencies than conventional ones.

2. In actually designing displays, we have inevitably found that there is never enough space to simultaneously display all the information that a pilot is capable of understanding and acting upon. (A corollary of this is the conclusion that there is no point displaying information that the crew member can do nothing about.) Time sharing of indicators reduces this problem. Measured parameters can be displayed in sequence, using the same indicators, or the system can be programmed to display only the most critical parameters or only those which have exceeded their redline value.

3. Some bisensory warning system need be combined with the displays. This is particularly important for an orbiting space laboratory or an interplanetary vehicle. (It would be unreasonable to expect a crew member to stare continuously at a panel for a year.) If several malfunctions occur simultaneously, a logic network will select the more important and interrupt any lower priority warning. An appropriate example is a voice warning system which uses a prerecorded female voice, speaking directly over the pilot's headset, to describe the exact nature of the hazard.

An example of the application of these guidelines is illustrated in Figures 3 and 4. The panel shown applies to a two-stage spacecraft. Each stage has two oxidizers and two fuel tanks and uses a single, pressure-fed engine. A primary example of the employment of time sharing is the use of the same propulsion panel (Figure 4) for display and control of both stages. The schematic arrangement of the displays facilitates a logical diagnosis of the condition of the propulsion system. Without further time sharing, the display shown would be adequate for a gross checkout but would not give enough information to pinpoint and isolate a fault in the system. For example, consider the helium pressure gauge shown in Figure 4. This gauge can be activated by a number of transducers, selected by the crew, to provide an indication of system pressures throughout the helium pressurization system (thus eliminating several additional gauges). Once a fault is located, the crew has the option of (a) isolating the troublesome elements in the system and switching to redundant elements, (b) attempting a repair of the fault, or (c) weighing a decision between abort or continuing the mission with reduced capability.

The same approach is applied to temperature readouts and, as shown on the panel, for a multiple tank configuration, display of two separate tank temperatures with a single gauge is easily accomplished. In addition, provision is made to use this same gauge to read temperatures at the top and bottom of the tanks by activating the switch located directly beneath the gauge. Propellant freezing is a pernicious problem in this example. Thermocouples are located in critical regions of the tank (i.e., near heat shorts) and an additional provision is made to display their readings sequentially on the same indicator.

Critical components of the pressurization and propellant feed system are also controlled by switches located on the panel. The pilot can select automatic control of redundant pressure regulation systems or manually override the automatic system. The propellant flow control valves are controlled in the same manner. Selection of automatic or manual switching depends on the mission phase. A critical maneuver may require the pilot to be occupied with other functions and manual switching of supporting systems would then be a redundant backup.

The display shown for a pressure-fed engine (Figure 4) only includes a chamber pressure monitor for steady-state operation. One can question whether this is adequate.

Present-day engines differ so substantially in their malfunction modes that no general conclusions can be drawn. (Obviously, pressure-fed engines do not throw turbine blades.) Experience has generally been that combustion instabilities are most likely to occur during engine start up transients, valve and turbopump failures during steady-state running, and valve malfunctions during shutdown.

A gas generator driven, turbopump-fed engine might then require a display of turbopump speed and inlet temperature. But not all turbopump engines are the same. A topping cycle engine, such as Pratt & Whitney's RL10, may not require turbopump speed and inlet temperature displays. This type of engine seems to respond to major malfunctions by shutting itself off. This is inherent in that it derives its pumping energy from regenerative chamber heat, with the warmed propellant driving the turbopump. Any interruption in the flow of either propellant reduces the chamber heat, and the pump tends to run down.

It appears that the only malfunction mode which might justify an automatic shutdown system would be combustion stability. This is still a somewhat controversial question. Sometimes the effects are mild. The Thor propulsion system, for example, sometimes exhibits a slight but harmless 20 cps resonance. Some engines, particularly in development, exhibit instabilities which lead to an extremely rapid chamber pressure excursion, sometimes resulting in destruction.

PROPELLANT MANAGEMENT

If an aircraft propulsion system fails entirely, the craft can return the crew safely by gliding back to earth. A lunar spacecraft, coasting in the translunar trajectory after earth escape, is in an analogous situation. If the trajectory is properly planned, the spacecraft will simply circumnavigate the moon and re-enter the earth's atmosphere without the propulsion system operating again. But this analogy breaks down once the spacecraft commander has committed himself to lunar orbit (or earth or Mars orbit for that matter). Assume that the spacecraft has some reserve propulsion capability for abort. (This is typically supplied by the terminal propulsion stage.) Before commitment to each successive phase of the mission, the spacecraft commander must not only check out the workability of the propulsion system but also decide whether there is enough fuel remaining to continue.

The fuel gauges employed in contemporary systems, such as capacitance probes or point sensors, are entirely inappropriate to measurement under zero gravity. This problem is avoided in some current vehicles by using small auxiliary engines to supply settling force fields during the period of fuel measurement. If bladders are used to control the propellant, the measurement problem is aggravated.

As propellant quantity is a safety item, it is highly desirable that it be measured continuously or quite frequently. (An additional benefit accrues from use of a continuous propellant measurement as a gross leak detector.) But no presently-developed measurement device seems capable of continuous measurement under zero g. Three approaches to this have been suggested.

1. The propellant could be contained within metal bellows, measuring the extension of the bellows.

2. Acoustica Associates, Inc., has suggested measuring the acoustic resonant frequency of the gas volume within the tank.

3. The radiation gauging system being developed by North American Aviation for the RS-70 might be adapted to this purpose.

Each of these three has its inherent disadvantages. There is currently a controversy as to whether or not interplanetary vehicles should provide artificial "g's" by rotation. From the rather provincial viewpoint of the propulsion engineer, the rotating vehicle would seem to solve many problems.

IN-FLIGHT MAINTENANCE AND REPAIR

Detection, replacement, and repair of faulty components may turn out to be the most important contribution of man's abilities in achieving prolonged space flights. It is rather early to speculate on this, but some indication of the type of propulsion system designs amenable to in-flight repair has already been obtained from the design of boosters, such as the Saturn S-IVB program.

Consider the representative propellant pressurization regulation system schematic shown in Figure 5. In current practice, each of the valves, solenoids, and filters shown would be connected by tubing with welded (or possibly even B-nut) fittings. Each fitting would require leak check. In trying to meet the requirements of completely automatic ground checkout of the system, rapid replacement of malfunctioning components, environmental control of the entire system in space, we have tended toward modular design concepts. A representation of this approach is shown in Figures 6 and 7. All of the components of Figure 5 have been combined into one body. A small cavity of known volume is connected to all leakage points in the module. Monitoring the pressure at this point serves as a leak check for the entire module, eliminating the myriad of leak detectors necessary otherwise. Reducing the entire pressurization system to a small surface drastically reduces temperature control requirements.

While this design approach has resulted from booster requirements, it also seems to be the appropriate approach to in-flight maintenance and repair. Collecting all the active parts in a small volume simplifies the problem of providing access. The analogy between this and the "black box" approach to electronic system design and maintenance is clear and appealing. Perhaps the internal components of such a module will some day be as standard and interchangeable as vacuum tubes and resistors.

For some phases of a manned mission, such as during a landing maneuver, in-flight repair capability is of little value. For lunar or earth orbital missions of a year's duration, the crew has the leisure time to repair almost anything. We have suggested an appropriate approach toward feed system designs. More work is needed in other areas, particularly repair of propellant tankage. While the very presence of man in a spacecraft forces a high requirement for propulsion reliability, his presence also seems to provide the best means of meeting that goal.

CLUSTER RELIABILITY VS NUMBER OF ENGINES & SHUTDOWN RELIABILITY

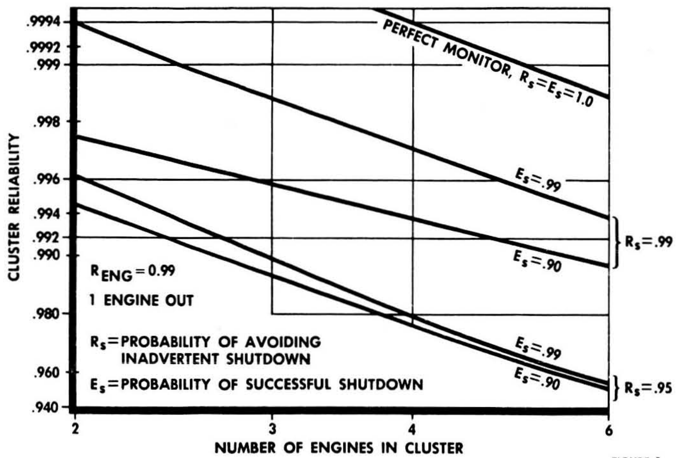


FIGURE 1

EFFECT OF MONITOR ON ENGINE-OUT RELIABILITY

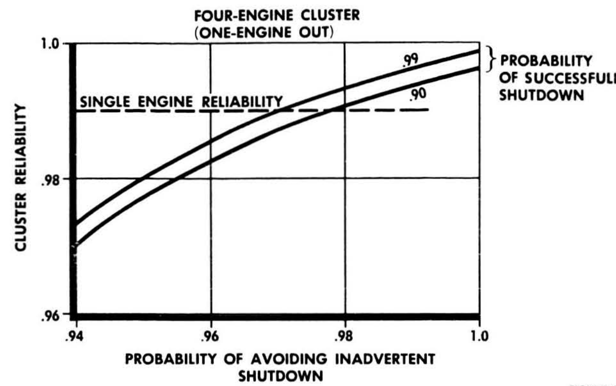


FIGURE 2

REPRESENTATIVE CONTROL & DISPLAY LAYOUT

FIGURE 3

DETAIL OF PROPULSION CONTROL & DISPLAY PANEL

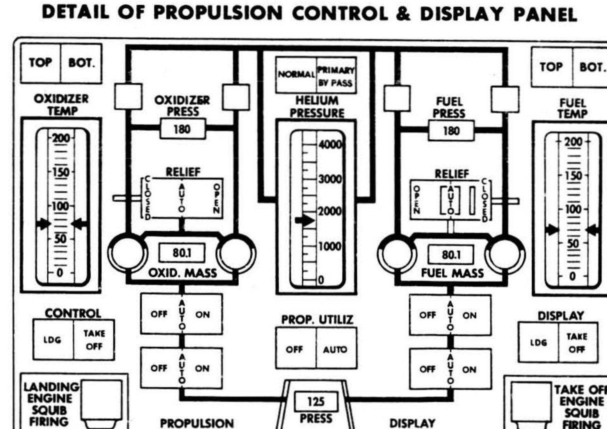
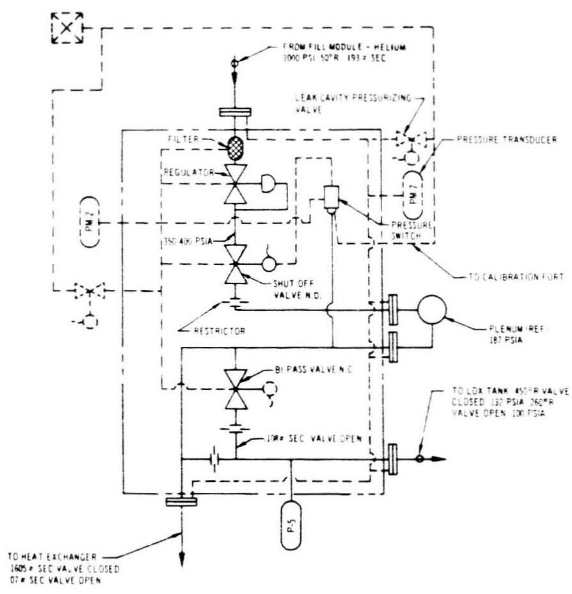
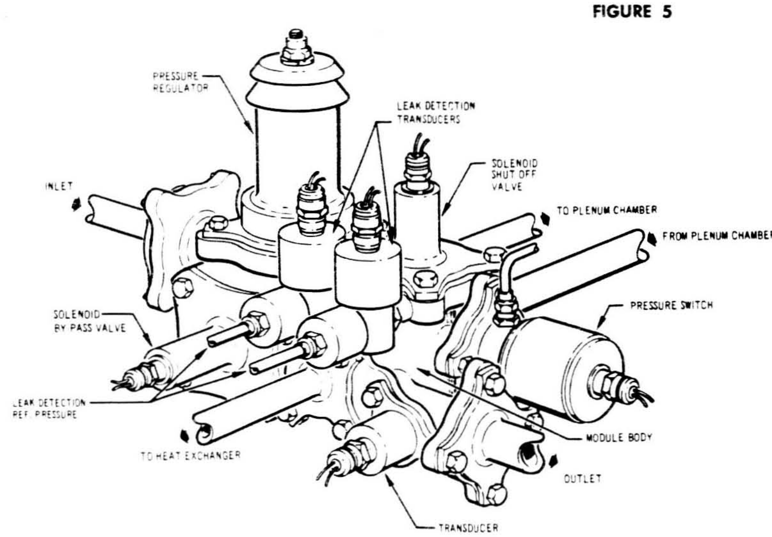


FIGURE 4



TYPICAL PRESSURIZATION REGULATION SCHEMATIC

FIGURE 5



MODULAR APPROACH

FIGURE 6

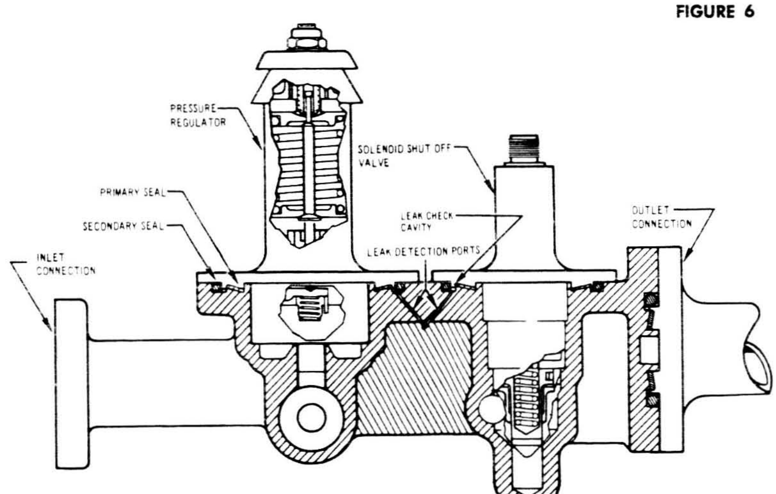


FIGURE 7
CROSS-SECTION OF FIGURE 6

EXTRA VEHICULAR MAINTENANCE AND TECHNIQUES

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General Electric ~~Company~~ Valley Forge, Pa.

13 35482 11

Orbital maintenance and repair can no longer be classified as a highly desirable and much sought after but hardly obtainable system characteristic. I am sure that most of us when considering this problem have shuddered at the weight penalty implied by considerations of spare parts, tools, diagnostic equipment and sub system design philosophy necessary to achieve this goal.

Achievement of the goal, however, is mandatory. The exceptionally long durations of the missions currently proposed for earth orbiting space stations, lunar base and interplanetary explorations by man offer the systems engineer no choice but to absolutely require in-flight maintenance and repair capability. Graphic illustration of this consideration is shown on figure 1. The solid line is typical of the shape of a sub system reliability curve plotted for a duration of one year. As you can see probability of successful operation at launch is high. A steady reduction of this probability occurs until at one year for instance the value is well below any reasonable acceptable limit. The dotted saw-tooth curve, however, indicates system reliability as a function of time considering the application of periodic preventative maintenance by the crew.

While the numbers shown here must be considered as relative they are representative of values which can be generated for any given sub system. The need for preventative maintenance to assure reliable operation of various sub systems for long periods of time is, therefore, easily seen. Examination of this particular problem leads us to the obvious conclusion that preventative maintenance as well as replacement and repair is one of the major and most significant roles that the crew of any long duration mission will be called upon to perform. It is also obvious that appreciation of this role as a necessary criteria for mission success must be recognized early and provided for in the design and development of the system and its related sub systems.

The type of maintenance and repair, the sub systems and the equipment which might be affected, the diagnostic needs and requirements as well as analysis of spare parts and tools has been studied on many occasions by many people. The chart shown here on figure 2 is representative of the types of failures which one might expect the crew of any long duration spacecraft to be required to correct. The criticallity rating shown is arbitrarily selected between 10 for catastrophic and 1 for inconsequential. Study of the trade-offs of replacement versus repair, self-monitoring versus manual diagnostic and adjustment, degree of redundancy versus degree of maintainability are, of course, necessary for the intelligent design of any sub system in any given spacecraft and depended, of course, upon the specific system and mission. There is a problem, however, which is universal to all systems and which is relatively inflexible in its need for solution. That is the problem of extra vehicular maintenance. With few exceptions maintenance and repair and the design problems associated thereof are amenable to engineering analysis and basic trade-off study, as long as we are discussing this activity taking place within the confines of the vehicle cabin. Although we are concerned about weightlessness here, it is not difficult to realize that restraint in seats or at work areas and the use of various types of equipment as retention aids can easily be designed. The problem of repair within a cabin, therefore, becomes little different than the problem which we face and daily solve in the maintenance of aircraft, missile systems, and other highly sophisticated pieces of machinery. Zero gravity is easily handled within a cabin and can in fact be an advantage in achieving those hard to get to places and difficult positions that we all hope to design out of our equipment but somehow invariably creep in.

Such is not the case in performing maintenance and repair external to the vehicle. Here the environment is hazardous. We are obviously encumbered by a pressure suit, a back pack, possibly some form of propulsion, and the way the man made radiation belts are beginning to look he will probably be equipped with some type of radiation protection. In addition this astronaut probably has to carry with him a satchel of tools, spare parts and test equipment and even so will probably be forced to operate like the proverbial plumber who constantly must return to the store for the correct equipment. Therefore, operating within the hazards of the environment and the limitations of his suit and his work place, this astronaut must in one way or another apply himself to the task of doing meaningful work in the weightless environment. Here obviously he is not so easily restrained as he was in the cabin and depending upon the location of his particular area of interest he can experience a minimum load path between himself and his work. This combination, pressure suit and frictionless with no restraint appears to be the most severe set of operating restrictions which will limit the astronaut and which will subject him to the most challenging development of his role of maintenance and repair.

Regardless of the size of future vehicles certain equipment and sub systems will by nature require service from outside the cabin. A partial list includes docking ports, radiators, sensors, solar paddles and associated gimbal mounts, antenna, fuel storage, fuel lines and valves, and propulsion systems. All of these to a greater or lesser extent may require repair. All of these most certainly will require at least inspection and probably preventative maintenance. It is in recognition of this problem, therefore, that a simulator was developed by General Electric to evaluate the ability of an astronaut to perform useful work while weightless and while encumbered by an inflated pressure suit. Analysis of this simulation problem suggested that the important parameters of this weightless environment as applied to the task of performing meaningful work are the dynamics. Frictionless in six degrees of freedom the astronaut will have little opportunity to work in the same fashion as he does on earth. Anticipation of this difficulty has promoted some concern and analysis of methods of overcoming this apparent difficulty. However, it occurred to us that it would first be beneficial to determine as best we could on earth what the basic capabilities and limitations of man might be in this unusual environment. The program for the development of this simulator was, therefore, undertaken and has resulted in the device shown in figure 3. The apparatus shown here is mounted on three commercially available air bearings. Compressed air at sixty pounds per square inch flowing through these bearings raises the entire device approximately .003" above the floor and provides a frictionless base through which translation in two directions as well as yaw can be achieved. I might note here that this .003" becomes difficult to work with and requires a special floor which in turn requires constant maintenance. As a result, for another similar device which we have since built, special air bearings were designed which raises the assembly approximately .012" and consequently permits its operation over almost any type of standard laboratory or factory floor. The yoke or "C" shape device is mounted to the upright frame directly through the use of one over-designed roller bearing assembly. This bearing provides 360° freedom in roll. At the ends of this yoke the vertical framework shown is attached at either side with two roller bearings equally over-designed. This gimbal provides pitch freedom. We can, therefore, achieve in this assembly five degree of freedom; roll, pitch, and yaw plus two directions in translation. Now while air bearings at the gimbal mounts would have provided a much lower coefficient of friction than the roller bearings; the roller bearings, because they are over-designed, offer a resistence which is well below the noise level of the ability of a test subject to sense the presence of friction. When perfectly balanced a subject can begin pitch rotation simply by sharply blowing air out of this mouth. Furthermore this well balanced subject, as he applies forces to a stationary object will duplicate within the limits of the simulation, the motions, displacements and reactions of the weightless astronaut.

The question which now arises is one which must be discussed during any simulation program; fidelity. How good is the simulation? How complete can we believe the data? How close is it to the real world are all questions which certainly deserve an answer. In the case of this machine the answer is in fact a difficult one. Certainly we do not simulate weightlessness. We do, however, simulate relatively well the dynamics of weightlessness in five of the six possible degrees of freedom. This in itself is fairly good dynamic simulation. The numbers are not absolute since the sixth degree of freedom as a parameter must be simulated in order to extract absolute numbers from your test results. However, for the testing which was conducted the quantitative data is considered to be valid within relatively narrow limits. It is not expected for example that the difference in adding the sixth degree of freedom will be as significant as the difference between subjects. Now it obviously is possible to cheat the machine. That is, one can learn to apply his body and his mass in a fashion which is contrary to the situation which one would expect in experience and thereby upset the data accordingly. However, with reasonable test design and with reasonable restriction upon the operation of the subjects one can reduce this to a point where the data is effected in a relatively small way. In addition to the quantitative data which can be extracted from this simulator there is a great deal of valuable qualitative information which is available. This will be discussed shortly. It is pointed out that the qualitative data is possibly the most significant that we have taken so far, since it has the effect of providing us with new confidence in the ability of man to learn to apply his body mass in a fashion which is to his advantage even in this weightless environment. At this writing the problem of correlation still remains. The quality of this simulation can only be determined after sufficient experience is developed by conducting similar tests either in space or in aircraft flying the Keplerian trajectory. Until that time, however, we feel confident that our data is real and of use in establishing equipment design criteria.

The test program we have conducted attempts to investigate three of the basic parameters influencing the ability of a man to perform extra vehicular maintenance. The first is his ability to apply maximum short duration push/pull and torque forces. This is, of course, a measure of his ability to accomplish such tasks as breaking a nut, snap or unsnap fasteners, inserting or removing modules or sub assemblies which are hung up and which require the application of high force level for a short time duration. The second parameter was his ability to apply these forces, push/pull and torque, as a relatively constant force application. By relatively constant we suggest a time of application sufficiently long to effect movement of a nut on a bolt, of a module from its rack or of a component from its container. Investigation of these parameters it was felt would lead us to some fairly good insight as to this man's capability of doing useful work. The high level short duration force application necessary to dislodge pieces of equipment and fasteners and the long duration force application necessary to move them about. The third parameter was that of time to accomplish a given hypothetical series of complex tasks. Here we were concerned not only with his ability to apply forces but also his dexterity and motor capabilities in performing certain activities with respect to this frictionless environment. This investigation, it was felt, would provide us with some insight into our ability to estimate the time that would be required by an astronaut to perform a given task as a ratio of the time it would take him to accomplish the same task on earth. In all of these tests the variables which were programmed were those of dynamic and suit restriction conditions. Our control in all cases was the conduct of these required tasks in shirt sleeves under 1 "g". Our variables included shirt sleeves in five degrees of freedom and pressure suited in five degrees of freedom.

It must be noted here that our operating pressure within the suits for the conduct of these tests was 1 psi. This pressure level was chosen as a value which hopefully simulates the mobility of the future Apollo space suit. The suit used for the tests was a Navy Mark IV pressure suit and as a cockpit oriented suit obviously provided mobility restrictions when pressurized to 3½ or 5 psi well beyond those being designed into the Apollo garment. Investigation of the design criteria for the Apollo suit and discussions with responsible personnel have indicated that use of the Mark IV suit with 1 psi pressure simulates the mobility of what will be achieved in the Apollo suit. As a result, this 1 psi operating pressure was used since it offered restrictions which appeared to be a good compromise between those ideally sought and those which are mandatory for extra vehicular space operation. The results of the first portion of our investigation, that of high application short duration force capability is shown in figure 4. This is a busy chart and is shown only for the purpose of summarizing the data which was accumulated. Not mentioned before was a systematic lock-out of roll, pitch, yaw, and translation in an investigation of the axis which was most critical to the performance of these tasks. However, as you can see no readily apparent indication of a critical restraint direction is in evidence. However, qualitative results indicate that elimination of translation is a great aid in improving performance.

Each data point represents six task replications for two subjects and relates the push, pull and torque capability in shirt sleeves and in a Mark IV pressure suit at 1 psi.

The most significant of this data is replotted on figure 5. Here we have divided the applied push force of the variable conditions by the push force which each subject could apply in shirt sleeves with his feet planted firmly on the ground. This percentage is plotted as an ordinate with the number of trials as the abscissa. As you can see, very little difference is in evidence between the pressure suited and shirt sleeve clothed subjects. The shaded band which includes nearly all data points indicates that a subject while weightless will be able to apply a magnitude of push force between 40 and 55 percent of his capability on earth.

Figure 6 is a similar curve plotted for pull force capability. The results here are similar to those above except the band ranges here between 50 and 65 percent of earthbound capability. It is noted that this data was collected on subjects who were well trained in the simulator and who reached a learning asmptote. Initial tests on these subjects when untrained recorded forces in the order of 10 percent. This is a rather significant indication of ability to learn and adapt to this frictionless environment.

Figure 7 indicates torque values plotted in a similar fashion. These values appear to be uncommonly high as compared to those recorded for push and pull. A logical explanation of this increase suggests that because the force couple in torque was applied in the vertical plane and because our simulation provided no vertical translation freedom "cheating" was possible with unusually high values recorded. This restriction has become critical in torque even though roll, pitch, and horizontal translation freedom was simulated. As a result, the torque data recorded cannot be considered real and has consequently been rejected.

Our second test program was designed to evaluate both constant force application and dexterity. The apparatus shown on figure 8 was constructed. The panel on the left contains switches, a standard jack, a commercial valve, a half inch diameter stud and nut, a simulated electronic module, and a hatch held in place by a hinge and two standard "Cleco" clamps. A rope representing a guideline was strung between the panel and an upright ten feet away. The subject in the simulator and pressurized was placed before the panel and was required to operate the switches, open and close the valve, crack the nut and back it off the stud, run it in and tighten it to 300 inch pounds torque, remove the Cleco's, open the hatch, disconnect the jack, remove the module, place it in the vehicle, insert a spare in its place reconnect the jack, close the hatch and re-install the Cleco's. He was then, using the guideline, required to translate ten feet, turn around and return.

This task includes the majority of motions and force/time applications that are representative of general extra vehicular maintenance activity. The subject was evaluated first as to whether he was able to accomplish the task and second against the time he required to perform it.

In all cases the subjects were able, without restraint, to accomplish the task. The results of the time to accomplish are shown on figure 9. Time to complete versus task number are shown for two subjects performing 34 replications over a four-day period. Although these curves have been smoothed the actual data points diverge only slightly from the slope shown. As you can see, a classic learning curve has developed which asymptotes at approximately two minutes. The control; task performance in shirt sleeves on the floor, is one minute time to accomplish or approximately one-half the time necessary in five degrees of freedom.

Within limits of the simulation this test program has generated a set of conclusions regarding man's ability to perform extra vehicular maintenance and repair.

1. His ability to apply a push force ranges between 40 and 55 percent of his terrestrial capability.
2. His ability to apply a pull force ranges between 50 and 65 percent of his terrestrial capability.
3. Simply by using his off hand for restraint he was able to perform a relatively complex maintenance task.
4. On a time base he was approximately 50 percent as efficient in five degrees of freedom as on earth.
5. A great deal of adaptation and learning occurs in a relatively short time in this frictionless environment.

Possibly the most significant conclusion we reached is that the problem, at least from the standpoint of performance of extra vehicular maintenance and repair, is not as awesome as we had supposed. With training a subject can handle himself reasonably well, learning how to apply his mass as required to do the job at hand. While personnel restraint is obviously desirable for safety and in order to permit two hand operation, it is not absolutely necessary. Special tools also fall into this category. While they may be desirable they most probably should be task and efficiency oriented rather than designed to eliminate entirely an unbalanced load path.

In summary, we might as a result of this test program, place more confidence and consequently more reliance upon man to provide, in the future, the ability to fulfill the maintenance and repair role that will be so critical to both mission and flight safety success in long duration spacecraft.

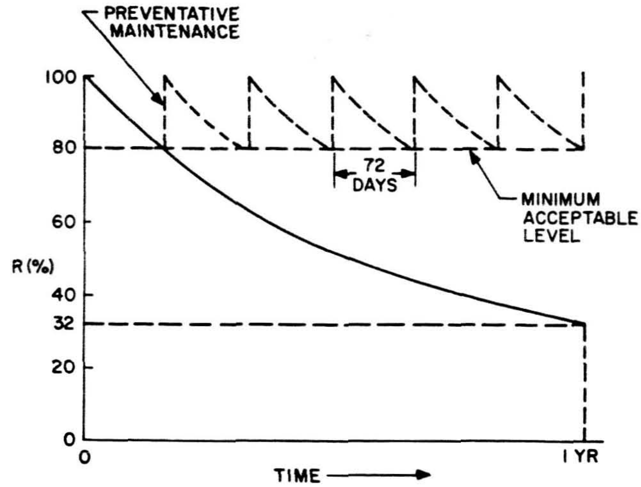

Figure 1

COMPONENT	PREDICTED DAMAGE	CONSEQUENCES	CRITICALITY
PRESSURE SHELLS	METEOROID PUNCTURE	ATMOSPHERE LOSS	8
WINDOWS & ASTRO-DOMES	METEOROID EROSION	LOSS OF VISIBILITY THROUGH WINDOWS	4
LIQUID TANKS	SPALLATION OF PRESSURE PULSE BURST TANKS	LOSS OF FUEL AND/OR SUPPLIES, FIRE, EXPLOSION	10
WINDOWS, ASTRODOMES	RADIATION DARKENING	PROGRESSIVE LOSS OF CLEAR VISIBILITY	4
PRESSURE SHELLS	WIDE SEAM CRACKS FROM THERMAL CYCLING	LOSS OF ATMOSPHERE, EXPLOSIVE DECOMPRESSION	10
ELECTRONIC CIRCUITS	SHORT CIRCUITS FROM MATERIALS SUBLIMATION AND CONDESATION ON CIRCUIT SECTIONS. ROUTINE ELECTRICAL BURNOUTS	EQUIPMENT OR CONTROLS FAILURES	9
BEARINGS	ACCELERATED WEAR AND SEIZURE DUE TO VACUUM REMOVAL OF ABSORBED GASES ON BEARING SURFACES	EQUIPMENT AND CONTROL SYSTEMS FAILURES	8
PRESSURE SEALS	THERMAL CYCLING, USE OF HATCHES, ETC. WILL CAUSE SEAL WEAR TO OCCUR PROGRESSIVELY	SLOW LOSS OF ATMOSPHERE, SUPPLIES OR FUEL	5
ELECTRONIC CIRCUITS	NORMAL ELEMENT BURNOUTS	EQUIP. OR CONTROL FAILURES	9
DOCK MECHANISM	ROUTINE IMPACTS AND MISHAPS COULD BUCKLE STRUCTURES, BEND LINK AND CAUSE WEAR OF ASSOCIATED ELEMENTS	DOCK. CAPABILITY LOSS. THIS IS LOSS OF PART OF RESUPPLY CAPABILITY	6
STRUCTURE & PRESSURE VESSELS	COLISION DAMAGES WITH FERRY, EQUIPMENT OR PERSONNEL	CATASTROPIC, EXPLOSIVE DECOMPRESSION, LOSS OF STATION	10
	METEOROID PUNCTURE OF OXYGEN TANK	LOSS OF OXYGEN, FIRE, EXPLOSION, POWER LOSS	10
FUEL CELL	LEAK IN FLUID LINE	LOSS OF FLUID, POWER DECAY	3
	INOPERATIVE VALVE, SOLENOID OR TRANSDUCER	POWER LOSS UNTIL REPAIRED, USE REDUNDANCY	4
ENVIRONMENTAL CONTROL	PUNCTURE OR LEAK IN THERMAL RADIATOR TUBING	LOSS OF COOLANT FLUID AND THERMAL CONTROL. SWITCH TO REDUNDANT CURCUIT	4
	INOPERATIVE VALVE, PUMP, SOLENOID OR CLOGGED LINE	LOSS OF THERMAL CONTROL. MAY NOT BE POSSIBLE TO USE REDUNDANCY	9
ATTITUDE CONTROL	FUEL TANK OR LINE LEAK, INOPERATIVE VALVE OR SOLENOID, CLOGGED NOZZEL, ETC.	REDUCTION IN OR LOSS OF ATTITUDE CONTROL CAUSING REDUCTION IN THERMAL CONTROL	8
GUIDANCE	GYRO BEARING "SEIZES", RADAR INOPERATIVE ETC.	GROUND TRACKING USED TO MAINTAIN LOCATION WHILE MALFUNCTION IS CORRECTED	6
COMMUNICATIONS	SHORTED ELEMENTS, OPEN OR SHORTED TRANSFORMER WINDINGS, FAULTY RELAYS, CONTACTS, BLOWN FUSES, ETC.	DISTORTION AND/OR LOSS OF SIGNAL. INABILITY TO TRANSMIT, ETC.	7

Figure 2

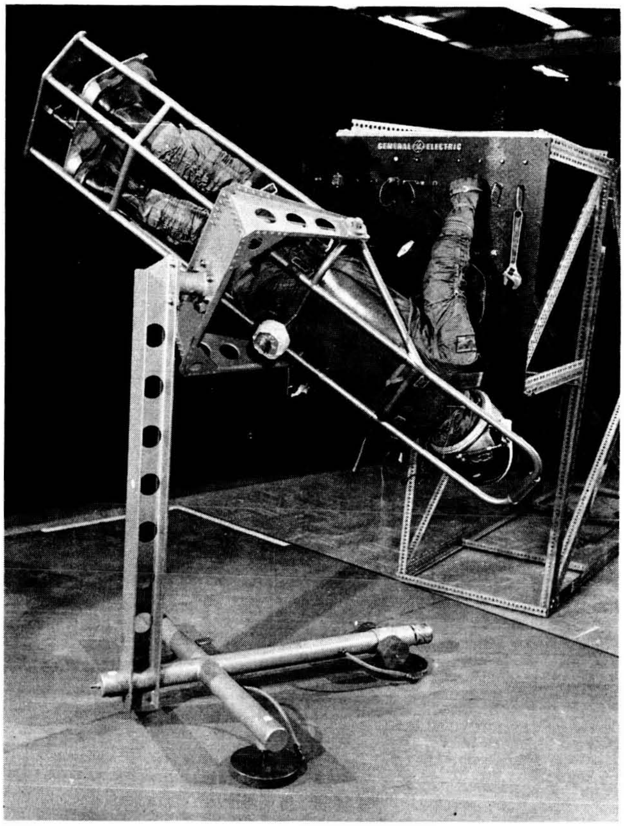

Figure 3

SHIRT SLEEVES

CONSTRAINTS	NO RESTRAINT			ONE HAND RESTRAINT		
	PUSH-#	PULL-#	TORQUE INCH-LBS	PUSH-#	PULL-#	TORQUE INCH-LBS
GROUND	170	221	311	—	—	—
5° FREEDOM	50	93	274	75	115	287
ROLL	65	101	277	75	115	271
PITCH	87	85	237	101	115	289
TRANSLATION	94	115	268	108	143	299
PITCH & ROLL	72	115	278	109	130	292
PICTH & TRANS.	81	125	271	117	138	309
ROLL & TRANS.	93	137	271	128	137	292

PRESSURE SUIT AT 1 PSI

CONSTRAINTS	NO RESTRAINT			ONE HAND RESTRAINT		
	PUSH-#	PULL-#	TORQUE INCH-LBS	PUSH-#	PULL-#	TORQUE INCH-LBS
GROUND	162	219	307	—	—	—
5° FREEDOM	71	120	236	86	129	250
ROLL	79	116	282	80	147	280
PITCH	71	96	226	83	103	238
TRANSLATION	69	100	232	83	114	272
PITCH & ROLL	84	113	244	107	116	287
PITCH & TRANS.	73	96	243	73	95	250
ROLL & TRANS.	77	97	247	81	112	258

Figure 4

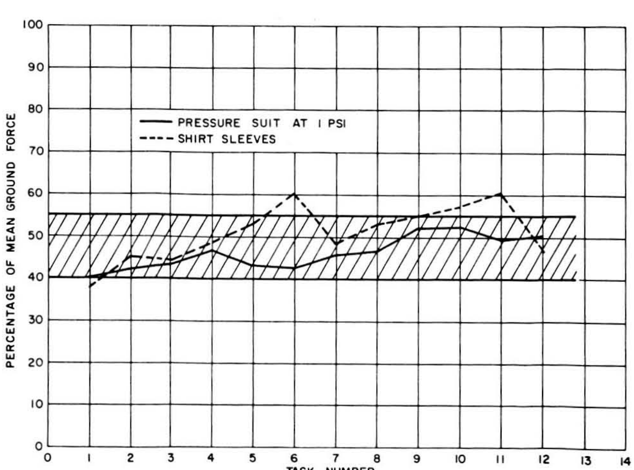

Figure 5

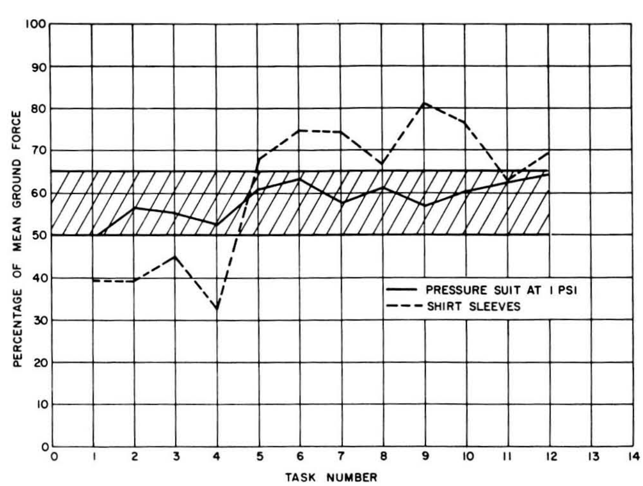

Figure 6

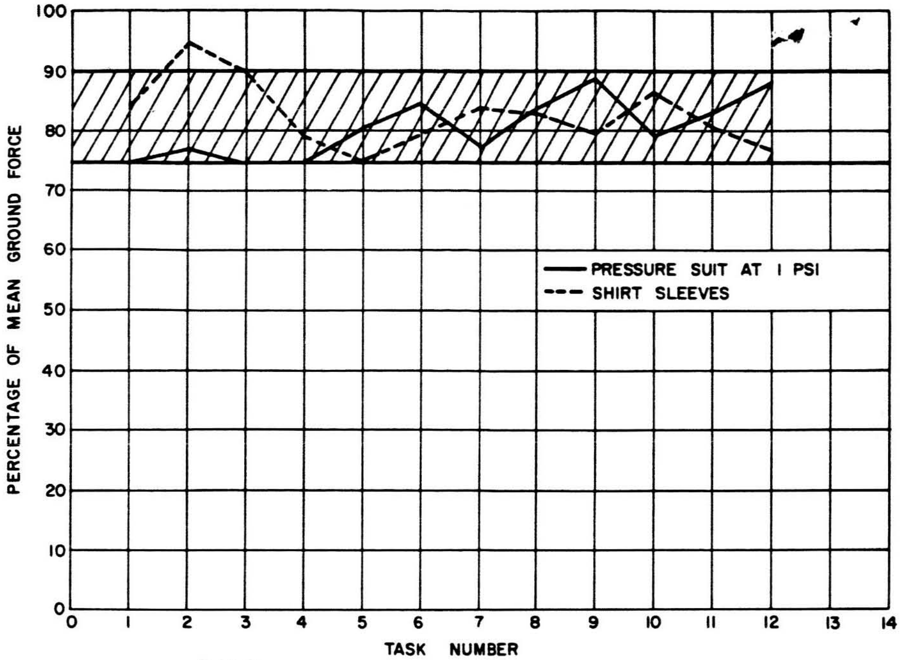

Figure 7

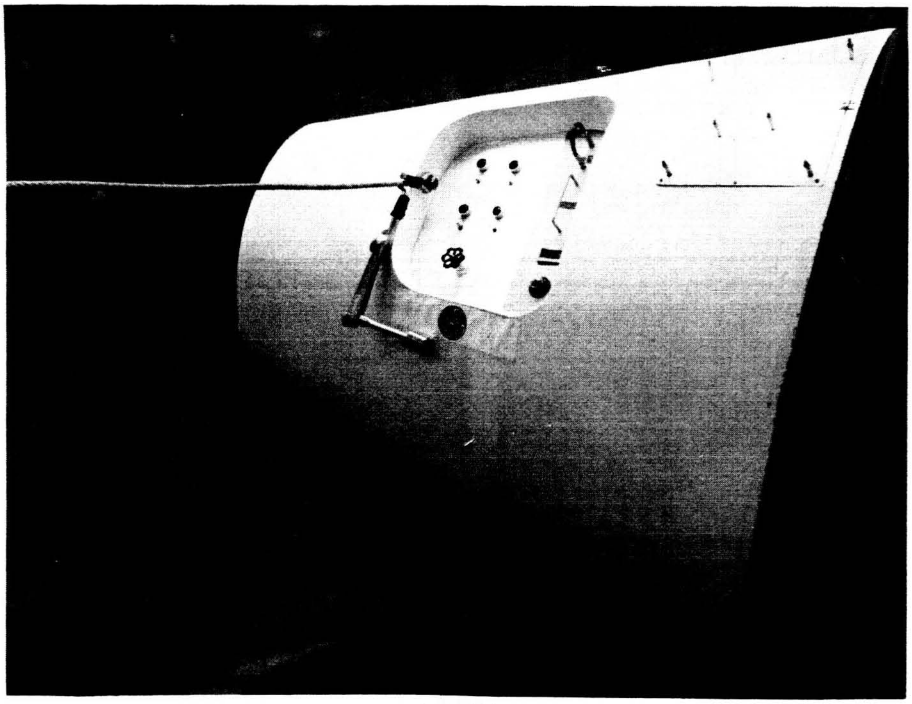

Figure 8

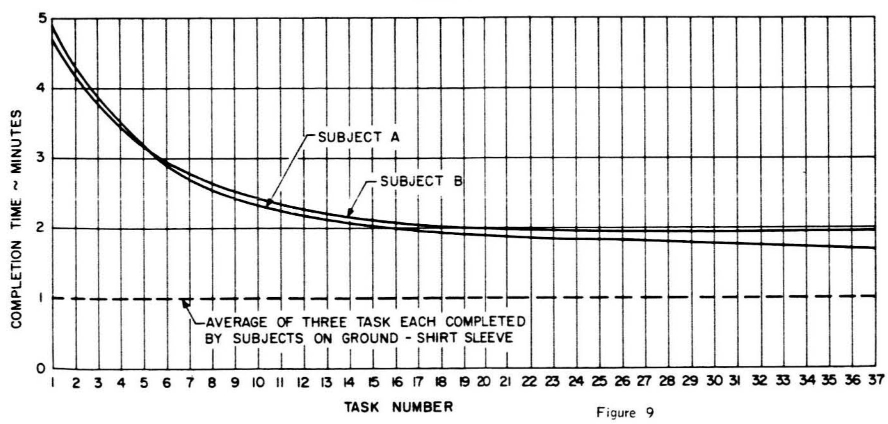

Figure 9

A VISUAL PRESENTATION SIMULATOR USED FOR INVESTIGATION OF VARIOUS PHASES OF LUNAR FLIGHT

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ABSTRACT 23214

This paper describes a Visual Presentation Simulator used for man-in-the-loop simulation investigations of various phases of the lunar landing mission. This simulator consists of a full-scale fixed-base cockpit mockup with operating instruments, a closed-circuit television system, a terrain model, a servo-powered six degree of freedom TV camera transport, and an analog computer.

Specific techniques and equipment used to enhance the realism and to adapt the simulator to various phases of the mission are discussed. Typical results which illustrate the type of system characteristics that can be investigated are presented and discussed.

Finally, on the basis of recent experience in simulation, refinements and new developments that may be expected in man-machine simulation techniques in the near future are presented.

AUTHOR

INTRODUCTION

Some of the most demanding problems anticipated in the lunar landing program are those associated with the deorbit and landing, and with the rendezvous and docking phases. Current thinking is to allow the astronauts to participate extensively in these phases, i.e. handle the guidance and control manually, with any automatic systems being used for reference or backup.

In any event, these phases of the mission are intended to be independent of outside aids, thereby giving the astronauts as much mission flexibility as possible, and raising their confidence level by establishing their ability to conduct the mission "on their own". The high degree of reliability required for those systems affecting crew safety adds further emphasis to this approach inasmuch as this approach generally allows the use of simpler and more reliable mechanical systems as opposed to more complicated electronic systems.

The success of a manned soft landing on the moon depends on the astronaut being able to select a suitable landing site, bring the vehicle to this site, and actually touch down on the surface within the design limits of the vehicle. Similarly, the success of the lunar orbital rendezvous concept depends on the astronaut's ability to perform a rendezvous and subsequent docking maneuver with the orbiting command module.

These tasks are significantly different from those which astronauts, and pilots in general, have performed to-date. For instance, current experience with vertical landing and take-off vehicles has been necessarily restricted to the earth's environment, which has a considerably different gravitational and atmospheric environment than the moon. These two factors alone make extrapolation of earth-operating VTOL criteria to lunar vehicles quite unsound. And added to these two factors is the extreme criticality of fuel consumption for the lunar vehicle.

With the astronaut being such an integral part of this lunar landing vehicle system, and the required tasks being of such an unusual and critical nature, it has become apparent that man-in-the-loop simulation must be used to a further extent than ever before. Only by making the man as integral a part of the system as possible can the over-all system be studied and refined. Analyses which do not take full account of the astronaut's capabilities and limitations under the actual conditions to be encountered will not possess the degree of validity or reliability required for this important mission.

The purpose of this paper is to

(1) describe typical man-in-the-loop simulation investigations of the previously noted phases of the lunar landing mission that have been conducted using a unique Visual Presentation Simulator,

(2) describe specific techniques and equipment used to enhance the realism of the simulation,

(3) indicate the type of system characteristics that were investigated and the nature of results that were obtained, and finally,

(4) indicate what improvements might be expected in man-machine simulation techniques in the near future.

Simulator Description

The investigations were conducted utilizing a Visual Flight Simulator shown in Figure 1. This simulator consists of a full-scale fixed-base cockpit mockup with operating instruments, a closed circuit television system, a terrain model, a servo-powered six degree of freedom TV camera transport, and an analog computer which contains the appropriate equations. In order to "fly" the simulator, the astronaut operates the controls in the cockpit, which produce proportional voltages that are transmitted to the analog computer. The computer then solves the equations of motion for the parameters describing the flight path and attitude of the vehicle. These parameters are sent to the TV camera transport, where they position the camera in the 3 translational axes and the 3 attitude angles above the terrain, and to the instrument panel where the information is displayed on the appropriate instruments. A diagram of the information flow is shown in Figure 2.

The basic cockpit mockup used for the investigations is shown in Figure 3. It is a full-scale fixed-base mockup, representative of a two-man lunar landing vehicle. It will be noted that the cockpit configuration differs for the two sides; this merely reflects the fact that the forward vision limitations can be changed readily from one investigation to the next.

The TV projector is also visible in Figure 3. This is a Schmidt type projector whose position can be readily changed to suit differing simulations. Generally, the projector nearly fills the 12' x 16' screen, resulting in a field of view to the astronaut of about 50°. It is recognized that this field of view is less than desired; however, it is a limitation which we feel is not so severe as might be thought at first. Note that in the case of the docking simulation, the astronauts attention is directed to one specific limited area, so this field of view presents no real limitation.

Also shown in the right hand corner of Figure 3 is a starfield projector. This picture is superimposed on the TV picture in the docking simulation investigations, as will be discussed later.

The cockpit instrument display is shown in Figure 4 for one of the earlier studies. Note 1) the relatively large number of instruments available, 2) the use of tape instruments to obtain the required accuracy for altitude, altitude rate, range and range rate, and 3) the Mercury type attitude - attitude rate indicator in the center. A modification of this same instrument display, as used in a later study, is shown in Figure 5. Note that the Mercury attitude instrument has been replaced by a conventional "8-ball" type (All Attitude Indicator), and that considerably fewer instruments are used in this investigation.

The cockpit controls generally consist of a right hand side stick or pencil stick for pitch and roll control, and foot-operated treadles for yaw control. The left hand throttle provides thrust control. Considerable variations in these controls are used; particularly in the docking investigations, where the left hand control becomes a 3-axis translation controller. An intercom system is also provided to the astronaut.

Figure 6 shows the terrain model and TV camera transport in position. The servo-driven camera transport assembly controls the attitude and position of the TV camera, and thereby provides a continuously moving real-time picture of the terrain model. The camera's six degrees-of-freedom have the following limits:

Translation			Angular Attitude	
Longitudinal,	x,	20 feet	Roll, ϕ ,	$\pm 60^{\circ}$
Lateral,	y,	6 feet	Pitch, θ ,	$\pm 20^{\circ}$-95°
Vertical,	z,	4 feet	Yaw, ψ ,	$\pm 270^{\circ}$

More detailed characteristics of the camera transport system, including maximum rates and frequency responses, can be found in Reference 1.

The terrain scales vary depending on the particular study; in the case of the deorbit phase, the scale is such (40,000:1) that considerable curvature is evident in the model. The terrain is installed in a removable cart 10 feet by 32 feet long.

The modeling of the lunar surface is somewhat arbitrary, particularly for the smaller scales, since the area simulated presents no terrain features within the resolution limits of current lunar topography. Accurate modeling is thus limited to distant features such as craters and mountains along the horizon. Minor craters and other local effects are cut and colored according to the latest information available and cemented in position.

The over-all results of this simulation setup are presented in Figure 7, which shows the astronaut approaching a touch down next to a "reference" space vehicle.

For adaptation to rendezvous docking studies, a model of the command module is substituted for the terrain, with the background being completely black. A picture of this is shown in Figure 8. With this black background, a starfield can now be superimposed on the TV picture. This projector is also servo-operated and controlled by signals from the analog computer. Its position was seen in Figure 1 and 3. The over-all effect can be seen in Figure 9.

The lack of motion in the cockpit is treated briefly in Reference 2 and will be discussed later; however, in general, it can be seen that the simulation is a rather complete one. The astronaut receives visual cues externally from the TV and also, in some cases, the starfield display; visual cues internally from any of a large number of instruments; force and motion "feel" from the various control sticks or pedals; and, in some cases, auditory cues via the intercom head set and engine noise generator. The over-all effect is to make the man as much a part of the system as possible, by simulating as many of the significant factors as practicable.

Test Procedure

Engineering test pilots or astronauts were used during the investigations. The subjects generally had previous experience in flight simulators. Prior to a series of runs, each of the subjects was given a cockpit orientation session which included the following points:

(1) The purpose of the experiment.

(2) Control modes being used.

(3) Objective to be accomplished.

To enable the pilots to accurately rate the suitability of the vehicle's handling qualities, a nominal task requiring precision control was established. The subject was then given a few practice runs to become familiar with the system and then he performed a specific series of runs having a combination of one of the control modes and/or instrument panel variations. Upon completion of these runs each subject went through a recorded debriefing session for recommendations and opinions.

While pilot performance was recorded and analyzed, the actual determination of the suitability of the characteristics being investigated was based largely upon pilot opinion. The standard NASA Cooper scale (Figure 10) was employed as the pilot rating basis, and regions of satisfactory, acceptable and unacceptable characteristics were established.

Specific Investigations and Typical Results

Deorbit Phase (From orbit to hover) - The purpose of this investigation was to determine the astronaut's capabilities of performing this phase of the mission under full manual control and with minimum pilot aids and instrument displays. Different deorbit schemes, control modes, and instrument panel displays were investigated.

The results indicated that the astronaut should be able to perform this phase with relatively simple displays and controls, and that the propellant used was only slightly higher (approximately 5%) than for a "perfect steering" or automatically controlled trajectory.

Regarding the instrument displays, the subjects in general preferred the simpler panels and were equally as successful in flying the mission with the "minimum" or "essential" panels than with the "plush" panel. The "plush" panel is shown in Figure 4, and represented what the first "open loop" engineer's analysis indicated the astronaut would need. The "minimum" panel (obtained by masking this panel) by comparison, consisted of about one-half this number of instruments.

A conclusion derived from this phase of the study concerning the external visual display was that the astronaut, looking straight down at the surface of the moon (as he is scheduled to do from deorbit to roundout) cannot really utilize the external visual information available to him. This was primarily due to lack of the normal horizon reference, and required the pilot to scan his attitude indicators and his out-of-orbit plane indicator at a much higher rate than would be used under normal "VFR" conditions. Thus, if the deorbit to roundout portion of the landing mission is to rely on the visual flight conditions, the pilot must be able to see the horizon more readily than by looking overhead.

Three different control system modes were investigated: acceleration command, rate command, and attitude hold. The evaluations of acceleration command controls were conducted at the beginning of the deorbit maneuver. The subjects were unable to control the vehicle for longer than 10 seconds with the acceleration levels used (full control maximum acceleration as low as $7^{o}/sec^{2}$ in pitch and roll.) It was therefore concluded that at this stage in the training period, the acceleration command control system, operating simultaneously about all 3 axes at the level noted, was unsatisfactory. Evaluation of the rate command and attitude hold control systems indicated there was little difference between them, and that they were both acceptable.

Final Approach and Touchdown Phase - The purpose of this investigation was to determine the control requirements in the hover, translation and vertical touch down phase.

Attitude command, rate command, and acceleration command systems were investigated. More detailed characteristics such as dead zones and thrust misalignment effects were also investigated, as well as the effects of instrument failures.

Performance data recorded consisted of RMS error during hover, time to hover, total flight time, range and cross-range displacement, and velocity and attitude angles at touch down. However, primary conclusions are again based upon pilot opinion rating.

The pilot opinion ratings indicated that the attitude command control system was superior for performing the given task.

The quantitative performance measurements, one of which is shown in Figure 11, substantiated this. This figure shows that the time (and hence, fuel) to establish hover at the desired landing point, is lowest for the attitude command system.

The effect of a dead zone about all 3 axes is shown in Figure 12 for a rate command system. It can be seen that only a $2\frac{1}{2}$% internal dead zone is sufficient to cause a deterioration in pilot rating from Satisfactory to Unsatisfactory but Acceptable, a significant change. This particular result is significant because it is the type of effect that is not amenable to analysis, but must be obtained with man-in-the-loop simulation of a representative task.

Since the standard maneuver was essentially an instrument task, several runs with rate command systems were conducted in which the pilot's landing reference was obtained visually. While it was difficult for the pilot to get exactly over the intended site (since downward visibility disappeared as the target was approached), the standard deviation for all landings was reasonable. This shows that visual performance could, in general, match or exceed, with the proper presentation, the touch down performance of the instrument landings.

In addition to visual approaches, control and instrument failures were simulated. The control failures were conducted with nominal rate command systems and progressively failed in one and two axes. At the time of failure, the control system in the failed axis reverted to a proportional acceleration command system. Instrument failures consisted of flying first with position and velocity information covered over, and second, with no instruments at all. It was concluded from these studies that pure visual and partial instrument panel runs could, with adequate learning time, be flown nearly as well as the complete instrument panel cases.

Here again is a truly significant conclusion, available with confidence only through the tool of man-in-the-loop simulation.

Docking Phase - The purpose of these studies was to evaluate the astronaut's ability to accomplish this lunar orbital docking task with no automatic mechanical system. Here again, the attitude control system modes investigated were attitude hold, rate command, and acceleration command. The translational control system was an "on-off" system. Translational thrust level, system deadband, and cross-coupling effects were also investigated.

For this investigation, it was necessary to convert the left hand throttle to a 3 axis translational control, as previously mentioned. Also, a starfield projector (Figure 1 and 3) was added; this enables visual attitude cues to be provided by projecting stars at infinity. Were these stars to be on the command module background, they would change angle as the camera moved into the command module model during docking. This starfield also allows angular motion of the command module in space to be simulated. For example, limit cycle motions (considered to be likely) of the command module or target vehicle could be introduced.

A significant refinement to the simulation was devised during the investigations. This consisted of mounting a portion of the pilot vehicle on the TV camera in the field of view; the astronauts task was then to mate the two bodies in the TV picture. This innovation essentially overcame the TV picture's lack of three-dimensional characteristics.

One result of this investigation was that the subjects were quite reluctant to use the treadles for yaw control; it was strongly recommended that a 3 axis attitude stick be used.

As in the previous phases, the control mode preferred was attitude command. However, in this case the acceleration command system could be made flyable as a backup if a low enough level of thrust were used.

One interesting aspect of this mode was that one astronaut considered that noise cues, supplied to indicate the operation of the thrust motors, were extremely helpful in certain control modes.

Pilot opinion was employed in the major part of the study as the primary criterion for determining the acceptability or unacceptability of the control system characteristics investigated. However, performance parameters were recorded and analyzed. A summary of performance data for the translational characteristics is presented in Figure 13 and indicates a reasonably good correlation between the performance and pilot opinion.

Significant data were obtained on the degradation of system rating due to cross-coupling effects (c.g. offset from thrust). In this case, it appeared that c.g. offset in the acceleration command mode would be difficult to handle without considerable training.

One final comment on this particular phase is in order. Besides being the best way of obtaining the required analytical results of the man-vehicle system capabilities for system design purposes, the simulation setup used was considered by the subjects to provide first-class training for conditions of space rendezvous and docking.

Concluding Remarks

It is hoped that the preceeding discussions have provided insight into the way that man-in-the-loop simulation can be used to help solve the engineering problems of the man-machine system which will soon be landing United States astronauts on the moon. It is realized that the simulation techniques and devices discussed are far from the ultimate, and that we will in a few years look back and consider many of these techniques obsolete.

Some of the significant improvements that might be expected to do this can be listed as follows:

(1) Wider angle projection systems - whether TV or real-time optical systems.

(2) Use of color and higher resolution (if TV) well beyond the current levels.

(3) Obtaining of a 3-dimensional effect.

(4) Introduction of motion "cues" to the astronauts; though adding considerable bulk, thus can be done with present state-of-the-art. (The cues would be limited to the initial acceleration, and then washed out at a rate just below "threshold").

(5) Addition of physical environmental factors such as appropriate pressure, temperature, radiation and acoustic levels; this would, for example, require the astronauts to wear and operate their "pressure" suits.

(6) Requiring the astronaut to perform the noted tasks in their proper time period - i.e. after being space-borne for several days, and confined with the other astronauts.

As each refinement is added, be it those noted above or others, the manned lunar landing program will utilize it and continue to demand even more improvements. This lunar program will depend to an increasing extent on comprehensive, detailed man-in-the-loop simulation investigations such as have been outlined. This dependency will promote the rapid development of even more realistic and sophisticated simulation devices and techniques.

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2. A'Harrah, R. C. and Kwiatkowski, S. F., "A New Look at V/STOL Flying Qualities", IAS Paper No. 61-62, January 1961

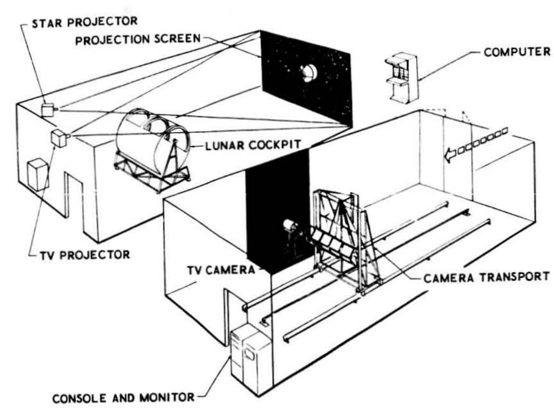

FIG 1-VISUAL FLIGHT SIMULATOR

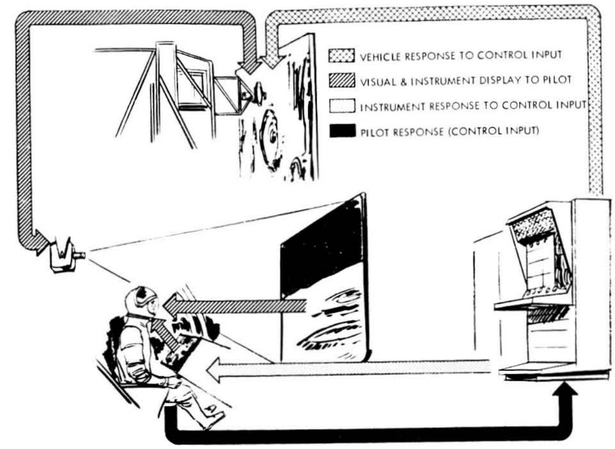

FIG 2-SIMULATOR INFORMATION FLOW

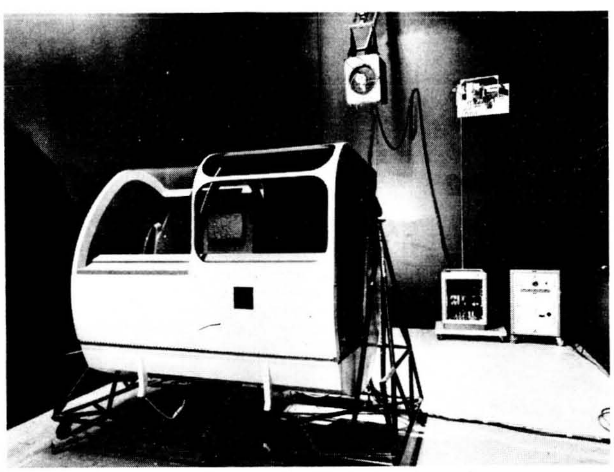

FIG 3-COCKPIT MOCKUP
COCKPIT, TV-PROJECTOR AND STAR PROJECTOR

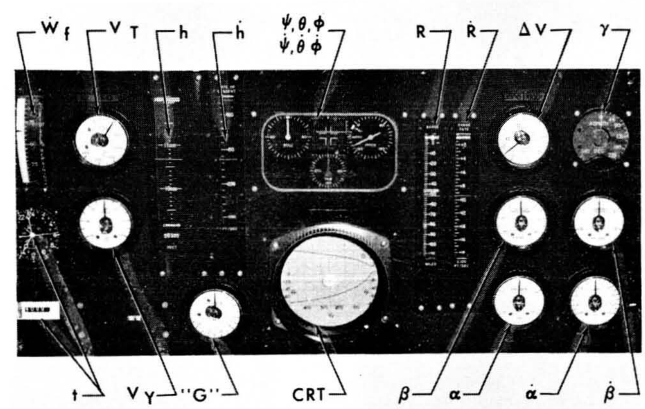

FIG 4-INSTRUMENT DISPLAY

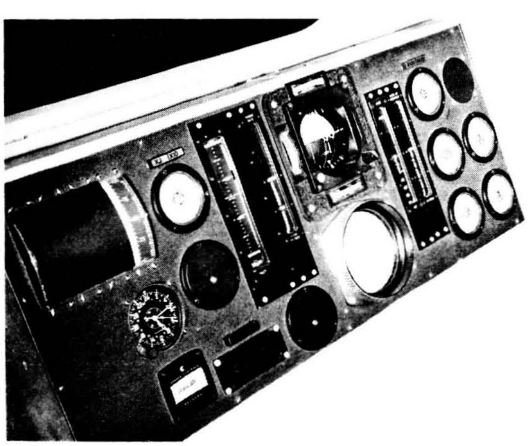

FIG 5- MODIFIED INSTRUMENT DISPLAY

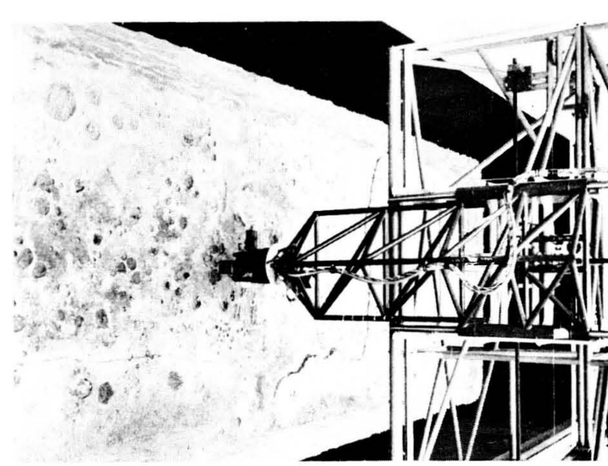

FIG 6-CAMERA TRANSPORT AND LUNAR TERRAIN MODEL

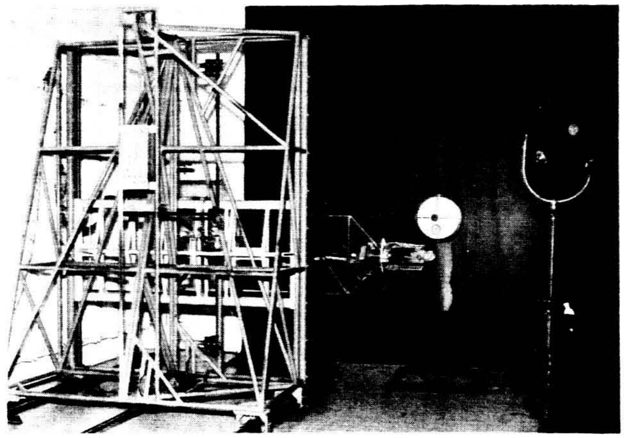

FIG 8-TV CAMERA RIG TRANSPORT AND COMMAND MODULE

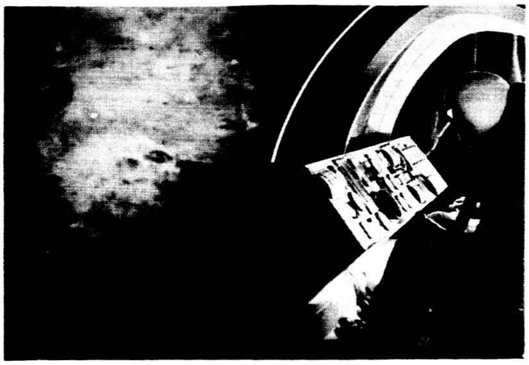

FIG 7- VIEW OF VISUAL PRESENTATION
COCKPIT AND TV-SCREEN

	ADJECTIVE RATING	NUMERICAL RATING	DESCRIPTION	PRIMARY MISSION ACCOMPLISHED?	CAN BE LANDED
NORMAL OPERATION	SATISFACTORY	1	EXCELLENT, INCLUDES OPTIMUM	YES	YES
		2	GOOD, PLEASANT TO FLY	YES	YES
		3	SATISFACTORY, BUT WITH SOME MILDLY UNPLEASANT CHARACTERISTICS	YES	YES
EMERGENCY OPERATION	UNSATISFACTORY	4	ACCEPTABLE, BUT WITH UNPLEASANT CHARACTERISTICS	YES	YES
		5	UNACCEPTABLE FOR NORMAL OPERATION	DOUBTFUL	YES
		6	ACCEPTABLE FOR EMERGENCY CONDITION ONLY*	DOUBTFUL	YES
NO OPERATION	UNACCEPTABLE	7	UNACCEPTABLE EVEN FOR EMERGENCY CONDITION*	NO	DOUBTFUL
		8	UNACCEPTABLE - DANGEROUS	NO	NO
		9	UNACCEPTABLE - UNCONTROLLABLE	NO	NO
	CATASTROPHIC	10	MOTIONS POSSIBLY VIOLENT ENOUGH TO PREVENT PILOT ESCAPE		

* (FAILURE OF A STABILITY AUGMENTER)

FIG 10- PILOT OPINION RATING SYSTEM FOR UNIVERSAL USE

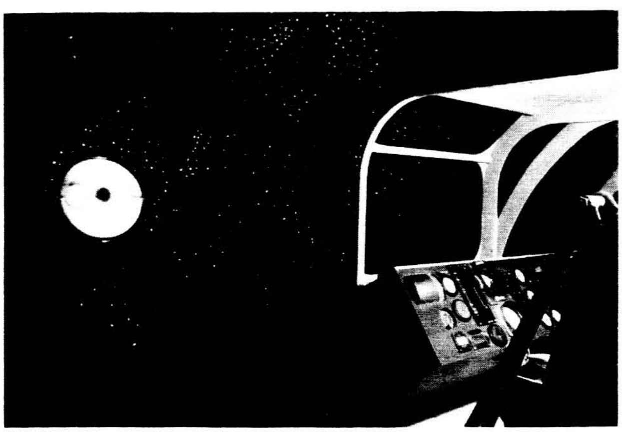

FIG 9 -VISUAL PRESENTATION SIMULATOR ROOM
COCKPIT AND TV-SCREEN

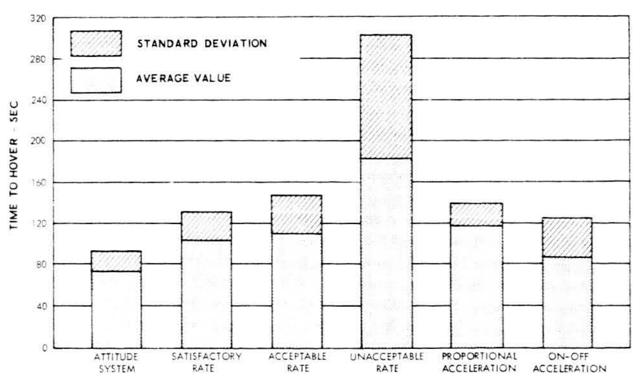

FIG 11-SYSTEM PERFORMANCE
TIME TO HOVER

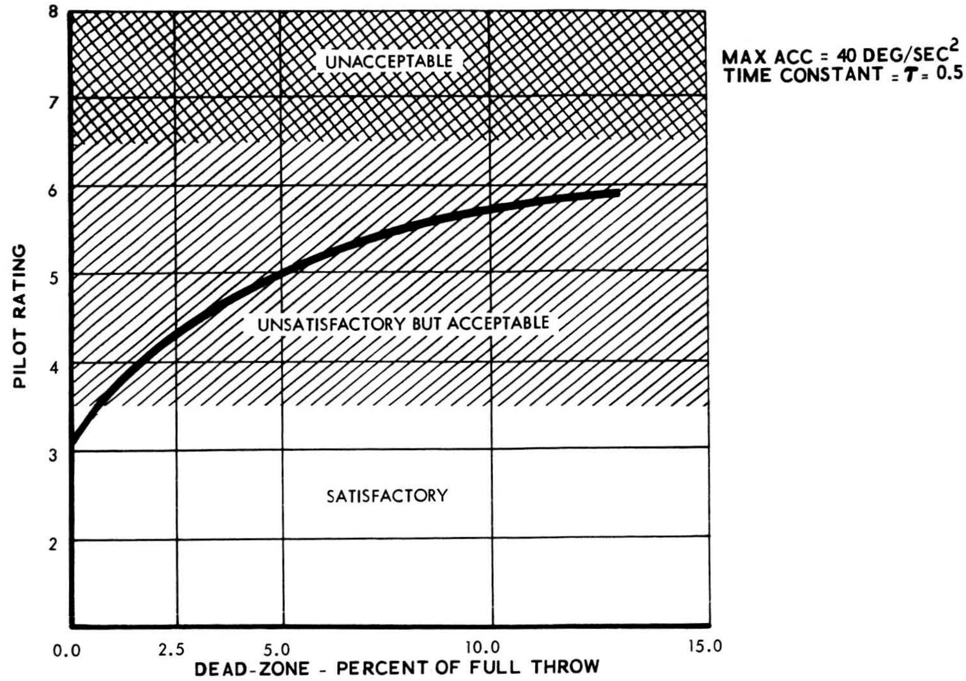

FIG 12-EFFECT OF CONTROL SYSTEM DEAD-ZONE

SATISFACTORY ATTITUDE RATE SYSTEM

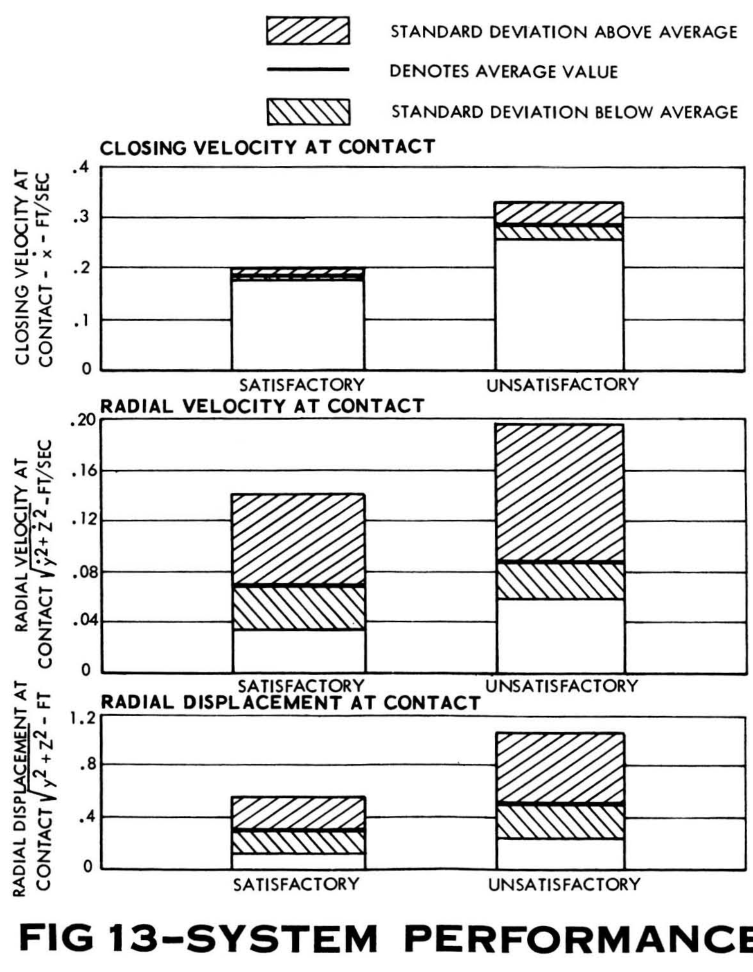

FIG 13-SYSTEM PERFORMANCE

EFFECT OF MATERIALS ON ATMOSPHERIC CONTAMINATION IN MANNED SPACECRAFT

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ABSTRACT

When selecting materials for use in manned spacecraft, outgassing and flammability characteristics must be given greater consideration than when the same materials are used in earthbound applications. In the confined atmosphere of a space cabin the outgassing products from surface coatings, electronic equipment, plastics, and other materials, may accumulate to toxic levels. In the event of fire, the atmospheric contamination may rapidly become serious. The sudden contamination could overload the air purification system creating a potential hazard to the occupants of the cabin.

Information on the relative atmospheric contaminating potential (ACP) of various materials is needed in the design of spacecraft that are to be occupied by man for long periods under "shirt sleeve" conditions. ACP values will aid in the choice of materials to be used and help establish the requirements of air purifying equipment.

This paper describes an experimental program which will determine the relative outgassing and flammability characteristics of over 100 materials, including surface coatings, wire insulations, tying cords, molding compounds, adhesives and casting compounds. Experimental apparatus and analytical techniques are described.

Preliminary results obtained from the experimental testing programs will be presented with special emphasis on flammability characteristics of materials in pure oxygen.

AUTHOR

INTRODUCTION

This paper describes a program at Honeywell to determine the suitability of a number of materials for use in spacecraft that are to be occupied by men under "shirt sleeve" conditions for prolonged periods. Many materials, especially plastics, give off quantities of volatile substances into the atmosphere. On planet Earth these contaminants are diluted in our atmosphere to such a low concentration that they do not present a problem. However, in the confines of a space vehicle these materials accumulate and may reach toxic concentrations.[3] In the event of a fire, the buildup of a harmful atmosphere is very rapid and especially serious.[2]

This paper describes the equipment and procedures used to measure the atmospheric contaminating potential (ACP) of various materials such as surface coatings, adhesives, tapes, wire insulations and molding compounds. The measurements are made under two conditions: (1) simulated use conditions at 200°F, and (2) simulated accidental fire.

Knowing the relative potential for atmospheric contamination and fire hazard is useful for two reasons. First, materials having low ACP values can be selected and second, contamination levels in the atmosphere of the spacecraft can be estimated from the amount and type of materials that are used. This estimate will aid in determining the requirements and capacity of the air purifying system for the vehicle.

The Navy has pioneered studies on the problems of living in the closed environment of submerged submarines.[3, 19, 20] They have made significant advancements in the selection and development of paints, plastics, and other materials having low ACP characteristics.[7] They have found ways of removing carbon dioxide, carbon monoxide, hydrocarbons and other contaminants from the submarine atmosphere,[3] and at the same time, they have learned how to minimize contaminants. For

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example, restrictions are placed on living habits of the crew. Crew members are not permitted to use aerosol type shaving creams because of contamination caused by the Freon propellant. The hobby kits brought aboard to occupy the sailors' leisure hours must not contain lacquers, solvent based adhesives or other materials having high ACP values.

Work on the problem of contaminants accumulating in closed environments was done when materials were selected for use in the Project Mercury Capsule.[6] For this program, a simple but adequate test procedure was used to choose materials for the relatively short trips planned for Mercury. Test materials were sealed in a desiccator under 5 psi oxygen pressure for 24 hours at various temperatures. The atmosphere in the desiccator was swept through an activated charcoal trap cooled in a dry ice-alcohol bath using helium or nitrogen as carrier gas. The contaminated charcoal was placed on a watch glass in a covered one cubic foot aluminum box and heated on a 300°F hotplate. The odor evolved in this box was evaluated by having at least three persons sniff the air in the aluminum box. Each material was rated satisfactory or unsatisfactory on the basis of the intensity and type of odor.

For longer space missions, it is natural to extend the Mercury test procedure to find out what the outgassing products are and in what quantities they are evolved. This is the objective of our present program.

In this presentation it is convenient to divide the subject into two sections: outgassing studies and flammability studies.

OUTGASSING STUDIES

Honeywell's outgassing tests consist of sealing test material in a glass chamber under 5 psi oxygen and about 10 psi helium. The sample is then heated and, after a specific period, gas samples are withdrawn for analysis. Figure 1 is a photograph of the equipment and Figure 2 shows a diagram of the apparatus we use for this test. The test specimen is suspended from a stainless steel support. A Pyrex chamber is put in place and the system is sealed by turning a threaded ring into a rubber "O" ring, forcing the "O" ring to make a vacuum-tight seal against the glass. The system is evacuated to about 1 mm pressure and then filled with pure oxygen to a pressure of 5 psi. Helium is added until the total pressure in the system reaches ambient atmospheric pressure. The purpose of the helium is to minimize the effect of possible leaks. Over a period of 48 hours even a small leak could appreciably alter the composition of the sealed atmosphere. The sample is then heated to 200 ± 10°F.

The temperature of the trap outside of the oven is kept at approximately room temperature. The temperature differential between the part of the system inside the oven and the trap on the outside causes convective circulation. It is estimated that the total volume of gas circulates through the trap in somewhat less than one hour. Any outgassing products which can condense at room temperature collect in the trap. After about 48 hours, samples of the enclosed atmosphere are withdrawn with a hypodermic syringe for analysis and odor classification.

Carbon monoxide and carbon dioxide are determined using a gas analysis kit.[22] The total organic content is determined by passing a measured volume through a flame ionization detector. The detector has been calibrated using pentane and the total organic content is reported as parts per million pentane equivalent. The moles of CO_2, CO and organics per gram of material are calculated for each contaminant as the ACP of the material. The odor is classified by having three persons carefully sniff the sample in the syringe. The odor is given a numerical rating of zero to three. Zero indicates no detectable odor; one, detectable but not objectionable; two, mildly objectionable, but tolerable for short periods; three, intolerable for even a short period.

Materials that are eliminated because of high carbon monoxide, carbon dioxide, or total organic content or because they have an objectionable odor are not tested further. The better materials, or those for which no substitutes are presently available, are analyzed further using gas-liquid chromatography.[5] Figure 3 shows the gas chromatograph used in the studies. Figure 4 is a sample chromatcgram.

Chromatograms typically contain 10 to 30 peaks. We are now in the process of identifying major peaks by comparing retention times with that of known materials. Three different chromatographic columns are being used in which the liquid phases are apiezon, carbowax and didecyl phthalate, respectively. If a good correlation is obtained between the retention times of an unknown peak with that of a known compound the unknown is identified with reasonable certainty. In cases where more positive identification is desired the material in a given peak is collected from several cycles of the chromatograph, sealed in a capillary "U" tube and submitted for mass spectrograph analysis.

Table I gives outgassing test results for several plastic materials. A quantitative value for the amount of a given contaminant produced per unit weight of material is defined as the ACP of the material. ACP is expressed as moles of contaminant per gram of material. When using this data, the test conditions under which it was obtained must be kept in mind. The 200°F temperature is higher than the anticipated use temperature, but was chosen for this work to accelerate outgassing and provide a margin of safety. Volatiles contained in the plastics come off most rapidly at the beginning of the test and should be coming off at a much reduced rate after 48 hours. Products that are produced by chemical reaction, such as CO, CO_2, and thermal degradation products, would be expected to continue coming off as the test is continued longer. The possibility of reactions having long induction periods is a possibility that should also be considered. Therefore, great care should be used in extrapolating these data to temperatures or time periods different from that of the test.

Note that carbon monoxide is frequency produced under these relatively mild conditions. Some materials, such as the polycarbonate and the nylon molding compounds, increased in weight, suggesting that oxygen combines with the polymer molecule instead of producing volatile oxidation products.

FLAMMABILITY TESTS

Figure 5 shows the flammability test equipment. This test consists of heating the test material under standardized conditions and noting the time required for ignition and the characteristics of the fire. The atmosphere is analyzed for toxic combustion products. The equipment and procedure is a modification of that used by the Navy in evaluating materials for use in submarines.[22]

Figure 6 schematically shows the equipment plumbing. A Pyrex bell jar 18 inches in diameter and 18 inches high standing on an aluminum base provides a closed environment. The composition and pressure of the atmosphere in this environment are obtained by evacuating the system to about 1 mm pressure and then back filling with the desired gases. Most of our tests were made in an atmosphere of pure oxygen at 5 psi (which is one of the atmospheres considered for use in spacecraft). A few tests were made in an atmosphere of 80 per cent nitrogen and 20 per cent oxygen to determine how atmospheric composition and pressure affect test results.

Two test setups were used. The first, shown in Figure 7, was for materials that could be molded or cut into a bar form. Molding compounds, casting resins, and bulk plastics were usually made into one-half inch square bars, five inches long. The test bar was set on a pedestal and centered within a 1.5 inch diameter nichrome heating coil. Thermocouples are attached to both the test bar and the heating coil. Two spark plugs are positioned about 0.125 inch from the test bar, and 0.5 inch above the coil. The purpose of the spark is to ignite combustible gases close to where they are formed to prevent dangerous accumulation of explosive mixtures.

The second type test setup, Figure 8, was for materials such as surface coatings, tapes, tying cords and solvent based adhesives. This setup consists of an aluminum tube 1.25 inches in diameter and nine inches long with 0.020 inch thick wall. These tubes are coated following the different specifications for each surface coating material. Tapes, tying cords, and other thin film stock are wrapped onto the tube. The tube is then placed on a pedestal and centered within a nichrome coil 2.25 inches in diameter. Thermocouples and spark electrodes are adjusted as with bar specimens.

The following test procedure then takes place: 1). The test specimen (bar or tube) is in position; 2). the desired atmosphere in the system is attained; 3). the spark is turned on; and, 4). an electric current is passed through the heater coil. A current of 50 amperes is used for the smaller coil and 56 amperes for the larger. The coil reaches a temperature over 800°C (1472°F) in 100 seconds. The interval between turning on the current and the time the sample starts burning is recorded as the ignition time for that particular material. Figure 9 shows a very active sample of burning material. This particular sample was an epoxy adhesive.

Continuous records of the coil and sample temperatures are made with Honeywell Electronik 17 recorders. The coil temperature record indicates how constant the heating cycle is from test to test. The "sample temperature" is not a true measure of the surface temperature of the sample, but it is useful in recording ignition time. When the sample ignites, the slope of the temperature versus time curve increases sharply, providing an automatic record of ignition time.

The system pressure is read immediately after the burn. Two factors will affect this pressure:

1. An increase or decrease in the number of moles of gas in the system;
2. An increase in temperature within the system from the coil heat and the heat of combustion.

The reading immediately after the burn is a more or less complex combination of these two effects, but it does give a qualitative value of pressure change due to non-explosive fires in a constant volume environment. Typical increases in pressure range from 50 to 100 mm.

The pressure is then brought to ambient atmospheric pressure by back filling with helium. Gas samples are withdrawn with a large hypodermic syringe and analyzed, following essentially the same procedure used in analyzing outgassing products. The remaining contaminated atmosphere is drawn through a liquid nitrogen trap with a vacuum pump. The collection of water and CO_2 is nearly quantitative when the system is evacuated to less than 10 mm pressure. The trap is weighed before and after the carbon dioxide is evaporated. The liquid condensate (mostly water) is poured into a small vial and saved for possible future testing and odor characterization. This method of obtaining the weight of CO_2 and water is reliable to 0.1 gram as determined by introducing known amounts of CO_2 and H_2O into the system.

DISCUSSION OF RESULTS

The time required for the test material to ignite under the standardized heating condition of this test is considered to be a relative measure of the ease with which the material ignites. There are a number of factors that affect the ignition time and these must be remembered when interpreting the data. Several of these factors are listed below:

1. Degradation temperature of test material.
2. Convective cooling of sample and heater coil.
3. Variability of heating conditions.

4. Absorptivity - emissivity characteristics (a/e) of various materials.
5. Specific heat.
6. Thermal conductivity.
7. Sample geometry.

Degradation Temperature

To ignite a sample, the surface of the test material must reach a temperature at which low boiling volatiles are evaporated or at which it degrades to produce combustible volatiles. The thermal stability of plastic materials varies widely. The plastics having greatest thermal stability should have the longer ignition times; however our test results show that this is not always the case.

Convective Cooling

The effect of convective cooling was investigated using an aluminum sample tube without any test material applied. Figure 10 shows heating curves of the coil and the untreated aluminum sample tube under three different pressures. The ultimate temperature reached by the coil and especially the sample tube is considerably less at higher pressures. Comparing the sample tube temperatures at equal heating times of 600 seconds we have 530, 424, and 370°C, respectively, at pressures of 0.8, 260, and 650 mm. Figure 11 shows the effect of placing a cap on top of the sample tube. Without the cap the cooling due to the "chimney" convection resulted in the tube being 136°C cooler after 600 seconds heating, that is 288°C without the cap versus 424°C with the cap. After learning this, the flammability tests were made with the tube capped.

Variability of Heating Conditions

The heat output to the coil is controlled (to within ±0.5 ampere) by regulating the current input. The corresponding heat output of the coils is 341 watts ±2 per cent for the small coil and 947 watts ±2 per cent for the larger coil (calculated for 600°C). Assuming that emissivity changes of the nichrome resistance wire in the coil are minor, the maximum variation in heating conditions due to current fluctuations is estimated to be less than five per cent.

Absorptivity - Emissivity Considerations

We have no exact information on the absorptivity-emissivity characteristic of the various materials tested. In the infra-red region of the electromagnetic radiation spectrum, plastic materials approach the characteristics of a gray body and a/e is assumed to be close to unity.

Specific Heat, Density, and Thermal Conductivity

Materials having a high thermal capacity per unit volume (specific heat times density) will warm up more slowly and tend to have longer ignition times. Likewise, materials with higher thermal conductivity will tend towards longer ignition times by virtue of the heat carried away from the surface into the material. Mineral fillers increase the thermal capacity and conductivity of composite materials and would be expected to increase ignition times.

Sample Geometry

The corners of a square bar heat up faster than the flat surface because the ratio of heat absorbing surface to the mass being warmed is highest at the corners. Most of the bar specimens were standard one-half inch square and five inches long. Some materials were available only as round rod stock. Those materials tested as round specimens are noted. Just how much the ignition time is increased by this geometrical factor was not determined, but it is probably significant.

The thickness of surface coatings and other materials tested as thin films affects the ignition time. It was noted that coatings less than 0.001 inch thick were difficult to ignite, presumably due to the rapid heat transfer to the aluminum tube through the thin film. Wherever possible a film thickness greater than 0.003 inch was used.

Table II shows burn test data on a number of plastic materials in the form of bar specimens. The atmosphere is 100 per cent oxygen at 5 psi pressure (260 mm). It is difficult to correlate the results with the individual factors discussed above. Because this was a screening program, most of the materials were tested only once. In three instances where experiments were repeated, the ignition times agreed reasonably well -- 76 and 80 seconds, 52 and 55 seconds, and 63 and 65 seconds for materials 6278, 6020G, and 7159, respectively. The Navy test procedure specifies four runs on each material to obtain an average value of test results. (22)

Once the sample is ignited the burning time and weight of material burned serves as relative measures of the material's tendency to burn under the controlled experimental conditions. The toxic reaction products produced in greatest quantity are carbon monoxide and carbon dioxide. The atmospheric contaminating potential (ACP) is calculated for CO, CO_2 and total organic matter produced in the burn. The units are in moles per gram of material consumed by the fire (except for silicone materials where the initial sample weight is used). These values can be used to estimate the potential hazard of burning a given weight of a material in a closed environment. For example, if one ounce of phenolic resin from an instrument panel were consumed by fire in a 100 cubic foot cabin, the atmosphere would contain more than 0.3 per cent CO_2, 0.1 per cent CO and 10 ppm total organics (pentane equivalent). Breathing this atmosphere would cause asphyxia within 30 minutes. The astronaut would have to don a rebreather unit or gas mask immediately after the fire starts. Incidentally, the smoking of two packs of cigarettes would produce about the same quantity of CO as the burning of one ounce of phenolic resin.

Table III gives burn test data on several surface coatings, tapes, tying cords and miscellaneous items. Table IV compares burn test data on a number of materials in 100 per cent oxygen at 260 mm pressure with that of normal air at 650 mm pressure. The 650 mm pressure was chosen to avoid a pressure buildup during the burn which exceeded ambient atmospheric pressure and thereby lifting the bell jar. Furthermore, the partial pressure of oxygen at 650 mm air pressure is exactly one-half that of 100 per cent oxygen at 260 mm.

The results show that the ignition time in air is consistently longer than in 100 per cent oxygen. This is attributed more to the greater convective cooling at 650 mm pressure than to the concentration of oxygen. The burning time and quantity burned are consistently higher in pure oxygen than in air. Less convective cooling and higher oxygen concentration in the pure O_2 burns are probably both contributing factors.

The amounts of CO and organics produced per gram consumed in the air burn is generally greater than in the corresponding oxygen burn. The efficiency of CO_2 production shows no consistent tendency in one atmosphere over the other.

ACKNOWLEDGEMENT

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Material Type	Ident. No.	Test Temp (°F)	Test Time (hours)	Sample wt. (g)	Weight loss (g)	Odor	Analysis ppm			ACP[1] $\left(\frac{moles}{gram}\right) X10^{-6}$	
							CO_2	CO	org.	CO	org.
Polycarbonate	7157	215	72	51.31	gain 0.001	2	none	none	84	0	0.33
Acrylic	6025	217	48	11.28	0.0440	1	none	25	15	0.44	0.31
Polyamide molding cpd	6820	210	48	21.45	gain 0.0359	2	trace	25	5	0.21	0.05
Phenolic molding cpd	6343	199	64	34.26	0.1147	2	trace	25	250	0.15	1.5
Diallyl phthalate molding cpd	7525	214	65	32.00	0.0091	2	1200	250	267	1.6	1.7
Grey vinyl primer	7573	220	72	11.94	1.5431	3	none	300	2842	5	48
Epoxy varnish	7134B	218	45	0.761	0.0167	2	none	40	139	1.1	37
Epoxy varnish	7134B	208	72	0.781	0.0163	2	trace	50	127	1.2	33
Polyester tape	6856	200	24	0.9599	0.0039	1	trace	25	12	5.2	2.6
Epoxy casting cpd	7083A	326	24	58.35	0.2989	3	28,000	1400	296	4.8	1
Epoxy casting cpd	7083A	207	74	36.06	0.1617	2	1000	none	123	0	0.7
Urethane potting cpd	7363A	213	48	25.03	0.0900	1	trace	300	63	2.4	0.5
Urethane foam	7212A	200	74	3.907	--	2	trace	none	60	0^+	3.1
Silicone rubber	6891	219	48	26.63	0.2231	2	trace	100	197	0.75	1.5
Glass epoxy laminate	6473	202	58	17.37	0.0111	2	trace	25	42	0.29	0.48

[1] ACP is the atmospheric contaminating potential per unit weight of material

$$ACP \left(\frac{moles}{gram}\right) X10^{-6} = \frac{4.8 \text{ X ppm}}{24.5 \text{ X weight sample}} = \frac{0.2 \text{ X ppm contaminant}}{\text{weight of sample}}$$

4.8 = volume of closed system (liters)

24.5 = ambient gram molecular volume (liters)

Table I. Outgassing Data on Several Plastic Materials

Material Type	Ident. No.	Ignition Time, sec.	Burning Time, sec.	Sample wt. g	Weight loss g	Odor	ACP[3] $\left(\frac{moles}{gram}\right)$ X 10^{-6}		
							CO	CO_2	Org.
Phenolic	6343	93	392	34.3	10.6	2	>1300	4300	13
DAP Molding Compound	6938	60	225	24.3	8.48	2	778	49,500	29
DA1P Molding Compound	7525	60	105	32.6	12.5		53	41,600	39
Nylon Rod[1]	6278	80	290	22.4	9.1	2	60	38,000	7
MoS_2 Filled Nylon[2]	7387	87	346	12.8	8.60	3	26	31,500	6
Epoxy Casting Compound	7553	45	105	31.1	9.62		NA	6150	64
Epoxy Casting Compound	6293F	81	225	43.9	10.5	3	0	29,200	13
Epoxy Adhesive	6020G	55	137	35.8	12.8	2	1000	6400	10
Epoxy Adhesive	6020M	110	190	41.2	11.5	3	803	51,400	84
Epoxy Casting Compound	6020Q	85	190	45.5	12.4	2	1000	17,800	19
Epoxy Foam	6293C	46	234	7.3	7.3	2	2200	5600	28
Urethane Foam	7212A	47	68	4.1	4.1	3	160	20,000	9
Urethane Foam	7616	47	88	7.9	5.7	3	30	60,000	14
Silicone Rubber	7147A	60	260	24.2	9.9	2	63	32,000	4
Acrylic Sheet	6985	85	125	32.4	12.8	3	309	41,400	75

[1] Round rod 1/2" in diameter

[2] Round rod 3/8" in diameter

[3] ACP is the atmospheric contaminating potential per unit weight of material consumed by fire.

$$ACP \frac{moles}{gram} \text{ X } 10^{-6} = \frac{53 \text{ X ppm}}{24.5 \text{ X weight loss}} = \frac{2.2 \text{ X ppm}}{\text{weight loss of sample}}$$

53 = volume of closed system (liters)

24.5 = ambient gram molecular volume (liters)

Table II. Burn Test Data - Bar Samples in 100 Per Cent Oxygen at 5 psi Pressure

Material Type	Ident. No.	Ignition Time, sec.	Burning Time, sec.	Sample wt. g	Weight loss g	Odor	ACP (moles/gram) X10^{-6} CO	CO_2	Org.
Red Enamel	6676	50	32	4.9	total	2	0	23,000	9
Red Screening Enamel	6913	no ignition		0.8	0.2	2	0	68,000	165
Epoxy Enamel	7202	52	35	7.61	near total	2	290	23,000	7
Vinyl Primer	7573	75	50	10.8	near total	3	1600	9200	290
Lacquer	1557	52	25	6.2	6.2	3	250	37,400	14
Black Enamel	16429	no ignition		4.2	1.0	1	4400	27,300	97
Electrical Tape	6768	27	43	8.14	total	2	700	41,900	68
Fiberglass Tape	6771	105	35	9.69	3.63	2	1800	61,400	23
Fairprene Rubber Sheet	6793	72	70	24	total	3	174	14,200	31
Neoprene-Cork Sheet	6690	35	80	12.2	4.9	3	1320	86,800	69

Table III. Burn Test Data - Surface Coatings, Tapes, Tying Cords, and Miscellanea in 100 Per Cent Oxygen at 5 psi Pressure

Material Type	Ident. No.	Ignition Time, sec.	Burning Time, sec.	Sample wt. g	Weight loss g	Odor	ACP (moles/gram) X10^{-6} CO	CO_2	Org.
Polytetrafluoroethylene	6008								
	oxygen	260	220	41.3	14.0	3	2500	12,000	580
	air	375	55	41.4	0.7	3	1900	81,000	720
Polychlorotrifluoro-ethylene	7173								
	oxygen	150	210	41.7	11.0	3	4300	Note 1	67
	air	no ignition		41.8	1.76	3	96	Note 1	120
Epoxy Adhesive	6020A								
	oxygen	65	120	29.3	11.7	3	620	42,000	37
	air	80	70	27.8	3.1	3	2000	110,000	160
Epoxy Casting Compound	6020R								
	oxygen	71	141	33.3	11.3	3	270	38,000	34
	air	96	133	33.3	7.7	3	1300	52,000	76
Epoxy Adhesive	6293G								
	oxygen	60	120	29.0	10.5	3	590	44,000	34
	air	100	35	31.7	2.5	3	1900	35,000	200
Silicone Rubber	7041								
	oxygen	76	449	22.7	4	1	240	11,000	18
	air	100	160	22.5	0.5	1	69	810	7
Epoxy Foam	7609								
	oxygen	61	171	20.2	10.7	3	350	46,000	22
	air	125	80	20.6	1.94	3	5200	47,000	220
Epoxy Foam	VX								
	oxygen	45	110	4.35	4.35	3	51	42,000	18
	air	72	43	4.36	1.20	3	3100	68,000	190
Silicone Adhesive	6851								
	oxygen	76	19	7.0	total	3	0	11,000	3
	air	148	44	7.1	4.4	3	100	12,000	35
Screw Sealant	7045								
	oxygen	76	221	20.2	13.4	2	624	36,000	35
	air	86	99	20.5	7.1	2	1270	19,000	130

[1]Condensate in liquid nitrogen trap (8.1g and 1.4 gram from oxygen and air burns, respectively), mostly chlorine

Table IV. Burn Test Data in 100 Per Cent Oxygen at 5 psi Pressure and Air at 12.6 psi Pressure

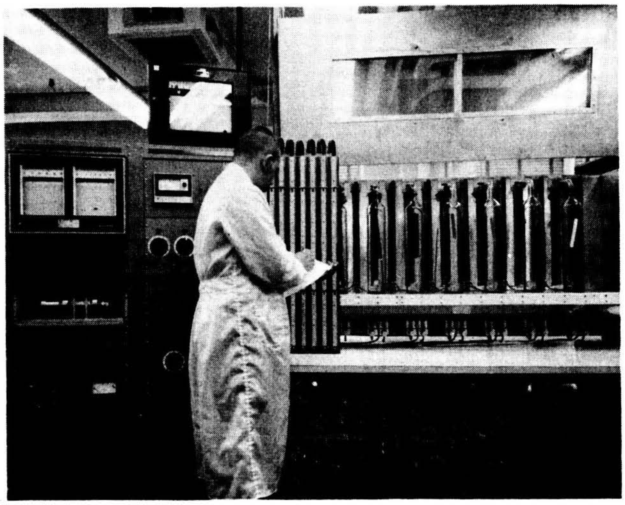

Figure 1. Outgassing Equipment With Oven Door Raised

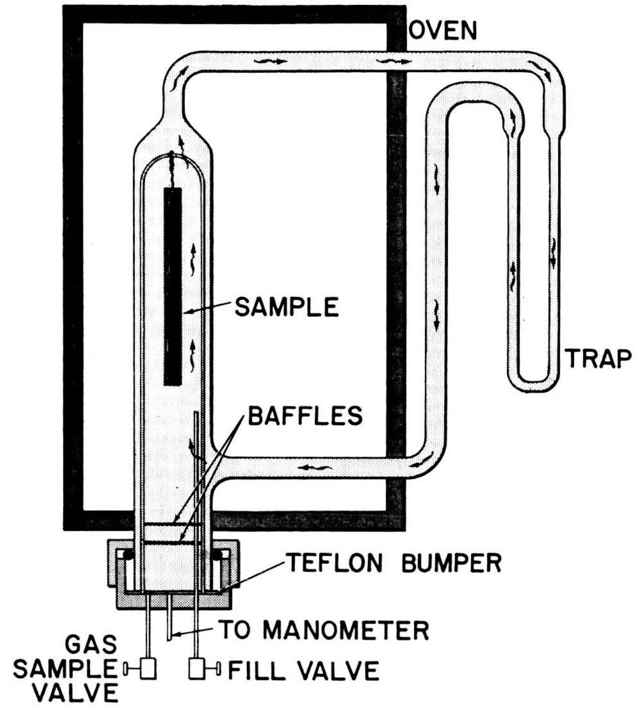

Figure 2. Outgassing Equipment Schematic

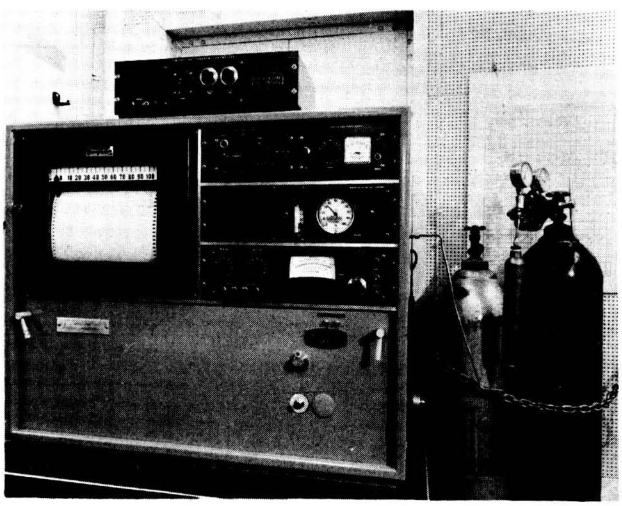

Figure 3. Gas Chromatograph

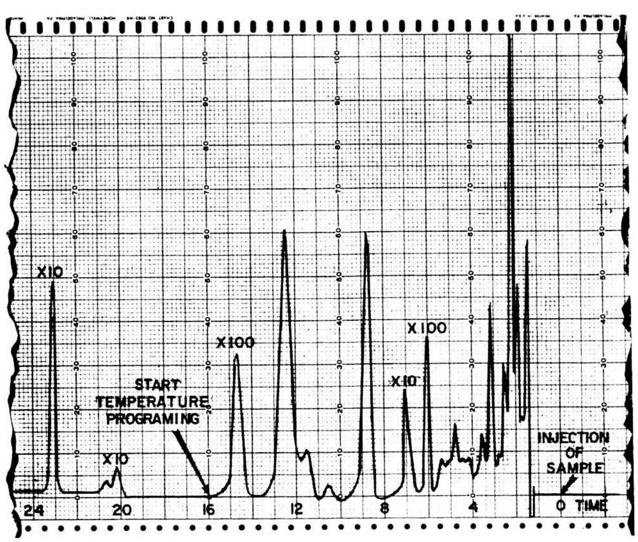

Figure 4. Sample Chromatogram

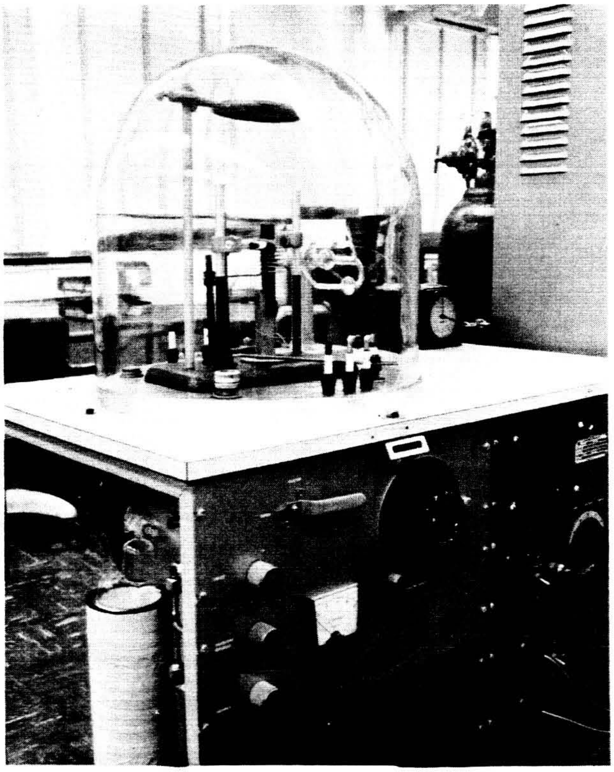

Figure 5. Flammability Equipment

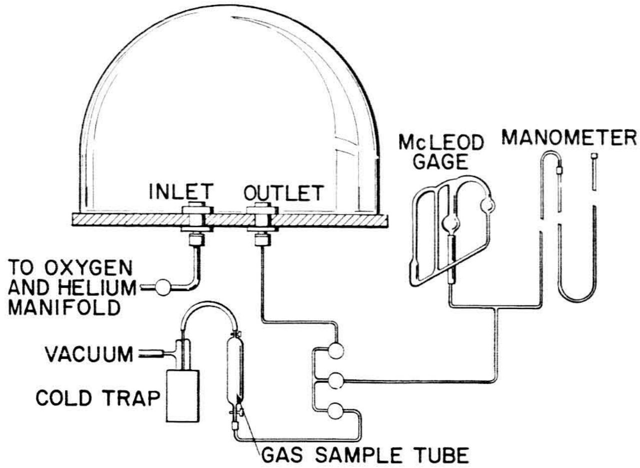

Figure 6. Flammability Equipment Plumbing Schematic

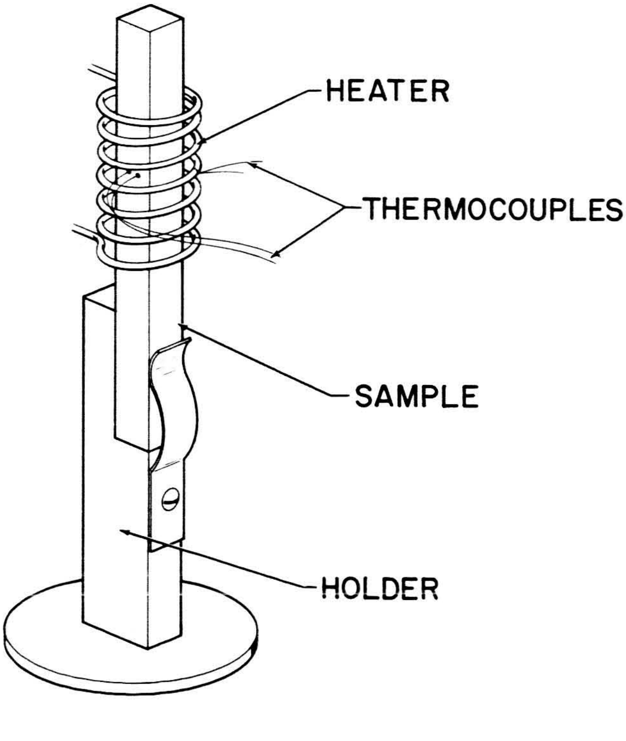

Figure 7. Bar Test Setup

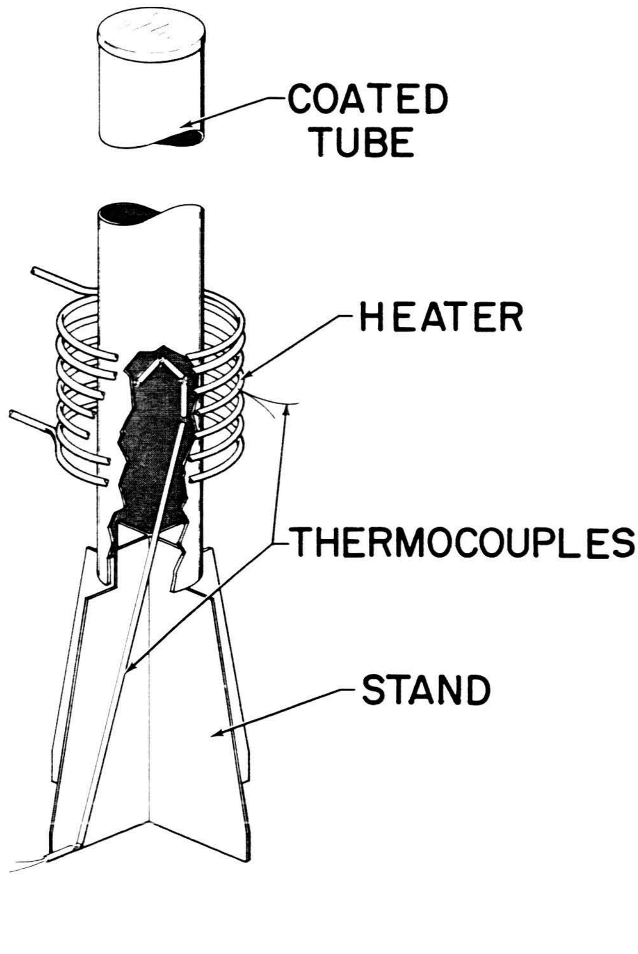

Figure 8. Tube Test Setup

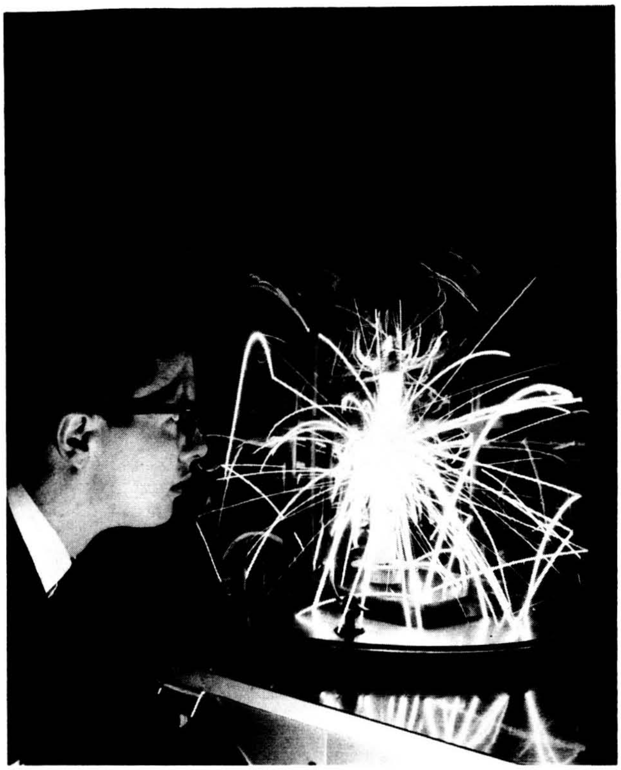

Figure 9. Material Under Test

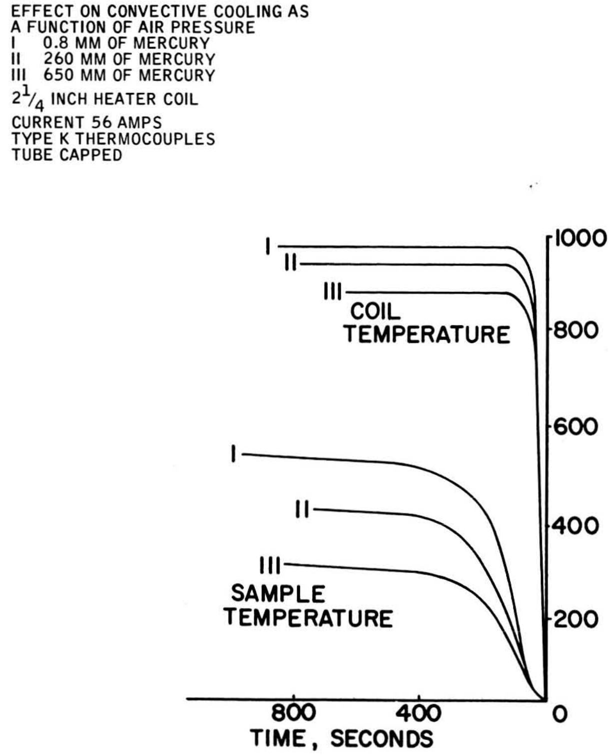

Figure 10. Heating Curves - Capped Tube

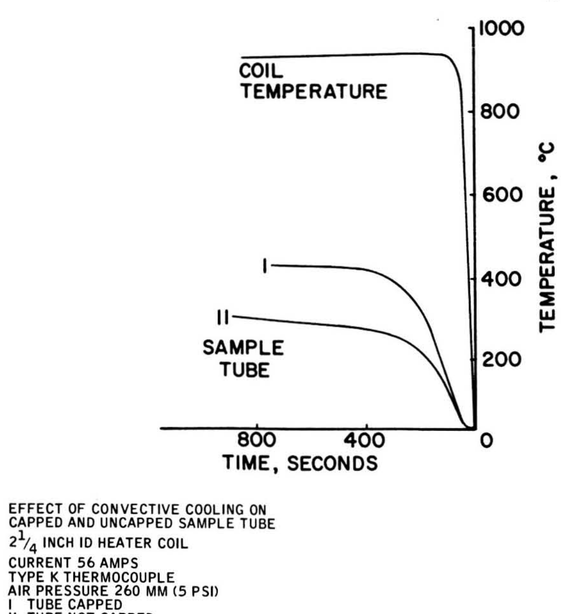

Figure 11. Heating Curves

Control-Display Design in Manned Space System Development

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Abstract ~~93746~~

The problem of designing a control-display system for a vehicle is that of organizing information and getting it to the operator so he can understand clearly what the situation is and what controls he may use to effect his decision. The work sponsored by the Air Force on this problem has yielded some tentative conclusions. One is that the role of the man in vehicle control is determined, deliberately or not, by the early crystallization of system design. Required overall system performance can only be assured if that role is based on sound detailed prediction of man's performance in the specific system design. Another conclusion is that there is no substitute for a sound experimental development program in control-display. Newly required equipment must be anticipated and developed by the time it is needed in a specific system.

The control-display equipment in any manned system is that equipment which provides the interface between the man and the rest of the system. It is the key to man's usefullness in a system because it is the means by which he gets information about the system and its situation in relation to some desired condition, and the means by which he imposes his desires upon the system. Therefore the control-display equipment is that which makes the difference between a vehicle in which the man is a passenger and one of which he is in command.

This implies that the control-display equipment is designed as an integral part of the system, or as an integral subsystem and not on a piece meal basis.

One way of looking at the requirement that the control-display system be designed as an integral part of the rest of the system is to note that the design of the control-display equipment determines the role of the human operator in the total system. This role of the operator becomes very important in assessing the overall reliability of the system in its operation.

Space vehicles are complex systems and assuring reliability of performance requires detailed study of system interrelation and predictions of conditions for operation. From the standpoint of reliability a completely automatic system has a singular characteristic; that is, it is capable of performing only those functions and taking into account only those inputs which were anticipated in the design and for which connections were made in the design process. There is therefore a considerable stress upon the requirement that the designers anticipate all relevant aspects of the system's operation. On the other hand a system which includes a human operator with a somewhat flexible control system allows the operator to take into account things which were not anticipated in the design. An automatic system can only respond with control modes which are specifically connected to given expected conditions. The human operator can use any available control mode to respond to conditions he finds in operation whether or not these conditions were anticipated. In this sense an operator can perform functions which allow second guesses on the part of the designer or an extension of the designer's function.

The problem of system design is reliability of performance. There does not appear to be any short cut in terms of a philosophy or a simple kind of decision which will allow a general answer to the problem of what functions should be made automatic and what functions should be made manual. In general it appears that any function which is performed in exactly the same way each time to the same set of inputs which are present in the situation can usually be made quite reliable when automatic. On the other hand there are functions which will be performed in different ways depending upon local conditions. In some cases these conditions could not have been anticpated or required information is not present in the situation. Such functions will probably be more realiably performed by the human operator than by a piece of equipment. For some time now it has been apparent that the role of the vehicle operator has been shifting from that of tracking in continuous control to a new primary use as a decision maker, situation determiner, and state selector. There will be even less employment of the man in the inner control loops and a more sophisticated employment of him in the outer loops. This appears to be a sound prediction of future trends in the role of the operator. However, this is not a simple answer to all design problems. Even if one determines that those functions will be automatic which are highly determined then one still has to identify these functions one by one in the prospective system design. Computation, for example, is one of these determined functions. Once it can be decided that a certain computation is needed then it is generally true that a piece of equipment can do the computation better than a man. An example of the type function which will usually be allocated to a man is the decision about time to retrofire. While the trajectory following retrofire may be highly determined there are many factors which enter into the decision to select a time. For example, the level of the oxygen supply, radiation level etc. may become more important to the astronaut than the optimum choice of landing area for speedy pickup.

In the past it has frequently been true that a great deal of design decision in a given system is crystallized before any significant thought is given to the control-display system and hence to the detailed role of the man in the system. It should be clear that a continuation of this procedure precludes the possibility that a rational determination can be made of the contribution of the man to overall system performance reliability. That this procedure was at all tolerable in the past is partly due to the fact that aeronautical systems were simpler and had less stringent operational boundries than space systems, and partly that we had more experience in operating similar systems in similar environments. Not only will the consequences of error be more stringent on the part of space systems but for some time we will have less direct knowledge about the operating environments and circumstances.

At this point it should be clear that the most reliable system is neither the all-manual nor the all automatic. In general the most reliable system lies somewhere in between. At present there is no rule of thumb for deciding just where in between the design should fall. Each individual function must be decided on its own merits in view of the limitation on this particular system and mission. This implies a need for a detailed study in the early design phases showing each required control action and decision[1].

In the history of display design it was typically true that a sensor picked up one aspect of vehicle's performance or position and fed this into an indicator designed for displaying just this parameter; one indicator per displayed parameter. For example, in the case of altitude the actual momentary value of altitude was shown on the indicator. It was left up to the operator to derive from some source information about how high he wanted to be. If there were any general or transitory limits on the altitudes at which he could operate he needed to bear these values in memory or else derive them from some other source. What is being said is that the actual altitude is only one of several things about altitude which the operator needs to know in order to use the information. These other things are such as desired altitude, temporary limits, long term limits, predicted values of altitudes and perhaps others. In the last several years the development of aircraft cockpits has seen a number of these computed functions being added to the information displayed on the panel. It seems clear that this trend toward more information about the operating situation will be continued in future system design. This seems especially true for those systems like space vehicles for which the operating limitations will be very strict.

The need for more information in the cockpit rides hand in hand with the need for better interpretability of the display systems. There is no simple way to resolve these somewhat conflicting demands, just as aerodynamicists have found no general solution to the problem of reducing drag while getting adequate lift. In a precisely analogous manner display and control systems must be designed to some extent uniquely for each given application and setting. While some general functional relationships can be identified, in the end it is true that displays are in the cockpit to provide information required for control decisions. A vehicle for a given mission will have a particular set of control decisions required of the operator. Only when this set of decisions is made specific can the designers be sure that they know what information must be provided in the display system. This information content must be specified before there can be a rational design of a system to provide this information.

One specific problem in display design appears worth special mention at this time. This is the problem of designing displays to make clear some specific and perhaps complex relationship which is important to the control task at the moment. One historical example of such a display is the Instrument Landing System which gave specific relative bearing information required for the job of maneuvering into landing position relative to the runway. Another example is to be found in the 3-Axis Attitude Indicator. In this case the complex relationship which is shown by the display is that between the control effects of pitch, roll and yaw and the resulting effects on pitch angle, bank and heading. The complexities in these relationships occur mainly at very high pitch attitude angles.

1. Kearns, John H. & Ritchie, Malcom L. Cockpit control-display subsystem engineering. U.S. Air Force, Aeronautical Systems Division, Technical Rep. No. 61-545.

One current example of work on illuminating specifically complex relationships is that of the terminal phase of orbital rendezvous. In recent work sponsored by the Air Force an approach to this problem has been made resulting in a concept which allows very ready interpretation of the relative motion between rendezvous vehicle and target vehicle. This relative motion concept allowing a clear interpretation of the control action required to correct simultaneously for altitude error, cross range error and range rate error. The key display of this relationship was called the Rendezvous Vector Display. The detailed study of such specific maneuvering problems can be expected to result in more specific display solutions to elucidate these complex relations.

In the space vehicles which have flown and those which are next to go into space the instruments were strongly influenced by a search for those indicators which were already developed and had demonstrated reliability. While this procedure may appear to limit the systems in some respects there are powerful reasons why this procedure can be expected to recur.

When looking at an actual system development with time schedules and dollar limitations, it is necessary to consider how expected time to accomplish proposed termination of requirements, design of equipment, and building of equipment for the specific system can be brought in line with the system development schedule. The most time consuming and costly aspect of this cycle is designing and building of the equipment required to make up the system. Therefore it is natural that there will always be some pressure to use existing hardware and in fact there is a tendency to warp requirements to make this possible. Thus it is inevitable that, to some extent, system design will be dictated by equipment available and thereby the operator's role in the system will be determined in like manner. If this pressure is acknowledged, then the solution is to provide enough advanced work that newly developed hardware is available by the time it is needed. This means that new systems must be studied in advance in such a way that the interface problems are revealed. The only feasible way that this situation can be handled is to have Experimental Development and Advance Technology work, by which the Control Display equipment is developed to be system oriented from the start. That is to say, it is necessary that efforts in the Experimental Development area not be undertaken from the standpoint of developing a single parameter or single indicator. The requirements for new and improved Control Display equipment can be determined only by considering total system needs of some expected manned system. This can be done by hypothesizing classes of vehicles or systems and analyzing their requirements. The procedure is much like that which should be done if the system were actually to be developed. These classes of system would be made up of representative configurations and mission phases involving those problems of special concern to the Control Display subsystem and related subsystems.

In the absence of a real system with all its parts visible, a realistic system may be formulated with as much detail specified as will be needed. This realistic system is valuable because the critical control problems arise in the interaction of several subsystems. The subsystems are much easier to specify than are the interactions. Making reasonable but specific assumptions about the subsystems allows the interactions to be studied in the hypothetical whole system. The important issue here is to set the problem up for an adequate study while it is still an anticipated problem. When the problem becomes real and pressing it will usually be already too late to work out an adequate solution.

Only in this way can Control Display equipment of the proper design be made available to system development programs on an off-the-shelf basis. There is one point which system development groups must understand and accept. The phase "off-the-shelf" is going to have a different meaning in the future. It will not necessarily mean a piece of equipment which has been used in a system before, but will be "off-the-shelf" in that capability of the equipment will have been demonstrated in an Advanced Technology Program which will include flight demonstration in a vehicle having characteristics as near as possible to the type for which equipment is intended.

SELECTION AND EFFECTIVENESS CONSIDERATIONS ARISING FROM ENFORCED CONFINEMENT OF SMALL GROUPS[1]

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Introduction

For many centuries man has lived under adverse environmental conditions and banded together to operate weapons. His performance usually has been adequate for survival in these situations, and at times he has been unusually effective in spite of the fact that behavioral scientists did not assist in selecting and organizing the crews. In view of the adaptability and ingenuity which man has demonstrated throughout history in meeting the demands placed upon him in stressful situations, it is not at all out of order to raise a question as to why so much emphasis is being placed today on the interpersonal aspects of small crew performance and the emotional effectiveness of the individual group members. An effort will be made in the present paper to trace the development of this problem area and outline the general nature of some of the broader aspects presently being explored. While the focus of the Navy's current research effort is on future sea-based deterrent weapon systems, the work is equally applicable to the closed ecological systems of space vehicles.

The specific problem area of concern here extends beyond the usual considerations of human engineering, personnel selection, and technical training. It is assumed that the man-machine interface problems will be solved and highly trained and skilled personnel can be selected to man the vehicle in question. However, evidence has been accumulated which indicates that the above factors are not sufficient to insure mission success. The closed ecological systems with which we are concerned introduce a combination of factors which until recently had not been encountered in man's conquest of his environment. It is the unique combination of enforced interaction of small groups of men, each of whom is critical for mission success, and the sensory poor or impoverished environment resulting from prolonged physical isolation which assumes increased significance as a critical factor in mission success.

History is replete with situations where men have lived for prolonged periods of time in isolated, sensory poor environments, early Antarctic expeditions being cases in point. Small groups of men also have worked and lived in situations of prolonged and enforced interpersonal contact since the beginning of society. In fact, voyages of the early whaling ships were far more extended in duration than the presently contemplated missions of space craft and future weapons systems. However, to date man has not faced a situation involving the combination of: (a) prolonged total isolation in a sensory poor environment; (b) intensive, enforced interaction of small groups; (c) total interdependence of all individuals for group survival; (d) total impossibility of removing an ineffective crew member; (e) sustained demands for vigilance and reaching rapid though irreversible decisions; and (f) a major financial and idealogical commitment by society in the accomplishment of the mission.

Although technological advances are just now beginning to produce the unique set of environmental and psychological conditions outlined above, the implications and significance of certain less stressful combinations of these conditions clearly has been indicated in several well studied military programs. A brief review of findings from these programs will give an indication of the nature of the problems to be faced in future closed ecological systems.

Military Experience With Small Groups Under Stress or in Isolation

Research findings from several rather divergent operational programs form a core of information from which to approach the present problem. These are the Navy's experience in the Antarctic with Operation Deep-Freeze; isolation studies in the submarine service; and Army and Air Force studies of crew composition which were conducted during World War II and the Korean conflict.

Operation Deep-Freeze

In 1956, just prior to the International Geophysical Year, the Navy Medical Department was requested to undertake the psychiatric assessment of all men who would winter over in the Antarctic as part of Operation Deep-Freeze. This request was made as a result of an individual's developing a serious and disrupting psychiatric illness at the height of the wintering over period when there was no possibility of obtaining relief or assistance from outside sources. The continuing clinical and research program conducted during the subsequent seven years has yielded a wealth of information on the psychological response of healthy, highly select groups of men living in extreme isolation under conditions of genuine rather than simulated stress.

[1] The opinions and statements contained herein are the private ones of the writers and are not to be construed as official or reflecting the views of the Navy Department or the Naval Service at large.

The number of personnel at the five U. S. Antarctic stations ranges from roughly twelve to one hundred. The men are totally isolated for periods of time ranging from approximately five to ten months, four months of which is characterized by total darkness. Even communication by radio frequently is impossible because of adverse weather conditions. Temperatures exceeding one hundred degrees below zero have been recorded at the South Pole station. In addition to the bitter cold and darkness, extreme winds and drifting snow discourage free physical movement outside of the shelter buildings. Somewhat remote, nevertheless anxiety-producing, threats of death exist in the form of possible fuel shortages, fire, or serious injury.

Detailed results of the psychiatric assessment program are reported elsewhere[9]. However, in passing it might be noted that not a single man accepted for this program through the psychiatric assessment procedure is known to have been hospitalized and separated from service for psychiatric reasons. On the other hand, current Medical Department statistics indicate that approximately 15 men would have been so hospitalized and discharged from a comparable size sample of the general Navy population serving in less stressful environments[2]. Although relative concern over serious psychiatric illness has diminished somewhat on the basis of this experience, related findings point up an area of serious weakness in our current knowledge of psychiatric assessment. Mission success in isolation depends upon more than individual effectiveness. In fact, based on the Antarctic experience, there is good reason to believe that effective personality interaction and group cohesiveness probably is as important as individual proficiency to mission accomplishment[10].

Effective personality interaction becomes increasingly critical as the size of the group becomes smaller. Actually, one is faced with two different conditions of isolation in the Antarctic. At large stations a number of sub-groups can develop so the individual rejected by one group of men usually can find acceptance in another. However, this is not true at the small stations. When enforced interaction occurs over a prolonged period of time, among a small group of highly skilled and trained men, mission effectiveness will suffer unless the group is personally compatible. This extends beyond the team compatibility considerations usually encountered in weapon system operations. In essence, with the passing of time under extreme isolation emotional feelings and biases distort reason and judgement in situations where there is constant interpersonal friction.

Certain phenomena which seem to be related to the sensory poor environment of the Antarctic also have been identified. For the most part, these are individual rather than group phenomena; however, because of the widely differing rates with which the conditions are reported from one station to another and from year to year, there is some suggestion that there may be a certain degree of group contagion involved.

Rohrer[11] has postulated three primary phases of adjustment to prolonged Antarctic isolation. First is the period of heightened anxiety which accompanies arrival in the Antarctic. During this time the individual is finding himself in an unknown situation and evaluating the future in this new environment. This anxiety diminishes as the isolation becomes a reality and the men become involved in the extensive physical activity of preparing for the winter night.

The second stage begins after the last aircraft have departed and routines have been established. It continues throughout the winter night. This adjustmental phase might be characterized as one of depression; however, it should be emphasized that the depression is not necessarily of pathological proportions nor is it experienced in the same way or to the same degree by all individuals. In fact, many men are consciously unaware of depression until the signs and symptoms are brought to their attention. The subtle indications of depression take the form of insomnia, moodiness, forgetfulness, and concentration difficulties. Headaches appear to show a marked increase during this period, particularly among the more intellectual members of the group[8]. Both Mullin and Rohrer have commented on the increased repression of hostile feelings toward other members of the group which occur during this adjustmental phase. In fact, Mullin raises the possibility that this repression of interpersonal hostility underlies the increase in headaches.

Rohrer's third adjustment phase may be designated as the anticipatory phase. Here, there is an increase in activity and decrease in the depressive phenomena. Further, repression of hostility begins to lift so there are more arguments and overt displays of irritability. Less concern and pride is shown with the work remaining to be done and in maintaining the station; maintenance which can be deferred tends to be left for the relief party to accomplish.

An illustration of the behavioral changes outlined above is seen in Figure 1, which is based on data analyzed at the U. S. Navy Medical Neuropsychiatric Research Unit, San Diego. The significance of such disruptions for successful accomplishment of a mission requiring high degrees of technical skill and coordinated effort is self-evident.

	Period in Months		
Behavior	1-4	5-8	9-12
Disruption of sleep cycle	2	15	3
Apathetic, indifferent	1	5	1
Tense, restless	3	8	19
Complaining, whining	0	1	3
Irritable, hypersensitive	6	9	13
Suspicious, mistrustful	0	7	16
Uncooperative	1	2	13

FIGURE 1

FREQUENCY OF CHANGES IN INDIVIDUAL'S CONDUCT AND EMOTIONS AT ONE SMALL STATION DURING ANTARCTIC ISOLATION

Submarine Studies

Present day nuclear submarines are manned by large crews, and the period of isolation is not as extended as that found on Operation Deep-Freeze. Therefore, the situation is not directly comparable to the Antarctic. There are several studies, however, which shed further light on the problem of men living in closed ecological systems. These studies have been summarized in some detail by Weybrew[16]. Particularly germane to the present discussion is evidence of a phenomenon somewhat akin to the three adjustmental phases noted in the Antarctic. This has been noted in Operation Hideout[7], on an early habitability cruise of the U.S.S. Nautilus[15], and on the world circumnavigation of the U.S.S. Triton[16]. Generally speaking, the changes in group behavior became apparent approximately one week after the beginning of isolation. The greatest difference between the phases noted in the Antarctic and in submarine studies is in the final or anticipatory stage. The submarine studies report little or no evidence of a deterioration of interest in and care of the vessel. Parenthetically, in the operational Polaris program it would appear that highly effective competition has been encouraged between the two crews to leave the vessel more ship-shape at the end of the voyage than it was at the beginning. Whether this would carry over with small crews isolated for more extensive periods of time is a matter of conjecture. An increase of headaches also is reported in the above studies. Again this is consistent with Antarctic findings; but, a question remains as to whether the headaches reported during early isolation studies aboard submarines was attributable to increased CO_2 levels or other toxicologic agents.

It is unwise to place the submarine experience in the same category as the studies of small Antarctic stations with regard to information on group functioning. The crews of present day nuclear submarines contain in excess of one hundred men; therefore, they are more similar to the large Antarctic stations than the small groups of concern in the present paper. As indicated earlier, isolation in large "communities" apparently involves a different and less stressful set of interpersonal variables than that encountered in small groups.

The foregoing not withstanding, throughout history submarine crews traditionally have been among the most effective and best integrated groups ever assembled. This degree of excellence has been accomplished empirically through a system which has slowly evolved over a period of half a century. Thus, in spite of their size, knowledge gained from our nuclear submarine experience may have particular value in developing relevant techniques for sustaining high levels of crew effectiveness over long periods of time and anticipating the emotional impact of a sensory poor environment on the emotional effectiveness of individual crew members.

Air Force and Army Experience with Crew Composition

Another source of information regarding the importance of small group phenomena in the operation of military units stems from the efforts of various Army and Air Force research laboratories, dating from World War II and the Korean conflict, which were responsible for conducting research on social psychological factors affecting the performance of bomber crews in combat. Although not specifically concerned with isolated or confined crews in the strictest sense of the words, the research results are clearly relevant to the operation of small space craft or underseas vehicles. As has been suggested earlier in this paper, there is reason to believe that the prolonged and enforced interactions among a small group of men magnifies the importance of interpersonal behaviors. Much of this work, and its special relevance to the performance of teams in flight, has been summarized by Sells[13], among others, and will therefore be only very briefly reviewed here.

Perhaps the most intensive of these efforts was that of the Crew Research Laboratory at Randolph Air Force Base. B-29 bomber crews were assembled at Randolph and completed their crew training there prior to proceeding to combat in Korea. A wide variety of measures was obtained in training school and related to subsequent performance in combat. These measures ranged from individual ground school grades to simulated bomb drops over stateside targets. Among the many measures obtained were attitude scale scores describing crew member attitudes toward the Air Force mission, their crew in particular, the Korean conflict itself, and so on. Subsequently, various measures of combat performance were obtained, including superior officer's ratings and the percentage of scheduled missions completed.

Analysis of these data -- reported by French, Knoell, and Stice[4] -- indicated that, whereas training performance measures predicted crew performance in combat with only chance success, mean crew attitude measures obtained in training predicted rated combat performance significantly. That is, crew attitudes in training predicted combat performance better than did crew performance measures in training. Furthermore, crew attitudes in training predicted rated combat performance better than they predicted training performance.

DeGaugh and Knoell[3] collected attitude survey data from crews while they were in the Far Eastern theater, either in Japan or Okinawa. Their questionnaire, which had been administered to crews which had been in combat for at least thirty days, measured three primary attitudes: satisfaction with the Air Force; pride in work groups; and individual adventurousness. Satisfaction with the Air Force and pride in work groups were found to be significantly correlated with ratings of combat effectiveness, but individual adventurousness was not found to be related. These and other studies (for example, by Berkowitz[1], Schachter[12], Haythorn[6]) indicate rather clearly that attitudes are significant in determining performance of small military units.

It has also been fairly well established that a high degree of interpersonal liking within the group facilitates the acceptance of group standards. Thus, a group of men who are individually motivated to perform their assigned missions and who in addition have a strong team spirit is likely to perform very well, while a group of men equally well-motivated individually but lacking the team spirit may or may not perform well. In addition, the interpersonal relations among group members can provide powerful emotional support. The importance of this factor is stressed in the current psychological view that membership in groups is needed as a means of maintaining personal adjustment. Acceptance by group members was shown in World War II to be significantly related to measures of individual adjustments of naval recruits. Also, the importance of interpersonal loyalties among military group members was indicated in a research study of combat infantrymen during World War II in which it was found that loyalty to his buddies was one of the chief factors in keeping the infantryman going in combat[14]. This factor was also emphasized in a report by Grinker and Spiegel[5] of psychiatric experience with Air Corps personnel during World War II. It would seem, then, that a great deal of concern for the compatibility of a group of men is justified, both on the grounds of its importance in determining performance effectiveness and in its relationship to individual adjustment. A considerable amount of research directed toward a better understanding of the factors determining compatibility has been conducted, both in military research organizations and university laboratories. Our knowledge is still very meager, however. In the remainder of this paper, we shall discuss very briefly the research problem areas which seem to require attention.

Implications for Closed Ecological Systems

Based on a general analysis of the operational problems of closed ecological systems, derived research requirements and the existing state of knowledge, the following are seen as fruitful areas of research regarding team and crew behavior: criteria of small crew effectiveness, crew composition, maintenance of crew effectiveness, closed space environment problems. Fortunately, a good deal of information already exists in these problem areas. The primary task is one of synthesizing that knowledge which already exists, filling in basic theoretical as well as empirical gaps, and translating the information into usable operational selection and preventive psychiatry programs.

Criteria of Crew or Small Group Effectiveness

One of the most vexing problems in neuropsychiatric research is that of adequate criterion measures. Without such measures it is not possible to assess the performance effectiveness of small groups of men serving in closed ecological systems. This goes beyond gross measures of mission accomplishment and includes the "emotional cost" involved. The highly trained and skilled crews under consideration here are not expendable. Thus, while this research problem area might be considered much more basic or conceptual in nature than those which follow, it is extremely important in the development of psychiatric selection and preventive psychiatry programs which are adequate to cope with the problems of closed ecological systems.

In research to date, considerably more attention has been given to the investigation of how and why small groups function than to measures of group effectiveness. Moreover, for the most part, these studies have been concerned with descriptive measures of group action and interaction. Unfortunately, few attempts have been made to determine how closely the group functioning approaches the most highly desirable pattern of interaction as considered from the standpoint of maximizing performance effectiveness. Consequently, such studies can lead to identification of correlates (composition, organization, etc.) of differences in group process but cannot lead to establishment of correlates of most effective or efficient group functioning, even for a specific crew or group and a specific type of duty.

One of the more important considerations involved in the criterion area is the necessity for development of a systematic statement of the basic performance functions required in execution of small group duties in general, and a point by point delineation of how these functions are involved in the particular duties of a specific small group. Along with the development of generic small group functions, there is the need for specifying the performance dimensions appropriate for describing crew performance of those functions. Specification of the basic performance dimensions descriptive of crew task performance, for crews in general and for any particular type or section of a crew, will greatly facilitate the determination of criterion standards on those dimensions. Such criterion standards then can serve as reference values for assessing the performance effectiveness of operating crews.

Group Composition and Organization

The first problem in this area is the specification of conditions under which careful and deliberate composition of crews, on the basis of personality and/or other psychological factors, can contribute to the effectiveness of the operating mission to which the crew is assigned. On the basis of research to date, it has not been unequivocally demonstrated that psychological factors in crew or team composition should be considered a primary variable in performance effectiveness under all operating conditions. On the other hand, there is evidence that certain aspects of operational conditions under which crews work -- nature of duties, operating environment, degree of isolation, etc. -- are critical in determining the degree to which composition factors are of significance in achieving crew effectiveness.

The research problem here is envisioned as presenting two major tasks: First, to specify the operating conditions which maximize the importance of personality factors in crew composition; and secondly, to identify the individual personality patterns and combinations thereof which maximize crew or small group effectiveness under given operating conditions. An answer to the first task should specify the conditions under which deliberate composition of crews, on the basis of psychiatric and psychological factors, could profitably be effected. Answers to the latter task would specify the criteria to be used in selecting individual crew members and the explicit rationale to be employed in actually assigning men to crews.

Intimately related to the problems of crew or group composition are those concerned with organization of the group. In general, organization is used here to refer to many aspects or dimensions of group structure; e.g., leadership, communication authority, status, etc., all of which are probably inter-dependent. Research to date has been concerned primarily with identifying these major dimensions of structure and in illustrating how variation along such dimensions has important effects upon some aspects of group functioning. Less attention has been given to the inter-dependencies between the structural dimensions. It is here that considerable progress may be made.

The end result of work in this area is to fill the present gap in our knowledge of how to combine highly trained and skilled individuals into crews which will be emotionally as well as technically compatible under the stress of prolonged isolation. On the basis of Air Force and other studies, it appears likely that considerable pay-off may be expected in this area.

Maintenance of Crew Effectiveness

Once an effective crew has been formed, steps must be taken to insure that it remains effective over time. In the systems under consideration, the potential for disruption and ineffectiveness of small crews through interpersonal friction will be increased while the opportunities for outside appraisal and help become more limited. Under such conditions it becomes imperative to develop resources within the small isolated units themselves for diagnosing trouble spots and correcting potential difficulties. Training and indoctrination principles useful in the development of such group diagnostic and self-maintenance skills also are required. In essence, the quasi-therapeutic potential of small crews must be fully exploited.

The development of effective principles to accomplish these objectives requires the identification of, and control over, the factors underlying group cohesion and disruption. Fortunately, a considerable body of basic research also is available on this problem and can be brought to bear in future work. Unfortunately, most of this research has been carried out in laboratories with experimental groups which have no history and little future. Studies are needed which focus upon well-established groups under conditions of isolation. Such studies may be expected to yield information as to the specific psychological responses normally used in maintaining the integrity of small groups and the effectiveness of these responses. Moreover, this research area should include investigation of the consequences of limiting the availability of certain psychological responses or mechanisms, e.g., withdrawing from the group, social ostracism, etc. Studies of groups isolated in the Antarctic indicate that certain psychological mechanisms used by individuals and/or groups under such prolonged stress may be responsible for precipitating potentially serious psychiatric disturbances.

Environmental Restructuring and Enrichment

With technological advances, the role of man in future systems is shifting to include a heavier emphasis on information processing and monitoring functions and a decreasing emphasis on physical activity and personal mobility or "changes in scenery." As a consequence of these developments, the environment of closed system crews may be characterized as one of reduced sensory stimulation. Moreover, there is little opportunity for increasing the level of stimulation because of confinement and isolation.

It is clear from the Antarctic research that there are rather marked changes in behavior which occur under conditions of isolation and confinement. Recent work with total sensory deprivation also has produced quite dramatic behavioral changes. However, to date, the exact significance of reduced environmental stimulation for the effective operation of closed ecological systems remains unknown. From the point of view of military and space environments, it is likely that the situation of critical importance is the imposed structuring or monotony of the sensory environment.

Primary attention in this research problem area should be directed to an investigation of the effects of long-term exposure to such structured environments on individual functioning as well as the effects on the interaction and performance of small isolated groups. In addition, once the conditions of stimulus structuring and patterning which produce the behavioral response to reduced environmental stimulation are known, systematic investigation should be undertaken to determine ways of overcoming the deleterious effects of such deprivation. It is likely that deprivation effects may be attenuated by enriching the environment, by modifying the individual's tolerance for limited environments, or by more effective utilization of certain aspects inherent in the environmental situation per se. However, the exact approach to the attenuation problem must await the identification of variables responsible for the behavioral changes under conditions of reduced environmental stimulation.

Summary

The closed ecological systems of manned space vehicles and future weapon systems will engender specific individual and interpersonal stresses, the overall significance of which will exceed anything heretofore encountered. The significance of these problems has been recognized, but very little systematic effort has been focused on evaluating their magnitude and devising techniques for coping with them. The problem area of concern is one which extends beyond the province of man-machine research as it is presently envisioned.

On the basis of the Navy's operational and research experience with small isolated groups in the Antarctic, as well as Air Force research on bomber crews, it is clearly evident that mission success is dependent upon more than adequate human engineering and the selection of technically qualified and trained personnel. Two additional variables are introduced by the use of small crews in the closed ecology of space and future weapon systems vehicles. Effectiveness of interpersonal interaction among personnel operating the system is a critical variable in mission success. Secondly, factors inherent in prolonged isolation and physical confinement assume a significant magnitude in both individual and group effectiveness. The nature, significance, and implication of these problems is discussed in terms of crew composition, maintenance of crew effectiveness, closed space environment problems, and performance criteria.

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A REAPPRAISAL OF THE RADIATION HAZARDS TO MANNED SPACE FLIGHT

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INTRODUCTION

With the discovery of the great radiation belts surrounding the earth, emphasis has been placed on determining the nature, extent and effects of the space radiations and to characterize differences between the inner and outer radiation zones, the auroral zones and interplanetary space. These latter studies are being conducted in large measure to define the properties of high energy particle radiations emanating from the sun during times of intense solar flare activity.

Similarly, with the interest generated in quantifying the physical properties of these radiations, the biological aspects emphasize man's ability to perform specific tasks in several radiation environments, and the nature of man's vulnerability to radiation. It is reasonable to argue that one of the major reasons for sending man into the aerospace environment is to realize the advantage of his abilities and skills as an equipment operator, trouble shooter, maintenance specialist, observer and interpreter of dynamic situations and a maker of decisions. Our Mercury flights have already shown an operational flexibility and a gain in reliability, which can be realized when man is a component of the system. Furthermore, operational plans for Gemini and Apollo have been modified to exploit this versatility. Man's capabilities are the composite of many determinants including the functional properties of biologic components and systems. This being true, it can be argued that these biologic components and systems then constitute targets for ionizing radiation and further that biologic damage might well influence the success of a manned system.

RADIATION ENVIRONMENT

Although the origin of many of the radiation particles of space, particularly those in the radiation belts, is still a controversial question, four reasonably discrete zones can be identified and the structure, composition,

(Figure 1 Van Allen Radiation Belts)

energy spectrum, and temporal characteristics defined in a rather accurate manner. The outer radiation zone extends to 8 to 10 earth radii in the equatorial plane and dips down to low altitudes with increasing geomagnetic latitude. For example, at 55 to 70 degrees geomagnetic latitude, the outer zone dips as low as 300 km in altitude.

This zone is composed primarily of electrons and protons as determined by satellites and sounding rockets. The electrons at the heart of the zone suggest energies greater than 500 kev and values in the order of 2×10^7 electrons/cm^2. Protons with energy greater than 10 Mev appear to number less than 100 protons/cm^2 and protons greater than 140 kev approximately 10^8 protons/cm^2. A soft proton component in the energy range of 140 kev to 4.5 Mev has been discovered as a result of the flight of Explorer XII and leads to the hypothesis that the regional extent of the soft protons and the outer zone electrons is very similar. On a temporal basis, the outer radiation zone is characterized by fluctuations in intensities an order of magnitude greater than those observed in the inner radiation zone. Additionally, there are large changes associated with magnetic storms.

Properties of the inner radiation zone are not as well known as the structure, composition, and energy spectrum of components in the outer radiation zone probably because of the presence of a penetrating proton component superpositioned on an intense electron component and the inherent difficulty in experimentally discriminating the properties of one component in the presence of the other. In the equatorial plane, the inner radiation zone begins at an altitude of about 600 km and extends to an altitude of about 10,000 km (2.6 earth radii). The energy spectrum of protons has been measured at the low altitude fringe of the inner radiation zone but a definite spectral shape throughout the inner zone does not exist. The spectrum at 1.6 earth radii based on

Note: All experiments referred to in this paper in which laboratory animals were involved were conducted according to the rules of laboratory animal care as promulgated by the National Society for Medical Research.

protons of $E > 40$ Mev has been deduced from Pioneer 3. Temporal variations in the inner zone involving as much as a factor of 3 have occurred and can be correlated with major solar proton events and/or major geomagnetic storms.

It was conjectured prior to "Starfish" that a high-altitude nuclear test might result in a loss of some particles from the inner natural radiation belt, but the predominant effect was the injection into the earth's magnetic field of beta particles resulting from the decay of charged or uncharged fission products. In this way a new electron belt was formed superimposed upon and extending below the natural belts. Data so far do not give a completely consistent picture of the distribution and persistence of the injected electrons; measurements disagree by a factor of 2 and occasionally 4.

The maximum intensity appears to occur at a mean altitude of about 4000 km and the contour for which the electron flux is 10^9 /cm^2/sec is approximately 5000 km thick and extends 6000 km in the north-south direction. Definitive data are still lacking concerning the decay of these trapped electrons.

From the observed distribution of electron flux, the external radiation intensity at the exterior of the capsule during a mission similar to Mercury 7, 8, and 9, would be between 50-100 r and would represent an internal cabin dose of .5 to 1 rad to the skin of an astronaut.

The auroral zone is most often observed during periods of intense solar activity and/or magnetic storms. The light is caused by energetic electrons; protons may or may not be present but in any case do not contribute significantly to the light. The occurrence of these events is between 60 - 65° N and S geomagnetic latitude. A great deal is known about these events and for our purpose need not be further discussed.

In interplanetary space solar flare corpuscular radiation may well be the significant contributor to biological effects. This radiation, ejected from the sun, has been identified as protons mixed with a very small number of alpha particles. The energy spectrum of solar flare particles varies significantly from flare to flare. For a large flare proton peak intensities can be described at N(30 Mev)=1.5×10^5 protons/cm^2/sec at t=1.2 days. For a 1-day period the flux of protons of 30 Mev was about 5×10^9 protons/cm^2 and the dose corresponding to 5×10^9 protons/cm^2 at E=30 Mev equates to approximately 1300 rad in 1 gm/cm, which is their range. In general, solar flare particles have an isotropic distribution. The temporal variations, at least for the number of low energy flares and the flux of particles produced in them on arriving at the earth are strongly modulated by the eleven year solar sunspot cycle.

Compared with the solar flare protons, the hazard potential of the alpha particles and of the galactic cosmic radiation appears of minor concern. Analyses of galactic cosmic radiation as measured by balloons and space probes indicate particle fluxes and dose rates well below those that would produce any demonstrable biological damage unless interplanetary flights were extended to time periods of many months. For instance, it is estimated that free space galactic cosmic radiation integrated weekly doses range from 0.1 to 0.3 rad.

This, then, as a statement of the space radiation environment for man clearly indicates that a reliable evaluation of the biological efforts of high energy proton irradiation is of prime importance.

BIOLOGICAL EFFECTS

Protons of immediate interest most likely are above 10 Mev and below 1000 Mev. Radiation biology effects, as they are now understood can for general purposes be conveniently divided into high, middle, and low dose responses. High dose effects - incapacitation-unquestionably result from involvement of the central

(Figure 2 Types of Radiation Death)

(Table 1 Dose and Clinical Effect)

nervous system rather than the typical radiation syndromes of gastrointestinal and hematopoietic injury. The early manifestations of radiation injury, ataxia, retching, vomiting, shock-like symptoms, iritis, nystagmus, hemorrhage, and subsequent death are all a function of dose. Massive doses of high-dose-rate radiation have demonstrated quite a consistent pattern in monkeys, dogs, and one human accident victim. These experiments suggest that exposure to 2500 rad will cause incapacitation within a time period of thirty minutes. Furthermore, though limited recovery takes place for several hours, degeneration and death are imminent. Death time at this dose level most probably will occur in a few days (4 - 7). In this regard it has not been demonstrated in the laboratory that radiation in any dose can kill instantaneously.

As the radiation dose is decreased, i.e., 600-1500 rad, there is a different mechanism of death--predominantly from gastrointestinal injury--and rapid incapacitation is not clinically apparent unless it be psychological. Survival times are increased to 1 - 2 weeks.

Furthermore, at even lower doses, 300-600 rad, if death occurs, it is most probably attributable to the hematopoietic death pattern of several weeks to months. Extending the dose vs effect to lower values, for conditions encountered in short term space operations, 200 rad can be considered as noncasualty producing, i.e., there will be no acute incapacitating effects. This may not be true for such long-term latent effects as carcinogenesis, cataractogenesis, temporary sterility, shortening of life span, and genetic changes. In space exploration 50 rad can be considered as an acceptable, safe dose, while a dose of 400-450 rad can be considered lethal to half the crew within 30 days. One must, however, consider the variability of biological response to the dose. Human exposure to therapeutic doses of radiation as well as animal experiments suggest a factor of 2 difference in response, i.e., 400 rad may in one person appear clinically as if the dose were 200 rad, while in another more nearly like 800 rad. As the dose goes up, however, less variance occurs.

There are other uncertainties besides biological variance: the endpoints used to measure incapacitating criteria; the unknown factor of extrapolating animal data to man; the complexity and demands of the task he is to accomplish; and perhaps greatest of all, his will to accomplish his task.

In addition, in estimating radiation effects, the influence of dose rate must be included because of the likelihood of considerable variation in this factor during space missions.

Because of these considerations, it is essential to identify the specific biological effects of protons of different energies. It is very important in evolving reliable proton dose vs effects schedules that the experimental subjects be well standardized in terms of radio-biological research experience. From this standpoint, we recommend the infrahuman primate because of the fact that extensive RBE data based on Co^{60} as a standard have been developed in all of our acute primate studies with fast and thermal neutrons, gamma rays, nuclear weapons and now, with protons.

(Table 2 RBE vs Co^{60} Gammas)

Focal irradiation to the eye of the primate with 14, 39, 185 and 730 Mev protons indicates that the RBE for cataract formation, iridocyclitis, erythema and epilation ranges between 1 and 2 when compared to Co^{60} gammas. It is interesting to note that 14 Mev neutrons have a similar RBE (2) for these same sequelae of acute radiation exposure.

Preliminary whole body proton exposure using 730 Mev protons yields an LD 50/30 of 315 rad, a figure that compares well with bomb spectrum neutron experiments, thus suggesting an RBE of approximately 1.6.

Until we can unequivocally relate particles, rate, and dose to tissue equivalent effects, in the actual space environment, there will be a factor of uncertainty, but one of no greater magnitude than any of the other problems of vehicle launch, orbit and recovery. Furthermore, 10 years of long-term latent effects research with primates on the effects of neutrons, gammas, and mixed neutrons and gammas, support the concept of a dose schedule such as the following:

Table 3 Estimated Hazards

Additionally, in primates, 50 rad of acute exposure to neutrons, gammas, and mixtures of these, has not produced cataracts; erythema and epilation have not been observed below 250 rad.

Although lacking the precision of systematic experimental studies, certain observations acquired through clinical studies of bomb casualties, accidental exposures, and therapeutic experiences deserve careful consideration because of their relationship to what Furchtgott (1956) has called "behaviorally significant effects." For example, Keller (1946) noted fatigability as an almost universal complaint in his study of bomb casualties; and Gerstner (1957, 1958) commented on the appearance of listlessness, apathy, headache, and drowsiness "within a few hours" of exposure to radiotherapy. Miller, Fletcher, and Gerstner (1957) found about 50 percent of their patients showing fatigue, anorexia, and nausea shortly after radiotherapeutic whole-body exposures ranging from 125 to 175 r. Further studies of therapeutic experience by Levin, Schneider, and Gerstner (1959) showed that whole-body exposures of 150 to 200 r left patients asymptomatic for about an hour, but thereafter precipitated feelings of fatigue, apathy, dizziness, and headache, and produced appearances of depression and energy depletion. Thoma and Wald (1959) reported similar findings in their review of accidental exposures.

Finally, Furchtgott (1952) reported studies, unavailable to him in original form, which suggested that radiation of the skin in "suberythemal doses" increased scotopic threshholds for several days and produced decrements in dark adaptation levels.

One, of course, cannot foresee with confidence what impact these effects might have on task performance, since high levels of training and motivation often sustain an operator to outstanding levels of achievement despite his infirmities. Furthermore, many of these symptoms may be reasonably well controlled by use of mild analgesics, stimulants, and perhaps antinauseants.

SHIELDING

If the range of doses suggested in Table 3 should be judged biologically unreasonable, then additional applications of shielding, vehicular and/or partial body, must be explored along with protective chemical compounds designed to ameliorate the mechanism of radiation damage.

The physical principles of shielding against both electromagnetic and particulate radiation are well known and understood, and it is not likely that entirely new shielding principles will be found. At least no startling advances can be expected in the near future. The complexity of the problem and the many unknowns still prevalent suggest that shield design today is consequently a mixture of science and educated guesses.

Optimization of shield design for size and weight may well define vehicle configuration, payload, and mission. Therefore, it is essential to examine the radiations of space in terms of the attendant shielding problems.

As a first approximation, it may be said that for high energy particles the shielding properties of all elements are nearly the same and depend only on the total amount of matter (measured in grams per square centimeter) in the shield.

Refining this approximation then, there are differences in radiation attenuating characteristics among different elements, but these differences do not exceed a factor of 4, from hydrogen to uranium. For radiations other than electrons and gamma rays, the elements having low atomic numbers are more effective as shielding materials on a weight basis--hydrogen being twice as effective as uranium. However, for electrons and gamma rays, elements having higher atomic numbers are more effective.

Upon closer examination, the interaction of high-energy radiation with matter is more complex, and a more complicated picture emerges. For purposes of discussion, it is convenient to divide the energy spectrum into five ranges.

(Table 4 Representative Radiation Events)

1. For energies less than 10 Mev: a complex picture of secondary radiations--gammas, neutrons, betas, etc., emerges as a result of nuclear interactions. However, in view of the shielding required for more energetic particles, extremely intense incident fluxes of this lower energy radiation are necessary before they would constitute a problem.

2. From 10 to approximately 300 Mev, the situation is somewhat simplified. No mesons are produced, and the interactions that do occur, such as spallation, fission, and nuclear excitation, followed by the emission of neutrons, protons, etc., produce secondaries whose range in shield is for the most part smaller than that of the primaries. If the shield is designed to stop the primaries up to a given energy, secondaries will be taken care of more or less automatically.

3. Above 300 Mev, mesons are produced as secondaries (among other particles). However, in the range from 300 Mev to about 1 Bev, much of the kinetic energy of the primaries is consumed to create the rest mass of the mesons and relatively little remains for their kinetic energy. The range of the mesons, therefore, is fairly small and again shielding designed to stop the primary radiation will, to a large degree, stop the secondary radiations.

4. Above 1 Bev the situation again gets complicated. The multiplicity of possible secondaries and further generations of reaction products--such as mu-mesons and electron-gamma cascades--as well as the effects of heavy primary thindown events, necessitate close evaluation: composite shielding might be preferable for particles in this energy range.

5. At very high energies (above 10 Bev) the build-up of secondaries and further generations contribute predominantly to ionization and dose, and the best philosophy may be to use no shielding at all, unless the primary flux is very strong--which would be the case for an intense solar flare.

Whatever the choice, the vehicle must carry substantial weight in the form of shield material. As shielding weight increases, new propulsion systems will find application and, conversely, with new and more powerful propulsion systems, perhaps more exotic shielding methods will come into use. Electric and/or magnetic deflectors as have been suggested by some then may be the system of choice. At present, however, it is most urgent that we focus our attention on obvious gaps in our knowledge: (1) There is a dearth of precise physical data on the space radiations themselves; (2) Physical data on the interaction of very high-energy particles with matter is incomplete; (3) Information on biological effects produced by higher energy radiations is insufficient.

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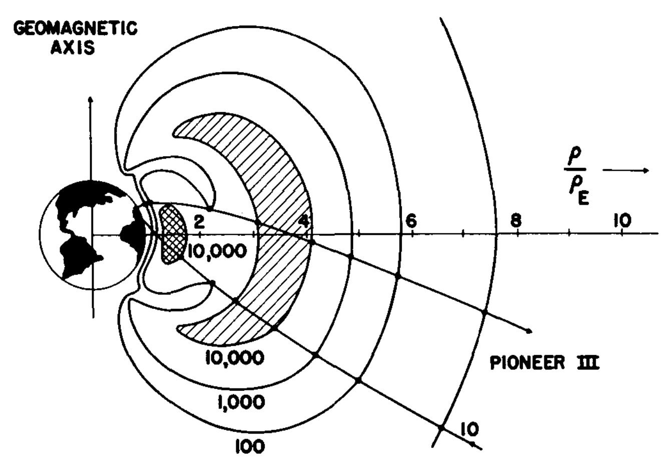
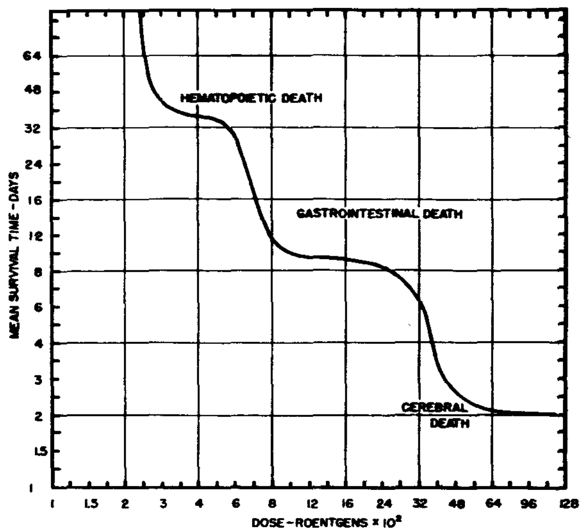


FIGURE 1 The Van Allen Radiation Regions with Pioneer III Trajectory



Causes of death and survival times as a function of dose, based on experimental work with primates, and roughly correct for man. The shortest survival times at the highest dose rates are caused by cerebral and central nervous system damage accompanied by coma and convulsions. At lesser dose levels, some deaths begin to be from gastrointestinal damage, while the incidence of cerebral death decreases. Gastrointestinal death comes from extensive damage to the lining of the intestines, with ulceration, bleeding, diarrhea, excessive water and mineral loss, and infection. Moving down from 1600 r, some deaths begin to appear which are primarily related to damage to the blood-forming tissues. The plateau at 300-500 r and 34 days is made up of deaths from the effects of severe depression of white blood cells, which reduces the ability of the body to combat infection, plus reduction in platelets, leading to continuous bleeding. (From Gerstner, A.F. Sch. of Av. Med. Research Report 58-6, 1957.)

FIGURE 2

Table 1

Dose and Clinical Effect

dose range:	0 - 100 r (subclinical)	100 - 1000 r (therapeutic)			1000 - ∞ r (lethal)	
		100-200r	200-600r	600-1000r	1000-5000r	5000-∞ r
therapy	psychological	surveillance	effective	promising	palliative	
leading organ		hematopoietic tissue		gastro-intestinal tract		CNS
characteristic sign		leukopenia		diarrhea		convulsions
cause of death			hemorrhage infection	circulatory collapse		respiratory failure
time of death			2 months	two weeks		2 days
human data	numerous	200	30	2	1	1

Table 2

RBE Compared with Co^{60} Gamma Radiation

type of radiation \ biol effect:	iridocyclitis	erythema	epilation	desquamation	cataract
Co^{60} gamma	1	1	1	1	1
14 Mev neutrons	14	2	2.3	2.3	1.2
thermal neutrons	.2	.6	.8	.3	.4
730 Mev protons	1	1.0	2.0	2.0	--
910 Mev alphas	2.0	.5	2.0	.7	--

Table 3

Estimated Hazards

Total Period	Dose (Rad)	Leukemia (x spon rate)	Longevity	Sterility	Cataract	Genetics (x spon rate)
1 year	50	3x	?	0	0	2x
3 years	95	6x	?	0	0	3x
5 years	130	9x	1	0	0	4x
10 years	200	12x	1-2 yr	?	0	5x

Table 4

Representative Radiation Events

	energy range	representative events
1.	<10 MEV	gamma rays, neutrons, electrons, beta particles, etc.
2.	10 MEV - 300 MEV	spallation, fission, nuclear excitation followed by neutrons, protons, etc.
3.	300 MEV - 1 BEV	mesons, etc.
4.	>1 BEV	mesons, electron-gamma cascades, thin down events for heavy primaries
5.	>10 BEV	build up of secondaries

THE ROLE OF SIMULATION IN THE DEVELOPMENT OF GEMINI GUIDANCE AND CONTROL

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Abstract

The Gemini Guidance and Control Systems, with their extensive use of man as both a mode selector and as a sensor-operator, have been established with simulation playing a major role. Simulation studies ranging from simple electronic devices duplicating display light sequences during the launch phase to complex optical-mechanical-analog simulation of the docking phase of rendezvous have been performed. This paper outlines the objectives of these studies and describes the special simulation equipment utilized.

A simulated crew station with all the pertinent displays and controls has been constructed adjacent to the analog computer laboratory to permit real time evaluation of guidance and control systems with man in the loop. This facility, used initially to evaluate hand controllers, displays, and attitude control system configurations, has been expanded to include a star projector and a target projector to permit rendezvous guidance studies, and has also been fitted with an abstract presentation of the target vehicle to permit studies of the docking phase. Currently, the system is under modification to provide high fidelity projection of targets and backgrounds by means of a TV projection system.

In addition to extensive all-analog and all-digital simulations, it has been found necessary to develop hybrid programs to achieve a satisfactory compromise between accuracy requirements and solution times. One example of this involves the re-entry phase where exacting navigation computations with a wide dynamic range must be solved concurrently with real time operations by the Gemini crew. Many of these simulations incorporate actual equipment, providing both added realism and rigorous evaluation of the design.

In order to evaluate manual controls and displays in a high gravity environment, a special Gemini cockpit gondola was constructed for use on the Johnsville centrifuge. Although the centrifuge tests of the controls and displays are vital to the program, the Docking Simulator is the most interesting in the field of dynamic simulators. This is a full scale replica of both the Gemini spacecraft and the target which allows six-degrees-of-freedom and permits actual evaluation of the last 100 feet of rendezvous. This simulator, utilizing analog equipment to solve the necessary equations, will permit techniques to be developed for docking from various initial conditions, lighting variations, and under conditions of minor equipment failures (such as target attitude control malfunction).

AUTHOR

Introduction

In the Mercury program, automatic systems were used almost exclusively in the Guidance and Control System with manual operations used solely as optional back-up modes. The experience gained proved that man is fully capable of making decisions and executing control during most phases of a space mission. There are obvious advantages to exploiting this proven capability - such as savings in equipment and providing mission flexibility - therefore, the Gemini project has attempted to fully integrate into the system such manual abilities as decision making, navigation, mode selection, and flight control. Since man relies on cues, displays, and controls to execute his task, development of the system could be accomplished only by using "man-in-the-loop" simulations.

Although manual systems are emphasized, automatic control modes still afford many of the same advantages to a space vehicle that they do to a high performance aircraft. For instance, solutions to problems of launch guidance, re-entry, and long term attitude control are better suited to automatic systems. Simulation played an equally vital role in the development and integration of these automatic systems - particularly the Inertial Guidance and Attitude Control Systems.

Simulations employed in the Gemini development have used digital, analog, and hybrid equipment. Also, use of special purpose simulators such as the Johnsville centrifuge and a full scale docking simulator are planned. Simulation, in the form of a complex mission simulator, will provide invaluable training for the astronauts and personnel in the entire manned ground complex.

Manual Controls and Basic Control Displays

In order to use man's capability for exercising control decision, execution, and feedback sensing, he must be provided with suitable controls and instruments. Sensing and control capability is directly related to the fidelity of the displays and to manual ability to operate the thruster controls. Analytic studies indicated that the attitude and translation thruster locations and magnitudes as shown in Figure (1) should satisfy the control and maneuvering requirements for the Gemini mission. Several manual modes of operating these thrusters had been suggested, and, therefore, an early simulation task was initiated to develop manual controls and basic control displays, determine manual control modes of operation, and verify that thruster levels and characteristics were suitable.

Astronaut control of spacecraft attitude is introduced through an "attitude controller." The following major design criteria were established for this controller:

1. Either astronaut must be capable of executing control in all manual modes.

2. Cross coupling, whether actual or apparent, must be minimized.

3. Must be of minimum weight with sufficient structural rigidity.

4. Must introduce no false inputs or demonstrate greatly degraded feel forces during high "g" environment.

5. Must be capable of efficient use during "hard" pressure suit conditions.

6. Must be capable of actuating six switches and three potentiometers in each control direction.

In the case of the translation controller (used for rendezvous maneuvering), design criteria included minimizing the controller size, weight, and cross coupling potential, and the detrimental effects due to a hard pressure suit. The translation controller was not expected to be used in a high "g" environment, and was designed for use by the command astronaut only. Only one switch operation in each direction is required. The requirements for crew station displays are fairly obvious - they must provide sufficient information in an understandable format to enable the pilot to perform the particular task.

Although designs to satisfy the requirements for the crew station devices were developed, the only sure method of design evaluation required that they be exercised under conditions approximating actual flight. To do this, a fixed base crew station, simulating the Gemini capsule, was constructed. A photograph taken early in the development program is shown in Figure (2). It can be observed that the crew station is tilted up at a 45° angle. This feature was introduced as a result of the 24° angle between the center lines of the two seats, and tends to reduce the discomfort and fatigue that would occur if one or both pilots had to sit at a "bank angle." The fixed base simulator was installed adjacent to the analog computer facilities. One of the first problems examined on the simulator was the evaluation of attitude controllers and displays. For these initial evaluations, a three-degree-of-freedom solution of the Gemini rotational equations (with disturbance moments) was adequate.

As a result of the experience obtained in Mercury, the design of the attitude controller started at a fairly advanced stage, and the development of the Gemini controller was not expected to be difficult. However, the hand controller evolution shown in Figure (3) demonstrates that the best "paper" designs frequently fail under the scrutiny of a realistic simulation. The design requirements for the hand controller proved somewhat contradictory. For example, some of the axes configurations best suited to minimum coupling tendencies were also the most difficult to mass balance for the high "g" situation. Controller (A) was simple but not suited for the "g" environment. Controller (B) was mass balanced, with an acceptable motion and axis location, but the horns interfered with the pressure suit. In Controller (C) an attempt was made to duplicate the axis and motion of Controller (B) without the "horns," using a four bar linkage, but the pitch motion was uncomfortable and the design was complicated. Finally, Controller (D), which features a palm pivot point for pitch and yaw, evolved. The roll axis of the mutually orthogonal system is located some four inches below the palm pivot point. The present design characteristics are shown in Figure (4). The rate command potentiometers have a deadband about the neutral point to eliminate rate inputs due to inexact stick centering.

During the evolution of attitude controls, yaw pedals were suggested and tested as a possible solution to the cross coupling problem, since only two axis hand controls would then be required. Simulation tests soon revealed that, for the particular configurations, pedals were uncomfortable, space consuming, and virtually impossible to use in hard pressure suit operation. The manual control modes selected for use with the attitude controller are:

1. Rate Command Mode - This is a primary manual mode which provides rates proportional to controller deflections.

2. Single Pulse Command Mode - A useful mode for orbit maneuvers. Each controller deflection provides one brief thrust pulse (10 milliseconds).

3. Direct Command Mode - This is a back-up mode which fires the thrusters directly when controller deflections exceed a threshold.

To more fully evaluate the attitude controller, and to permit studies of the translation controller, a six-degree-of-freedom simulation was developed. The simplified docking problem employed will be discussed in the section on docking studies. The evolution of the Gemini translation controller is, again, a demonstration of the effectiveness of simulation for manual systems development. This controller is designed for left hand operation by the command pilot to provide translational thrust in response to intuitive motions - in other words, push to go forward, pull to reverse, raise to go up, etc. Original concepts for this controller employed a fairly massive design which provided one inch deflections in all directions. Evaluation of this design on the simulator disproved these concepts, and the much simpler design shown in Figure (5) evolved. Full scale deflections were reduced to 1/2 inch, and the control forces vary from approximately 4 lbs. at breakout to 5 lbs. at full travel. Direct mode switches operate at 1/4 inch and require negligible force to operate. The controller can be folded for stowage under the panel. The button controller shown in Figure (5) represents another theory disproved by simulation. The removal of the button allowed replacement of the "T" handle by a knob. It was believed that a pulse mode, which would provide a short thrust pulse in the direction of control deflection for each button depression, would provide a desired vernier control capability for the docking maneuver. Simulator studies revealed however, that direct control afforded comparable operation, and also that a potentially dangerous situation could arise when closing in pulse mode due to inability to cancel closing rates rapidly.

As indicated in the photograph of the simulator interior in Figure (6), the control display evolved is a modified version of the Lear-Siegler 4060E Attitude Ball Indicator which had already proven successful for high performance aircraft such as the F4H. Although meters were considered for supplementing the basic attitude ball indications, flight director needle indicators mounted on the face of the indicator were selected. The horizontal and vertical needles across the face of the instrument, and a pointer at the top of the instrument, are used to display pitch, yaw, and roll information, respectively. In all modes, information is presented such that the astronaut's task is to null the needles, which can be driven by attitude, rate, or a mixture of rate and attitude, from platform, radar, or computer outputs. Simulator studies, conducted to establish the meter sensitivities, produced the unexpected results that these values were not critical. Full scale deflections have been set at 5° or $5\ {}^{\circ}/\text{sec.}$ for all modes.

The next simulation step in the development of the controllers, displays, and attitude control modes will be an evaluation in a high "g" environment. These simulation tests are currently scheduled to be run in June - using the Johnsville centrifuge facility. The appropriate controls and displays are being installed in a crew station mock-up which will be installed in the centrifuge gondola. The "g" environment during boost, boost abort, and re-entry will be simulated with a subject monitoring the displays and operating switches and controls to respond to the situations presented.

Boost Phase Abort Display Study

The boost phase of flight leading to Gemini injection into orbit is normally guided by radio commands to the booster, although the spacecraft Inertial Guidance System provides control, if required. Vehicle control is all automatic, but monitoring of the boost phase performance is a critical astronaut task. The astronauts must observe guages indicating fuel and oxidizer pressures, displays of attitude and rate, power system status, chamber pressures, sequence lights, and Environmental Control System displays. The cues available from displays are supplemented in the actual situation by motion and audio cues such as those arising from thrust disturbances. Obviously, a complete evaluation of this launch phase is a complex task. Studies have been conducted with a high degree of realism by Chance-Vought Corporation using their (moving base) Manned Aerospace Flight Simulator. However, early in the program an initial evaluation was made to determine pilot capability of interpreting and reacting to the three display lights which indicate the conditions during the critical period of staging. The simulation was unique in its simplicity and in the results obtained.

The simple display shown in Figure (7) was used to determine the ability of six test pilot subjects to properly interpret lights under both normal and malfunctioning conditions. The indicator on the left is lighted when the chamber pressure in the first stage drops below a threshold value, and, normally should not come on. This indicator is disabled prior to normal first stage burn out. The middle indicator is lighted when the staging command occurs, and remains lit until physical separation is achieved - normally, a period of about 1.8 seconds. The light on the right provides an indication of second stage pressure, and, normally stays lit until second stage reaches operating pressure. In other words, in the normal sequence at staging, the second stage chamber pressure light is on, the staging light comes on and stays on about 2 seconds, and then both lights should go out at about the same time. Should the lights not operate in this sequence, the astronaut has a very limited time in which to initiate abort action.

The first results obtained showed a high rate of both unnecessary aborts, and failures to react to malfunctions requiring abort. Examination of the problem showed that the tolerances for times of indicator illumination had been (erroneously) introduced in a completely random fashion, without any normal pattern or sequence observed. In addition, the pilot was obliged to estimate a fixed time period. When the problem was re-presented with properly correlated occurrences and improved instructions, the subjects developed a "feel" for the proper sequence, and were generally successful in making the proper action. The system was judged to be potentially usable, and the more rigorous tests followed.

Development and Integration of the Inertial Guidance and Attitude Control Systems

In developing the computer portion of the Inertial Guidance System, digital simulation has been used very extensively. This is true for the guidance modes: Back-up Launch Guidance, Rendezvous Guidance, and Re-entry Guidance. It also applies to the prediction modes: Orbit Prediction and Retrograde Firing Time Determination. The general techniques and results of these studies are collected for brevity in this section. Basically, the sequence was first to develop simplified guidance equations and prediction functions compatible with problem requirements and spacecraft computer capacity. The equations and functions were then tested to determine the effects of hardware and mission errors, and modified as required. The resulting computer program was then used to provide prediction data such as fuel requirements, touchdown envelopes, trajectories, operational procedures, etc. Also, the spacecraft computer flow diagram is usually set up and checked out by digital simulation.

In developing the attitude control system, analog simulation was used for studying the automatic control modes as well as the manual modes already discussed. However, an interesting aspect of the analog simulations employed, and planned, is depicted in Figure (8). It is the orderly integration of the Inertial Guidance and Attitude Control Systems by progressively introducing hardware into the closed loop analog simulation. The major test objectives are shown in the figure. The flight hardware expected to be included, ultimately, is indicated by an asterisk. So far, studies involving the Attitude Control Electronics and the Orbital Attitude and Maneuver Electronics packages have been conducted. The rate gyros, horizon sensor, and inertial measurement unit will be mounted on a 3-axis Carco table which will simulate the spacecraft.

Rendezvous Simulation

A key aspect of the Gemini program is the rendezvous in space with an orbiting target vehicle. In its simplest form, the rendezvous problem consists of matching the position and velocity of two bodies in space, and several guidance schemes have been proposed to accomplish this task. One of these schemes uses radar inputs to the Inertial Guidance System which computes the required maneuver from the orbital mechanics equations. This method can be treated as basically static during a computation cycle, and has responded to analysis by pure digital simulations. Other methods for Gemini rendezvous, however, require visual tracking and pilot decisions for successful intercepts, and have, therefore, required "man-in-the-loop" studies.

These simulations have, thus far, been conducted using the fixed base crew station as previously described, but with added equipment to provide the desired visual display. The star projector and target projector, shown in Figure (9), have been installed on the top of the fixed base simulator. The star projector, gimballed in three axes, provides simulated star brightness magnitudes down to that of third order stars without spot size increase. Only a general star field pattern is simulated at present. The target is projected from a two axis gimballed light source driven by computed target line-of-sight angles. The target image is displayed intermittently to represent the flashing light used on the actual target. A flat screen, at a distance of about five feet, is presently used for projection.

A semi-optical rendezvous method has been examined on the simulator, wherein the pilot applies thrust in a manner to stop the apparent motion of the target relative to the stars while maintaining a desired closing schedule based on radar measurements of range and range rate. In the simulator studies, the basic feasibility of this technique has been verified, and a number of performance figures have been provided. Also, some of the problems with such an approach have been pointed up, and, in one case, the simulation proved to be deficient. The latter condition resulted when trajectories were such that very low line-of-sight rates were produced, and the projector servos became inadequate because of granularity. A problem in the technique, which was observed in the simulation, is the difficulty in separating apparent target motions due to spacecraft rotations from the true motions when the line-of-sight rates are very low. An optical device to improve this condition will be evaluated in the next study.

Another rendezvous technique, in which all information on the target is obtained optically, has also been examined on the simulator. This system is still in the development stage, and results have, thus far, been primarily used in designing an optical device to assist in crew interpretation of the visual data. One interesting result was obtained in connection with observations of the target flash rate. When the rate becomes appreciably less than the 90 FPM design value, the target ceases to appear as a continuously tracked image, and the flashes begin to appear as unique targets.

A modification is currently underway to permit closed circuit television projection of the target vehicle for the rendezvous studies. This will provide a realistic image for short range conditions, and will enable an evaluation of the transition conditions leading to the docking phase. A gimballed Agena model, slaved to analog computer signals, will be projected through the TV system on a beam splitting optical mirror, and the star background will be observed through the beam splitter, which serves to blend the images.

Docking Simulation

When the Gemini closes within a few hundred feet of the Agena target, and the relative velocities have been reduced to about 5 ft/sec., the docking phase of the mission is initiated. This task is accomplished through manual spacecraft maneuvers using steering information provided by visual observation of the target. The astronaut also uses spacecraft attitude, body rates, range, and range rate information from the displays. The target vehicle has a radar antenna and other cues which can be used for a relative roll reference. The Agena will be equipped with a docking cone which is designed to absorb the impact forces. The damper system is designed to provide normal engagement under the following relative terminal conditions: one foot radial displacement, one-half ft/sec. radial velocity, one and one-half ft/sec. axial velocity, and ten degrees angular misalignment.

As previously mentioned, a simplified six-degree-of-freedom docking simulation has been used for early docking studies. The abstract oscilloscope display, used for presentation of the Agena target, is shown for various typical docking stages in Figure (10). The fore and aft ends of the target are represented by the large and small circles, respectively, and the horizon line provides a pitch and a roll reference. Circle sizes vary proportionally to range, and the relative circle positions vary to simulate proper aspect angles.

The "man-in-the-loop" simulator studies have indicated that astronauts can efficiently perform the dual task of controlling spacecraft translation and attitude to dock. The performance has been well within the docking cone design capabilities. Two man operation has also been tested with the command pilot controlling translation, and the second pilot controlling attitude. Although this approach proved to be simpler for untrained operators, trained pilots have demonstrated that sharing the control is unnecessary.

A well trained pilot can dock using the direct command mode of attitude control, although the task is complicated by disturbance moments introduced by the translational thrusters. In the rate command mode, these coupling effects are detected and controlled by the automatic operation. As mentioned previously, the results of the simulation indicated that the pulse mode of translation thrusting was an unnecessary feature, and this mode was eliminated.

The simulation described above is a useful design tool, but it is relatively crude and unrealistic, particularly for short ranges. While the planned closed circuit TV system will provide more realism, still, at close ranges, a three-dimensional system is required. This feature will be available in a full scale simulator, which has as its major objective the training of astronauts. An artist's conception of this docking simulator is shown in Figure (11). In its early use, the device will be used for design studies. It will then be shipped to NASA to be used primarily as a trainer. Crew compartments and visible portions of the full scale vehicle replicas will be as realistic as is feasible. An isometric drawing of the docking trainer is shown in Figure (12). Three degrees of relative translational motion are provided, with the Agena moving longitudinally and vertically, and the Gemini moving laterally. The Gemini will be gimballed for pitch, yaw, and roll rotational freedom. The Agena replica is mounted in a 40 foot tower on a counterbalanced sting, and has no angular freedom except for safety measures. An air bearing support system permits vehicle motion approximating the zero drag conditions of space. The Gemini yaw gimbal also uses an air bearing. In addition to providing the training capability, this simulator will allow the circumstances at spacecraft-target contact to be determined under conditions of high visual fidelity.

Re-entry Simulation

A major innovation in re-entry of manned spacecraft will be demonstrated in Gemini. Steering will be exercised during the re-entry to bring the spacecraft to a prepared land landing site. The heart of the guided re-entry is the Inertial Guidance System, which provides constant knowledge of spacecraft position and velocity, predicts the touchdown point if the trajectory is maintained, and computes the down range and cross range distances between predicted and desired landing points. Trajectory control is exercised by properly directing the lift vector produced by the offset center of gravity used in the re-entry module. This lift vector is oriented by rolling the spacecraft in response to bank command logic, which is based on computed range errors as shown in Figure (13). (Note that positive lift occurs in the head down position). The bank angle commands may be introduced to the automatic control system with no astronaut control required. The computed commands are also displayed, and the astronaut may manually control during re-entry. Because of inherent cross coupling of body rates and the high natural frequency of oscillation during re-entry, the manual control task is considered a difficult one, requiring "man-in-the-loop" simulation.

This simulation has been initiated using the fixed base crew station in a semi-closed loop program. The attitude modes are accurately simulated and respond properly to operator inputs, but trajectory data is programmed and does not react to simulator maneuvers. The trajectory information is provided by tapes which were obtained from all-digital programs where automatic attitude control was exercised. Mach Number, dynamic pressure, velocity, flight path angle, bank angle, and down range and cross range errors are provided by the tape, and the displays are appropriately driven. Unless the pilot makes drastic errors in responding to commands, the simulation is good and has served to establish the basic feasibility of manual control during re-entry. Although some of the expected coupling effects were observed, the astronauts proved adept at controlling the problem.

Full six-degree-of-freedom simulations are being developed to afford better evaluation of the trajectories as influenced by pilot action. The wide dynamic range of the trajectory variables, and the accuracies required in the navigation computations, indicate a need for solution on a digital computer. Whereas the solution of the rotational equations in real time with the dynamics involved, requires fast or continuous solution, as obtained with an analog computer. A compromise solution to the problem, using a hybrid (analog and digital) approach, has been designed. In this method, the translation equations are solved on a digital computer, and the rotational equations on an analog computer - with converter equipment providing the communication link. In addition, a completely digital solution to the problem has been programmed which provides a problem solution in faster than real time. This simulation, slowed to real time, is currently being investigated for use in "man-in-the-loop" re-entry studies.

Both the all-digital and the hybrid solutions become very expensive if they require full time use of the IBM 7094 digital computer. For the hybrid solution, a method has been developed by IBM-Federal Systems Division which permits the digital computer to be shared with other problems. This reduces the cost of the problem to approximately that of an analog computer solution.

Mission Simulator

The success of the Mercury program was due, in part, to the use of simulation in systems' checkout, and in training the astronaut as well as the ground complex. The increased complexity of the Gemini mission, particularly with regard to astronaut and ground network communications during launch and catch-up, requires even more training of the entire mission complex. For this reason, a mission simulator is being developed which is capable of operation in the following modes:

1. Alone for astronaut training only,

2. With the Mission Control Center,

3. With the Mission Control Center and ground tracking network computer complex,

4. With Mission Control Center, tracking network computer complex, and the Burroughs launch guidance computer system.

Voice communications, telemetry, teletype, and digital command system links, as well as the hard line connections to control and monitoring centers, are required. The mission from pre-launch through paraglider touchdown will be simulated.

To meet the training and checkout requirements which have briefly been outlined, the Gemini Mission Simulator shown in Figure (14) is being developed. It utilizes a fixed base flight trainer in conjunction with a digital computer complex, and peripheral "black boxes" simulating various onboard systems. A digital computer was selected because of reliability and accuracy requirements. Growth potential of the mission simulator includes provision for an out-the-window display which is not presently included. The mission simulator will allow instructor initiated and programmed fault insertion into the system simulations. The system is expected to meet the present design requirements, provide spare capacity for growth, and offer considerable flexibility through reprogramming to change equations, emphasize certain flight phases, etc. It should provide the capability to train the astronauts and ground network personnel, and also exercise as much operational equipment as is feasible.

Conclusion

It is difficult to oonceive where the Gemini program would be without simulation, but fortunately, this speculation is not necessary. The Guidance and Control System development has utilized various analog, digital, hybrid, and miscellaneous simulation techniques to:

1. Determine system design criteria and evaluate design concepts,

2. Evaluate man's capabilities,

3. Test flight hardware in closed loop operation in simulated environments,

4. Develop procedures and techniques,

5. Optimize controls and instruments,

6. Check spacecraft computer program,

7. Perform error analysis,

8. Train astronauts and ground crew personnel,

9. Predict performance.

The use of these techniques has facilitated the early and efficient development of a Guidance and Control System which takes full advantage of man's capabilities while providing advanced "state-of-the-art" Inertial Guidance System Modes to accomplish tasks never before undertaken in a space mission.

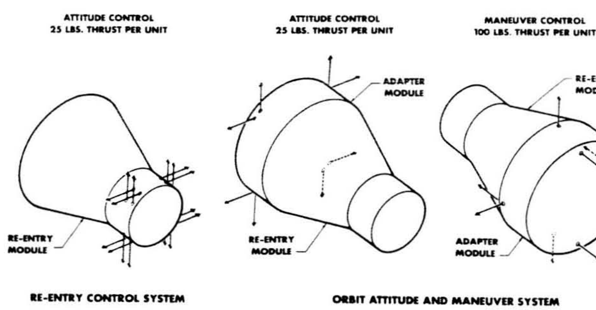


Figure 1 THRUST CHAMBER ARRANGEMENT

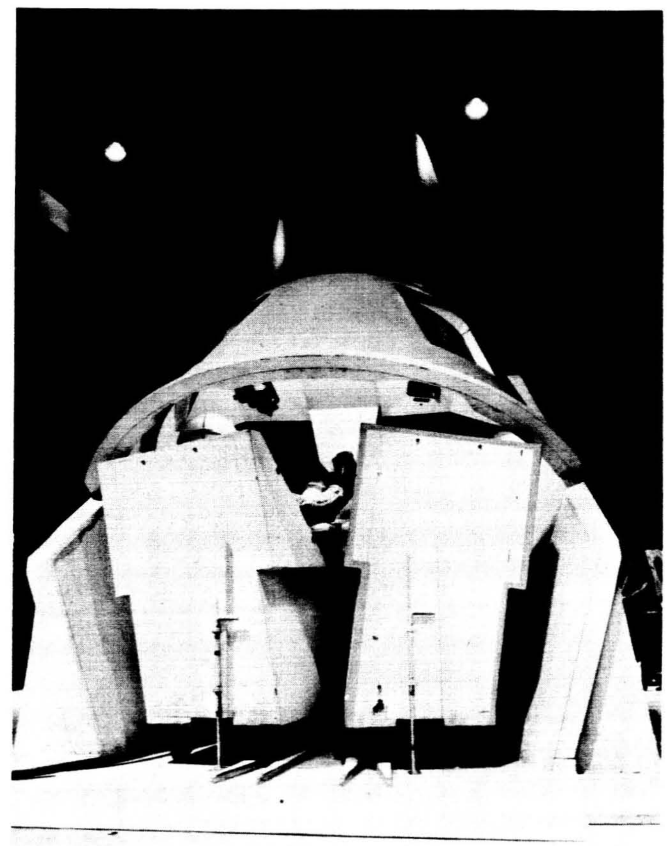


Figure 2 GEMINI FIXED BASE CREW STATION

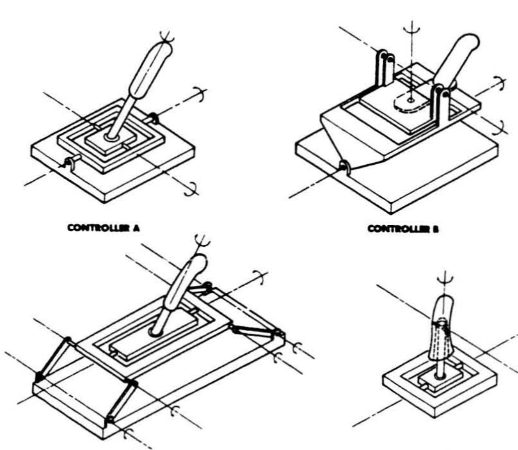


Figure 3 GEMINI ATTITUDE CONTROLLER EVOLUTION

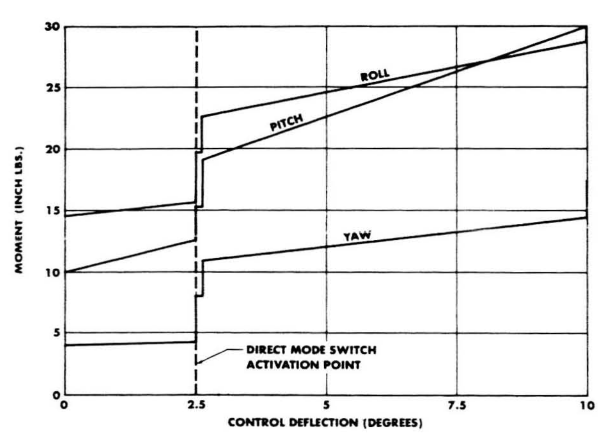


Figure 4 GEMINI 3-AXIS ATTITUDE CONTROLLER CHARACTERISTICS

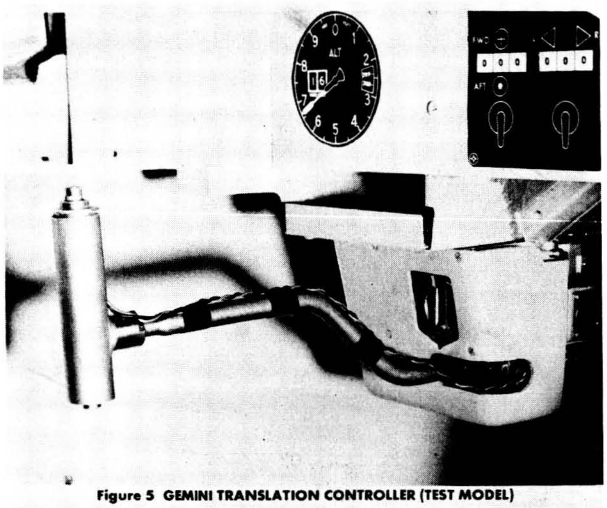


Figure 5 GEMINI TRANSLATION CONTROLLER (TEST MODEL)

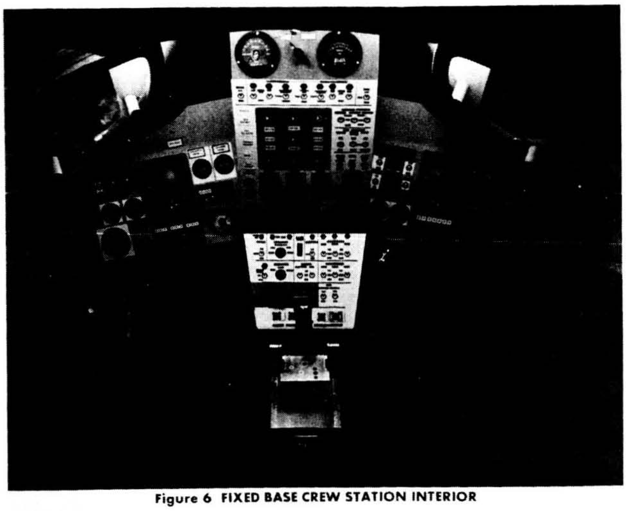


Figure 6 FIXED BASE CREW STATION INTERIOR

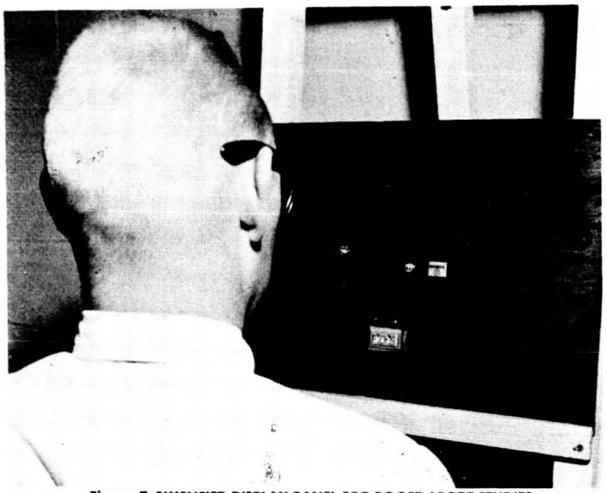
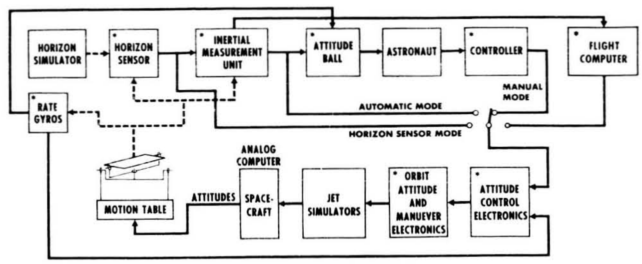


Figure 7 SIMPLIFIED DISPLAY PANEL FOR BOOST ABORT STUDIES



TEST OBJECTIVES

1. CONTROL SYSTEM DYNAMIC PERFORMANCE
   - FUEL USAGE
   - COUPLING AT EXTREME ATTITUDES
   - NON-LINEAR EFFECTS
   - GIMBAL FLIP
   - STABILITY
2. IN-FLIGHT PLATFORM ALIGNMENT
   - ALIGNMENT PROCEDURE
   - EFFECT OF VEHICLE DYNAMICS
   - EFFECT OF MANUAL CONTROL

Figure 8 3-AXIS MOTION TABLE TEST

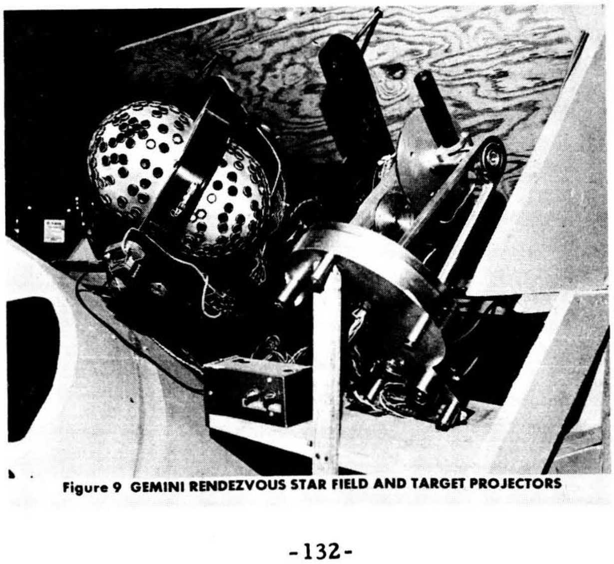


Figure 9 GEMINI RENDEZVOUS STAR FIELD AND TARGET PROJECTORS

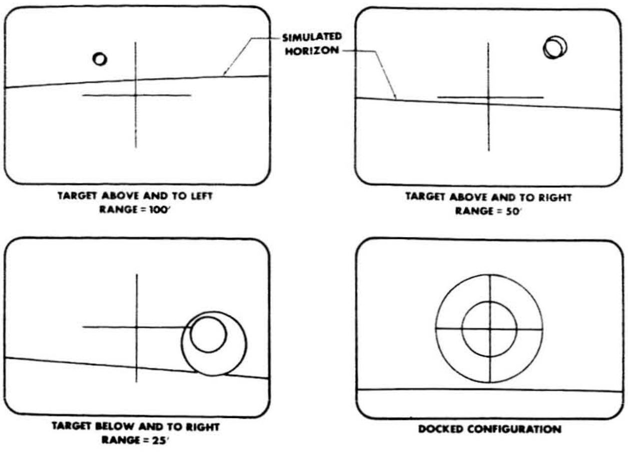


Figure 10 SIMPLIFIED DOCKING DISPLAY

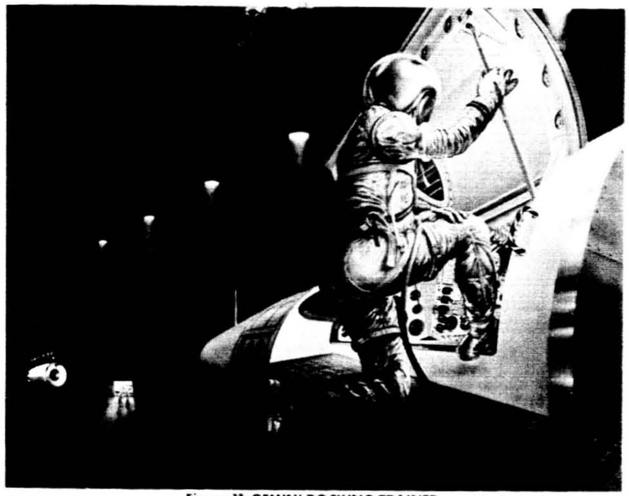


Figure 11 GEMINI DOCKING TRAINER

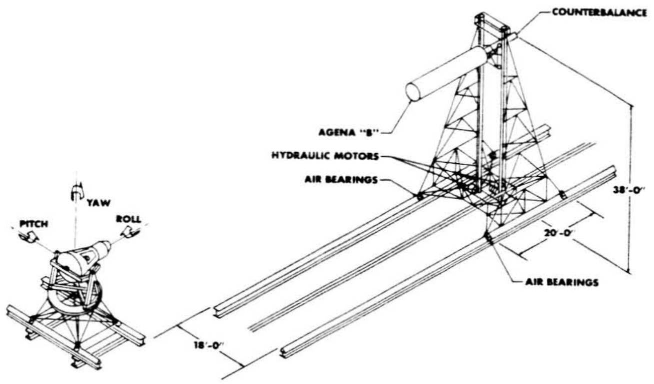


Figure 12 TRANSLATION AND DOCKING TRAINER

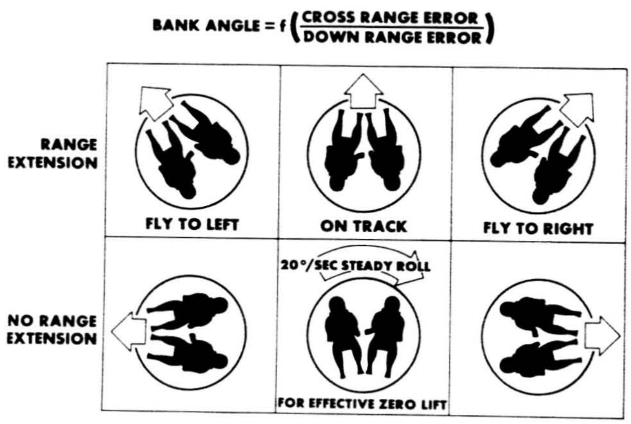


Figure 13 RE-ENTRY CONTROL LOGIC

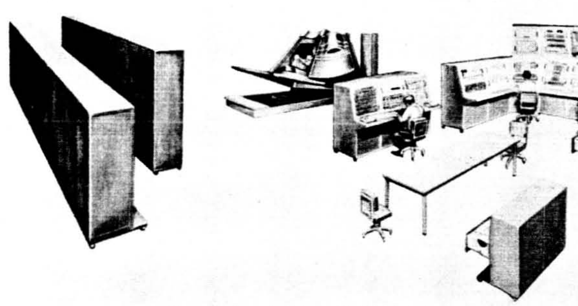


Figure 14 GEMINI MISSION SIMULATOR

A Study of Certain Aspects of Lunar Ascent and Rendezvous With an Orbiting Vehicle

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Introduction

In the investigation of space missions such as the lunar orbit rendezvous mission it is often necessary to investigate the relative motion betwenn two bodies which are on free fall trajectories in a central force field. Frequently one desires only preliminary results of limited accuracy and in such cases the use of the Keplerian equations of motion may be too time consuming. Often the area of interest concerns the motion of a body with respect to a second reference body which moves in a nominally circular orbit. Elsewhere, a set of linearized equations has been developed which describe the motion of a point mass with respect to a cartesian system of coordinates whose origin moves at orbital rate on a circular reference orbit.[1,2] These equations lend themselves to the study of the problem stated above. However, the choice of a cartesian system of coordinates requires that the relative motion be restricted to the vicinity of the origin which prevents the application of these equations to many situations of interest. In this paper a linearization of the equations of motion in a curvilinear system of coordinates is described whose origin also moves at orbital rate on a circular reference orbit. In this case the relative motion is restricted only to the vicinity of the reference orbit and not necessarily to the vicinity of the moving origin. The resulting equations can, therefore, be used to study a larger class of problems.

To illustrate one of the many ways these relations may be used in preliminary design the lunar launch and ascent phase of the lunar orbit rendezvous mission is considered below. In this phase a vehicle is launched from the lunar surface towards a rendezvous with a second vehicle which is in an orbit around the moon. In planning this maneuver one will, in general, have decided on some nominal ascent plan which will require that the first vehicle be launched at a precise moment in order to achieve the rendezvous. However, if one permits deviations from the nominal ascent plan this restriction can be relaxed and a tolerance on the launch time gained. It is the object of the second part of this paper to examine this question and to study the limitations which might be put on these deviations.

Equations of Motion

Consider a point mass which moves in a central force field. In this section equations will be presented which describe the motion of this point mass with respect to a rotating system of coordinates, whose origin moves at orbital rate on a circular reference orbit around the force center.

This particular problem has, of course, been solved previously.[1,2] Usually, a rotating system of cartesian coordinates is used as shown in Figure 1: The x-axis is along the tangent to the circular reference orbit, the y-axis is along a radial line and the z-axis is perpendicular to the reference orbit plane so that a right-handed system of coordinates is formed. If one restricts the relative displacements between the systems origin and the point mass to small values compared to the radius of the circular reference orbit, the differential equations describing the relative motion can be linearized and closed form solutions may be obtained. In this paper a rotating system of curvilinear coordinates is proposed which permits a linearization of the differential equations of relative motion under less restrictive conditions. Closed form solutions are, therefore, obtained which are valid over a larger domain.

The proposed system of rotating coordinates is shown in Figure 2. The origin O rotates at orbital rate ω on the circular reference orbit of radius R around an inertially fixed force center O_o. Let P be a point mass and let P' be the projection of P onto the reference orbit plane. The position of P with respect to the origin O is defined by the three components s, h, z which are shown on Figure 2 and which are defined as follows:

- s is the length of arc measured along the reference orbit circle from O to the line $O_o P'$
- h is the radial distance from the reference orbit circle to P'
- z is the distance $P P'$ measured perpendicular to the reference orbit plane and directed so that a right handed system of coordinates result.

It can be shown that the motion of P in the curvilinear x, h, z-system can be expressed by the following three differential equations.

$$\left(1 + \frac{h}{R}\right) \frac{\ddot{s}}{R\omega^2} + 2 \frac{\dot{h}}{R\omega} \left(1 + \frac{\dot{s}}{R\omega}\right) = 0 \qquad (1a)$$

$$-\left(1 + \frac{h}{R}\right) - \frac{\dot{s}}{R\omega}\left(2 + \frac{\dot{s}}{R\omega}\right)\left(1 + \frac{h}{R}\right) + \frac{\ddot{h}}{R\omega^2} = - \frac{R^2 (R + h)}{\left[(R + h)^2 + z^2\right]^{3/2}} \qquad (1b)$$

$$\frac{\ddot{z}}{R\omega^2} = - \frac{R^2 z}{\left[(R + h)^2 + z^2\right]^{3/2}} \qquad (1c)$$

Equations (1) can be linearized if one specifies that:

$$\frac{h}{R} \ll 1 \qquad \frac{z}{R} \ll 1 \qquad \frac{\dot{s}}{R\omega} \ll 1$$

It is important to note that no restriction has to be placed on the quantity s. After linearization equations (1) become:

$$\ddot{s} = -2\omega\dot{h} \tag{2a}$$

$$\ddot{h} = 3\omega^2 h + 2\omega\dot{s} \tag{2b}$$

$$\ddot{z} = -\omega^2 z \tag{2c}$$

Equations (3) can be integrated readily and with the initial conditions for $t = 0$:

$$s = s_o \qquad \dot{s} = \dot{s}_o$$
$$h = h_o \qquad \dot{h} = \dot{h}_o$$
$$z = z_o \qquad \dot{z} = \dot{z}_o$$

one obtains

$$s = s_o - (6h_o + 3\frac{\dot{s}_o}{\omega})\omega t + (6h_o + 4\frac{\dot{s}_o}{\omega}\sin\omega t) - 2\frac{\dot{h}_o}{\omega}(1 - \cos\omega t) \tag{3a}$$

$$h = h_o + \frac{\dot{h}_o}{\omega}\sin\omega t + (3h_o + 2\frac{\dot{s}_o}{\omega})(1 - \cos\omega t) \tag{3b}$$

$$z = z_o \cos\omega t + \frac{\dot{z}_o}{\omega}\sin\omega t \tag{3c}$$

One should point out here that a certain amount of care has to be exercised in interpreting the meaning of the rate $\dot{s}$. It is not the difference between the tangential speeds of P' and O, but rather a measure of the difference in angular rate between the two. The difference in tangential speed can be shown to be

$$\Delta v_t = \dot{s} + \omega h$$

It is interesting to note that these equations are formally identical to the equations in Ref. 1 and 2 which were developed for a system of coordinates as shown in Figure 1. These latter equations can be obtained from equations (3) simply by exchanging s and h by x and y. However, one must bear in mind, that they are valid only as long as all displacement components x, y, z are small, i.e. only in the immediate vicinity of the origin. On the other hand, in the derivation of equations (3) no restriction had to be placed on the magnitude of the coordinate s. Thus the equations are valid anywhere in the vicinity of the reference orbit and are not necessarily restricted to the vicinity of the origin. What one does, in fact, by going from the x, y, z system to the s, h, z system is to exchange a limitation on a displacement for a limitation on a rate. Whereas, in the first case x is limited but the rate $\dot{x}$ is not, in the second case s is not restricted while $\dot{s}$ must be small compared to $R\omega$.

In order to demonstrate the meaning of this difference consider the example illustrated in Figure 3. Let the origin of a rotating system of coordinates move on a circular orbit of radius R. A point mass originally located at this origin is given a tangential impulse so that it enters the elliptic orbit shown in Figure 3. The subsequent motion is now of interest as it appears in the rotating system of coordinates shown in Figure 2 (the s, h, z - system) and as it appears in the system shown in Figure 1 (the x, y, z - system). The relative position as it is measured in the two systems is illustrated in Figure 3.

Figure 4 shows the motion in the s, h, z - system as computed using equations (3) and also as computed using the equations of Keplerian mechanics. The computations were carried out assuming the circular orbit to be an orbit around the moon at an altitude of 50,000 ft. above the surface and the elliptic orbit to be an orbit with an apocynthion altitude of 100 n.mi. It is seen from Figure 4 that there is substantially good agreement between the two cases. On the other hand, Figure 5 shows the motion as it appears in the x, y, z - system computed with the equations of Keplerian mechanics and also with the linearized equations of Reference 1 (i.e., equations (3) with x and y replacing s and h). Clearly, after only a short time the discrepancies between the linearized equations and the equations of Keplerian mechanics become intolerably large. Figures 4 and 5 illustrate the motivation which led to the choice of the s, h, z system.

Since they have the property of not being restricted to the vicinity of the origin but being sufficiently valid anywhere in the vicinity of the reference orbit equations (3) are particularly useful for the study of orbital transfers. In what follows they will be applied to the study of certain aspects of lunar ascent and rendezvous.

Ascent to Rendezvous with Orbiting Vehicle

Consider two vehicle in coplanar circular orbits in a central force field as shown in Figure 6. Let the higher of the two orbits have a radius R and orbital rate ω, and let the lower orbit be of radius $R - \Delta R$. Assume, now, that it is desired to have the lower of the two vehicles ascend to the higher orbit for a rendezvous with the second vehicle. Let us further specify that all thrusting maneuvers be purely impulsive.

The optimal way to execute the maneuver defined above is to let the lower vehicle be transferred to the higher orbit by way of a Hohmann ellipse. In order to achieve a successful rendezvous the Hohmann ellipse must represent a collision course, which requires that the two vehicles have a specific relative position with respect to each other at the moment when the transfer is initiated. If a transfer is initiated when the relative position of the vehicles is different another trajectory has to be flown in order to achieve rendezvous.

In Figure 6 a typical transfer trajectory is shown. It intersects the high orbit at points A and B, thus offering two opportunities for a rendezvous. Since the flight times on the transfer trajectory are different for the two points, the required initial relative vehicle positions must be different for rendezvous at A or B, as is indicated on Figure 6.

If the impulse by which the vehicle is injected into the transfer is applied tangentially to the lower orbit then the periapsis of the resulting trajectory coincides with the point of injection. Such trajectories are said to have a clear periapsis; they are of interest because a vehicle which travels on such a path will never collide with the central body.

For the discussion that follows let the origin of the rotating s, h, z system of coordinates coincide with the vehicle which is in the higher of the two orbits. To begin with, it is useful to express in this system of coordinates the motion of the other vehicle while it is still in the lower orbit. Presuming that the two orbits are coplanar and circular with a radius difference ΔR one can obtain from equation (3) the following non-dimensional equations.

$$\frac{s}{\Delta R} = \frac{s_o}{\Delta R} + \frac{3}{2}\omega t \qquad (4a)$$

$$\frac{h}{\Delta R} = -1 \qquad (4b)$$

$$\frac{z}{\Delta R} = 0 \qquad (4c)$$

Now, consider the transfer of the vehicle in the lower orbit. Let subscript o denote conditions just after the injection into the transfer and subscript 1 conditions at the time when the two vehicles meet. Also, let the development again be restricted to planar maneuvers.

The initial relative position of the two vehicles is given by the quantities s_o and $h_o = -\Delta R$. If, in addition one chooses the time of flight $t_a = \varphi a/\omega$ from injection to rendezvous the transfer is fully defined. In general, this transfer will of course not have a clear periapsis.

In order to achieve rendezvous at t_a the quantities s and h must become zero at $\omega t = \varphi_a$. Thus, from equations (3) and their derivatives one obtains the non-dimensional expressions:

$$\frac{\dot{s}_o}{\omega \Delta R} = \frac{\frac{s_o}{\Delta R}\sin\varphi_a + 6\varphi_a \sin\varphi_a - 14(1-\cos\varphi_a)}{3\varphi_a \sin\varphi_a - 8(1-\cos\varphi_a)} \qquad (5a)$$

$$\frac{\dot{h}_o}{\omega \Delta R} = \frac{-2\frac{s_o}{\Delta R}(1-\cos\varphi_a) + (3\varphi_a\cos\varphi_a - 4\sin\varphi_a)}{3\varphi_a \sin\varphi_a - 8(1-\cos\varphi_a)} \qquad (5b)$$

$$\frac{\dot{s}_1}{\omega \Delta R} = \left(6 - 3\frac{\dot{s}_o}{\omega\Delta R}\right) - \left(6 - 4\frac{\dot{s}_o}{\omega \Delta R}\right)\cos\varphi_a - 2\frac{\dot{h}_o}{\omega\Delta R}\sin\varphi_a \qquad (5c)$$

$$\frac{\dot{h}_1}{\omega \Delta R} = -\left(3 - 2\frac{\dot{s}_o}{\omega\Delta R}\right)\sin\varphi_a + \frac{\dot{h}_o}{\omega\Delta R}\cos\varphi_a \qquad (5d)$$

The ΔV required for injection into the transfer trajectory is obtained from equations (4a), (5a), (5b)

$$\frac{\Delta V_o}{\omega \Delta R} = \left[\left(\frac{\dot{s}_o}{\omega\Delta R} - \frac{3}{2}\right)^2 + \left(\frac{\dot{h}_o}{\omega\Delta R}\right)^2\right]^{\frac{1}{2}} \qquad (5e)$$

The ΔV for insertion into the higher altitude orbit simply becomes

$$\frac{\Delta V_1}{\omega \Delta R} = \left[\left(\frac{\dot{s}_1}{\omega\Delta R}\right)^2 + \left(\frac{\dot{h}_1}{\omega\Delta R}\right)^2\right]^{\frac{1}{2}} \qquad (5f)$$

Equations (5) are valid for any type of ascent and are not restricted to clear periapsis trajectories. In order to obtain a clear periapsis a particular relation must exist between the quantities $s_o/\Delta R$ and φ_a which may be obtained from equation (5b) by letting $\dot{h}_o/\omega\Delta R = 0$.

For the special case of the Hohmann transfer it can be shown that

$$\frac{\Delta V_0}{\omega \Delta R} = \frac{\Delta V_1}{\omega \Delta R} = .25$$

In addition, the equations of a Hohmann transfer in the s, h system can be derived from equations (3) and one obtains the dimensionless expressions:

$$\frac{s}{\Delta R} = -\frac{3}{4}\pi + \frac{3}{4}\omega t + \sin \omega t$$

$$\frac{h}{\Delta R} = -\frac{1}{2}(1 - \cos \omega t)$$

It appears from these equations that a Hohmann transfer takes place over $\omega t = \pi$ i.e., over the time it takes the target vehicle to complete one half revolution. This is, of course, in error and is a consequence of the linearization process.

Equations (5) were evaluated numerically and the results are shown on Figures 7 and 8 where the quantity $\Delta V/\omega\Delta R = \Delta V_0/\omega\Delta R + \Delta V_1/\omega\Delta R$ is plotted versus initial relative position $s_0/\Delta R$ with φ_a as parameter. Figure 7 gives results for $\varphi_a \leqslant 180^o$. On Figure 8 results are plotted for $\varphi_a \geqslant 180^o$. These two figures show the characteristics of all possible ascent trajectories which result in a rendezvous within the ranges of the independent variables given.

Consider now only ascent trajectories which are characterized by clear periapsides. Since a vehicle which is on such a path will never collide with the central body these trajectories are attractive for manned missions, where safety is of prime interest. As mentioned before, for each value of $s_0/\Delta R$ there is only one value of φ_a which will result in a clear periapsis trajectory. The loci of all points which have this property are shown in Figure 7 and 8 as broken lines. Both of these lines start at the Hohmann point but their behavior thereafter is quite different. For a rendezvous during the first 180^o (i.e., at a point of type A on Figure 6) the ΔV-requirements rise very steeply as one moves away from the Hohmann point. There is, therefore, only a very narrow region of the initial relative position $s_0/\Delta R$ where a clear periapsis ascent is possible. Even within this region large Δ V-savings might be achieved if one were to abandon the clear periapsis requirement. On the other hand it is seen from Figure 8 that for rendezvous during the second 180^o (i.e., at a point of type B on Figure 6) clear periapsis trajectories can be obtained over a large range of initial relative position $s_0/\Delta R$ and with relatively little cost in Δ V.

From the data contained in Figures 7 and 8 one may derive some information about the launch window for ascent from the lunar surface to rendezvous with an orbiting vehicle, such as the ascent of the lunar excursion module (LEM) to the orbiting Command and Service Module (CSM) in the Apollo Mission. To that end it is first necessary to define what is meant by the term launch window.

Assume that the ascent consists of two distinct phases; during the first phase the LEM is launched into a circular orbit of low altitude; the second phase consists of injection of the vehicle into a transfer trajectory towards rendezvous with the CSM. For the nominal case it is assumed that the transfer trajectory is a Hohmann ellipse and that both phases are accomplished by one single continuous thrust application. This maneuver requires precise timing in order to achieve the proper relative vehicle position so that the Hohmann is a collision course.

Should launching not occur at the nominal time the maneuver has to be changed in order to achieve rendezvous. For such cases it is assumed that the launch into the circular orbit remains unchanged and that all timing errors are made up by changes in the transfer trajectory. Let it also be stipulated that only trajectories which have a clear pericynthion are acceptable. With these rules it is possible to determine the ΔV-penalties which are associated with improper launch times. One may now define the launch window as the time span during which the vehicle may be launched without exceeding a specified maximum permissible ΔV-penalty. However, as will be demonstrated below, this definition is not sufficient and will be modified.

Assume that the LEM is launched from the lunar surface at a time which differs from the nominal time by an amount Δ t. Upon reaching circular speed it will miss the proper position for a Hohmann transfer by the amount $\Delta s_0 = R \omega \Delta t$. Knowing this quantity and referring to Figures 7 or 8 one may determine the Δ V penalty associated with this miss.

The results of such an investigation are shown on Figure 9. The LEM low altitude lunar orbit was assumed to be 50,000 ft; CSM orbit altitudes of 50 n.mi., 100 n.mi. and 150 n.mi. were considered. Inspection of Figure 9 shows, that depending on the permissible Δ V-penalty, a fairly substantial launch window exists for early launches. Compensation for an early launch is achieved by injection of the vehicle into a trajectory with $\varphi_a > 180^o$ (points B on Figure 6) right after it has reached circular speed at the low altitude. Such trajectories, however, cannot offer compensation for late launches. Also included on Figure 9 are curves which show Δ V versus launch time for the cases where one injects the vehicle, immediately after it has reached circular velocity at low altitude, into a trajectory with $\varphi_a < 180^o$ (points A on Figure 6). Clearly these flight paths are not useful; they require high ΔV and permit practically no compensation for late launches. Late launches can be accommodated if one permits the LEM to remain a certain length of time in the low altitude orbit before one injects it into the transfer trajectory, i.e., by the well known phasing orbit technique. In this technique one takes advantage of the fact that, if launched late, the LEM will eventually move into the position which is proper for a Hohmann transfer, by virtue of the difference in period between the high and the low orbit. Obviously, this technique does not involve a ΔV-penalty. The longer the launch is delayed the longer the vehicle will have to remain in the low altitude circular orbit. Eventually a delay time will be reached where it is no more time-consuming simply to remain and wait on the lunar surface until the next opportunity arises for the nominal launch, approximately one CSM revolution later.

One may now define the term launch window differently for early and late launches. For early launches the definition above still holds; for late launches one may consider the limit of the launch window to be the time at which waiting on the lunar surface for the next launch opportunity does not introduce any loss of time.

Returning to Figure 9 the following concluding remarks may be made. As the CSM park orbit altitude is increased the launch window for late launches increases greatly, i.e., from 420 sec for 50 n.mi. to 1580 sec. for 150 n.mi. On the other hand the launch window for early launches decreases with increasing CSM altitude although this altitude dependence is not strong. If one, as an example, permits a ΔV penalty of $\Delta V = 200$ fps the launch window decreases from 170 sec. at 50 n.mi. to 150 sec. at 150 n.mi.

Acknowledgement

The author would like to express his appreciation to Dr. L. E. Matson of Radio Corporation of America for suggesting that the curvilinear coordinates discussed above be used and for his advice in executing the work reported in this paper.

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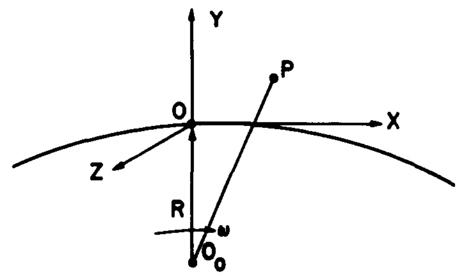
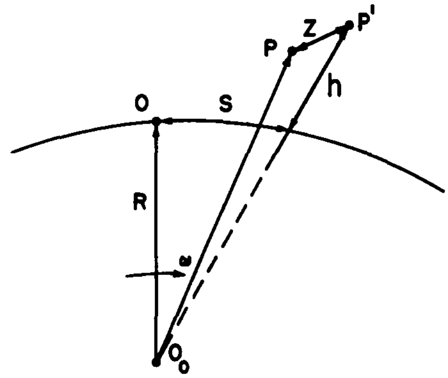


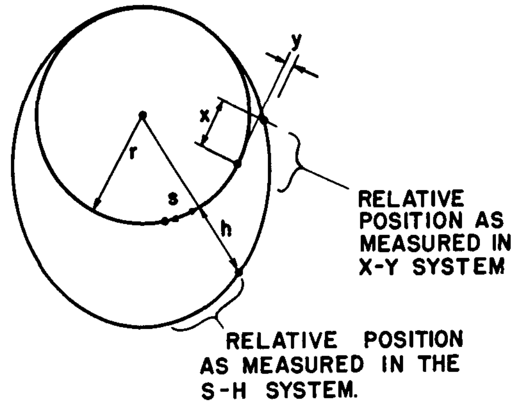
ROTATING SYSTEM OF CARTESIAN COORDINATES

FIG.1



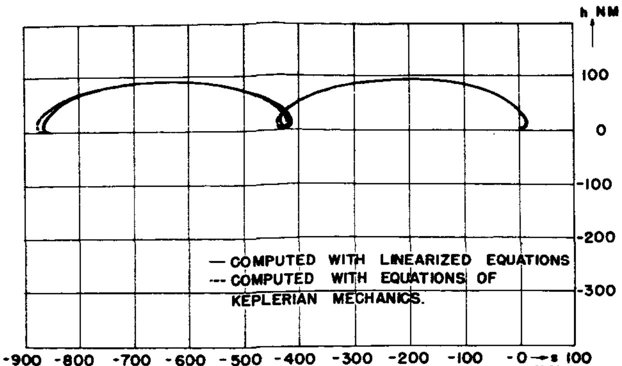
ROTATING SYSTEM OF CURVILINEAR COORDINATES

FIG.2

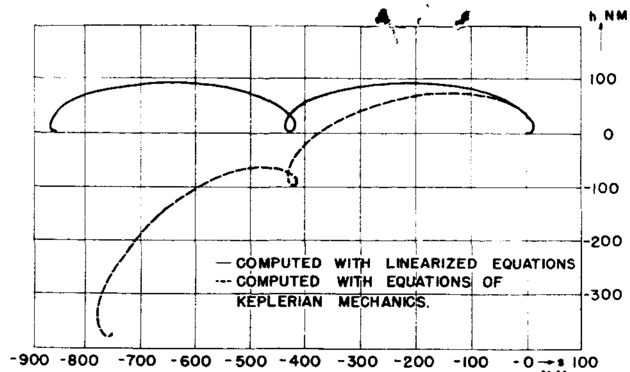


POINT MASS IN ELLIPTIC ORBIT

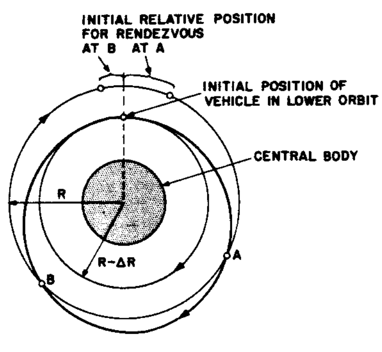
FIG.3



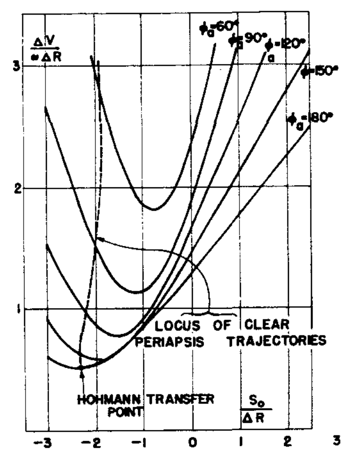
ELLIPTIC PATH OF FIG.3 AS SEEN IN ROTATING s-h SYSTEM OF COORDINATES
FIG.4



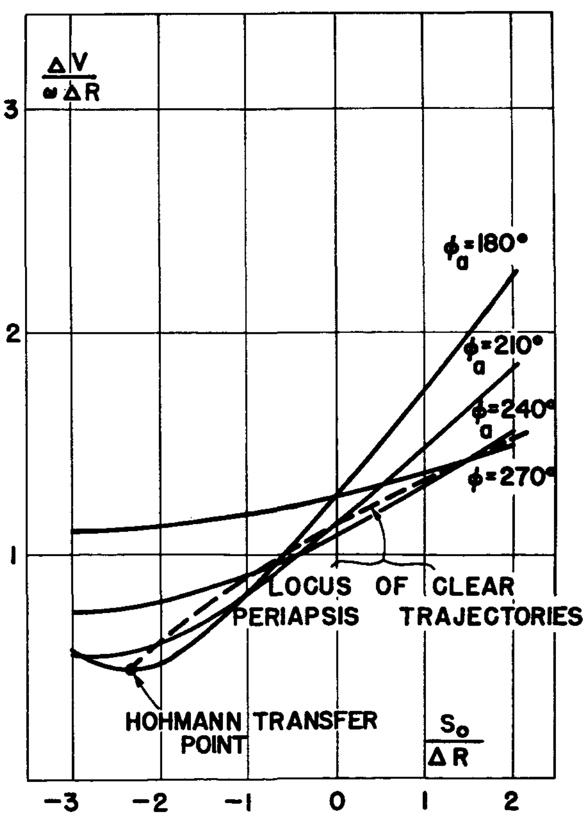
ELLIPTIC PATH OF FIG.3 AS SEEN IN ROTATING x-y SYSTEM OF COORDINATES
FIG.5



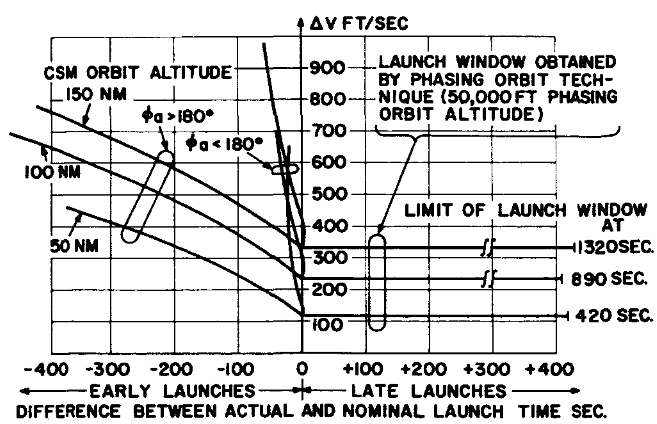
CO PLANAR CIRCULAR ORBITS WITH TYPICAL TRANSFER TRAJECTORY
FIG.6



CHARACTERISTICS OF ASCENT TRAJECTORIES
FIG 7



CHARACTERISTICS OF ASCENT TRAJECTORIES CONTINUED
FIG.8



LAUNCH WINDOW FOR LUNAR LAUNCH OF LEM TOWARDS RENDEZVOUS WITH CSM
FIG.9

LUNAR LANDING AND LONG-RANGE EARTH REENTRY GUIDANCE BY APPLICATION OF PERTURBATION THEORY

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Summary

A guidance scheme has been investigated which has as its basis linear perturbation theory. An improved capability of the scheme has been achieved by the proper choice of independent variable and by appropriate weighting of the guidance gains computed on the basis of the linear theory.

The capability of this guidance scheme applied to the descent-to-hover phase of lunar landing is demonstrated for two different types of nominal trajectory; a constant-thrust gravity turn maneuver, and a constant-thrust, constant-pitch-rate maneuver.

For the purpose of demonstrating the performance of this type of guidance scheme for atmosphere entry, it has been applied to the guidance of a vehicle entering the earth's atmosphere at parabolic velocity. The guidance capability of this control scheme is evaluated for entries from abort conditions as well as for entries within the normal entry corridor; in addition, the effects upon the guidance capability of variations in lift-drag ratio and atmospheric density are investigated.

It is shown that for both lunar landing and atmosphere entry this guidance system, which uses a single nominal trajectory, permits guidance to a selected landing site from a wide range of initial conditions. Since a single nominal trajectory is used, only minimum storage capacity is required.

AUTHOR

Introduction

In the field of guidance of aerospace vehicles, the concept of guidance about a nominal or reference trajectory has received considerable attention. It has been investigated for use in the midcourse phase of the lunar mission, and for vehicles reentering the earth's atmosphere (e.g., refs. 1 to 5). The mathematical basis for this concept is perturbation theory[6,7]; that is, the analysis of conditions in a limited neighborhood of a nominal trajectory. Nonlinear systems may be handled by the theory because in the neighborhood of a known trajectory they can be described in terms of linear differential equations with varying coefficients. However, it is the restriction to the neighborhood of a known trajectory which forms the primary limitation on the usefulness of the theory, particularly in the case of atmospheric reentry. The various proposed methods employing this concept for reentry have, in general, range capability of 6,000 miles or less. Theoretically, the proposal of reference 5 to store multiple nominal trajectories and associated feedback gains should permit guidance to any range, but the increased storage capacity required makes this approach undesirable. The purpose of this paper is to show that by the proper choice of independent variable and use of empirical weighting factors it is possible to greatly increase the guidance capability of a linear perturbation scheme. This capability is achieved without increase in information storage requirements.

Notation

A	aerodynamic acceleration, g units
F_α^η	linear theory gain for the α state variable used to determine the magnitude of the control variable η, dimensions of η/α
g	surface gravity, ft/sec^2
h	altitude, ft
I_{sp}	specific impulse, sec
K_A, $K_{\dot{h}}$	empirical dimensionless weighting factors
L/D	lift-drag ratio, dimensionless
m	mass, lb-sec^2/ft
r	$r_e + h$, ft
S	vehicle reference area
t	time, sec
T	thrust, lb
V	total velocity
V_c	characteristic velocity, $gI_{sp}\ ln \frac{m_i}{m_f}$, ft/sec
W	earth weight, lb
x	downrange
y	crossrange
x_{TG}	range to go, $x_f - x$
X,Y,Z	inertial axis system positive north, east, and radially outward (see fig. 17)
$(\ )$	derivative with respect to independent variable
γ	flight-path angle (see fig. 17), deg
$\delta(\ )$	difference between actual and reference value of any quantity, $(\ ) - (\ )_r$
θ	thrust orientation, positive upward, deg
λ_α^η	$\left(\frac{\partial\eta}{\partial\alpha}\right)$ adjoint variable
Λ	angle of latitude, deg
ζ	heading angle (see fig. 17), deg
ψ	angle of longitude, deg
μ	product of universal gravitational constant and mass of planet, ft^3/sec^2

β atmospheric density decay parameter, 1/ft

ρ atmospheric density, lb sec^2/ft^4

Subscripts

i initial

f final

r reference or nominal

v vertical component

h horizontal component

Theory

Development of Control Equation

In this section we will derive the basic equation used in linear perturbation guidance. Somewhat similar developments may be found in the literature (e.g., ref. 7). Consider the set of nonlinear differential equations

$$\dot{x}_m = F_m(x_n,u_p,v) \tag{1}$$

where $1 \le n \le M$

F = M known functions
x = M state variables
u = P external force variables
v = independent variable (such as time, velocity, etc.)

Expanding equation (1) in a Taylor series about some desired nominal or reference trajectory and retaining terms to first order only gives

$$\delta\dot{x} - \sum_M a_{mn}\delta x_n = \sum_P b_{mp}\delta u_p \tag{2}$$

This is a set of M linear differential equations with varying coefficients $a_{mn}(v)$ and $b_{mp}(v)$, the solution of which describes the motion about the reference trajectory, where

$$\delta x_n(v) = x_n(v) - x_{n_r}(v)$$

$$a_{mn}(v) = \left(\frac{\partial F_m}{\partial x_n}\right)_r(v)$$

$$b_{mp}(v) = \left(\frac{\partial F_m}{\partial u_p}\right)_r(v)$$

The set of equations adjoint to equation (2) is defined by

$$\dot{\lambda}_m + \sum_M a_{nm}\lambda_n = 0 \tag{3}$$

Multiplying equation (2) by λ_m, equation (3) by δx_m, summing over M and integrating over the interval v to v_f ($v_i \le v \le v_f$) gives

$$\sum_M \lambda_m\delta x_m \Big|_{v_f} = \sum_M \lambda_m\delta x_m \Big|_v + \int_v^{v_f}\sum_M\sum_P b_{mp}\lambda_m\delta u_p dv_1 \tag{4}$$

This is the basic equation for control about a reference condition, and was called by Bliss[6] the fundamental formula. Equation (4) may be particularized by identifying the single sum at $v = v_f$ with the state variable x_q, which it is desired to control ($1 \le q \le M$). Thus, identify

$$\sum_M \lambda_m\delta x_m \Big|_{v_f} = \delta x_q \Big|_{v_f} \tag{5}$$

Then

$$\lambda_m \Big|_{v_f} = \frac{\partial x_q}{\partial x_m}\Big|_{v_f} \tag{6}$$

To indicate the proper partial derivative, the following notation has been introduced in the literature. Equation (6) is written

$$\lambda_{x_m}^{x_q}(v_f) = \frac{\partial x_q}{\partial x_m}\Big|_{v_f} \tag{7}$$

Equation (7) defines the boundary conditions necessary for the solution $\lambda_{x_m}^{x_q}(v)$ of equation (3). Equation (4) may now be written

$$\delta x_q(v_f) = \sum_M \lambda_{x_m}^{x_q}(v)\delta x_m(v) + \int_v^{v_f}\sum_M\sum_P b_{mp}\lambda_{x_m}^{x_q}\delta u_p dv_1 \tag{8}$$

Equation (8) is the basic equation by means of which an estimate can be made of the first-order change δx_q of the state variable x_q from its reference value at the final condition v_f, due to (a) a change δx_m of any state variable x_m from its reference value at a prior condition v, and (b) a change δu_p of any external force variable u_p from its reference value during the interval v to v_f.

For simplicity, consider u_p to be control variables, and assume the number q of state variables it is desired to control is equal to the number P of control variables. Then, given a desired final value $\delta x_q(v_f)$ and given certain departures $\delta x_m(v)$, there is an infinity of control variable functions which will accomplish the desired final value. In particular, there is a constant value δu_p over the interval $v \le v_1 \le v_f$ which will accomplish the desired final value, and, with the notation,

$$I_{u_p}^{x_q}(v) = \int_v^{v_f}\sum_M b_{mp}\lambda_{x_m}^{x_q} dv_1 \tag{9}$$

equation (8) may be written

$$\delta x_q(v_f) = \sum_M \lambda_{x_m}^{x_q}(v)\delta x_m(v) + \sum_P I_{u_p}^{x_q}(v)\delta u_p \tag{10}$$

Solution of equation (10) for the control variables u_p then gives

$$u_p(v) = u_{p_r}(v) + \sum_M F_{x_m}^{u_p}(v)\delta x_m(v) \qquad (11)$$

Equation (11) is applicable to a complete three-dimensional analysis. All the guidance results obtained in this paper are two-dimensional. It is shown in the appendix that to first order, these results are valid for three-dimensional applications.

Insofar as the theory is concerned, the particular set of state and independent variables chosen is completely arbitrary. There are practical considerations for using a state variable as independent variable rather than time, since this reduces M in equation (11) by one, thus simplfying the guidance through reduced information storage requirements. In the present investigation the state variables chosen were somewhat arbitrary, but total velocity instead of time was used as independent variable because of the simplification just noted, and because it appears that it has additional advantages as well. The most significant advantage is that the neighborhood of the nominal trajectory appears to be larger in terms of velocity, or, perhaps more correctly, the excursions of the state variables on the actual trajectories relative to those on the nominal trajectory generally are smaller when compared on the basis of velocity. Obviously, the advantage of using the independent variable for which the δx_m of equation (11) are minimized is that less violence must be done to the linear theory to make it operate over the range of conditions desired.

Equation (11) defines a terminal control system in that the system makes no attempt to eliminate present errors, but instead acts to prevent the propagation of present errors of all variables into errors of the controlled variables at the final condition $v = v_f$. As formulated, the system defined by equation (11) attempts to use minimum control excursion for a maximum length of time. If the information possessed by the system is correct in the sense that all pertinent variables have been accounted for, and if the system is in the neighborhood of the nominal trajectory where linearization of the equations is valid, equation (11) will command a control increment just sufficient to achieve the desired result if the increment is maintained to the final condition. Another formulation which has been used is to command the maximum available control excursion for a minimum amount of time[3]. This approach has a certain appeal because the desired end result is always brought to what is thought to be the center of the vehicle's capability as quickly as possible. However, during the earth reentry portion of the present investigation, it was found that (when attempting to operate outside the region wherein linearization is valid) this type of command tended to cause erroneous trajectory excursions from which it was later impossible to recover, and so was not satisfactory. The form of control finally used was intermediate to these two extremes; the guidance gains were adjusted through the use of empirically determined weighting functions as will be described subsequently. By these two means - the use of velocity as independent variable and empirical weighting of the guidance gains - it was possible to greatly extend the operating range of the basic linear theory.

Lunar Landing

The guidance scheme just described will now be applied to the descent-to-hover phase of lunar landing. The main features of the descent from orbit to the lunar surface are indicated in figure 1. A 100-mile circular orbit was assumed, with the gross descent accomplished by means of a Hohmann transfer orbit whose perilune determined the initial conditions for that portion of the descent considered here - a guided letdown to an altitude of less than 1000 feet.

Two maneuvers previously considered in the literature[8,9] were chosen as reference trajectories, the gravity turn and the constant-pitch-rate maneuvers. The fuel required, in terms of the characteristic velocity, is shown in figure 2 for the two maneuvers as it is affected by initial or perilune altitude and thrust level. Any desired value of thrust may be used for the constant-pitch-rate maneuver for a given initial altitude, in contrast to the single value of thrust necessary for the gravity turn. Both maneuvers require a large increase in fuel with increase in initial altitude. As a compromise between fuel requirements and avoiding the mountainous lunar surface, an initial altitude of 75,000 feet was chosen. This prescribed a $T/W_i = 0.42$ for the gravity turn. The thrust ratio for the constant-pitch-rate maneuver was chosen to be 0.56, the optimum value for this altitude.

The characteristics of the resultant reference trajectories are shown in figure 3 in terms of the state variables chosen for use in the control equation; altitude h, flight-path angle γ, and range x. It is desired to control the final values of two quantities, range and altitude. The two control variables are thrust, T, and thrust orientation, θ. Then the two control equations from equation (11) are

$$\left.\begin{aligned} T(V) &= T_r + F_h^T(V)\delta h(V) \\ &\quad + F_\gamma^T(V)\delta\gamma(V) - F_x^T(V)\delta x_{TG}(V) \\ \theta(V) &= \theta_r(V) + F_h^\theta(V)\delta h(V) \\ &\quad + F_\gamma^\theta(V)\delta\gamma(V) + F_x^\theta(V)\delta x_{TG}(V) \end{aligned}\right\} \qquad (12)$$

The guidance gains associated with equation (12) are presented in figure 4. The gains associated with flight-path angle remain finite over the velocity range; however, all other gains for both reference trajectories have singularities at zero velocity. Since this investigation was performed on an analog computer, rather severe storage limitations were necessarily imposed, a circumstance, however, which conforms with the original intent of developing a guidance system with minimum information storage requirements. The dashed lines in figure 4 indicate the maximum values of the gains actually used in the investigation.

The guidance capability using the constant-pitch-rate reference trajectory is summarized in figure 5 in terms of the corridor of initial altitude and range limits from which it is possible to guide to a target area 1,000 feet in altitude and 10,000 feet in range, the center of which is located 770,000 feet downrange, the range for this particular reference trajectory. Changes in initial range and altitude were accompanied by the appropriate initial velocity and flight path angle changes corresponding to the Hohmann trajectory passing through that point.

Two corridors are shown, the smaller one corresponding to the use of the guidance gains shown in figure 4. The boundaries of this corridor are defined entirely by the inability of the guidance to meet the altitude limits of the target area. Multiplying the gain $F_{\dot{h}}^{\theta}$ by a factor of two, more than doubled the size of the corridor. Further increases in guidance capability were found to be possible by the same means, but characteristics such as increased fuel requirements, excessive angular rates, and other factors made the results unsatisfactory.

Also shown in figure 5 are the fuel requirements for a perilune altitude of 75,000 feet. Only a moderate fuel increase occurs for initial range errors. The curve shown results from use of the linear theory gains; however, the fuel requirements due to using the adjusted gain is not significantly different.

Figure 6 shows the guidance capability obtained with the gravity turn reference trajectory. Again, the smaller corridor, obtained using the guidance gains of figure 4, had boundaries defined entirely by the inability of the guidance to meet the altitude limits of the target area. By a moderate adjustment of the guidance gains, namely, the use of $1.5F_{\dot{h}}^{\theta}$ and $0.85F_{x}^{\theta}$, the extreme increase in the guidance capability shown in the figure was obtained. The fuel requirement is shown in the lower part of the figure for the nominal altitude of 75,000 feet. A 100-percent range error requires a fuel increase of about 70 ft/sec, which is equivalent to about 14 seconds of hover time.

Typical guided trajectories for the gravity turn reference trajectory are shown in figure 7. The guided descents were initiated from a Hohmann transfer orbit with a perilune altitude of 50,000 feet. This figure emphasizes the extreme initial errors which the guidance system is capable of handling.

Earth Reentry Guidance

In the investigation of guidance for atmosphere reentry at parabolic speed, a vehicle with a maximum $L/D = 0.4$ and $W/C_DS = 50$ was chosen. The nominal atmosphere used was the 1959 ARDC model.

The characteristics of the nominal trajectory chosen are shown in figure 8. Several factors were considered in choosing this trajectory. One factor was the easing of the restrictions upon the time of return from a lunar mission to a single earth site, by seeking ranges up to one-half the earth's circumference. A 6,000-mile nominal range was chosen since it is approximately in the center of the desired range envelope. A high-skip type of trajectory was chosen because it imposes low total heat loads, and because the final range is less sensitive to state variable errors than it is for trajectories which have relatively low skip altitudes. Although not of concern in the present investigation, these considerations are of great practical significance in the design of the heat shield, and the backup and monitoring system for the primary guidance.

The nominal L/D for this trajectory is equal to 0.1. Because of the two-dimensional nature of the investigation, the nomenclature used in appendix A to indicate the vertical component of L/D will be eliminated. Figure 8 shows the characteristics of the reference trajectory in terms of the state variables chosen for use in the control equation: altitude rate $\dot{h}$, aerodynamic acceleration A, and range x. It is desired to control the single quantity, range, by means of the control variable L/D; then the control equation (11) is:

$$\frac{L}{D}(V) = \left(\frac{L}{D}\right)_r + F_{\dot{h}}(V)\delta\dot{h}(V) + F_A(V)\delta A(V) - F_x(V)\delta x_{TG}(V) \qquad (13)$$

The guidance gains associated with equation (13) are shown in figure 9. These gains, determined by means of the linear theory, did not define a guidance scheme capable of handling the nonlinearities resulting from the large departures from the reference trajectory desired, even when account was taken of the multivalued nature of velocity evident in figures 8 and 9. The gains actually used in the results to be presented are indicated by the dashed lines. Associated with these gains are the empirically determined weighting factors shown in figure 10 that enable the guidance system to operate over virtually the full range of vehicle capability. The combination shown is not unique; other combinations also permit full guidance capability, a fact which will allow for optimization studies. The final form of the control equation is now

$$\frac{L}{D}(V,x_f) = \left(\frac{L}{D}\right)_r + K_{\dot{h}}(V,x_f)\delta\dot{h}(V) + K_A(V,x_f)\delta A(V) - F_x(V)\delta x_{TG}(V) \qquad (14)$$

This equation with the modified linear theory gains and the weighting factors just described were used to obtain all the results to be presented subsequently.

Figures 11(a) and 11(b) show two typical guided trajectories for entry angles corresponding to the extremes permitted by the vehicle's capability, the uncontrolled skip boundary and maximum acceleration boundary. The uncontrolled skip boundary is defined as the shallowest reentry angle at which the vehicle can acquire sufficient aerodynamic force to control the subsequent trajectory. For the vehicle considered in this study the boundary is $\gamma_i = -4.7^{\circ}$. The maximum acceleration boundary is defined as the steepest reentry angle for which the acceleration will not exceed the maximum desired. For the 10g limit chosen for this study, the boundary is $\gamma_i = -7.3^{\circ}$.

These boundaries are shown in figure 12; for ranges greater than approximately 6,300 miles another vehicle capability boundary is defined by the maximum range possible at a given reentry angle. Also shown in figure 12 are the data points indicating guided trajectories calculated to delineate the guidance capability. It can be seen that the guidance system is capable of operating over virtually the entire corridor defined by the vehicle itself.

An item of significant interest in the evaluation of a guidance system is its ability to handle off-design conditions. Three types of off-design conditions were considered in this study:

(1) Reentry from abort conditions
(2) Variations in the vehicle L/D
(3) Atmospheric variations

The abort conditions considered were reentries from circular orbit and from a velocity of 32,000 ft/sec. The two reentries shown in figure 13(a) were initiated from a circular orbit at an altitude of 600,000 feet. The range traversed from the time of leaving orbit altitude until an altitude of 400,000 feet was reached was 4,700 miles greater for the $\gamma_i = -0.41^o$ entry than for the $\gamma_i = -1.59^o$ entry. Comparison of this range increment with that possible through guidance as shown in figure 13(a) indicates that the thrust applied in orbit to initiate reentry must be used as the primary range control in this type of abort situation. The results of figure 13(a) show, however, that the guidance system is capable of utilizing almost full vehicle capability.

Two angles were considered in the reentries shown in figure 13(b) for a velocity of 32,000 ft/sec. At this velocity the vehicle has the capability, at the angle of -4.3°, of extremely long range. As shown, however, the guidance system is incapable of achieving a range greater than 7,500 miles for this off-design condition. At the steeper entry angle of -7.2° the guidance is again able to utilize almost full vehicle capability.

The density deviation from the 1959 ARDC atmosphere used in this study varied linearly from zero at 100,000 feet to ±50 percent at 400,000 feet altitude. The effects of variations in vehicle L/D and deviations of the atmosphere on the shape of a 6,000 mile trajectory entering at a steep angle are shown in figures 14 and 15. A summary of these effects on guidance capability at various ranges is presented in figure 16. The solid line is a repeat of the information given in figure 12, that is, the guidance capability under nominal conditions. It can be seen that the L/D variations affected the capability relatively little. At long ranges, however, the density variation caused a considerable loss in the guidance capability. It is anticipated that including a component in the control equation sensitive to density deviations (the adaptive feature of ref. 5) will make a marked improvement. This will be investigated in the near future.

Concluding Remarks

In this study a modified perturbation theory has been applied to the problems of lunar landing and earth reentry guidance. It has been shown that if velocity is used as the independent variable in the guidance equation and if the linear theory gains are appropriately weighted, then one reference trajectory can be used successfully in spite of large errors in nominal or initial conditions. The use of a single reference trajectory in each problem means that the guidance method requires little storage capacity.

In the lunar landing study, the guidance capability for a control system formulated with the gravity turn as the reference trajectory was far superior to one formulated with a constant-pitch-rate reference trajectory. With a single gravity turn reference trajectory, the guidance system could compensate for initial range errors of 100 percent of the reference value with a small additional fuel increment, equivalent to a characteristic velocity of 70 ft/sec.

In the earth reentry problem, it was found that with a single reference trajectory it was possible to obtain a guidance capability from 1,500 to 12,000 miles for a range of entry conditions which utilized virtually all of the vehicle's capability.

For the abort conditions considered in this paper, the guidance system was generally able to make almost full use of the vehicle's range capability.

Errors of 5 percent in vehicle L/D had little effect on the capability of this guidance scheme.

Density variations from the nominal affected the long-range guidance but had little effect on guidance capability for ranges less than 6,000 miles.

Appendix

In this appendix the specific control equation used in the earth reentry portion of the paper will be developed, and it will be demonstrated that to first order the two-dimensional results presented in this study are valid for three-dimensional applications. The equations of motion used were

$$\left.\begin{aligned}
\dot{h} &= V \sin\gamma \\
\dot{x} &= r_e\dot{\psi} = \frac{r_e}{r}\,\frac{V\cos\gamma\cos\zeta}{\cos\Lambda} \\
\dot{\gamma} &= \frac{L_V}{mV} + \frac{V}{r}\cos\gamma - \frac{1}{V}\frac{\mu}{r^2}\cos\gamma \\
\dot{V} &= -\frac{D}{m} - \frac{\mu}{r^2}\sin\gamma \\
\dot{\zeta} &= \frac{L_h}{mV}\,\frac{1}{\cos\gamma} - \frac{V}{r}\tan\Lambda\cos\gamma\cos\zeta \\
\dot{y} &= r_e\dot{\Lambda} = \frac{r_e}{r} V\cos\gamma\sin\zeta
\end{aligned}\right\} \qquad (A1)$$

where the geometry is shown in figure 17. The assumptions made in the development were

(a) C_D = constant

(b) C_L = constant

(c) Planar reference trajectory

With these assumptions the control variable may be considered to be the vertical component of lift, and the coefficients of the perturbation equations (2) of the text are found to be, for those which are other than zero,

$$a_{13} = V_r \cos\gamma_r$$

$$a_{14} = \sin\gamma_r$$

$$a_{21} = -(r_e V_r \cos\gamma_r)/r_r^2$$

$$a_{23} = -(r_e V_r \sin\gamma_r)/r_r$$

$$a_{24} = (r_e \cos\gamma_r)/r_r$$

$$a_{31} = -\frac{\beta}{2}\rho_r V_r \left(\frac{C_D S}{m}\right)\left(\frac{L_V}{D}\right)_r - \left(\frac{V_r}{r_r^2} - \frac{2}{V_r}\frac{1}{r_r}\frac{\mu}{r_r^2}\right)\cos\gamma_r$$

$$a_{33} = \left(\frac{1}{V_r}\frac{\mu}{r_r^2} - \frac{V_r}{r_r}\right)\sin\gamma_r$$

$$a_{34} = \frac{\rho_r}{2}\left(\frac{C_D S}{m}\right)\left(\frac{L_V}{D}\right)_r + \left(\frac{1}{r_r} + \frac{1}{V_r^2}\frac{\mu}{r_r^2}\right)\cos\gamma_r$$

$$a_{41} = \frac{\beta}{2}\rho_r V_r^2\left(\frac{C_D S}{m}\right) + \frac{2}{r_r}\frac{\mu}{r_r^2}\sin\gamma_r$$

$$a_{43} = -\frac{\mu}{r_r^2}\cos\gamma_r$$

$$a_{44} = -\rho_r V_r\left(\frac{C_D S}{m}\right)$$

$$\pm a_{51} = \mp\frac{\beta}{2}\rho_r V_r\left(\frac{C_D S}{m}\right)\frac{1}{\cos\gamma_r}\sqrt{\left(\frac{L}{D}\right)^2 - \left(\frac{L_V}{D}\right)_r^2}$$

$$\pm a_{53} = \pm\frac{\rho_r}{2}V_r\left(\frac{C_D S}{m}\right)\frac{\sin\gamma_r}{\cos^2\gamma_r}\sqrt{\left(\frac{L}{D}\right)^2 - \left(\frac{L_V}{D}\right)_r^2}$$

$$\pm a_{54} = \pm\frac{\rho_r}{2}\left(\frac{C_D S}{m}\right)\frac{1}{\cos\gamma_r}\sqrt{\left(\frac{L}{D}\right)^2 - \left(\frac{L_V}{D}\right)_r^2}$$

$$a_{56} = -\frac{V_r}{r_r r_e}\cos\gamma_r$$

$$a_{65} = \frac{r_e}{r_r}V_r\cos\gamma_r$$

$$b_{31} = \frac{\rho_r}{2}V_r\left(\frac{C_D S}{m}\right)$$

$$\pm b_{51} = \mp\frac{\rho_r}{2}V_r\left(\frac{C_D S}{m}\right)\frac{1}{\cos\gamma_r}\frac{\left(\frac{L_V}{D}\right)_r}{\sqrt{\left(\frac{L}{D}\right)^2 - \left(\frac{L_V}{D}\right)_r^2}}$$

where the $\pm$ signs indicate the possibility of a left or right orientation of the horizontal component of lift. Then equations (2) and (3) of the text become

$$\begin{bmatrix}\dot{\delta h}\\ \dot{\delta x}\\ \dot{\delta\gamma}\\ \dot{\delta V}\\ \dot{\delta\zeta}\\ \dot{\delta y}\end{bmatrix} = \begin{bmatrix}0 & 0 & a_{13} & a_{14} & 0 & 0\\ a_{21} & 0 & a_{23} & a_{24} & 0 & 0\\ a_{31} & 0 & a_{33} & a_{34} & 0 & 0\\ a_{41} & 0 & a_{43} & a_{44} & 0 & 0\\ \pm a_{51} & 0 & \pm a_{53} & \pm a_{54} & 0 & a_{56}\\ 0 & 0 & 0 & 0 & a_{65} & 0\end{bmatrix}\begin{bmatrix}\delta h\\ \delta x\\ \delta\gamma\\ \delta V\\ \delta\zeta\\ \delta y\end{bmatrix} + \begin{bmatrix}0\\ 0\\ b_{31}\\ 0\\ \pm b_{51}\\ 0\end{bmatrix}[\delta(L_V/D)] \tag{A2}$$

$$\begin{bmatrix}\dot\lambda_1\\ \dot\lambda_2\\ \dot\lambda_3\\ \dot\lambda_4\\ \dot\lambda_5\\ \dot\lambda_6\end{bmatrix} = -\begin{bmatrix}0 & a_{21} & a_{31} & a_{41} & \pm a_{51} & 0\\ 0 & 0 & 0 & 0 & 0 & 0\\ a_{13} & a_{23} & a_{33} & a_{43} & \pm a_{53} & 0\\ a_{14} & a_{24} & a_{34} & a_{44} & \pm a_{54} & 0\\ 0 & 0 & 0 & 0 & 0 & a_{65}\\ 0 & 0 & 0 & 0 & a_{56} & 0\end{bmatrix}\begin{bmatrix}\lambda_1\\ \lambda_2\\ \lambda_3\\ \lambda_4\\ \lambda_5\\ \lambda_6\end{bmatrix} \tag{A3}$$

and equation (4) of the text becomes:

$$[\lambda_1\delta h + \lambda_2\delta x + \lambda_3\delta\gamma + \lambda_4\delta V + \lambda_5\delta\zeta + \lambda_6\delta y]_{t_f} = [\lambda_1\delta h + \lambda_2\delta x + \lambda_3\delta\gamma + \lambda_4\delta V + \lambda_5\delta\zeta + \lambda_6\delta y]_t + \int_t^{t_f}(\lambda_3 b_{31} \pm \lambda_5 b_{51})\delta(L_V/D)dt_1 \tag{A4}$$

It is desired to control the final downrange, x, and crossrange, y. By the strict identity (5) of the text, the left side of equation (A4) will equal the downrange change at the final value of the independent variable if

$$\left.\begin{aligned}\lambda_1 = \lambda_3 = \lambda_4 = \lambda_5 = \lambda_6 = 0\\ \lambda_2 = 1\end{aligned}\right\} t = t_f \tag{A5}$$

A slightly different formulation holds if the stopping condition is other than the independent variable acquiring some specified value (see, e.g., ref. 7). In the present type of problem the results are not significantly different. In the notation of equations (7), equations (A5) are

$$\left.\begin{aligned}\lambda_h^x = \lambda_\gamma^x = \lambda_V^x = \lambda_\zeta^x = \lambda_y^x = 0\\ \lambda_x^x = 1\end{aligned}\right\} t = t_f \tag{A6}$$

Solving equations (A3) using the boundary conditions (A6) gives the values for all t. By inspection,

$$\left.\begin{aligned}\lambda_\zeta^x = \lambda_y^x = 0\\ \lambda_x^x = 1\end{aligned}\right\} \text{for all } t \tag{A7}$$

That is, to first order, there is no effect of heading angle, ζ, or crossrange, y, on the downrange, x. This result is true only for a planar trajectory, a restriction approximately fulfilled as a result of the nature of the vehicles considered in both parts of this paper; the strictly two-dimensional results presented should then remain valid if extended to a full three-dimensional investigation.

Equation (A4) now may be written

$$\delta x(t_f) = [\lambda_h^x\,\delta h + \delta x + \lambda_\gamma^x\delta\gamma + \lambda_V^x\,\delta V]_t + \int_t^{t_f}\lambda_\gamma^x\,b_{31}\delta(L_V/D)dt_1 \tag{A8}$$

Transformation to any independent variable and combination of state variables is a simple matter. As noted in the text, velocity was chosen as the independent variable, and altitude rate, $\dot{h}$, aerodynamic acceleration, A, and range, x, were the state variables chosen. With these variables, equation (A8) becomes

$$\delta x_f = \delta x(V_f) = [\lambda_{\dot{h}}^{x}\delta\dot{h} + \delta x + \lambda_A^{x}\delta A]_V + \int_V^{V_f} \lambda_\gamma^{x}\, b_{31}\delta(L_V/D)\,dV_1 \qquad \text{(A9)}$$

and equation (11) becomes

$$\left(\frac{L_V}{D}\right) = \left(\frac{L_V}{D}\right)_r + F_{\dot{h}}\delta\dot{h} + F_A\delta A + F_x(\delta x - \delta x_f) = \left(\frac{L_V}{D}\right)_r + F_{\dot{h}}\delta\dot{h} + F_A\delta A - F_x\delta x_{TG} \qquad \text{(A10)}$$

where

$$I_{L_V/D}^{x} = \int_V^{V_f} \lambda_\gamma^{x}\, b_{31}\,dV_1$$

$$F_x = \frac{1}{I_{L_V/D}^{x}}$$

$$F_A = \frac{\lambda_A^{x}}{I_{L_V/D}^{x}}$$

$$F_{\dot{h}} = \frac{\lambda_{\dot{h}}^{x}}{I_{L_V/D}^{x}}$$

and the superscript has been left off the F functions because of the single control variable. Equation (A10) is the same as equation (13) given in the text as the unmodified linear theory control equation for earth reentry guidance.

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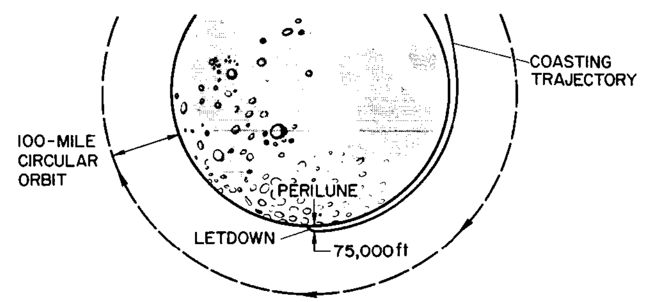


Figure 1. - Lunar landing approach.

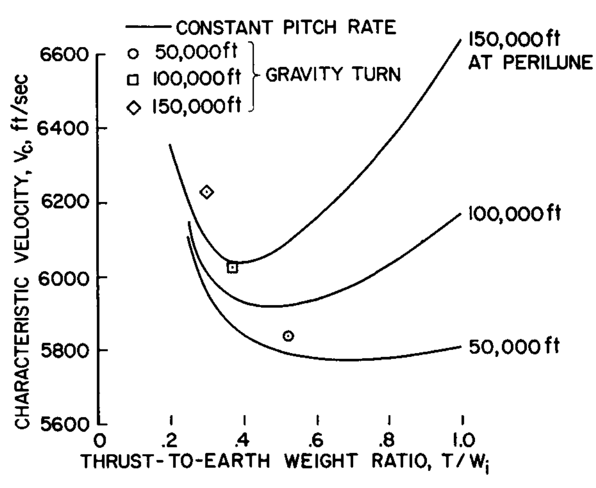


Figure 2. - Fuel required for gravity turn and constant-pitch-rate maneuver.

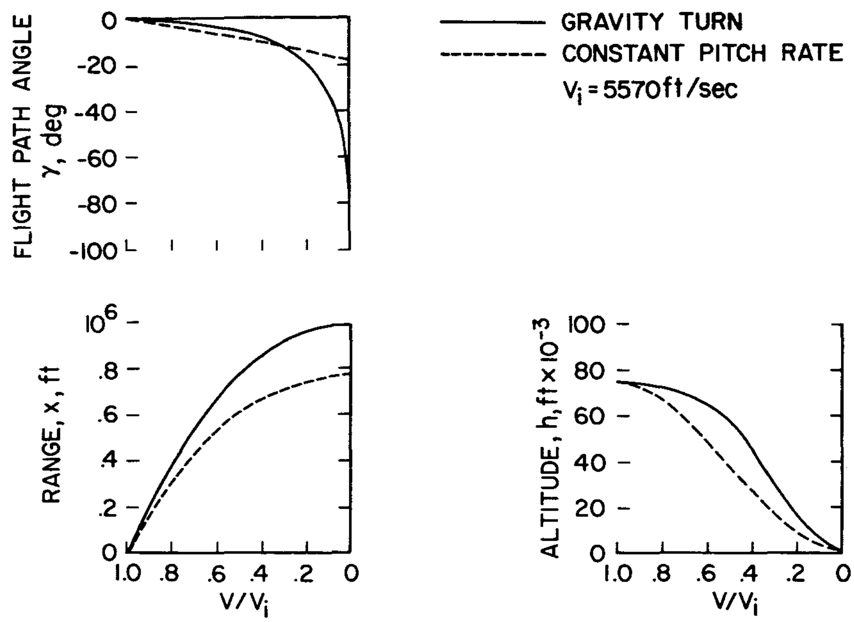
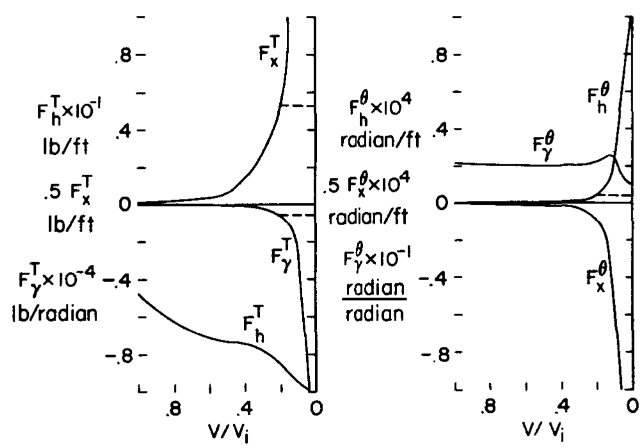
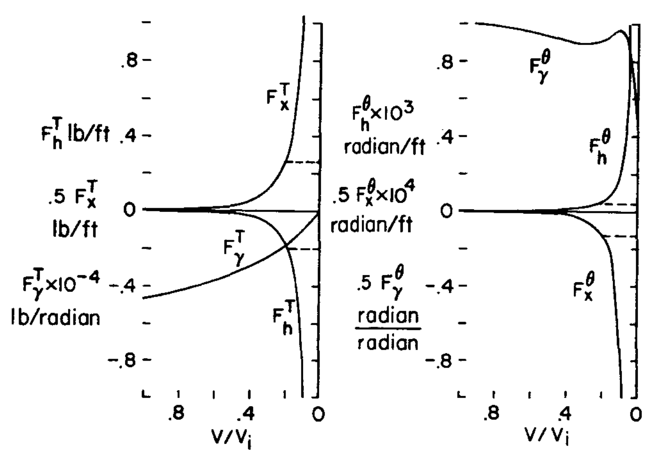


Figure 3. - Peference trajectory state variables.



(a) Constant pitch rate.



(b) Gravity turn.

Figure 4. - Linear theory guidance gains.

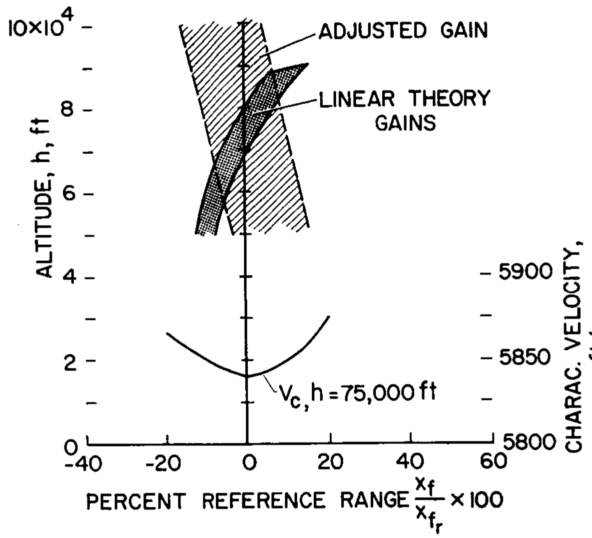


Figure 5. - Guidance capability: constant-pitch-rate reference trajectory.

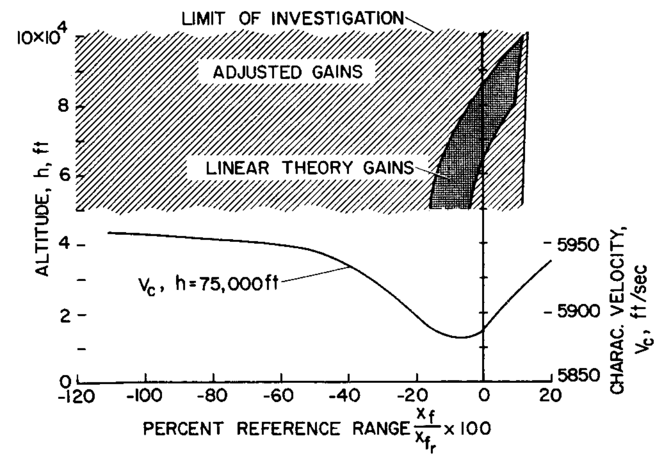


Figure 6. - Guidance capability: gravity turn reference trajectory.

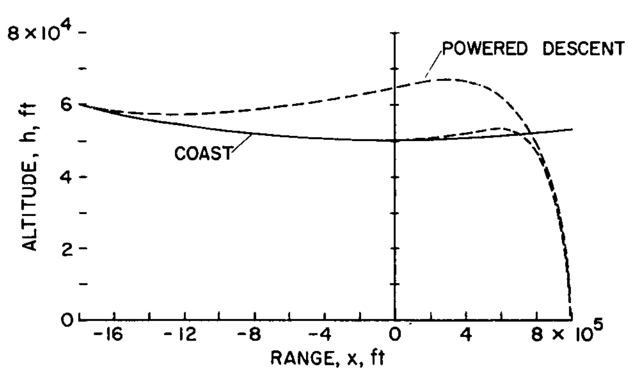


Figure 7. - Typical guided trajectories: gravity turn reference trajectory.

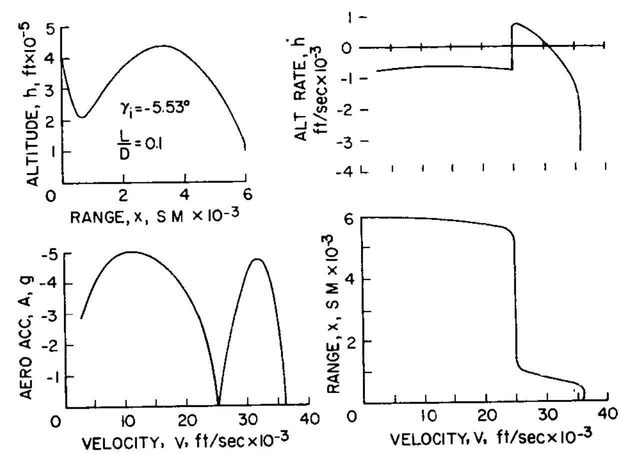


Figure 8. - Reference trajectory state variables.

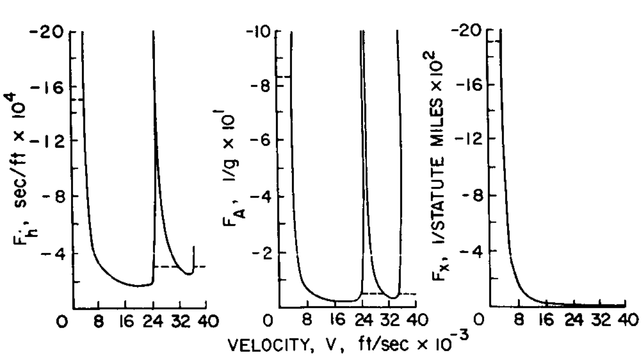


Figure 9. - Linear theory guidance gains.

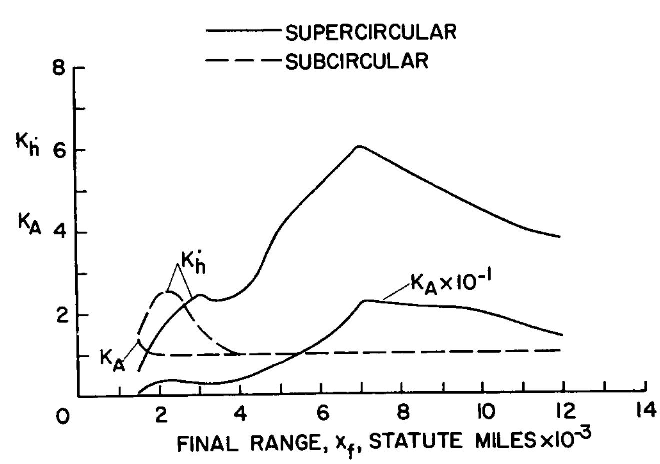
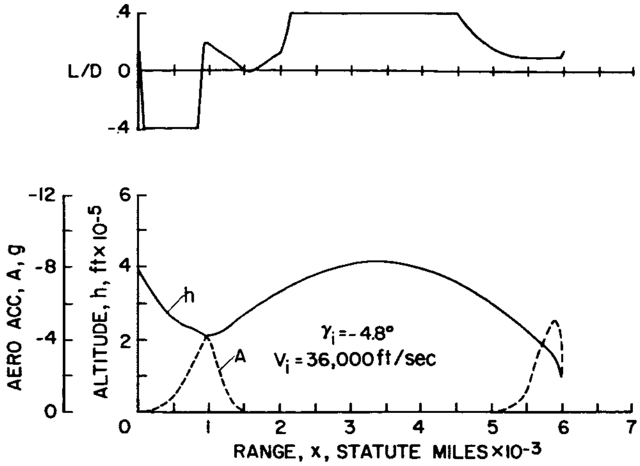
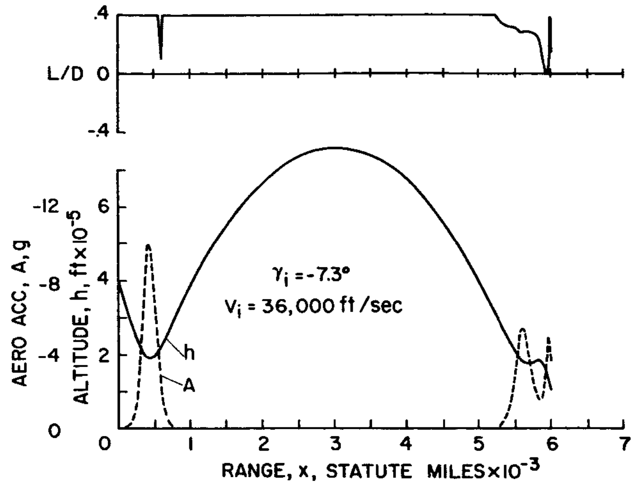


Figure 10. - Empirical weighting factors.



(a) Shallow entry.

Figure 11. - Typical guided trajectories.



(b) Steep entry.

Figure 11. - Concluded.

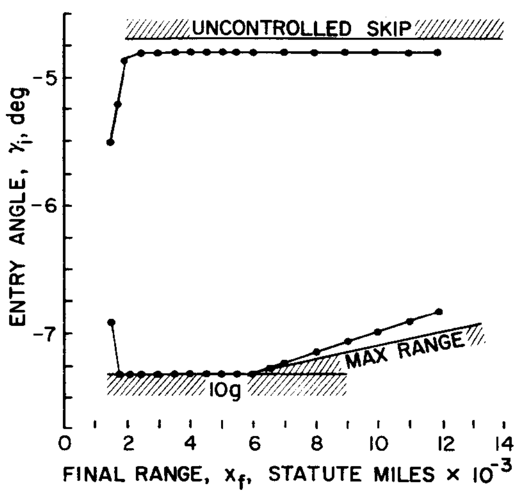
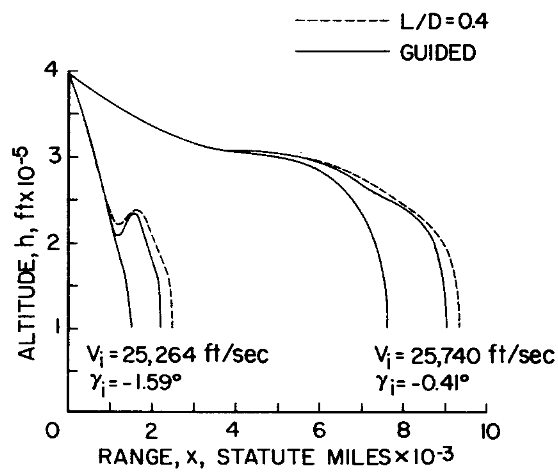
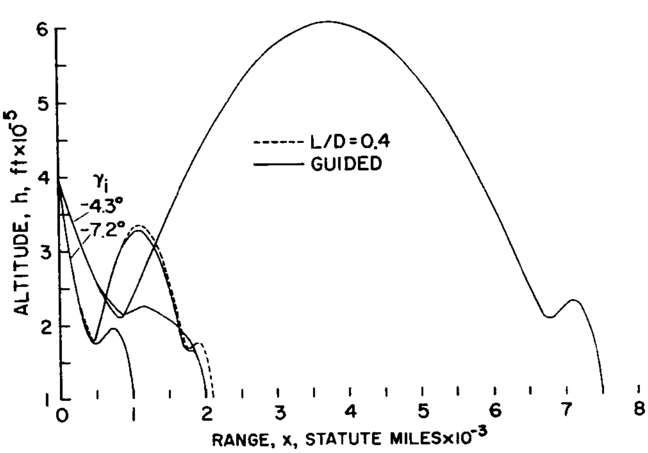


Figure 12. - Guidance capability.



(a) Reentry from circular orbit.

Figure 13. - Abort conditions.



(b) Reentry at 32,000 feet per second.

Figure 13. - Concluded.

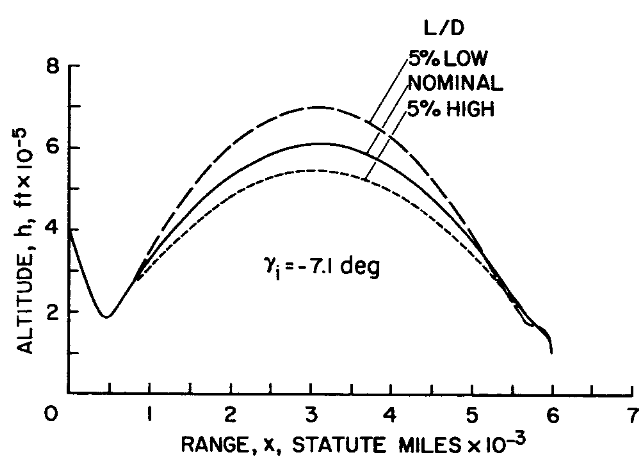


Figure 14. - Effect of L/D variation on guided trajectory.

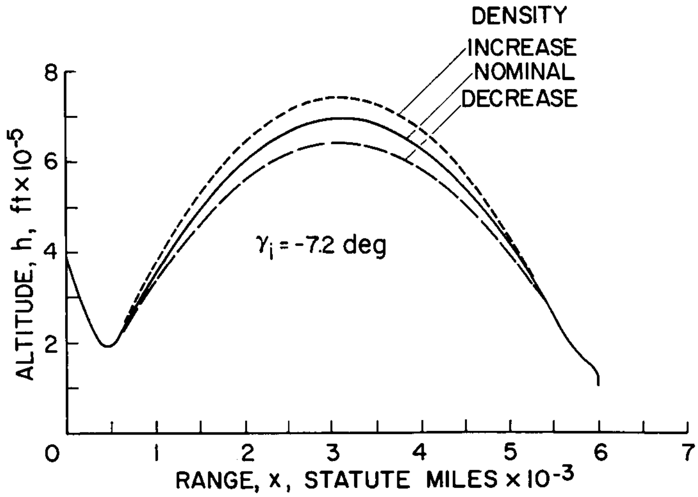


Figure 15. - Effect of density variation on guided trajectory.

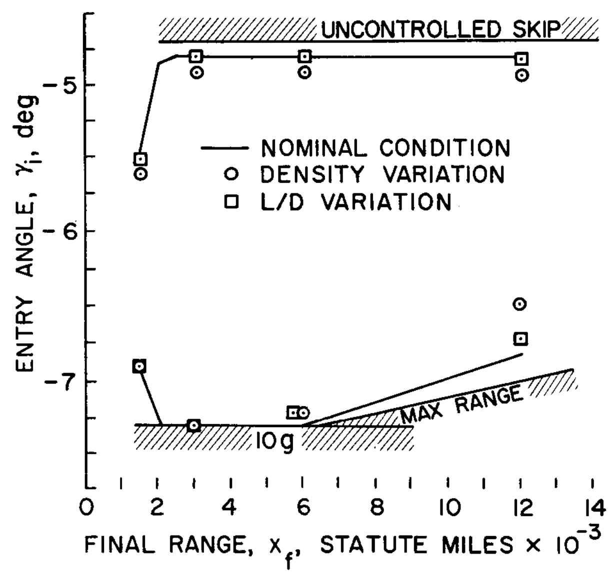


Figure 16. - Effect of off-design conditions on the guidance capability.

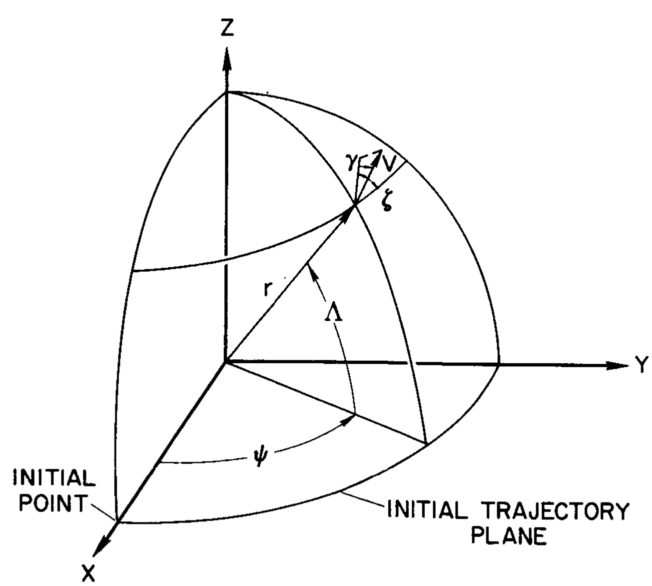


Figure 17. - Axis system for trajectory equations.

FUTURE OF ON-BOARD COMPUTERS FOR SPACE VEHICLES

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I. INTRODUCTION

The application of on-board digital computers for space vehicle systems is in its infancy. Primary applications have been used for timing and control functions on unmanned vehicles. Several manned systems are in development where on-board computers will be used.

In a field in its infancy, it is difficult to predict what future computers, future applications, and future capabilities are going to be. However, information is available from the earlier application of on-board computing equipment in aircraft, missiles, and submarines which can be extrapolated to space applications. In these earlier systems, it has been found necessary to incorporate computing capability for essentially four reasons: (1) there was no man in the system so a computer had to be included to perform the control and decision making functions required, (2) the function being performed was too complex and rapid for a man to perform, (3) the function had to be controlled more precisely than a man could readily do, and (4) the man had so many things to do and such a short time to do them in, that he needed the help of a computing system.

Tracing the history of computer applications to missile systems, it is found that the first digital computers were used for navigation. As more missile systems have been developed, the on-board computer performs not only the navigation functions but the steering computations, the stability augmentation computations, the instrument calibration functions, preflight checkout, in-flight performance monitoring, staging control, thrust cutoff computations, and warhead pre-arming functions. Thus, the computer has gone through evolutionary stages; first performing a single function and then performing the computations for a highly integrated, highly sophisticated but very simple and consequently, very reliable system. Similarly, in aircraft applications the first on-board digital computers were for single-purpose functions such as fire control or navigation. As aircraft systems have become more sophisticated, the on-board computer has accumulated the combined functions of navigation, fire control, display control, performance monitoring, autopilot steering functions, air data processing, premission checkout, radar control, and fuel management. Since computational functions in both aircraft and missile systems have evolved from single function applications to system integration applications, it is reasonable to predict that computers for space vehicles will evolve in the same manner.

II. SPACE MISSIONS

A. COMPUTATIONAL FUNCTION

The functions of computers for space vehicles initially have been for guidance and control. Systems currently in development have digital computers which possess functions for guidance, attitude control, steering, star tracking and display. These computers also will perform functions of premission checkout, performance monitoring, and it can be expected that in the near future the on-board computers also will be doing in-flight checkout and other self-test functions leading to in-flight maintenance. In the DYNA SOAR system, the on-board digital computer is involved with the re-entry and energy management functions as well as the guidance and control functions. In lunar missions such as that of the APOLLO, the on-board computers will perform all the previously mentioned functions and, in addition, will be aiding in orbit injection, mid course and terminal guidance functions. The problem of rendezvous will add requirements for antenna control, radar tracking, and radar control. As space vehicles probe farther and farther into space, the on-board computing function could be expanded to include data correlation, communications control, data compression, data storage, pattern recognition, antenna pointing, and telemetry control functions. Even in vehicles which can be controlled from the earth for boost and mid course guidance functions, it may be desirable to have an on-board computer for exact terminal control. This is necessary because the communication time lag from the earth may be several minutes. This would make precise control impossible.

The computing functions on early space vehicles will be simpler than those required for later more sophisticated systems. Consequently, the computers required for these early systems will be simpler than those required for the more complex systems. At the Spaceborne Computer Engineering Conference held in Anaheim, California, in October of last year, a half dozen digital computers were described which are being designed for early space vehicles. As the space missions become more sophisticated, the computer systems will require higher speed, more memory capacity, and more computational sophistication. For functions of the vehicle velocity vector and which will be constant for the remainder of the thrust period, are loaded into special registers and the computation indicated by

$$S = K - Kx\,dx - Ky\,dy - Kz\,dz \qquad (2)$$

is performed continuously in a simplified digital differential analyzer. When the function, S, passes through zero, the cutoff signal is provided.

The logical designer of a vehicleborne computer will provide high-speed computing capability through other techniques. One of these might be the utilization of parallel arithmetic. The arithmetic center in a parallel computer generally is complex because all digits in the number are processed simultaneously. In this case, to simplify the computer, the logical design may shorten the computer word, restrict the number of operations that the computer is capable of performing, and use several programming steps at high speed to provide the capability required.

Another arithmetic technique sometimes used provides higher speed than the method of processing the digits serially. This technique utilizes a combination of the two methods whereby the digits are grouped and then, each group is processed in parallel.

There are several methods of paralleling other than arithmetic that can be used to provide high-speed computing capability. One of these is paralleling of instruction sequencing so that portions of several different instructions are processed simultaneously to give higher computing speeds without increased arithmetic speed.

Another form of paralleling that can increase computational speed is the paralleling of processing centers. (See Figure 5). In this case a common memory is shared by several computing centers, or by a computing center and an input-output processing center.

Because of the variability of computer system organization, it is difficult to compare computers on the basis of the time required to add or multiply, or even to compare them on the basis of memory capacity. The only way that the computation capabilities of computers for space applications can be compared is to compare their performance, speed, memory capacity required, and ancillary equipment required for the particular application in question.

C. RELIABILITY THROUGH REDUNDANCY

To achieve the reliability which will be required for deep space missions even though highly reliable integrated circuits and sophisticated design techniques are used, it may be necessary to utilize some form of redundancy. Redundancy techniques have received considerable study in the past few years and can be incorporated in future systems at a variety of levels.

For relatively simple devices, redundancy at the circuit level is attractive. However, when the amount of hardware required in a device as complex as even a simple digital computer is considered, then circuit level redundancy is not as appealing. It requires many more circuit elements than a device with no redundancy; it poses some serious problems during manufacture and maintenance in isolating redundant components which have failed, and the power, weight, and volume required for the hardware is significantly greater than for a nonredundant computer.

A second level of redundancy which can be considered is redundancy at the module level. This requires voting circuits to identify the proper output from the module and at least three identical modules at every point where redundancy is utilized so that an automatic vote can be taken. The voting circuitry also requires hardware so this method is expensive.

A third level of redundancy is that of redundancy at the subsystem level where more than one computer would be supplied to perform the desired functions. This is shown in Figure 6. In a manned system, it may not be necessary to supply more than two computers, one performing the function and the second being redundant. One would expect that in order to vote, a majority vote would be required. However, by using two computers which can each be performing self tests and monitoring the performance of the other computer and, if there is any discrepancy, the astronaut can make a decision as to which computer is performing properly.

Another approach to redundancy which may be promising is a combination of several small, single-purpose processors into a multiple computing system. These computers would be integrated so that the function of a computer which has failed could be transferred to a second computer which is performing a lower priority function. If a spare computer were provided and the astronaut had the capability of repairing the units, extremely long missions can be accomplished with a high probability of success. In this case, a computer which has malfunctioned can be electronically replaced by a spare computer and the malfunctioning computer repaired at the leisure of the astronaut. The astronaut would isolate the malfunction through a combination of diagnostic and automatic checkout techniques. A multiple computer system with switching and repair capability is shown in Figure 7.

IV. SUMMARY

In less than six months, there will be computers for space applications designed for single functions which will be 10 lb in weight, 0.15 cu ft in size, will require approximately 50 w of power, elements also are in development. These include pneumatic elements, cryogenic elements, bionic devices, and a combination of lasers and fiber-optic elements. Most of these latter techniques are in an early state of development. Thus, it is difficult to predict what their impact on space computers is going to be.

Currently the most promising circuit elements, which are being incorporated in the Improved MINUTEMAN system, are integrated circuits. An example of an integrated circuit is shown in Figure 1. Theoretical studies and actual reliability measurements indicate that the integrated circuits in the Improved MINUTEMAN will be a factor of from 10 to 100 more reliable than the circuits used in the current MINUTEMAN production program. This is extremely significant considering the high reliability components used in the current program have been demonstrated by test data to be 100 times more reliable than conventional MIL-Standard components. A reliability growth curve comparing the same circuit using MINUTEMAN high reliability components with the integrated circuit form used in Improved MINUTEMAN is shown in Figure 2. The data from which this material is plotted was obtained from accelerated life test measurements and indicates that the integrated circuit is already a factor of two more reliable than the same circuit when implemented in the highest reliability conventional components available.

There is the immediate possibility of an even greater reliability growth through a combination of thin film and integrated circuit techniques. It is not clear at this time whether it will be thin films deposited on integrated circuits or integrated circuits deposited on thin films. Either technique will result in more circuit flexibility and fewer interconnections. These advances will lead to improved reliability.

The memory technology currently in use in vehicleborne computers includes rotating magnetic memories, core memories, multiaperture core memories, rope core memories, delay lines, and in the near future, thin film memories. It is expected that these memory devices will continue to be used for some time even though cryogenic memories are currently in development and there is some hope for a combined laser and fiberoptic memory device in future systems.

B. COMPUTER ORGANIZATION CONTRIBUTIONS TO RELIABILITY

The problem of meeting the requirements of future space systems computers with increased computing capability, higher speed, easy programming, large memory, and at the same time meet the requirements of high reliability, imposes tough design problems for computer designers. At the same time, the requirements for more computing capability imply more hardware while the requirements for higher reliability imply less hardware.

The function of the designer of a spaceborne computer is to provide the simplest design to do the job required. Because there is a wide variety of functions that might be performed in space vehicles for different applications, it is expected that at least two major classes of computers will be required. One class will consist of very simple computers designed to perform more or less single functions such as navigation, antenna stabilization. A second class will consist of computers designed to integrate all of the computations required in the system into a single computing device. The internal organization of these computers is difficult to predict at this time. However, the designers of the spaceborne digital computers may time share circuits to a greater extent than is required for the design of ground-based computers. Designs also may provide special features in these vehicleborne computers to solve the necessary computations at the required rates with a minimum amount of complexity.

One method of providing high-speed computing capability without complexity is to provide a digital differential analyzer incremental computing capability, in the computer. There are a number of different computational algorithims used in digital differential analyzers but the one most commonly used is

$$dz = (Y_O + \sum dy)\, dx = Y\, dx \qquad (1)$$

The digital differential analyzer is composed of a number of integrators, each capable of performing the computation of Eq 1. A commonly used symbol for an integrator is shown in Figure 3. The integrators are usually implemented in four or five delay lines. Information is read from the delay lines serially. It then is processed and recorded back into the memory. Because the incremental computer processes all integrators in the same manner and is serial, very little circuitry is required.

Because of the simplicity of generating some functions with incremental techniques, variations of a digital differential analyzer are sometimes used in conjunction with a general purpose computer. For instance, sine and cosine generation with a digital differential analyzer is very simple. (See Figure 4.) Because only two integrators are required to continuously generate sine and cosine functions, the logic designer may provide special registers in the computer for performing these computations.

Incremental techniques also will facilitate high-precision thrust cutoff computations. In this case the general purpose computer performs continuous computation until the computer predicts that during the next computational cycle thrust cutoff will occur. At this time parameters, which are instance, in the APOLLO mission with a lunar excursion module, the mission can be divided into a number of relatively independent operational periods. These periods include preflight calibration, boost guidance, orbital, transfer orbit injection, mid course guidance intervals, parking orbit injection, orbit determination, rendezvous, and a similar set of functions for the return flight. To perform these functions, the on-board computer requires more than 12,000 storage words. In this situation, the computations performed are relatively simple during each phase, so extremely high computing speeds are not necessary. However, for more complex missions (planetary landings on Mars or Venus where gravitational constants and atmospheric and surface conditions are not known accurately), it may be necessary to provide both large memory and high speed on-board computing.

B. IMPACT OF MISSION VARIATION

A significant difference between aircraft and missile applications and space vehicle applications is the high degree of variability in the space mission requirements. In aircraft and missile systems, vehicles were produced with identical equipment aboard, identical functions to be performed in the computer, and were designed for identical missions. Programming for each computer was not performed individually, but the programs were written for the weapon system and were used in all of the vehicles. However, in the space systems even though there may be identical systems, the mission requirements for each system may be different. Consequently, one programming job will not be sufficient for the whole family of identical or nearly identical space vehicles, but each computer program must be designed to the specific mission requirements of each space vehicle. Trade offs are always possible between programming complexity and hardware simplicity.

In aircraft and missile systems, the designers usually have made the decision to use simpler hardware at the expense of programming complexity because the programs once written were applicable to all of the vehicles. However, in the spaceborne applications with each space vehicle having its own mission requirements, the trade off in favor of simplified hardware must be examined more closely. Another requirement that may influence the design of computers for deep space probes is the additional data processing capability beyond that required for real-time control functions. The desirability of extremely low powered systems may make the systems design such that the complexity and hardware requirements in the computer may be expanded to reduce the hardware complexity and power requirements in the telemetry system with a net savings in power and reliability. It may be possible by preprocessing the measurement data on-board the space vehicle to reduce the amount of data which must be transmitted back and hence, reduce the bandwidth and power requirements in the telemetry equipment.

III. RELIABILITY CONSIDERATIONS

A major difference between the aircraft system application and space system application of digital computers is required reliability. Aircraft system reliability is highly desirable because of economy and weapon system effectiveness. However, a mean-time-between-failure (MTBF) of between a few hundred and a few thousand hours has been acceptible because of the relatively short mission times in an aircraft system and the availability of the equipment for scheduled or nonscheduled maintenance between flights. In special applications when mission times may range between weeks and perhaps years, the reliability requirements will be much more stringent. In an APOLLO mission of approximately two weeks, if a probability of success of 0.95 for the total system is desired, a system MTBF of 7720 hours is required. If the probability of success for the computer portion of the system is 0.99, the reliability requirement for the computer is 33,600 hr MTBF. If one looks beyond the APOLLO type lunar missions to, for instance, a Saturn probe mission of approximately 50,000 hr mission time, the computer system must have a MTBF of 5 million hr to have a probability of success of 0.99.

Except for radiation hazards, the space environment is more benign than the aircraft environment and this will be extremely helpful for system reliability. A submarine environment is nearly like the environment that would be expected in a space vehicle. Actual operational experience with identical digital computers which are used in both submarine and airborne applications indicates that the reliability in the submarine application is approximately 10 times better than that in the aircraft.

Reliability in digital computers has been the subject of intensive efforts for several years. Reliability can be obtained through a combination of several techniques. These techniques consist of using the most reliable basic electronic components that are available in circuits which are designed to be tolerant to deviations in component parameters. These circuits are then combined to provide the simplest computer which will do the job. Using manufacturing processes which are carefully controlled so that incipient modes of failure are not introduced during assembly also is important in achieving reliability. If the above techniques do not provide for enough reliability to do the job, some form of redundance will be incorporated.

A. COMPONENT CONTRIBUTIONS TO RELIABILITY

Electronic components that are likely to be used in the next few years will include conventional resistors, capacitors, diodes, transistors, etc, that have been designed and developed for high reliability in combination with thin film circuit elements and integrated circuits. Other circuit and will have a predicted MTBF of approximately 50,000 hr.

Within one year, there will be at least one large central computer for space applications which will weight approximately 20 lb, occupy approximately 0.25 cu ft, will require under 200 w, and will have a predicted MTBF of approximately 20,000 hr. This computer will have a computational capability comparable to large scale scientific and data processing computers.

Both of these computers will have integral analog-to-digital and digital-to-analog conversion capability which will facilitate integration with a variety of devices in the space vehicle.

If they can be used, special purpose devices having little flexibility, even lower power requirements, and higher reliability than either of these computers, can be provided.

If the 20,000 hr MTBF computer is used in a multiple computer system application similar to that shown in Figure 7, and the mean time to repair is 1 hr and reduced system capability is defined as function A or function B having to be curtailed, the mean time between reduced system capability is 66,700,000 hr or 7610 years. The significance of this number is not that anyone contemplates a system which will operate for several thousand years but that for a manned deep space mission the probability of successful completion of the mission is very high.

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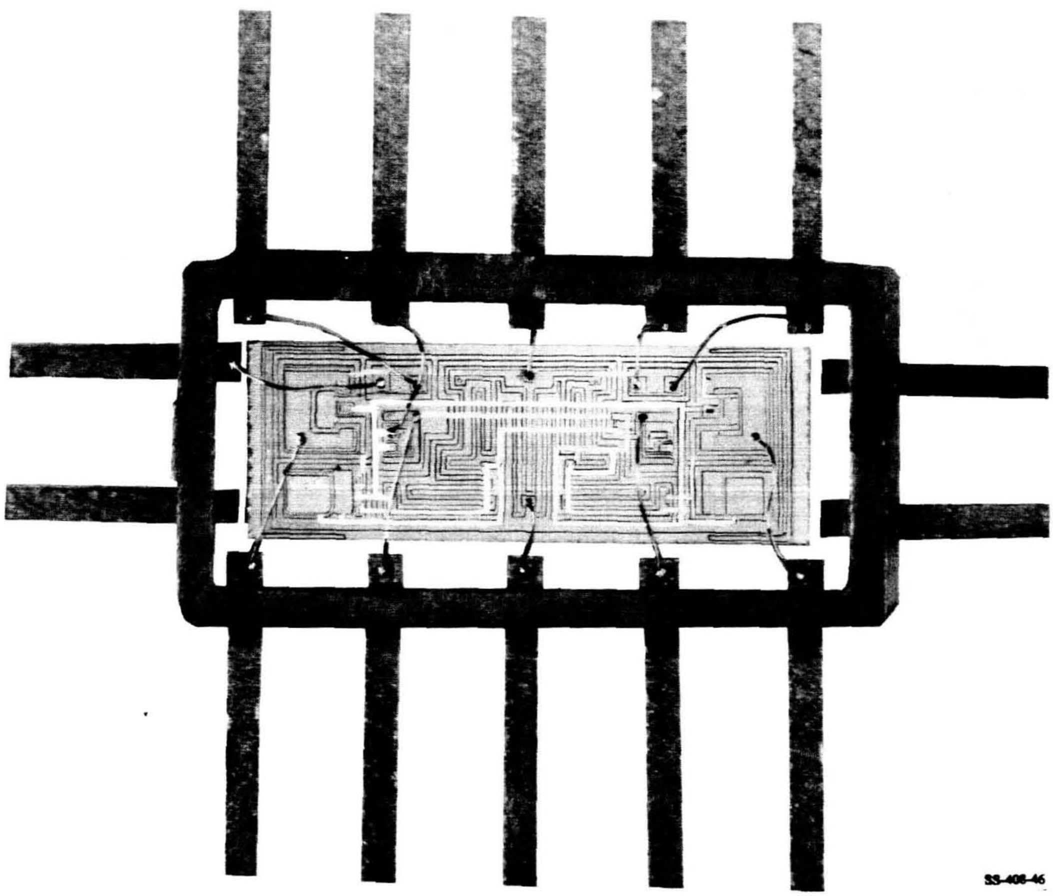

Figure 1. Integrated Circuit

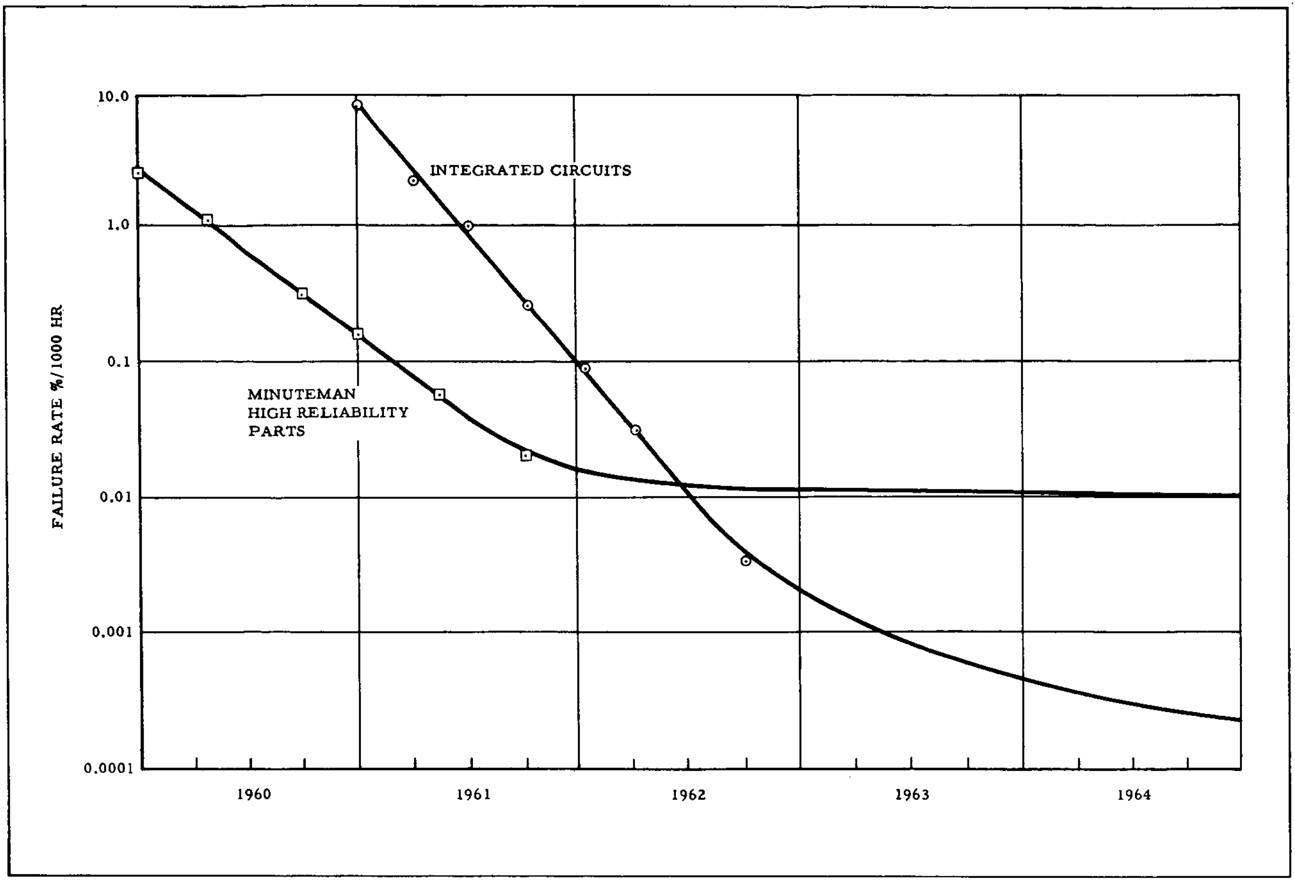

Figure 2. Reliability Comparison: Integrated Circuits Versus MINUTEMAN High Reliability Component

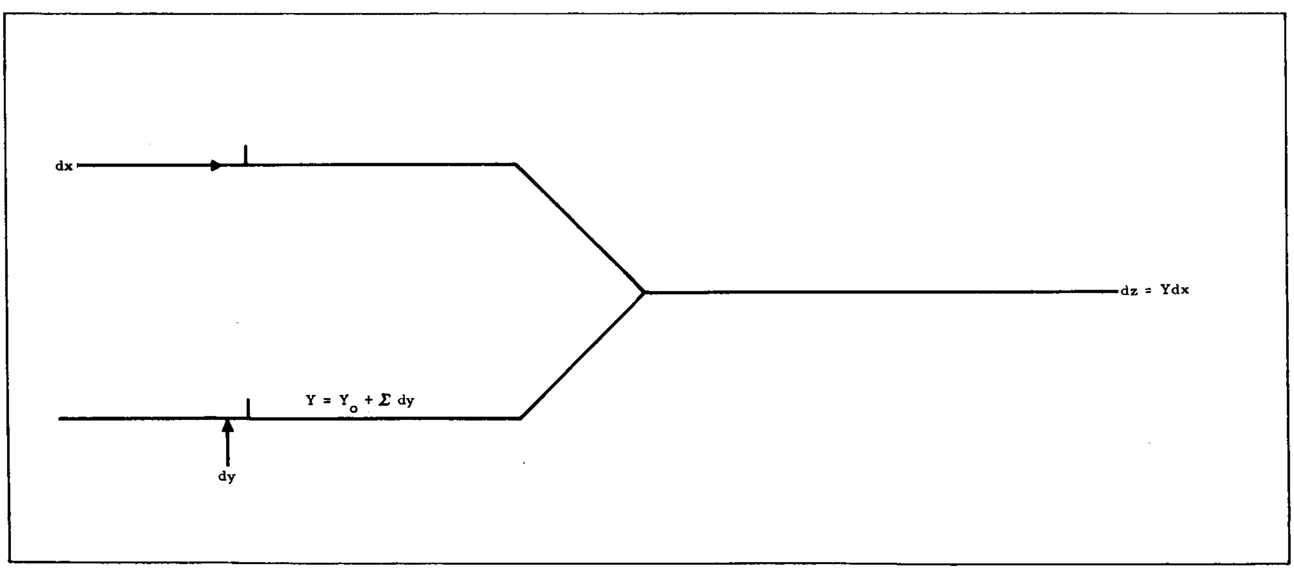

Figure 3. Integrator Symbol

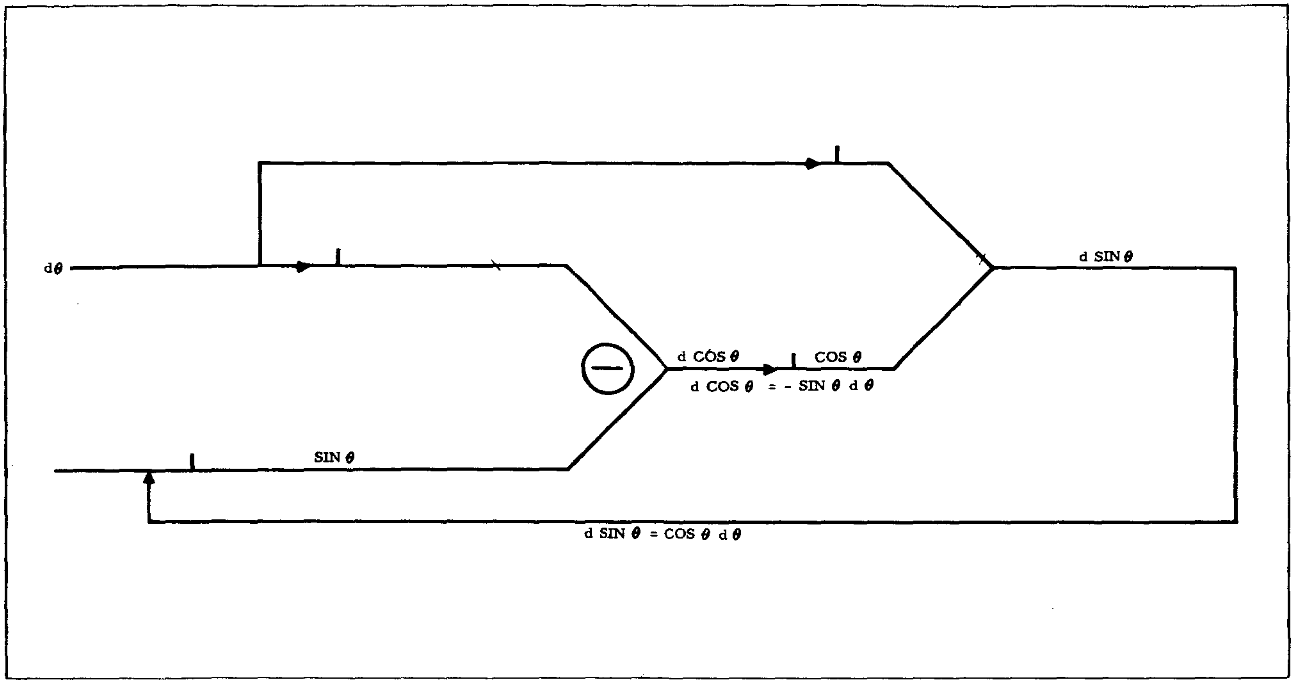

Figure 4. Sine and Cosine Generation with a Digital Differential Analyzer

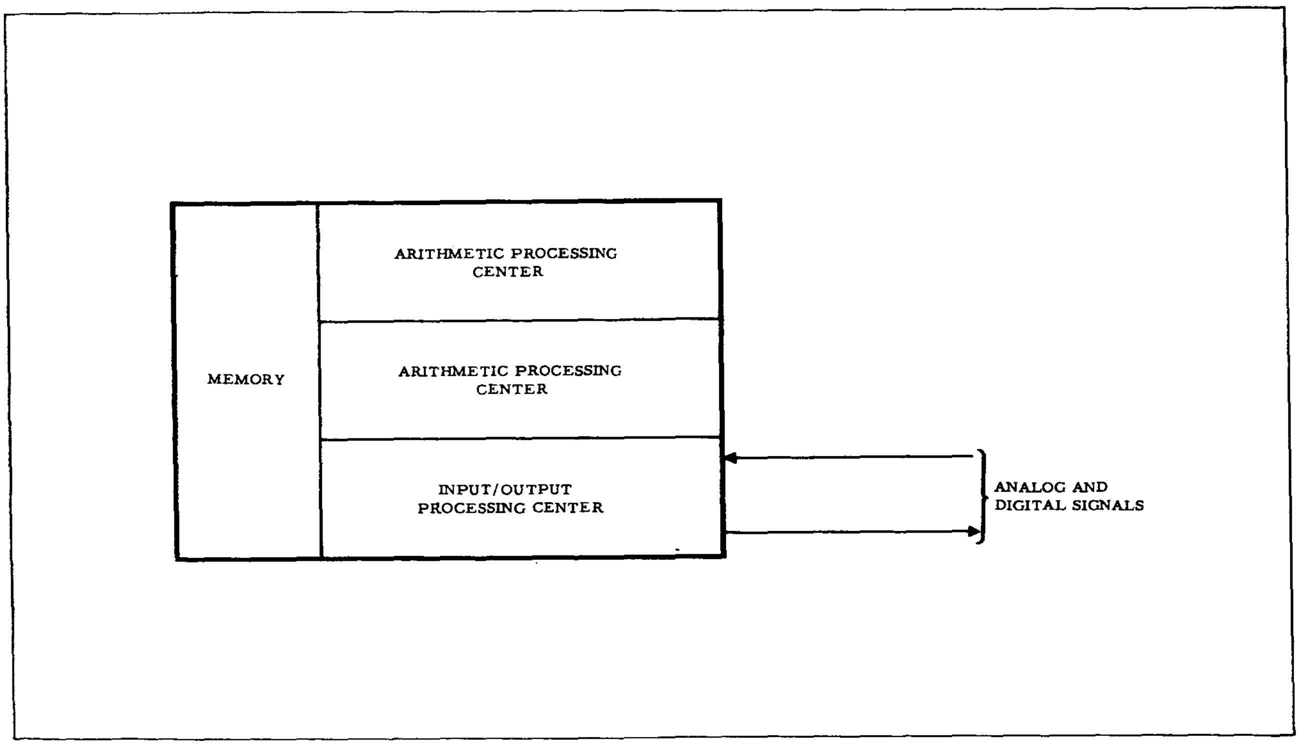

Figure 5. Parallel Processing Center

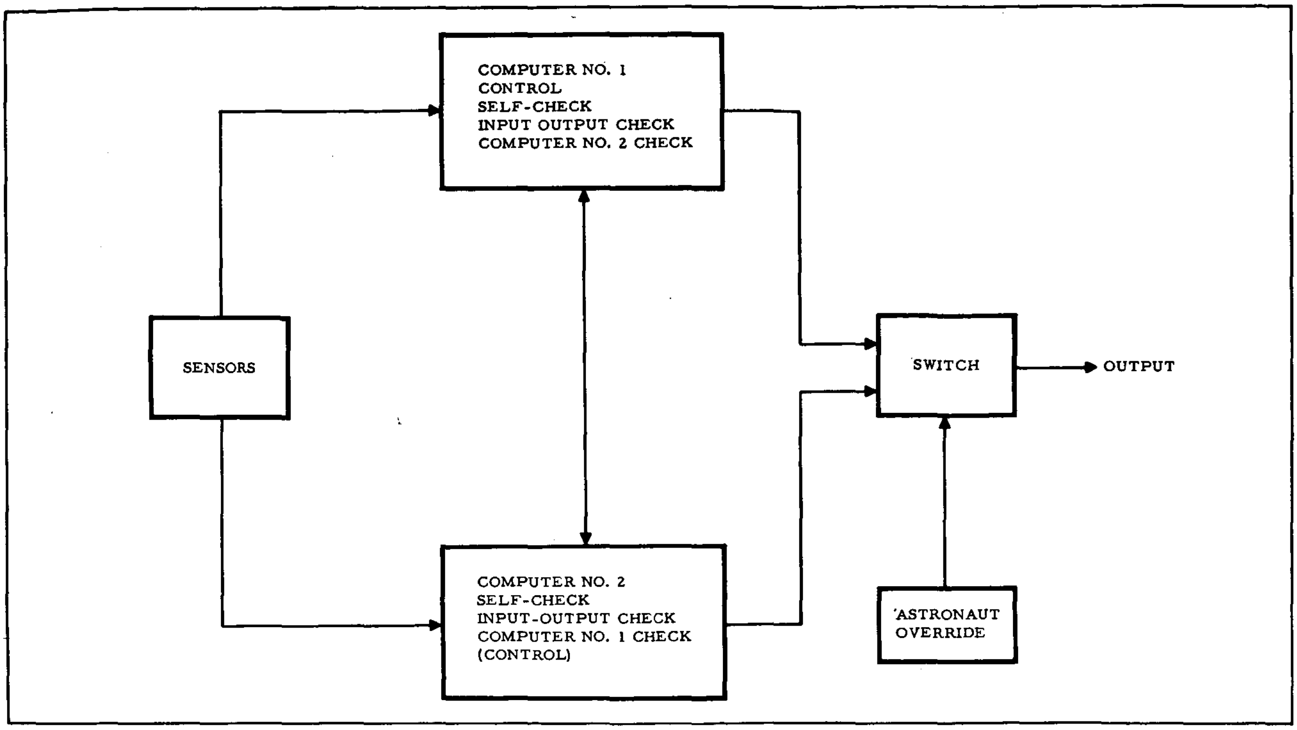

Figure 6. Subsystem Redundancy

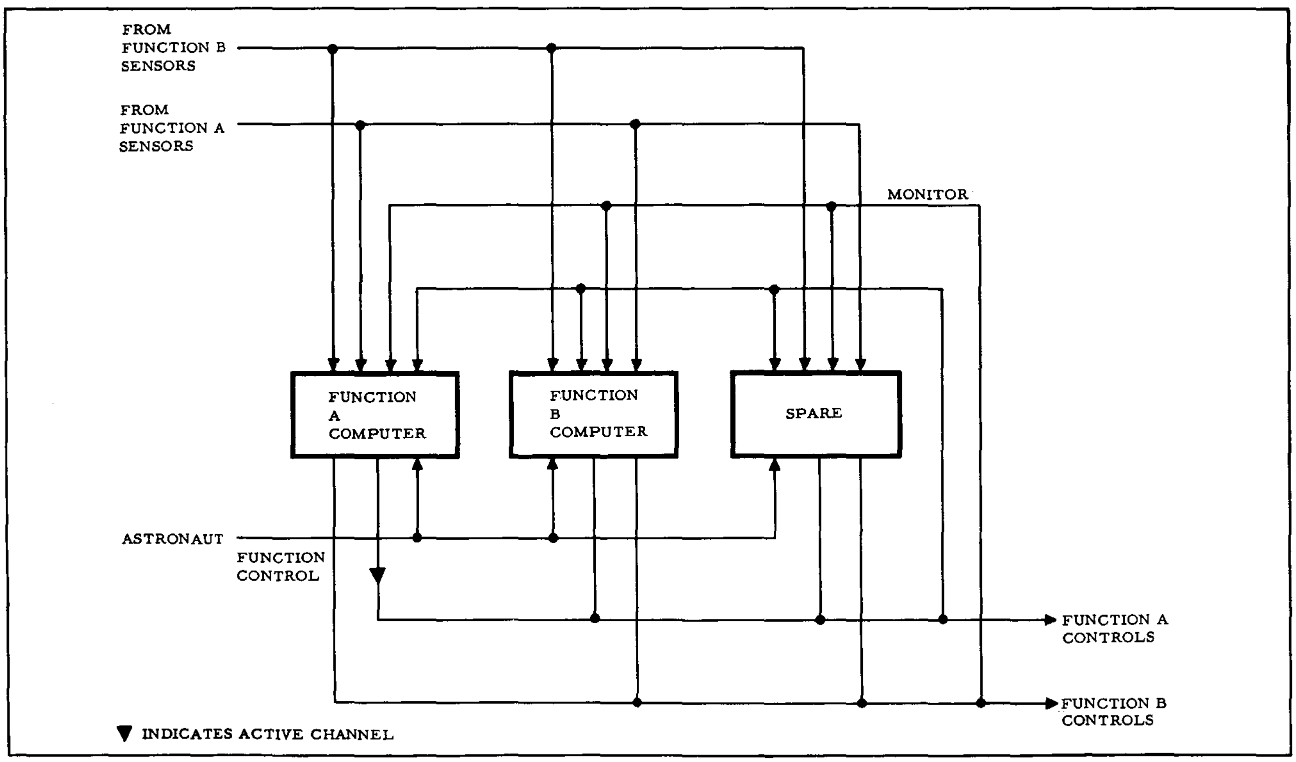

Figure 7. Multiple Computer System

LATERAL-RANGE AND HYPERSONIC LIFT-DRAG-RATIO REQUIREMENTS FOR EFFICIENT FERRY SERVICE FROM A NEAR-EARTH MANNED SPACE STATION

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Introduction

The operation of any long-term manned space station will require some type of ferry vehicle to transport men and equipment to and from the station with regularity and reliability. Such a vehicle, designed for entry at near-orbital speeds, could also be useful in the return from any deeper space mission if either an earth-orbit rendezvous terminal maneuver or a maneuver combining atmospheric braking and a near-earth parking orbit is used.

This study was undertaken to determine the class of vehicle which could be most efficiently used as a ferry vehicle between a near-earth space station and the earth. One measure of this efficiency is the ability of the vehicle to reach prechosen landing sites with some prescribed frequency.

In considering this frequency of return it is necessary to consider not only the normal mode of operation in which only infrequent returns are scheduled at desirable times, but also operation under various degrees of emergency, which dictate quick or even immediate return to earth. In extreme emergencies, when immediate return to earth is necessary, choice of landing site becomes impractical. In most cases, however, although it might be required to abandon the station quickly, the ferry vehicle could remain in orbit for some time before initiating reentry in order to land at a prechosen site. The allowable delay time in orbit would be determined primarily by the capabilities of the ferry life-support system.

This paper will examine the geometry of the ferry ranging problem, that is, the lateral ranges required to reach chosen landing sites from various near-earth orbits, and will investigate and compare several means of achieving these ranges. The particular case considered is that of returning from a space station which is in a circular orbit at an altitude of 200 statute miles, but the results obtained are not sensitive to orbit altitude for orbits within a few hundred miles of the surface[1]. From this orbit the vehicle will retro and reenter at very close to satellite velocity. The down-range problem can be handled by proper timing of the retrofiring, and the desired lateral range can be achieved by aerodynamics, space propulsion to change orbit plane, atmospheric propulsion, or combinations of these methods. The relative cost in terms of weight of using these different methods to achieve lateral range will be discussed.

Lateral Range by Aerodynamics

In order to tie the lateral-range requirements to vehicle aerodynamics, an investigation of the cross-range capabilities of vehicles with various hypersonic lift-drag ratios was undertaken. The results of this investigation are presented in figure 1.

The maximum lateral range achievable is presented in terms of the vehicle maximum hypersonic L/D. For this investigation a value of W/C_DS of 200 lb/ft^2 was chosen. At the lower values of L/D a value of 75 lb/ft^2 was also considered and was not found to have a significant effect on the values presented. To achieve these lateral ranges, the vehicle performs a shallow reentry and holds a constant altitude at the bottom of pullup by rolling about the wind vector until an appropriate equilibrium glide path can be followed. This mode of operation has been discussed previously[2,3].

Requirements for "Quick" Return

In the discussion that will follow, "delay time" is taken as the time between decision to reenter and initiation of reentry, and a "quick" return is defined as a return with a delay time of less than one orbit. For a quick return, therefore, the vehicle initiates reentry some time in the first orbit after the decision to reenter.

The maximum lateral range that could be required for quick return from a given orbit to a landing site at a given latitude is determined, as shown in figure 2, to be given by

$$Y_{req} = r_E(\alpha_1 + \lambda_2) \quad \text{or} \quad r_E\left[\pi - (\alpha_1 + \lambda_2)\right] \qquad (1)$$

whichever is the lesser. Here α_1 is the orbital inclination to the equator, λ_2 is the latitude of the landing site, and r_E is the radius of the earth. These range requirements are shown in figure 3. Also shown are the values of hypersonic L/D required to achieve these ranges. Several geographic points are shown to indicate the limits of the United States and of the U.S. Mainland.

The ability to reach any point on the globe once each orbit from any orbit inclination is seen to require a hypersonic L/D of about 3.6. For equatorial or polar orbits no lateral range is required to reach points on the equator or the poles, respectively, once each orbit. Although such landing sites are usually impractical, they do serve to point out the fact that the minimum lateral ranges required for quick return from equatorial or polar orbits approach zero as the location of the landing sites approaches these points. For other inclinations no choice of landing site can assure quick returns with zero lateral-range capability.

Effects of Delay Orbits on Range Requirements

For an equatorial orbit the range requirements to reach a base at a given latitude are fixed. For orbits of other inclinations, the requirements can be reduced through use of delay orbits or multiple landing sites.

The effect of delay orbits on lateral range and associated L/D requirements is illustrated in figure 4, which presents the maximum holding time required to reach a landing site near Edwards AFB from orbits of various inclinations and for several values of the hypersonic L/D. The solid curves represent boundaries or discontinuities in the picture, while the dashed curves are representative of the members of the family. As seen in figure 4, a vehicle with an L/D of 3.6 could reach Edwards AFB within one orbit for any orbital inclination. This quick return is possible for a vehicle with L/D = 2.7 from orbits inclined at less than 22° to the equator or from a polar orbit. With less L/D some holding time is required for a polar orbit and the orbits for which quick return is possible are limited to low inclination orbits as illustrated by the L/D = 2 curve. With an L/D of greater than 1.9 quick return from an equatorial orbit is possible, while with less L/D the landing site cannot be reached from an equatorial orbit. The curve for L/D = 1 shows that at least daily return is possible to a base such as Edwards for all orbits which pass over the continental U.S. As L/D is decreased further, the holding times for all inclinations increase and the band of low inclinations, for which no return is possible, grows wider. With values of L/D greater than 0.84, holding times of less than 12 hours are required to reach the site from a polar orbit, but for values of L/D less than 0.84, the delay time for a polar orbit jumps to 24 hours and remains there for values of L/D down to 0.66 after which no regular return to Edwards from a polar orbit can be counted on.

These discontinuities are shown more clearly in figure 5 where a cross plot of maximum delay time to reach Edwards AFB is plotted against L/D for three orbital inclinations. As can be seen, from an equatorial orbit it is a quick-return or no-return proposition, with the break point coming about L/D = 1.9. For the polar orbit, quick return is possible down to an L/D of 2.7, after which there is a steady increase of holding time with reduction in L/D down to an L/D of 0.84 where a holding time of about 12 hours can be required. For values of L/D between 0.84 and 0.66 once daily return is possible, and for lower values of L/D, no regular return is assured. The curve for an inclination of 30° shows some of the characteristics of each of these limiting curves.

We must keep in mind that in discussing these delay times we have made no restriction as to the time of day the return is to be made. If we restrict our landing to the daylight hours, approximately 12 hours or more additional delay time could be required for single site landings.

Return From Polar Orbits

Choice of landing site. In considering space stations for reconnaissance, surveillance, weather-watching, or other applications, the complete earth coverage provided by polar or near-polar orbits is attractive. In considering return from such orbits, proper choice of landing site location and use of multiple landing sites can serve to alleviate ranging requirements for either quick or delayed returns.

The effect of landing site choice on the L/D requirements for return from a polar orbit is shown in figure 6. The interior of the table presents the values of L/D required to assure reaching the representative landing sites shown on the left utilizing various numbers of delay orbits. The values presented for zero delay orbits are those for quick return, that is, ability to return once each orbit. The sites at Alaska and Thule were included on this chart for purposes of emphasizing the decrease in L/D requirements with increasing latitude of landing site. Although the unfavorable climates at these latitudes might preclude their use as prime landing sites, their possible use as emergency landing sites might warrent consideration. As may be seen from this chart, for return from a polar orbit, the L/D required can be reduced considerably by proper choice of landing site when a number of delay orbits are allowed.

Use of multiple sites. While still considering polar orbits, it is obvious that considerable reductions in lateral-range requirements can be obtained through use of multiple landing sites as illustrated in figure 7. In this figure we are looking down on the northern hemisphere, with the orbital plane appearing as a straight line. The landing sites are placed on the equator for figure simplicity.

If we have one landing site, it can be as much as 90° in longitude away from the plane of the orbit. If two sites are considered, chosen so that they are separated by 90° in longitude, one of the sites will always be within 45° of the orbit plane. This distance is reduced to 30° for three sites and 22.5° for four sites spaced as shown.

The decrease in maximum possible separation from the orbit plane is, of course, directly reflected in a decrease in the lateral-range requirements to reach at least one of these sites, and, therefore, a decrease in the vehicle L/D requirements. In figure 8 the L/D required to reach several landing site combinations are shown for various numbers of delay orbits. Edwards AFB is included as a single site for comparison purposes. The other combinations of sites shown, reading downward, are two bases within the continental United States, two bases with near optimum spacing, three bases across the continental United States, three near optimally spaced bases, and four bases approaching optimal spacing. The degree to which multiple bases properly spaced can reduce vehicle L/D requirements is apparent.

Interim Stations and Relaxed Return Requirements

The discussion to this point has dwelt on the requirements for quick returns and returns with delay times up to one-half day and has been primarily concerned with polar orbits. The values of L/D that have evolved from this analysis represent what we would like to have for efficient, regular service to a future space station. If we consider ferry requirements for an earlier space station, the questions that arise are how much do we have to relax our return requirements in order to use the low L/D and ballistic vehicles which will be available, or, conversely, just how much L/D or how large a landing area is required to meet minimum return requirements.

The minimum lateral-range angle η_y which will allow twice daily return from a polar orbit is given by

$$\left(\eta_y\right)_{II} = \frac{\omega_E \tau_S}{2} \cos \lambda \qquad (2)$$

where ω_E is the earth rotation rate, τ_S is the orbital period, and λ is the latitude of the landing site. For the near-earth orbits considered in this paper, $\omega_E \tau_S$ is taken as 22.5°. For the ability to return once daily the minimum lateral-range angle required is given by

$$\left(\eta_y\right)_{I} = \left(\frac{\pi + 2\lambda}{2\pi}\right)\left(\frac{\omega_E \tau_S}{2}\right)\cos \lambda \qquad (3)$$

These lateral-range requirements and their associated hypersonic L/D values are presented in figure 9. As was shown previously, a vehicle with an L/D of about 1 can reach any point on earth twice daily from a polar orbit, while an L/D of about 0.7 would give the capability of daily return to any site. The approximate limits of the U.S. Mainland are indicated to show the ranges that would be required for return to mainland landing sites on a daily or twice daily basis from a polar orbit.

Early space station orbits may well be other than polar, inclined so as to take advantage of existing launch and tracking facilities. For such orbits we can determine the minimum range which will assure the ability to return to a properly positioned base at least once daily by the method illustrated in figure 10. Shown as solid lines are three consecutive orbit traces mutually separated by a distance $\omega_E \tau_S$. The maximum latitude of these orbits is given by $\lambda_1 = \alpha_1$ and the intersection of the traces can be shown to occur at λ_2 which is given by

$$\tan \lambda_2 = \cos\left(\frac{\omega_E \tau_S}{2}\right)\tan \alpha_1 \qquad (4)$$

If a landing site is located at a latitude λ^* halfway between λ_1 and λ_2, then this site will always be within an angular distance of $\frac{\lambda_1 - \lambda_2}{2}$ from one or other of these traces. The maximum lateral range required to reach this site at least once a day is therefore given by

$$Y_{req} = r_E \eta_y = r_E\left(\frac{\lambda_1 - \lambda_2}{2}\right) \qquad (5)$$

These values are shown in figure 11 for all orbital inclinations.

It should be borne in mind that these values apply for a landing site at the latitude λ^* which is located slightly under the maximum latitude which the orbit reaches. For a polar or equatorial orbit, λ^* becomes 90° and 0°, respectively, and no lateral range is required as was indicated in an earlier figure. For other inclination angles the ranges required are small, reaching a maximum value of less than 20 miles for an inclination of 45°. These small ranges should be within the capability of any controllable landing device, such as a paraglider, or could be achieved through use of a small amount of space propulsion, thus placing the landing sites within the reach of nonlifting reentry vehicles. These small range requirements also serve to indicate the size of landing area that would be required if the vehicle had no control over its set down.

On the right of figure 11 are presented the fuel requirements which would be required to achieve these cross ranges by use of space propulsion to change the inclination of the orbital plane. These values were obtained assuming a specific impulse of 300 seconds. It can be seen that the weight of propulsive fuel required for bases within the continental U.S. is about 2 to 3 percent of the reentry vehicle weight.

Weight Penalties Associated With Lateral Range

As mentioned previously, the lateral ranges required may be obtained by several methods in addition to the use of aerodynamic glide which has been discussed to this point. The other methods considered here are:

1. The use of space propulsion

2. The use of combinations of space propulsion and aerodynamic glide

3. The use of aerodynamics and atmospheric propulsion

Structural or fuel weight penalties are, of course, associated with these various methods and one purpose of this paper is to evaluate the cost in terms of weight of these approaches.

The weight of vehicles capable of reentering the atmosphere with high values of L/D will, of course, be greater than that of vehicles capable of only ballistic reentry. Some data to this effect are presented by E. S. Love[1] and have been used to obtain figure 12. The abscissa on figure 12 is the lateral range that could be obtained aerodynamically by the vehicles and the upper nonlinear scale presents the vehicle hypersonic L/D. The shaded area encompasses the majority of the data presented in reference 1. The weight increase is very rapid for the first few hundred miles of lateral range, primarily because of the ineffectiveness of the low L/D vehicles in obtaining lateral range. The exact weight values will, of course, change for each study, and the solid line which was chosen for use in the remainder of this paper represents a mean of all the data rather than a curve of minimum weights.

Space Propulsion

Propulsion can be applied either in the atmosphere or in space to obtain the desired range. If atmospheric propulsion is to be used, it is most efficient to use air-breathing engines with their relatively high values of specific impulse. If pure rocket propulsion is to be used, this can be used best in orbit where the fuel and propulsion systems do not have to execute the reentry. The initial weight in orbit then consists of the normal reentry vehicle, the fuel for orbital transfer, the tanks, engine, and any special guidance system required for the orbital transfer.

Assuming a circular orbit with an impulse applied 90° before the point at which maximum range will be required, the ratio of the initial to final weight will be given by

$$\log \frac{W_i}{W_f} = \frac{2r_E \sin \frac{Y}{2r_E}}{I_{SP}\sqrt{g_E r}} \tag{6}$$

where

W_i initial weight

W_f final weight

g_E acceleration due to gravity at r_E

r_E radius of the earth

r orbit radius

Y lateral range

I_{SP} specific impulse of rocket

The weight ratio W_i/W_f is plotted against range for several orbit altitudes in figure 13. The effect of specific impulse is also shown in this figure. A value of I_{SP} of 300 seconds will be used for the remainder of the study of space propulsion. Because of the lower orbital velocity at the high altitudes, the energy required to change the orbit is lowered. For low orbits, however, a change of several hundred miles in altitude has little effect on the weight ratio.

In actual practice the initial weight W_i should be increased by the weight of the tankage, engine, and associated control system. No detailed study has been made of this weight; however, a preliminary review indicated that 20 percent of the fuel weight should be more than enough to cover these items. The dashed curve in figure 4 shows that this is not a very large factor in the total weight.

In this analysis it has been assumed that the orbit change is independent of any retroimpulse used to initiate reentry. For small amounts of lateral-range change (on the order of 200 to 300 miles) the retro-rocket, when properly oriented, may be able to supply a large part of the energy required to obtain the desired lateral range; however, for lateral-range requirements over 1,000 miles, its effect is small. The exact effect of the retro-rocket on these curves would, however, be a strong function of the initial orbit and reentry conditions required which make the inclusion of these effects impractical as they are beyond the scope of the present investigation.

Aerodynamic-Glide-Plus-Space Propulsion

Combinations of space propulsion and aerodynamic glide might be practical in some cases. The weight required can be found by

$$\frac{\text{Weight in orbit}}{\text{Weight in orbit for zero lateral range}} = \left(\frac{W_i}{W_f}\right)\left(\frac{\text{W in orbit}}{\text{Weight in orbit for zero lateral range}}\right)_{L/D} \tag{7}$$

and

$$\text{Lateral range} = (\text{Lateral range})_{L/D} + (\text{Lateral range})_{\text{propulsion}} \tag{8}$$

These relations give the curves shown in figure 14. For large lateral ranges, L/D appears the more economical approach and combinations of the two would not be designed into the system just to obtain range capability. For small values of lateral range, space propulsion appears to be more economical and should be studied further for the specific configurations and tasks considered.

Atmospheric Propulsion

The use of propulsion in the atmosphere allows the higher specific impulse of air-breathing propulsion systems to be utilized. However, the efficient use of propulsion in the atmosphere requires aerodynamically efficient configurations since

$$R = I_{SP} V (L/D)_{eff} \log\left(\frac{W_i}{W_f}\right) \tag{9}$$

where

R range

V velocity

$$(L/D)_{eff} = \frac{\text{Weight}}{\text{Drag}} = L/D \frac{1}{\left(1 - \frac{V}{V_{orb}}\right)^2}$$

The specific impulse values that can be expected in the atmosphere with ram jet, turbojet, or combinations of various jet engines are shown in figure 15. The shaded area represents the current predictions for the performance of future engines.

Propulsion in the atmosphere can be applied either at subsonic or supersonic speeds. For the purposes of this paper it is assumed that the L/D is constant throughout the supersonic speed range. While this is not strictly true, it will be seen later that secondary variations in L/D will not alter the conclusions. For the subsonic speeds, the L/D corresponding to a given hypersonic L/D class of vehicles is not easily determined since it depends strongly on the design philosophy.

Since range is proportional to L/D, it appears unattractive to consider atmospheric propulsion for very low values of L/D. On the other hand, because of the inherent range capability of very high L/D vehicles, they will not require the use of propulsion. The analysis has therefore been carried out only for an L/D of 1 vehicle with the expectation that the analysis could later be extended to higher L/D values if atmospheric propulsion looked promising. It was found that for this vehicle it is more efficient to use atmospheric propulsion supersonically than subsonically for subsonic L/D values up to about 8.

A maximum range is obtained for a given weight ratio at about 11,000 feet per second for the high I_{SP} values and at 7,000 feet per second for the low I_{SP} values. Solving the range equation for these values gives two sets of weight ratios which do not make any allowance for containing the fuel or the weight of the engine and other associated equipment. The effect of these factors is very strong on the overall weight ratio since the vehicle must reenter the atmosphere with all of this weight. However, it is interesting to compare these weight ratios (starting at an L/D = 1 vehicle) with the weight predicted for range obtained by the L/D alone. The area between the two curves just above the L/D alone curve in figure 16, therefore, is the band of the weight ratio including only the original L/D = 1 vehicle and the fuel. Even this area is well above the L/D alone curve and indicates that, neglecting the increase in vehicle empty weight, this approach is still uneconomical from the weight standpoint.

In order to show the effect of the vehicle modifications required to reenter with the extra fuel and equipment, an elaborate but approximate weight analysis was made. This analysis attempted to account for variations in the percentage of weight tied up in heat protection and structures as the vehicle becomes larger to include fuel and propulsion equipment. The analysis is subject to much question concerning the exact values used, but since the use of atmospheric propulsion does not look promising even in the case where this added weight is neglected, the details of the weight analysis are not of importance and therefore have not been included in this paper. The results of this analysis, however, are included in figure 16 as the band between the upper two curves to illustrate the large effect that can result from modifying the vehicle so that it is capable of reentering with the fuel and associated equipment.

Conclusions

It has been shown that the lateral range required for return is strongly dependent on the location and number of landing sites, on the inclination of the orbit to the equator, and on the amount of delay time that can be tolerated. For the ability to reach any point on earth once each orbit from any orbit, a vehicle with a hypersonic L/D about 3.6 is required. Although it would be nice to have this capability, it is questionable whether it is actually needed, and in any event it is probable that such a vehicle will remain beyond our grasp for quite some time to come. If we are satisfied with selected single or multiple bases and are willing to accept reasonable delay times in orbit prior to initiation of reentry, the hypersonic L/D requirement for return from a polar orbit can be reduced to the neighborhood of one. For other than polar orbits, larger ranges are, in general, required for returns with delay times of one-half day or less, but for properly selected landing sites, very small lateral ranges are necessary to assure daily return.

This study has also shown that for large lateral ranges, the use of space propulsion is very expensive and not competitive with L/D alone. A combination of space propulsion and L/D is not attractive and such a combination should not be considered for the sole purpose of obtaining lateral range. For small amounts of lateral range space propulsion with an L/D = 0 vehicle appears to achieve lateral range more economically than aerodynamic glide. The differences, however, are not great enough to eliminate the possibility that efficient L/D vehicles can be made competitive with space propulsion in this range.

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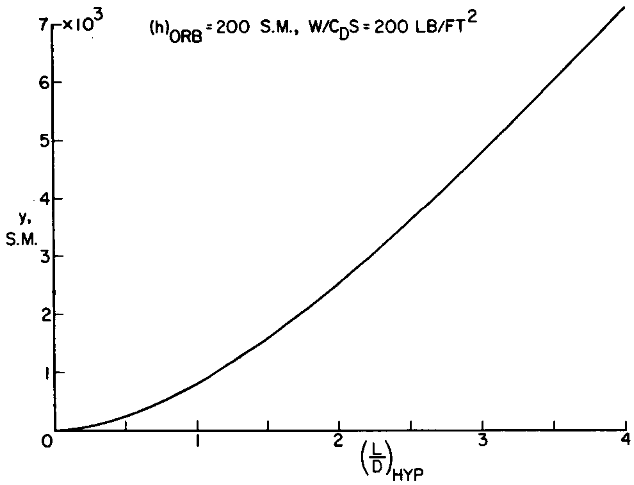

Figure 1.- Lateral range obtainable by aerodynamics.

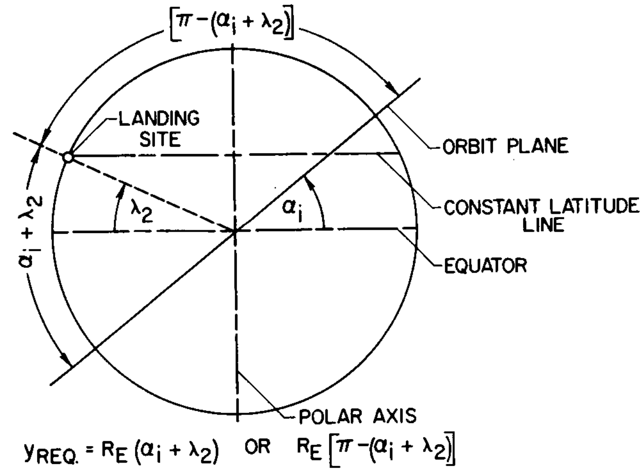

Figure 2.- Maximum lateral range required for "quick" return.

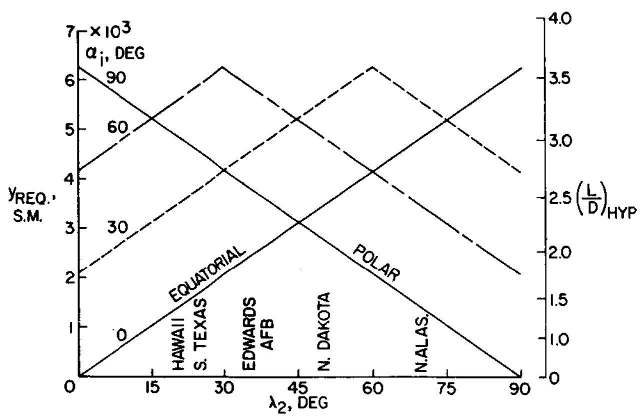

Figure 3.- Lateral range required for "quick" return from space station orbit.

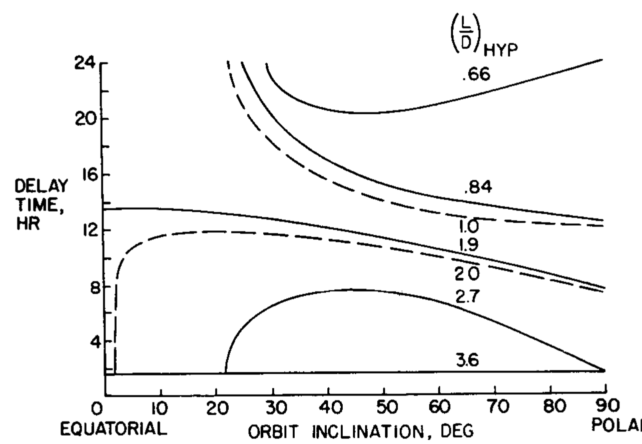

Figure 4.- Maximum delay time for return to Edwards AFB.

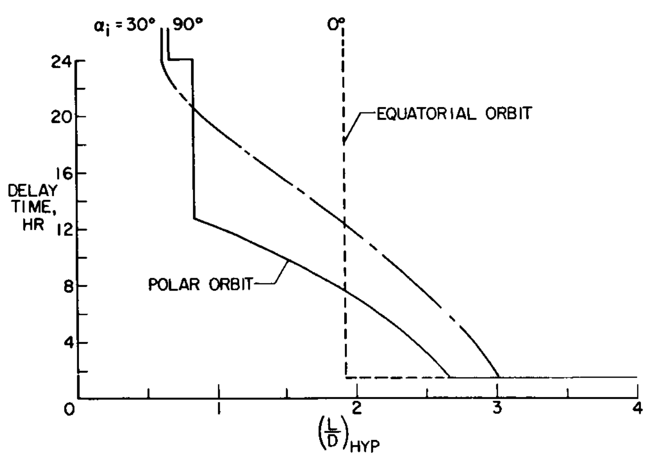

Figure 5.- Maximum delay time for return to Edwards AFB.

		NUMBER OF DELAY ORBITS							
λ_2	LANDING SITE	0	1	2	3	4	5	6	7
35°	EDWARDS AFB	2.7	2.6	2.4	2.2	1.9	1.6	1.3	.8
42°	NEW YORK STATE	2.4	2.3	2.2	2.0	1.8	1.6	1.2	.8
50°	NORTH DAKOTA	2.2	2.1	2.0	1.8	1.7	1.5	1.1	.7
70°	ALASKA	1.4	1.4	1.3	1.2	1.1	1.0	.8	.5
77°	THULE	1.1	1.1	1.0	1.0	.9	.7	.6	.4

Figure 6.- Hypersonic L/D required to reach various landing sites from a polar orbit.

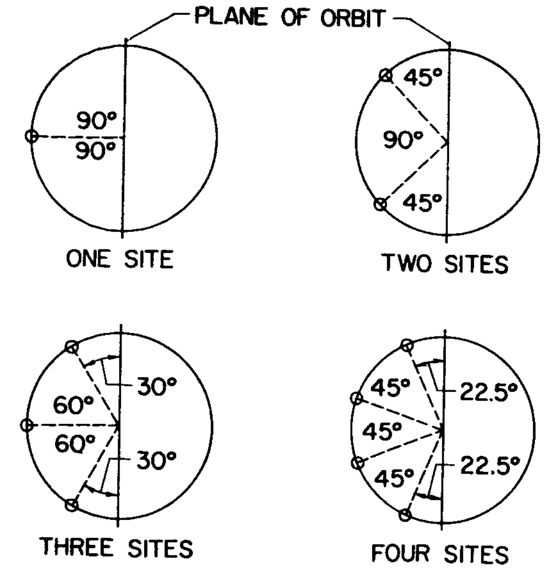

Figure 7.- Optimal spacing of multiple landing sites for return from polar orbit.

	NUMBER OF DELAY ORBITS							
LANDING SITE	0	1	2	3	4	5	6	7
EDWARDS AFB	2.7	2.6	2.4	2.2	1.9	1.6	1.3	.8
EDWARDS + NEW YORK	2.4	2.3	2.1	1.8	1.6	1.2	.8	.8
* EDWARDS + S.E. AUSTRALIA	2.1	1.9	1.6	1.2	.8	.8	.8	.8
EDWARDS + HOUSTON + NEW YORK	2.4	2.3	2.1	1.8	1.6	1.2	.8	.8
* EDWARDS + HAWAII + S.E. AUSTRALIA	1.8	1.3	1.0	1.0	1.0	1.0	.9	.8
* NEW YORK + EDWARDS + HAWAII + S.E. AUSTRALIA	1.6	1.1	1.0	.8	.8	.8	.8	.8

* NEAR OPTIMAL SPACING

Figure 8.- Hypersonic L/D required to reach multiple landing sites from a polar orbit.

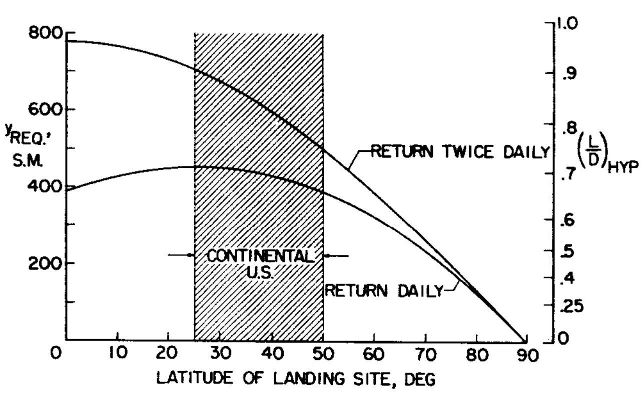

Figure 9.- Lateral range required for daily or twice daily return from a polar orbit.

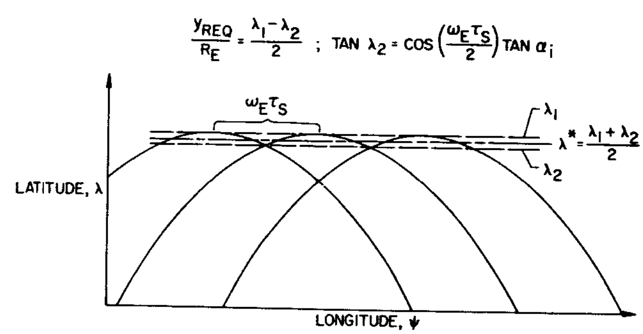

Figure 10.- Location of landing site to achieve daily return with minimum ranging capability.

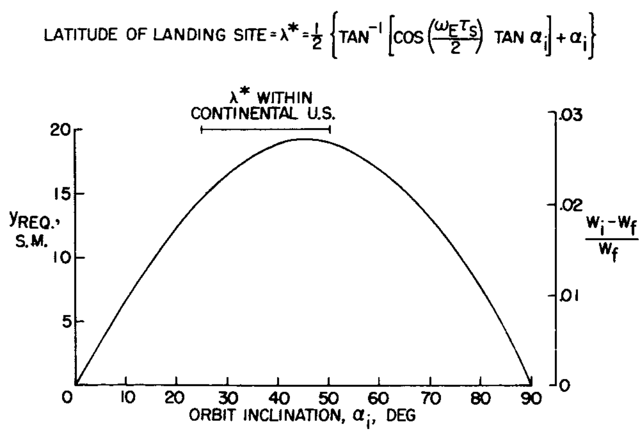

Figure 11.- Minimum lateral range required for daily return from inclined orbits.

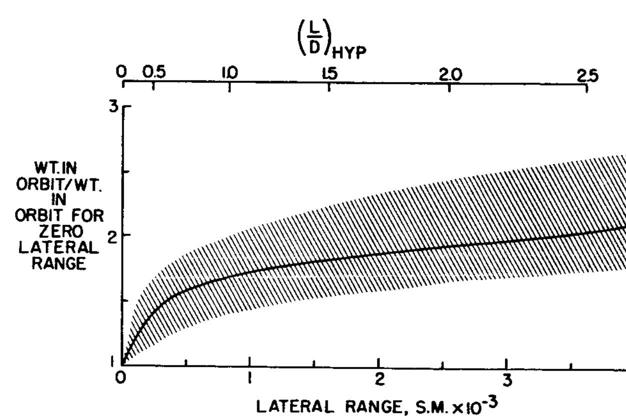

Figure 12.- Weight required to achieve lateral range by aerodynamics.

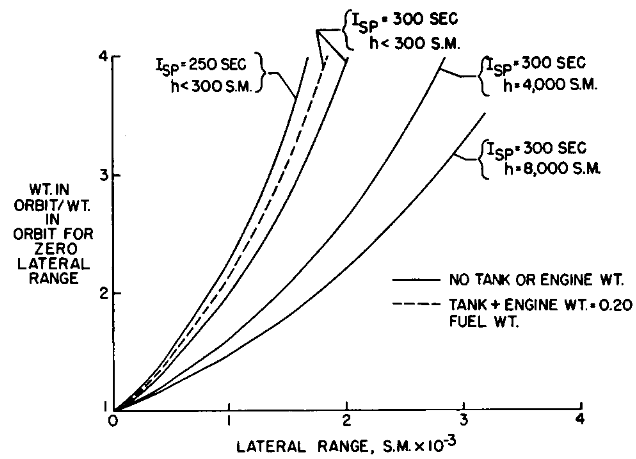

Figure 13.- Weight required to obtain lateral range by space propulsion.

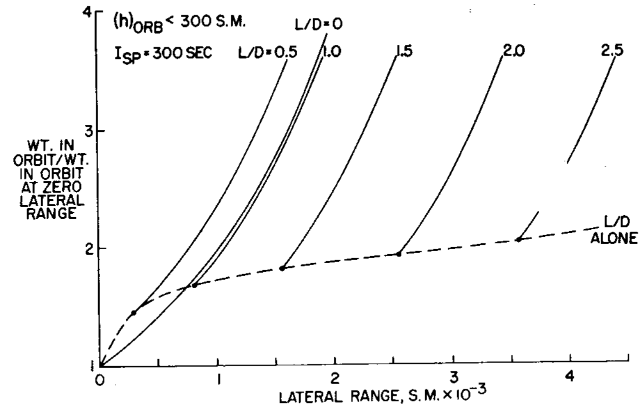

Figure 14.- Weight required to achieve lateral range by aerodynamics plus space propulsion.

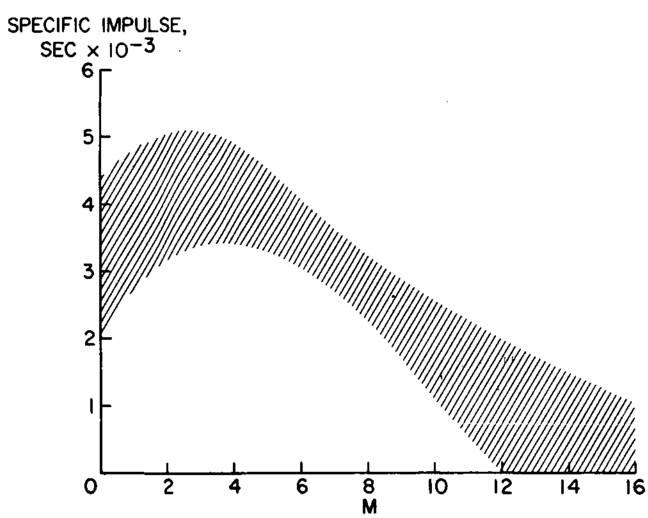

Figure 15.- Estimated specific impulse of air breathing engines.

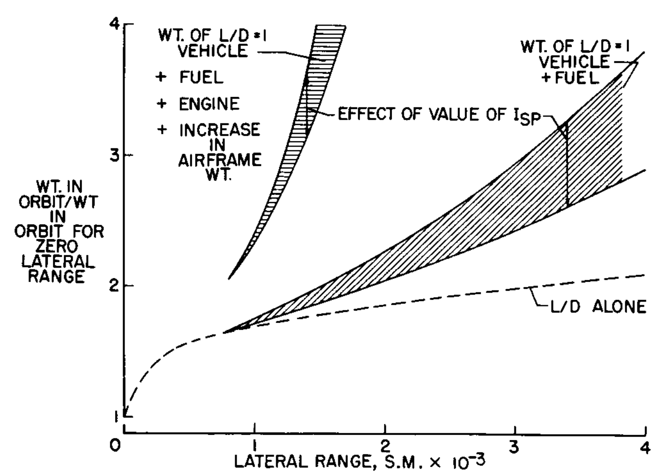

Figure 16.- Effect of lateral range on weight using atmospheric propulsion - vehicle L/D = 1.

A LOOK AT MANNED ENTRY AT CIRCULAR TO HYPERBOLIC VELOCITIES

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Summary

The environment associated with manned entry is reviewed with the objective of hopefully arriving at an entry vehicle class that shows promise of future mission versatility and a capability of entry over a range of entry velocities from circular to hyperbolic without large penalties for aerodynamic performance. Such a multipurpose entry vehicle is shown to be feasible to the degree that it would appear to merit further study in consideration of future entry vehicle needs.

AUTHOR

Symbols

A	ablation as main heat protection mode
C_L	lift coefficient
C_D	drag coefficient
g	deceleration, earth referenced
L/D	lift-drag ratio
M,P,F,G	relative evaluations for landing approach: marginal, poor, fair, good
Q	heat load
$\dot{q}$	heat rate
r	nose radius
R	radiative cooling as main heat-protection mode
S	reference area
T	temperature
2-D	two dimensional
V_E	entry velocity
W	weight
ϵ	emissivity
γ	flight-path angle
γ_E	entry angle
$\Delta\gamma_E$	corridor width from undershoot to overshoot

Subscripts:

max	maximum value
rad	radiative heating
conv	convective heating
s	stagnation point
eq	radiation equilibrium
o	at L/D = 0
p	payload
26	at entry velocity of 26,000 ft/sec

Introduction

The theme for this meeting, as given in the meeting announcements and summarized in figure 1, is to review the trends and requirements for next generation space programs, using the status of today's programs and technologies as the focal point. Moreover, to assist in future planning, four questions have been posed. These are:

1. Where have we been?
2. What have we learned?
3. Where are we going?
4. What will we need?

The studies from which the results to be given in this paper are drawn were stimulated by similar questions directed toward and confined to manned entry vehicles. For the most part we will deal with the second question - what have we learned - but with an eye toward the others.

As for where we stand now, figure 2 gives the picture of the manned entry vehicles involved in current programs in terms of their hypersonic lift-drag ratio and nominal entry velocity. At the far left with design entry velocities at orbital speed are Mercury with L/D of zero, Gemini with L/D of about 1/4, and X-20 or Dyna-Soar with an L/D of slightly less than 2. At escape velocity is Apollo with L/D of 1/2. We will not bother to describe these vehicles further, since status reports on each are scheduled for this meeting, and their basic features are well known.

This picture raises some interesting questions for the future if we assume that manned activities in space are not only going to continue but will become more frequent and of greater variety. For example, what is the comparative growth potential of these vehicles in terms of ability to adapt to greater entry velocities (also lesser velocities for Apollo) and to different missions? What L/D class, if any, including those outside the values for these vehicles, would appear to merit special attention in consideration of future entry vehicle needs? The question of mission growth potential will be left to those giving reports on these vehicles, although the results to be presented herein may aid in such an assessment. By and large it is difficult to answer any of these questions without some knowledge of what type of missions will be undertaken beyond those associated with these vehicles. In this respect, most of us are uniquely gifted in our inability to predict the future. Nevertheless, we can within certain restraints examine the problems broadly with the hope of gathering at least some useful indications, if not specific conclusions.

For a number of reasons, among which are included economics, the several years that transpire from conception to flight for any new entry vehicle, and the acceleration that could be afforded manned space activities, there would appear to be a place in the future for a multipurpose entry vehicle that would satisfy the essential requirements of a wide variety of missions, including variable entry velocity. In the variety of missions, we include not only the peaceful and scientific, but possible military applications as well. This is perhaps not as Utopian as it may first sound, if we accept the idea of using an entry vehicle that, although it might not be ideally suited for a particular mission, would be sufficiently versatile to do the job without major sacrifice in the mission objectives. Clearly, reuse would be a prime requirement in the vehicle design. Of course there is recognition that in the spectrum of manned space activities there will be certain missions requiring certain specialized entry vehicles, even should such a multipurpose vehicle become a reality.

Although discussion of future manned missions is beyond the scope of this paper, one point is worth noting. A look at possible future missions quickly reveals an interest in a wide range of entry velocities. For example, figure 3 presents the effect of orbit altitude (circular orbit) or apogee altitude (elliptic orbit) upon entry velocity. Entry is assumed to begin at 400,000 feet. Within the range of entry angles of interest for manned entry, about 0^{o} to 10^{o}, there is little effect of entry angle upon entry velocity, except when the orbit or apogee altitude is within about 1,000 miles or less of the earth. Once the altitude exceeds several hundred miles, there is initially a marked increase in entry velocity. At 6,000 miles out, the entry velocity exceeds 30,000 ft/sec. At the higher altitudes the velocity increase with altitude lessens, but note that at the altitude for a 24-hour orbit a vehicle will face an entry velocity near 34,000 ft/sec. Moreover, lunar and most planetary missions will involve entries at escape speeds or beyond, unless propulsive braking in some form is used. For these reasons, entry velocity is considered to be one of the prime variables in studies of multipurpose entry vehicles.

No attempt has been made to list the reference works that have already examined one or more aspects of the entry problems that were reviewed in the course of the present study. An extensive survey of the literature was part of this study, as will be evident from the compilation of data in some of the figures. Accordingly, the list of references[1-45] appended hereto should be regarded as typical rather than exhaustive. In some respects, therefore, this study has been a review of what has been learned from studies by others; in most respects it has been a new examination; and in all respects it has been an independent examination, as will likely be evident. The objective has been to examine the environment associated with manned entry at circular to hyperbolic velocities with the hope of arriving at a general class of multipurpose entry vehicle that shows promise of future mission versatility and a capability of entry at circular to hyperbolic velocities without large penalties for aerodynamic performance. Should this objective not be reached, then hopefully the study would give a better feel for what we are up against in attempting to arrive at such a vehicle class. It is beyond the scope of this paper to give a summary of results of all facets of the entry problem that were examined in the course of this study. In what follows, we have selected examples that are believed to convey reasonably well a picture of the overall results. For these selected areas we will try to summarize the results broadly, avoiding detailed delving into the specialty areas.

Scope and Assumptions in Entry Calculations

The entry velocities generally range from orbital speeds, or nominally 26,000 ft/sec, to 46,000 ft/sec. The latter value was chosen somewhat arbitrarily; however, it is representative of entry velocities for a number of planetary missions. Primary attention is given to values of maximum L/D at hypersonic speeds between 1/2 and 2; in some instances the limits are extended to 0 and 3.

Although the advantages and disadvantages of a variety of entry modes and maneuvers have been examined, it is sufficient for the purpose of this paper to summarize the results for one representative mode. Accordingly, all results shown herein are derived from trajectory calculations that conform to the following conditions. The undershoot boundary is assumed to be deceleration-governed by a maximum of 12g. The overshoot criterion is a no-skip entry completed within one pass. Entry is considered to be initiated at 400,000 feet with the vehicle in a trimmed condition at either $(L/D)_{max}$ or $C_{L_{max}}$. A constant L/D trajectory is maintained from entry to pullout. At this point the vehicle is rolled so as to maintain a constant altitude flight path, i.e., roll or lift vector modulation is assumed. This maneuver is maintained until the vehicle is unable to generate sufficient lift to sustain flight at that particular altitude. An equilibrium glide maneuver at either $(L/D)_{max}$ or $C_{L_{max}}$ is then initiated and flown to the landing point.

The earth is considered to be spherical and nonrotating with an atmosphere defined by the ARDC 1959 model atmosphere.

The results were obtained by machine calculations for the region from entry to pullout and by analytic methods from the pullout point to landing.

Discussion

Deceleration

The maximum deceleration to be expected during entry and its relation to entry angle and entry velocity is reviewed in figure 4. Consider first the hatched areas which encompass entry speeds from 26,000 to 46,000 ft/sec. The values of g_{max} indicated by the curves in this area correspond to the maximum deceleration experienced during that portion of the trajectory involving deceleration from superorbital to orbital speed. It is evident that as orbital speed is exceeded, peak g becomes increasingly sensitive to change in entry angle and that for manned lifting entry at superorbital speeds, entry angle will be limited to roughly 10° or less.

Also shown are several curves falling outside the hatched areas. These correspond to suborbital entry velocities and are included merely to bring attention to the important role that abort conditions may have upon peak g. It is evident that there are marked effects of the velocity at which entry is initiated following abort, and of vehicle $(L/D)_{max}$. Other factors which will not be treated here, such as altitude at abort-entry, retro application[23,24], etc., have important bearing on abort peak g. Our examinations indicate that with a capability to withstand the order of 12g, a lifting vehicle would be able to handle a wide range of abort conditions.

All of the results given in figure 4 are for entry at $(L/D)_{max}$; however, the picture is essentially the same for entry at $C_{L_{max}}$.

At the steeper entry angles, the peak g's experienced at overshoot (positive lift is employed in these examples) is shown to be significantly lower for $(L/D)_{max} = 2$ than for 1/2. Figure 5 takes a closer look at the peak g's at overshoot as affected by L/D and entry velocity. These results do not correspond precisely to the overshoot peak g's indicated in figure 4, but essentially so. These are the peak g's experienced when the pullout achieves a flight-path angle of 0°, and the velocity at pullout is as shown. The main conclusion here is that increasing entry velocity brings about marked increase in peak g's at pullout, particularly at low L/D. It is equally important to note that there are sizeable benefits in alleviating peak g's by increasing L/D to about 1; however, further increase in L/D does little to alleviate peak g at pullout until very high entry velocities are encountered. The same remarks apply for any given value of entry angle between undershoot and overshoot. We cannot overlook the implication that the peak g's shown here are going to be difficult to avoid without resort to other modes of entry or maneuvers during pullout, such as skip, pitch-modulation, etc., some of which may be less desirable.

The dashed line in figure 5 represents the peak g's that would be experienced in an equilibrium glide following deceleration to orbital speeds. This merely serves to indicate that for L/D's of about 1/2 and greater the subortibal-speed portion of the entry will, at approximately escape entry speeds or less, generally produce peak g's greater than those experienced in overshoot pullout, whereas the overshoot pullout will usually produce the greater peak g's at hyperbolic entry speeds.

Corridor Width

The width of the entry corridor between undershoot and overshoot is of interest primarily from the standpoint of guidance and flexibility of operation. Figure 6 summarizes this subject in terms of the difference in entry angle between undershoot and overshoot, $\Delta\gamma_E$, as a function of hypersonic $(L/D)_{max}$. Increasing entry velocity drastically reduces the width of the entry corridor, from about a 10° maximum at orbital entry speeds to about a 2° maximum at 46,000 ft/sec. The latter width is generally considered to be acceptable without excessive demands on guidance requirements. It is apparent, however, that at much higher velocities the corridor width will become undesirably small and some widening by resort to different entry modes, negative lift, propulsion-induced negative lift, or the like, bears consideration. In these studies, we have regarded positive lift as the normal mode of entry, but with negative lift capability to be available, the latter to be treated as a safety feature to be used only in emergencies.

The effect of $(L/D)_{max}$ is much the same as the case for deceleration, namely, that there is little to be gained by exceeding an L/D of about 1. Entry at $C_{L_{max}}$, as contrasted with entry at $(L/D)_{max}$, is of interest because of the reduction in total heat load that is effected. Note that there is no serious reduction in corridor width by entering at $C_{L_{max}}$. This might be expected because of the variation of L/D at $C_{L_{max}}$ with $(L/D)_{max}$, as will be discussed later.

Heating

Any examination of the heating picture for a multipurpose entry vehicle as advanced herein is confronted with the possibility that radiative heating may have a major if not dominant place in the heating picture at hyperbolic speeds. While good progress has been made over the past year in the understanding of radiative heating and its prediction, there remains much room for improvement. In particular, accurate methods of prediction for areas considerably removed from the stagnation region and on afterbodies are urgently needed. In spite of these difficulties, some conclusions can be drawn as to the role of vehicle type in radiative heating when we impose the requirement that it must have capability of entry at hyperbolic speeds. The latter is an important factor to bear in mind in what follows, for termination of the entry velocity requirement at subescape speeds would alter the conclusions decidedly.

Attention is directed first to the right-hand portion of figure 7. Here is shown in a qualitative way the heat load input to an entry vehicle in terms of Btu's per pound of vehicle weight as a function of entry velocity. Two ballistic body types are considered; one is blunt with large nose radius, and the other is conical with small nose radius. The curves indicate the convective and radiative heat inputs. At entry velocities generally falling in the subescape regime, the advantage lies with the blunt body since the heating is essentially all convective and the nose radius is large. As entry velocity increases the radiative contribution first becomes significant on the blunt body, but over a narrow range of velocities the sum of the radiative and convective heat load still remains less than that for the conical body which is still deriving nearly all of its heat load from convective heating. Further increase in velocity soon brings about a reversal in the picture as the radiative load begins to dominate. (Note that the ordinate is to log scale, the abscissa to linear.) The result is that at hyperbolic velocities the conical shape is more efficient from the viewpoint that it has less total heat input. Reference is made to the work of Allen[14] for detailed treatment of this subject.

Guided by the foregoing considerations and the results of Allen[14], Bobbitt[25], and others, one can draw a fair analogy between the pointed or slightly blunted cone of fairly large half-angle at zero angle of attack and a pointed or slightly blunted highly swept delta shape at fixed high angle of attack, to conclude that if the former strikes an optimum in the tradeoffs between radiative and convective heating, the latter should have fair merit in this regard. Accordingly, we have selected the latter class of vehicle as one deserving special attention in the heating studies. Moreover, as has already been pointed out in the discussion of decelerations and corridor widths, some lift is desired. These two features, small nose radius and lift, act to reduce both the magnitude of the radiative heating and its relative importance to the total heat load. This is illustrated in the left-hand portion of figure 7, where the ratio of radiative heat load to convective heat load is shown as a function of entry velocity for three vehicle types representing different combinations of L/D and nose radius. Note that the undershoot condition is considered; this tends to accentuate the role of radiative heating. Estimates of the ratios for the stagnation point and entire body are shown. These estimates do not have the confidence level one could wish for, particularly the radiative contribution and the entire-body estimates. Generous use has been made of the work of Allen[14], Wick[20], Bobbitt[25], and others in arriving at these values, and where comparisons could be made, these estimates were found to be in reasonable agreement with those from other studies. The overall picture is felt to be fairly reliable and at least capable of illustrating the marked reduction in the importance of radiative heating to the total heat load as L/D is increased and nose radius is decreased. The $L/D = 0$ vehicle is a hemisphere with a short cylindrical afterbody; the $L/D = 1/2$ vehicle is of the Apollo type entering at the attitude shown; and the $L/D = 1$ vehicle is a highly swept, delta-planform, lifting body entering at $C_{L_{max}}$.

In the heating results to follow a vehicle with a loading W/S of approximately 35, a nose radius of 1 foot, and similar in shape to the $L/D = 1$ vehicle has been assumed. The results of figure 7 and other examinations indicated that for this assumption, convective heating accounts for nearly all of the total heat load, and that the convective heat rates far exceed the radiative, although at the highest velocities the radiative heat rate is significant at undershoot. For our purposes, then, convective heating should give a reasonable picture of the environment. In the results to follow, the value of W/SC_L for entry at $C_{L_{max}}$ and the value for entry at $(L/D)_{max}$ were assumed to be invariant with $(L/D)_{max}$.

Figure 8 summarizes the maximum stagnation point convective heat rates as a function of entry velocity. It is evident that regardless of whether entry is made near overshoot, undershoot, at $(L/D)_{max}$, or at $C_{L_{max}}$, entry velocity has a major effect on the heat rates. As has been observed in earlier studies by others, undershoot produces maximum heat rates much larger than experienced at overshoot; and, entry at $C_{L_{max}}$, as contrasted with entry at $(L/D)_{max}$, has a decided advantage in reducing maximum heat rates, moreso at the higher entry velocities. Moreover, the maximum heat rates for entry at $C_{L_{max}}$ are less sensitive to the value of vehicle $(L/D)_{max}$. For entry at $C_{L_{max}}$ a multipurpose lifting entry vehicle having a 1-foot nose radius and capable of covering the velocity range shown would be required to handle heat rates up to about 1000 Btu/ft^{2}sec. There is the overall indication that in considering heat rates, it is advantageous to keep $(L/D)_{max}$ small.

Figure 9 is merely a conversion of the heat rates of figure 8 to radiation equilibrium temperatures with an assumed emissivity of 0.85. For entry at $C_{L_{max}}$, the maximum radiation equilibrium temperatures would range from about 3000° F to 6500° F, and to higher temperatures for entry at $(L/D)_{max}$.

Figure 10 presents the stagnation point heat loads. The hatched band for any value of $(L/D)_{max}$ covers the heat loads from undershoot (bottom of band) to overshoot (top of band). The large reduction in heat load that is realized by entry at $C_{L_{max}}$ rather than $(L/D)_{max}$ is quite evident; and, as in heat rate picture, there is strong justification for keeping $(L/D)_{max}$ small, especially for entry at $(L/D)_{max}$. These factors direct interest toward a design that gives $(L/D)_{max}$ near $C_{L_{max}}$, and that enters near $C_{L_{max}}$ and employs roll modulation to govern ranging. It is encouraging to note that for entry at $C_{L_{max}}$ the increase in stagnation point heat load with entry velocity is not overly large over this velocity range, and in light of the localized areas subjected to this increase, would appear to be tolerable from the standpoint of increasing heat protection weight on a multipurpose vehicle with increasing entry velocity, additional support of which will be given later.

Some mention of the possible effect that use of pitch modulation[4,26-28] in the pullout might have upon the results presented thus far seems appropriate at this time. In this connection, modulation restricted to the regime between $C_{L_{max}}$ and $(L/D)_{max}$ is the mode of possible interest, with both lift and drag being modulated. The prime justification for pitch modulation in pullout is reduction of decelerations and widening of corridors. However, modulation of this type, as compared with pullout at $C_{L_{max}}$, increases both heat rates and heat loads. Estimates indicate that for $L/D \approx 1$, the penalty in heat load is relatively moderate, but by no means insignificant; further increase in L/D soon brings about sizeable increase in the heat load penalty. The prime questions are: in the absence of pitch modulation in pullout are the accelerations so large and the corridor width so small that pitch modulation in pullout is a necessity; or second, do the advantages of incorporating pitch modulation in pullout justify the heating penalty incurred? For the highest velocity considered in this study, it is doubtful that this point has been reached. However, in light of the desirability of entering at high C_L with subsequent roll modulation, the addition of some pitch modulation capability for use during and after pullout - with roll modulation remaining the primary mode - might offer a fuller realization of the aerodynamic performance potential without a prohibitive heating penalty, provided L/D does not exceed about one and the pitch modulation is sufficiently restricted.

Heat Protection

Figure 11 summarizes the results of figures 8 to 10 in the form of heat load versus heat rate so as to establish in a general way the relation of the stagnation point heating to the materials picture. The left end of each curve corresponds to entry at 26,000 ft/sec and the right end to 46,000 ft/sec. The hatched boundary is that suggested by Roberts[17] for approximating the limits to which metallic shields can operate; for example, the refractory metals can be expected to cope with some 40 to 50 Btu/ft^2/sec, and a copper heat-sink approach would be so heavy in handling heat loads greater than about 10,000 Btu/ft^2 that it would probably not be feasible. An extensive review of the current state of heat protection technology indicates that barring rapid developments in ceramics, transpiration cooling, and one or two other approaches, ablation materials will be the most likely choice for the stagnation region of the multipurpose vehicle treated herein.

Of greater concern, perhaps, than the heating to the nose or stagnation region is the heating of the major surface areas of the vehicle. An estimate of the maximum radiation equilibrium temperatures that would exist along the streamwise centerline of a delta-planform lifting body with $(L/D)_{max} = 1$ and entering at $C_{L_{max}}$ is shown in the left-hand portion of figure 12. The assumed emissivity is 0.85 as before. The hatched bands indicate the range of temperatures to be expected on the lower surface between undershoot (top of band) and overshoot (bottom of band) for entry velocities of 26,000, 36,000, and 46,000 ft/sec. Note that there is a drop of only a few hundred degrees in progressing 20 feet rearward from the tangency point of the surface with the hemispherical nose.

The curves showing the rapid decay in temperature with distance rearward are for the upper surface centerline and the condition of overshoot. Although these estimates are subject to considerable uncertainty because there is insufficient knowledge on how to handle the expansion over the upper surface, they should give some feel for the near-minimum temperatures to be expected on the vehicle.

In the right-hand portion of the figure is given the status of the life of coated refractory metal sheet as summarized recently by Mathauser[18]. The different curves represent different refractory metals; it is not essential to our purpose to identify each, but they include tungsten, tantalum, molybdenum, and columbium, and they represent a generally optimistic average of test information. The broad result is that present-day coatings can provide protection under continuous exposure of at least 1 hour at 3000° F to 100 hours at 2500° F, and that an order of magnitude or greater decrease in coating life is obtained under cyclic exposure conditions. This serious degradation under cyclic temperature exposure reflects directly on the reuseability of refractory metal components in entry vehicles. Added to Mathauser's compilation is a band indicating a probable improvement in the picture from future coatings and/or ceramics as indicated by current studies. This gain has not yet been realized for sheet-type application. Current work indicates that its achievement will likely be accompanied by short material life or inherent erosion, thereby inferring refurbishment after each entry flight, and in this respect would require a refurbishing technique similar to that for a surface protected by ablation material.

A comparison of the two sides of figure 12 indicates that while current refractory metals can adequately handle most of the surface areas of manned lifting vehicles entering at orbital speeds and perhaps slightly higher speeds, other modes of heat protection are required over much of the vehicle if it is to have a high growth potential in entry velocity. A review of the technology indicates that of the several approaches that show promise, a refurbishable ablation shield appears to offer the best approach to this objective at this time. Other approaches such as refractory metals might be satisfactory over large areas of the upper surface for entry at $C_{L_{max}}$, particularly if they have long life on a cyclic basis. On the other hand, the use of a refurbishable ablation covering over these areas as well may offer the greatest growth potential in terms of entry velocity, modes of entry, and therefore mission versatility.

In manned vehicles the greatest safety to the man, and generally the greatest versatility for the vehicle, come about by insuring that insofar as environmentally produced hazards and stresses are concerned, the limit factor is the man and not the vehicle, otherwise man's full potential has not been safely exploited. This would seem to be a desirable philosophy to follow in selecting the heat protection system and load-carrying structure of a multipurpose entry vehicle as envisioned herein, provided it is not carried to extreme. In this respect, our examinations have led us to conclude that there is much to be said for perfecting a refurbishable ablation covering technique in conjunction with a cold load-carrying structure. Such an approach allows the separation of the heat-protection job from the load-carrying job, thereby affording a choice of the optimum material and structure for each. A desirable and apparently reasonable goal is refurbishment at the operational site in a period of a few days or less by a technique that lends itself to use of coverings of different thickness, dependent upon the entry velocity of the mission to be undertaken or the safety factor desired. Hopefully, the technique would also be able to capitalize readily on new and more efficient ablation materials as they are developed.

The effect upon vehicle aerodynamics of the shape changes that accompany ablation might be an area of concern for vehicles with high hypersonic L/D if they require small leading-edge radii to achieve their aerodynamic performance. Our examinations indicate that for vehicles with L/D's at least as large as one this question can be largely circumvented through appropriate design.

As a final comment on the materials picture, much remains to be learned about the performance of coated refractory metals, ablation materials, and other thermal protection schemes during prolonged exposure to space environment. Both laboratory and flight data now in hand and being accumulated indicate some problem areas, but as yet nothing that appears to be insurmountable for the more promising approaches outside of the problems created by the impact of fairly large meteoritic particles; fortunately, impact by these larger particles occurs infrequently.

Weight of Entry Vehicles

Within the past year or so, a number of system studies have been made of entry vehicles by various industrial organizations and government agencies. Much of this information is of a proprietary nature, or classified. Nevertheless, some indications of the results of these studies can be presented if confined to a form that respects the interests of the source. For these reasons, the sources of the data points in much of what follows are not identified.

Figure 13 presents the results of a literature survey of entry vehicle system studies. At the top of the figure is shown the variation in the ratio of total entry vehicle weight at finite L/D to total entry vehicle weight at $L/D = 0$, i.e., W/W_o, as a function of hypersonic $(L/D)_{max}$. Each point represents a vehicle study. All of the points are for entry at or near orbital speed. The studies encompass 1 to 3 man vehicles and 1 to 14 day missions. In some cases, in order to form the ratios on a more common footing, it was necessary to increase or decrease vehicle size slightly; the same applies to equipment. Whenever this was done, the procedures employed by the group who made the original study were followed in making the revised estimates. In any event, it can be stated with confidence that the spread in the band at any value of $(L/D)_{max}$ is far less an effect of the range of the variables just mentioned, than a reflection of the vehicle type selected by those who conducted the original study, plus their differing views as to the procedures for estimating structural and equipment weights. The broad conclusion of this survey is that at a given $(L/D)_{max}$ there is considerable room to exercise ingenuity in lowering entry vehicle weight. The general indication that entry vehicle weight increases with $(L/D)_{max}$, and that at $(L/D)_{max} \approx 3$ is of the order of two times that at $L/D = 0$ for the same mission, seems reasonably valid.

The bottom half of figure 13 gives the ratio of payload weight to total vehicle weight as a function of hypersonic $(L/D)_{max}$. There are a few less data points than in the figure above since some studies did not give a breakdown of weights. The quantitative values are not overly important, since these are dependent on what one defines as payload. In this compilation, payload was assumed to include the crew, portable instruments, etc. - any equipment readily on-and-off loaded with each flight. Such items as on-board life support systems would not be included. The trend of the data is the important feature, in that it reflects the drop in payload efficiency for a given mission as L/D increases.

Following this survey of the literature, an attempt was made to get a more refined picture of the variation of W/W_o with $(L/D)_{max}$. Figures 14 and 15 present the results for the entry velocities indicated and for the particular vehicle types and attitudes sketched. It is recognized that selection of vehicles different from those shown could alter the results, but those selected should be fairly competitive in their respective L/D classes and entry velocities. Information from recent structure and material reviews, the aforementioned systems studies, and the existing manned programs was used in making the weight estimates. Details of the estimates will not be covered herein, but a few comments on volume and equipment weights can be given briefly. Internal volume was taken to be 100 cubic feet per man. The weight of equipment plus crew was assumed to be 2000 pounds for the 1-man-1-day mission, 4200 pounds for the 3-man-1-day mission, and 5200 pounds for the 3-man-14-day mission. These figures include a 500 pound unspecified payload item. Any equipment weight in excess of these figures that is required for a mission is assumed to be handled by appropriate distribution of items between the entry vehicle and a mission module.

Consider first figure 14. The letter A designates that ablation is the prime heat protection approach, R indicates a metallic radiation approach, and $A \sim R$ that the weight ratio is relatively unaffected by the choice of either the ablation or metallic radiation approach. The weight ratios shown are for a 1-day mission; however, other estimates indicated that the picture would be negligibly altered for a 3-day mission. The values of W/W_o cannot be defended any better than indicated by the symbol size; however, the overall results are believed to give a reasonably accurate picture of the relative positions of the different vehicles. Fortunately, the weight ratios are less sensitive to the assumptions involved in the estimates than are the weights themselves.

The results in figure 14 indicate that low to moderate $(L/D)_{max}$ can be achieved without major increase in weight. The overall reduction of the values of weight ratio in going from a 1-man to a 3-man vehicle is, for the most part, simply a reflection of the much larger values of W_o for a 3-man vehicle. However, there is an indication that the more marked increase in weight with $(L/D)_{max}$ that occurs near $(L/D)_{max}$ of 1/2 for the 1-man vehicles is delayed to about $(L/D)_{max}$ of 1 for the 3-man vehicles. This effect is associated primarily with the relation of crew size to the fineness ratio of the crew compartment that can be achieved with a specified volume, and to the flexibility in the choice of crew arrangement that is possible in the design of a multiman vehicle as contrasted with a 1-man vehicle.

Figure 15 presents results for an entry velocity of 46,000 ft/sec. Note that in contrast to the low L/D vehicles selected at the lower velocity in figure 14, the low L/D vehicles selected in figure 15 tend toward conical types with small nose bluntness, since they are believed to be more representative of types suitable for this velocity. All results are for a three-man crew, a 14-day mission, and all vehicles use ablation as the prime heat protection approach. It is doubtful that the estimates are more reliable than the height of the symbol bars. While these results must be considered tentative, they should give a fair picture of the relative weights. For reasons given earlier (decelerations, etc.) it is doubtful that vehicles with $(L/D)_{max}$ less than about 1/2 will be considered for manned entry at this velocity. With this in mind, these results indicate that an $(L/D)_{max}$ of about 1 can be realized without severe increase in weight.

As an adjunct to these examinations of trends in vehicle weight, figure 16 presents results of a literature search conducted with the object of exposing effects of entry velocity and heat protection approach. At the left is shown the ratio of total vehicle weight for a radiating metallic approach to that for an all, or nearly all, ablative approach. As would be expected, the trend in moving toward higher entry speeds, as indicated by the hatched band, is to shift the advantage to the ablation approach. These results are restricted to a maximum longitudinal ranging during entry of about 10,000 miles, which would appear ample for most missions. A much longer ranging requirement would alter the picture toward the favor of the radiating approach, or a combined ablation plus radiation approach. The $(L/D)_{max}$ of the vehicle also has important bearing, with the lower L/D vehicles tending toward the top of the band and the higher L/D vehicles toward the bottom, as roughly indicated by the wavy lines; however, the trend with velocity remains the same. As a general rule, the higher the $(L/D)_{max}$, the higher the entry velocity for which the radiating approach will remain competitive. No data are given beyond 36,000 ft/sec in that no system studies were found that gave serious consideration to other than ablation at hyperbolic velocities. For a multipurpose entry vehicle capable of entry over a wide range of velocities, the results of this survey lend support to the adoption of the ablative approach as the main mode of heat protection.

To the right in figure 16 is presented the results of a compilation directed toward defining the increase in weight associated with increasing entry velocity for a given vehicle class, $(L/D)_{max} \approx 1$ in this case. All vehicles represented here were mainly ablation protected. Again, more detailed system studies are needed, but the indication that the weight penalty for increased velocity potential is within the realm of practical consideration is encouraging, and tends to bear out the view expressed earlier in the discussions on heating.

Relation of $(L/D)_{max}$ to $C_{L_{max}}$ and to L/D at $C_{L_{max}}$

The review of the heating problem has indicated the advantages of entry at $C_{L_{max}}$ and the desirability of realizing a high value of $C_{L_{max}}$. One question that presents itself is: at what value of $(L/D)_{max}$ does further increase in $(L/D)_{max}$ produce little gain in $C_{L_{max}}$? This is, of course, not readily answered since it admits of an infinite number of entry vehicle shapes. An analytical attempt toward an answer was made by applying Newtonian procedures to a variety of body, wing-body, and all wing families. In each family of shapes one parameter was varied so as to cover a range of $(L/D)_{max}$. Only positive C_L and positive angle of attack (in the usual sense) were treated. An envelope was drawn about all the curves thus determined. The resulting Newtonian envelope is shown by the solid curve in figure 17. The peak in $C_{L_{max}}$ around $(L/D)_{max}$ of 3/4 or so reflects the input of several series of half-cone types as indicated by the sketch at top. To illustrate, the calculated curve for a series of pointed half-cones is shown by the dashed curve labeled P (denoting reference to planform area). The curve labeled B is for the same series of pointed half-cones and is included to bring attention to the importance of reference area to the overall results (B denoting reference to base area). The use of base area was found to be generally misleading in a broad comparison of configurations such as attempted here. While planform area was used as the reference area throughout, it is recognized that for blunt, chunky, vehicles that tend to dominate the picture at values of $(L/D)_{max}$ of about 3/4 or less, it is a moot point as to whether planform area or base area is the more logical choice.

A literature survey of experimental studies of entry vehicles at hypersonic speeds yielded the data points shown, where the open symbols indicate results for untrimmed vehicles and the closed symbols represent trimmed vehicles. Bear in mind that wing-body, all wing, and lifting body results are included. At first glance the picture is one of gross scatter, but if the points are ignored that correspond to vehicles whose shape was chosen with little desire for high $C_{L_{max}}$ capability, for example, the elliptic body type sketched at the bottom, most of the open symbols for $(L/D)_{max} \gtrsim 0.5$ and with $C_{L_{max}} \lesssim 0.5$ can be disregarded. The remaining data fairly well bear out the trend of the Newtonian envelope. Thus, the general indication from studies of vehicles made to date is that, for those vehicles having good $C_{L_{max}}$ potential, an increase in $(L/D)_{max}$ capability beyond about 3/4 to 1 cannot be counted on to yield significant gain in $C_{L_{max}}$. This, in turn, infers that beyond $(L/D)_{max}$ of about 1 increase in lift capability (not lift coefficient) comes about almost entirely by increase in planform area or vehicle size, which is generally accompanied by an increase in total weight. (This does not necessarily infer an increase in wing loading.)

It is perhaps worth noting that absolute values of $C_{L_{max}}$ higher than those shown are possible, but either $C_{L_{max}}$ or angle of attack will be negative in the usual sense. As an example, a point at $C_{L_{max}} \approx 1$ is shown which corresponds to a pointed cone with 45° semiapex angle and at large negative angle of attack (indicated by negative sign beside symbol). Aside from the question of practical interest in such a shape at the extreme angle of attack required to realize this $C_{L_{max}}$, the question of whether base area is a more logical reference than planform area for this and similar shapes is raised.

Another question to be considered is: at what value of $(L/D)_{max}$ does further increase in $(L/D)_{max}$ produce little gain in the L/D available at $C_{L_{max}}$? This is important from the view of realizing the advantage to the heating problem from entry at high C_L and at the same time having significant L/D for cross-ranging and maneuver during entry to a preselected landing point. Figure 18 presents the variation of L/D at $C_{L_{max}}$ as a function of $(L/D)_{max}$. The dashed line merely indicates a limiting boundary which is the maximum increase of L/D at $C_{L_{max}}$ with $(L/D)_{max}$, i.e.,

$(L/D)_{max}$ occurring at $C_{L_{max}}$. Two estimates are shown, one using a Newtonian polar, and the other exact two-dimensional pressures at infinite Mach number. Experimental data from the aforementioned literature survey are seen to follow the general pattern of these estimates.

It is evident that there is considerable to gain in L/D at $C_{L_{max}}$ with increasing $(L/D)_{max}$ until $(L/D)_{max}$ of about one is reached. Beyond this there is relatively little to gain by increasing $(L/D)_{max}$. It is worth noting that L/D at $C_{L_{max}}$ cannot be expected to exceed about 0.8, and that the Newtonian polar which was used in the analytical trajectory studies of this paper is indicated by these results to be a reasonable input.

Lateral Ranging

Aside from the factors already discussed, the mission objectives relating to lateral ranging during entry may have overriding influence on the choice of hypersonic $(L/D)_{max}$ for the entry vehicle. No attempt will be made to summarize this subject herein, in that the paper by Baradell and McLellan[29] to be given at this meeting serves this purpose. The paper[29] examines the problem at orbital entry speeds; however, other studies have shown that increased entry velocity introduces no particular difficulty in reaching a prescribed landing point, and generally increases the accessible landing area. Figure 19 is a sample of the information contained in the paper. In this example, the maximum lateral range, and L/D, required to return from a polar orbit to a spot within the continental United States (Alaska excluded) is examined. These results show that an L/D of about 0.7 will guarantee at least once-a-day return to any spot in the continental U.S. (Alaska excluded); an L/D of about 0.9 will assure twice-a-day return to the U.S.; an L/D of about 1 will provide at least twice-a-day return to any spot on earth. Other examinations show that an L/D of about 1 will give at least once-a-day return to the U.S. from an orbit of any inclination that passes over the U.S. (i.e., for the lowly inclined orbits any spot within the southern half of the U.S. would be accessible, and as orbit inclination increases the accessible area increases until the entire U.S. is accessible for orbits inclined greater than about 37°).

In a number of areas already discussed, the results point toward an $L/D \approx 1$ class entry vehicle as one meriting special consideration for a multipurpose vehicle. A more complete examination of the lateral-ranging picture than covered herein indicates that unless frequent or immediate recall is an overriding mission requirement, this same class of entry vehicle has adequate lateral-range capability. For a multipurpose vehicle frequent or immediate recall in the usual sense takes on less importance as entry velocity is increased, since the latter generally infers missions that are of longer duration and more remote from earth. Moreover, as the duration of even near-earth orbital missions is increased to the order of several days or more, immediate recall to a prescribed landing spot becomes more difficult to justify as a normal mode of operation.

Landing

Low-g impact at landing is desirable for a vehicle intended for reuse, such as the multipurpose entry vehicle. Although conventional tangential landing is not necessarily the preferred mode to accomplish this, it is of interest to see what conventional landing requires in the way of subsonic performance. In figure 20 a summary of landing approach criteria (just prior to flare) derived from pilots' evaluations is given in terms of wing loading and subsonic L/D. The results for values of C_L of 0.2 and 0.3 are shown; these values tend to bracket the range of interest. The hatched zones are defined by pilots' evaluations, M designating marginal, P poor, F fair, and G good. One must recognize that everything is relative in this regard, and the boundaries between the zones are fuzzy at best. Specific criteria for acceptability would have the final sayso. Nevertheless, these evaluations do bring out logical trends and should be generally adequate for the purpose of approximating desirable criteria for a multipurpose entry vehicle. For example, if we assume an entry vehicle with a hypersonic L/D of about 1 and a wing loading in the neighborhood of 35 or so, a subsonic L/D of about 4 or more would be a desirable but not an essential goal.

$(L/D)_{max}$ and Volume

The foregoing discussion raises the question: Is conventional landing attainable without undue compromise to useful volume? One aspect of this question is considered in figure 21, where a literature survey of experimental studies of fixed geometry entry vehicles gives a feel for the relation of hypersonic $(L/D)_{max}$ to subsonic $(L/D)_{max}$, and their relation to volumetric efficiency. The experimental points show that the higher the subsonic $(L/D)_{max}$, the greater the possible spread in hypersonic $(L/D)_{max}$. The boundaries denoting values of the ratio of volume to the two-thirds power to planform area were derived from the same source studies as the experimental data. The right-angle form of the boundaries is open to question but seems fairly reasonable. These boundaries are maxima in the sense that a given valued boundary could move down or to the left, but not up or to the right. The overall results show that hypersonic $(L/D)_{max}$ comes at greater expense to volumetric efficiency than does subsonic $(L/D)_{max}$. Of particular interest, in view of the previous results, is the indication that a fixed geometry entry vehicle with a hypersonic $(L/D)_{max}$ of the order of 1 is capable of achieving subsonic $(L/D)_{max}$ well in excess of 4 while retaining relatively good volumetric efficiency. Other examinations indicated that good volume distribution can be realized in a vehicle that has these characteristics.

Concluding Remarks

A review of the problems and environment of manned entry and entry vehicles for entry speeds between 26,000 and 46,000 ft/sec, some of the results of which have been summarized broadly herein, has indicated the following.

A reusable manned entry vehicle suitable for entry at circular to hyperbolic velocities appears practicable without incurring large penalties for aerodynamic performance. In the absence of propulsive braking or the like, lift would be an essential requirement for such a multipurpose vehicle. The general class of vehicle that shows considerable merit has a hypersonic $(L/D)_{max}$ of about one, and many of the arguments favoring this class are negligibly affected by entry velocity within the range considered. Entry at high C_L, coupled with a design that tends toward the realization of $(L/D)_{max}$ near $C_{L_{max}}$, appears desirable from consideration of heating and lateral ranging. The well-known alleviation of the heating problem by entry at high C_L is sizeable over the entire velocity range, and entry at high C_L produces no significant detrimental effect upon peak deceleration or corridor width. With entry at high C_L, indications are that the increase in weight of an ablation-protected vehicle in this class that is brought about by increasing entry velocity is within practical consideration. The recommended heat protection approach is refurbishable ablation; however, the use of a radiating metallic approach over significant surface areas is not ruled out for the highest entry velocity of this study. Use of roll modulation only as the entry mode is attractive from the heating viewpoint; however, it appears that the addition of some restricted pitch modulation capability, with roll modulation as the primary mode, might offer a fuller realization of the aerodynamic performance potential without prohibitive increase in heating. Conventional landing appears within reach of such a vehicle without excessive penalties in total weight or usable volume, and with good volume distribution.

While more detailed systems and configuration studies are and will be needed, particularly as future mission requirements become better defined, this review has indicated the feasibility of the multipurpose entry vehicle to the degree that such a vehicle would appear to merit further study in consideration of future entry vehicle needs.

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USING STATUS OF TODAY'S PROGRAMS AND TECHNOLOGIES AS THE FOCAL POINT, TO REVIEW TRENDS AND REQUIREMENTS FOR NEXT GENERATION SPACE PROGRAMS.

1. WHERE HAVE WE BEEN?
2. WHAT HAVE WE LEARNED?
3. WHERE ARE WE GOING?
4. WHAT WILL WE NEED?

Figure 1.- Meeting theme.

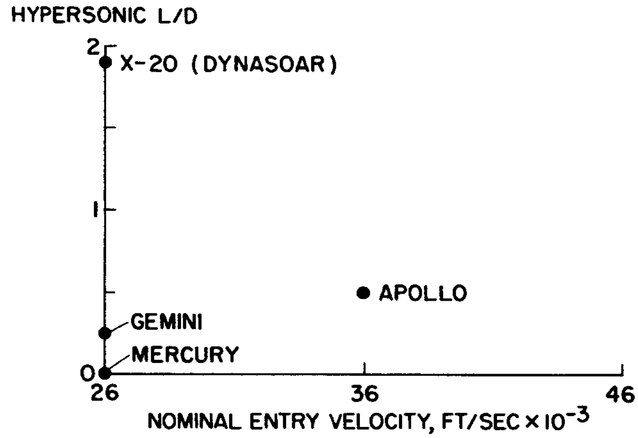

Figure 2.- Current manned entry vehicles.

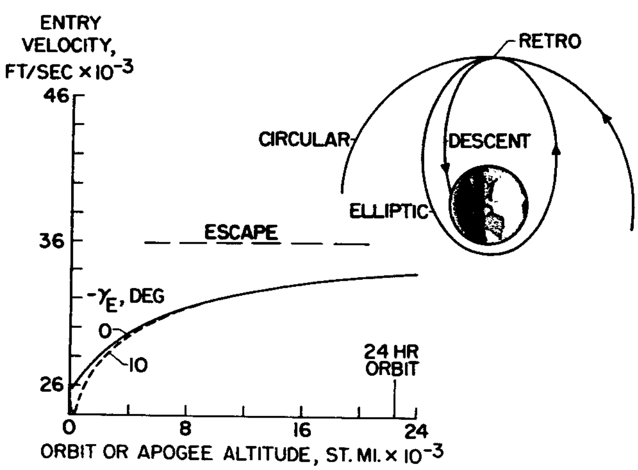

Figure 3.- Effect of orbit or apogee altitude on entry velocity.

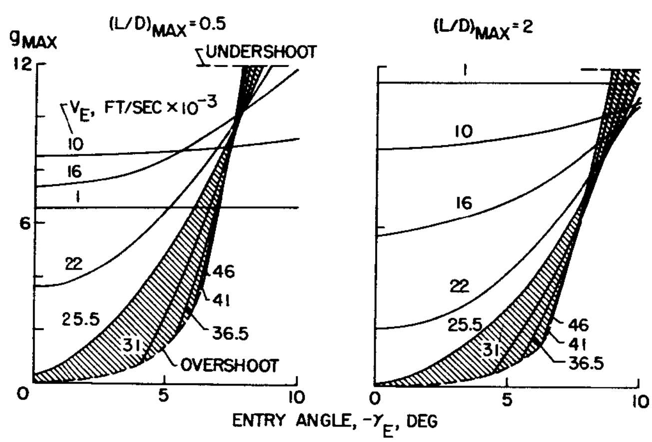

Figure 4.- Maximum deceleration during entry.

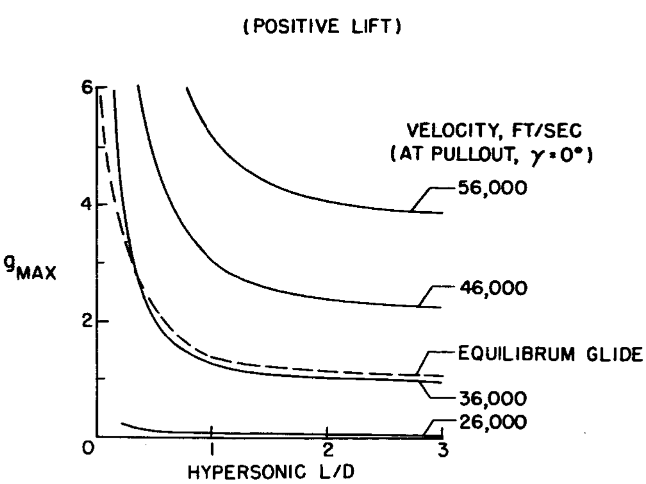

Figure 5.- Effect of L/D on maximum deceleration at overshoot.

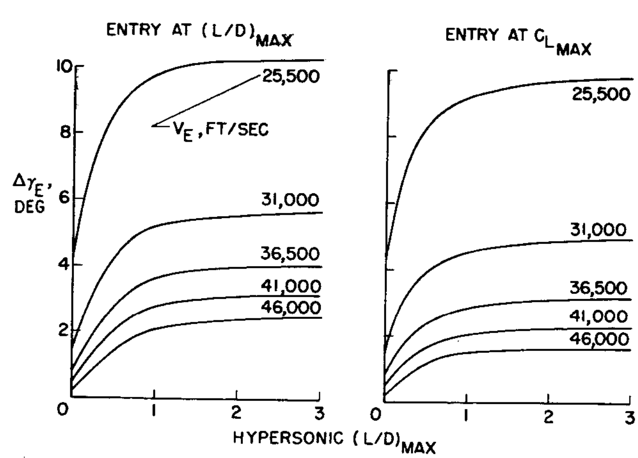

Figure 6.- Width of entry corridor.

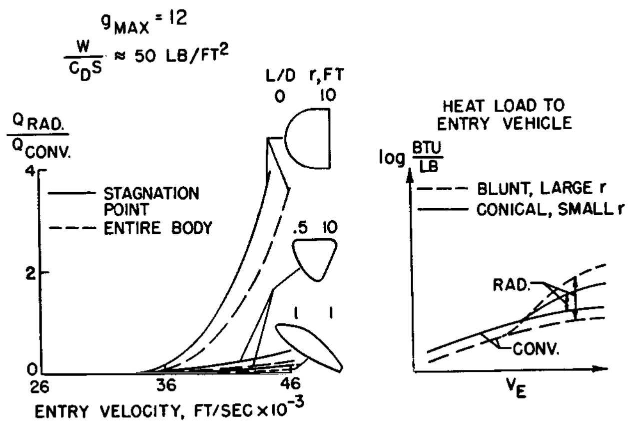

Figure 7.- Role of vehicle type in radiative heating.

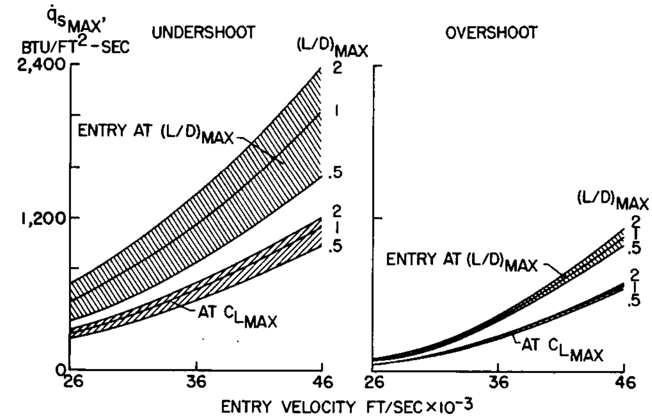

Figure 8.- Maximum stagnation point convective heat rates.

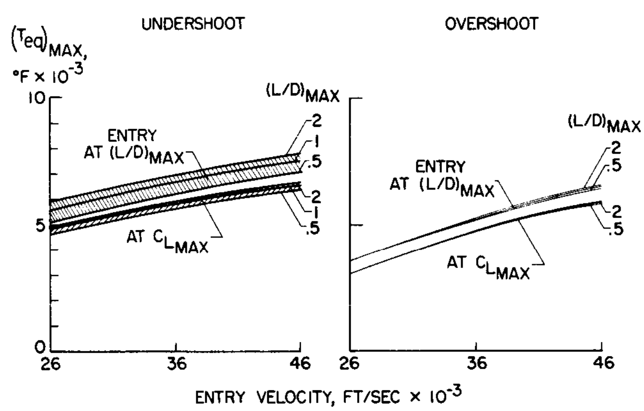

Figure 9.- Maximum stagnation point radiation equilibrium temperatures (ϵ = 0.85).

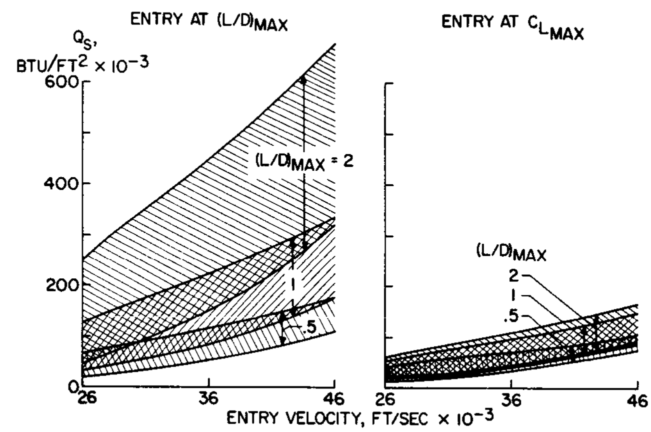

Figure 10.- Stagnation point convective heat loads.

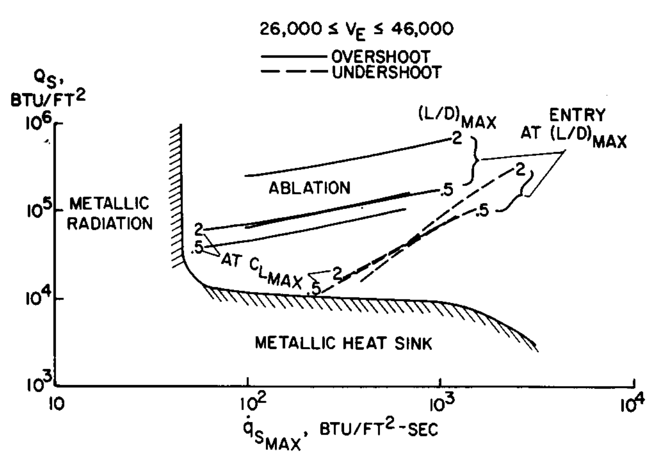

Figure 11.- Stagnation point convective heating in relation to heat protection.

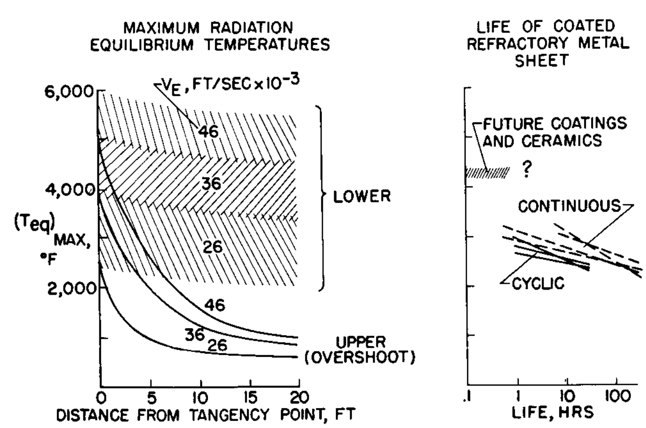

Figure 12.- Surface temperatures and material capability.

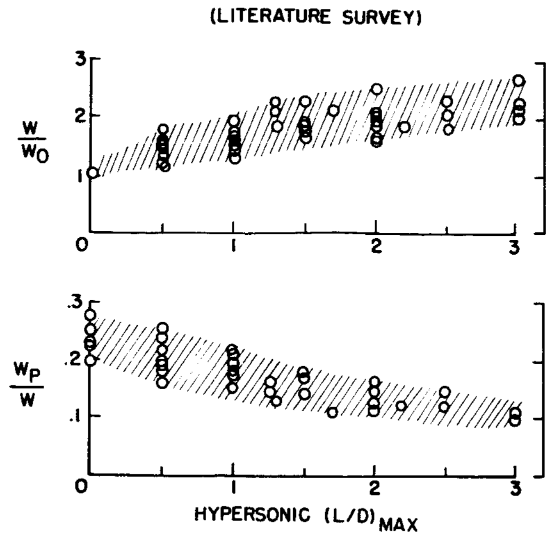

Figure 13.- Effect of maximum hypersonic L/D on entry vehicle weight.

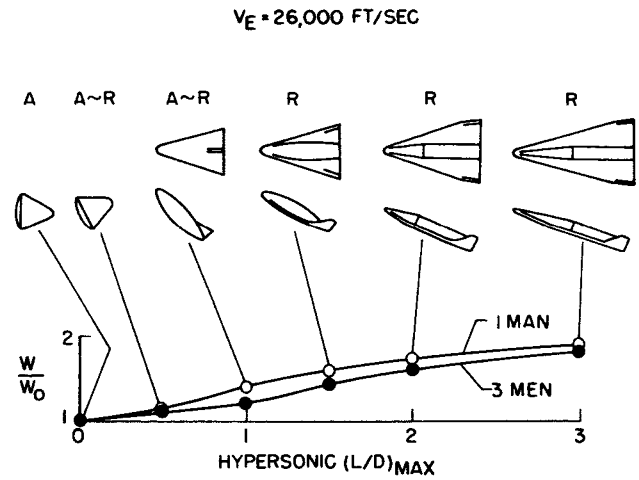

Figure 14.- Effect of maximum hypersonic L/D on entry vehicle weight.

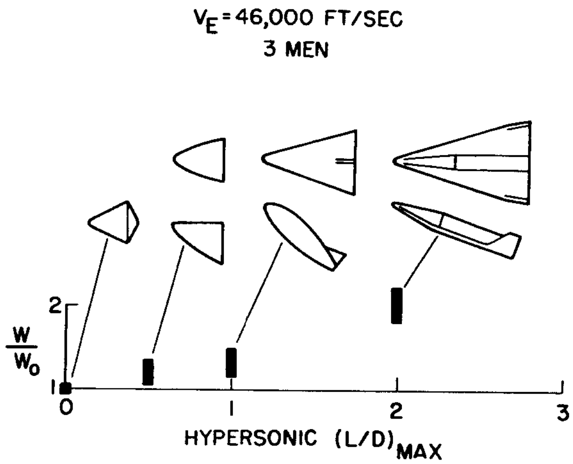

Figure 15.- Effect of maximum hypersonic L/D on entry vehicle weight.

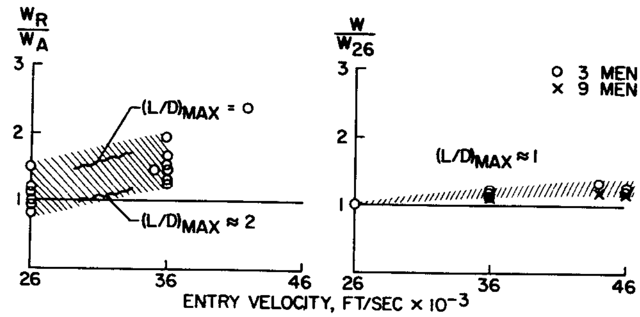

Figure 16.- Effect of entry velocity on total weight of entry vehicle.

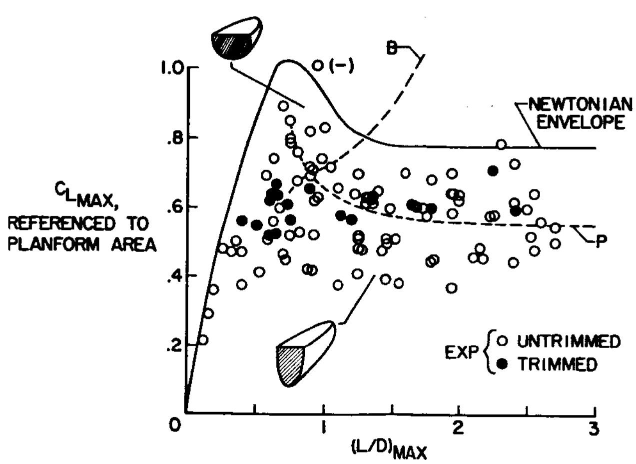

Figure 17.- Effect of $(L/D)_{max}$ on $C_{L_{max}}$ for entry vehicles at hypersonic speeds.

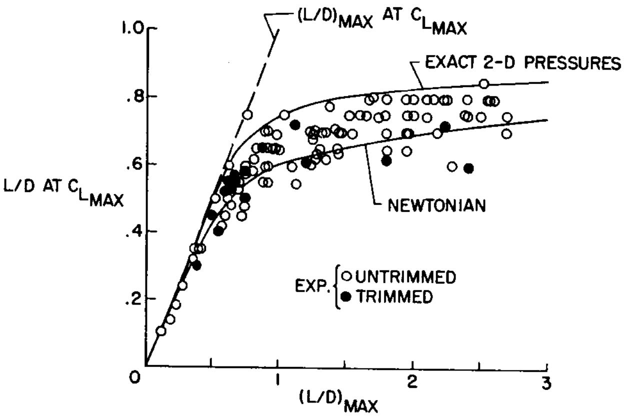

Figure 18.- Effect of $(L/D)_{max}$ on L/D at $C_{L_{max}}$ for entry vehicles at hypersonic speeds.

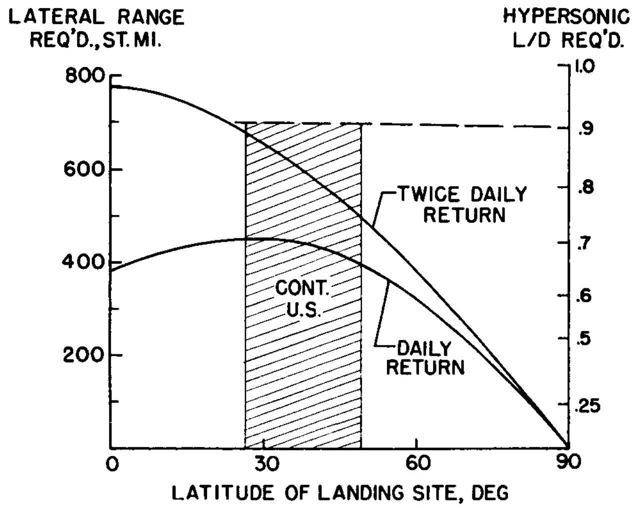

Figure 19.- Maximum lateral range and L/D required for spot return from polar orbit.

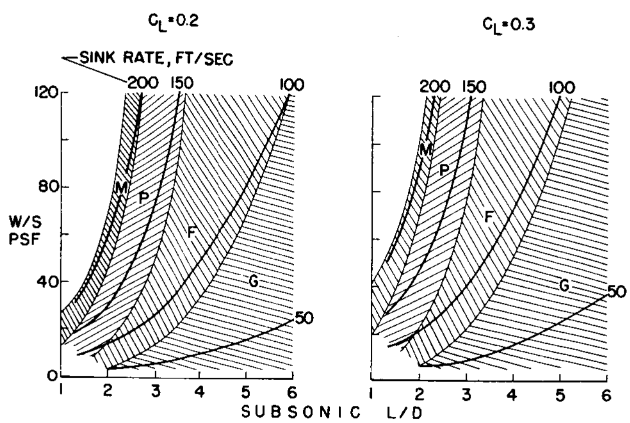

Figure 20.- Landing approach criteria from pilot evaluations.

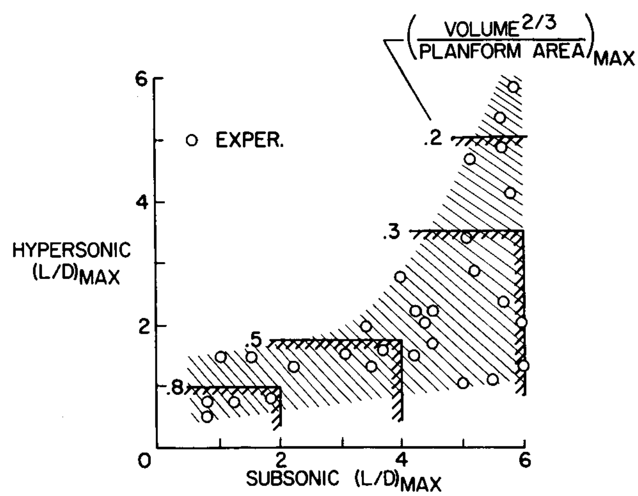

Figure 21.- Relation of hypersonic to subsonic $(L/D)_{max}$ for entry vehicles.

ENGINEERING MANAGEMENT TO ACHIEVE RELIABILITY IN MANNED SPACE SYSTEMS

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Abstract

Von Braun has stated, "Reliability is the result of an almost religious vigilance and attention to detail on the part of every member of a development team." Problems discussed in this paper are (1) how to manage people to attain such vigilance, (2) how to organize reliability knowledge.

When a system is engineered and built by a single, tightly knit team, vigilance can be achieved by inspiring leadership, and reliability experience can be communicated verbally. When developed by several large organizations and hundreds of suppliers, and when subject to a multitude of economic and political restraints, a formal management pattern is required. The author has developed such a pattern through ten years as line manager, management consultant, and Reliability Director of a multi-project aerospace organization. This paper outlines the system of engineering management practice that has been evolved to provide the extreme reliability necessary for manned space systems.

AUTHOR

Reliability Through Discipline

It is an old cliche that failure recurrence cannot be prevented by changing the laws of physics, but only by changing the actions of people. In the manned space program, one of the most urgent requirements is to change the actions of engineering people toward more intensive discipline. Wernher von Braun has expressed this requirement for personal discipline in the following words:

> "Reliability is the result of an almost religious vigilance and attention to detail on the part of every member of a development team, and the most important aspect of every successful reliability program is to keep this vigilance alive."

The need for improved engineering discipline is not limited to the exotic items required by the manned space program. It is serious, even in the long established branches of engineering. For example, in regard to metal forming in support of the nuclear submarine program, Admiral Rickover has published the following statement:

> "The most prevalent inadequacy found in our audits is failure to recognize that timely production of high quality components requires almost infinite capacity for painstaking care and attention to detail by all elements of the organization, both management and non-management; this is as true for a so-called conventional "old-line" product as for a new one."

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Purpose of This Paper

These two quotations are enough to establish that achievement of reliability in manned space flight is a serious problem and that the solution requires strict engineering discipline. In this paper, nothing more will be said about the seriousness of the problem. Rather, an engineering management system will be described. This system provides for development and application of formal discipline, wherever experience has shown that it is needed to help solve the space reliability problem.

To communicate an understanding of this system, it will be necessary:

1. to present a matrix of the activity groups that are critical to reliability,
2. to relate each activity group to the probability of failure, and
3. to discuss program management, contracting, and purchasing in relation to obtaining the required level of discipline.

Project Activity Groups

AUTHOR

All the activities in an engineering program, from conception to mission accomplishment, can be regarded as part of the following four primary functions.

1. Programming
2. Designing
3. Manufacturing
4. Using

Within each of these primary functions there are certain activities which are essential to building reliability into the product and into the mission. We will describe these activities by the term "creation activities".

The "Program Creation" activity group includes all those activities, such as program definition, planning, and funding, that are essential to putting people to work. The "Design Creation" group of activities covers those things that are essential to making design decisions and to creating the drawings and specifications. "Manufacturing Creation" converts the design documentation into hardware. "Use Creation" covers those transportation, storage, maintenance, and operation activities that are essential for achieving the mission. Again, it is the "Creation Activities" that build reliability into a product.

In addition to the "Creation Activities", management has found it necessary to add many types of "Assurance Activities". The assurance activities do not build reliability into a product directly. They do reduce the probability that errors or omissions in the creation activities will escape detection and cause failure. Qualification testing of components is an example of an activity that does not create, but does help assure, reliability.

The difference between a manufacturing creation action and a manufacturing assurance action can be illustrated by a micrometer measurement of the diameter of a steel shaft. When a machinist makes such a measurement to set a lathe tool, it builds reliability into the product. It is therefore part of "manufacturing creation". When an inspector makes an identical measurement, but only to check the work of a machinist, it is part of manufacturing assurance.

Combination of the four primary functions, program, design, manufacture, and use with the two aspects, creation and assurance, results in eight "Project Activity Groups". These groups are illustrated by Figure 1.

Total Engineering Programs

In a total engineering program, project work is preceded by "Development of Resources". Also, the project activities, including the reliability assurance activities, are subjected to an "Operations Evaluation" or audit system.

The "Resources" that must be developed and made available for project application consist of:

1. Procedures
2. People
3. Facilities
4. Suppliers

"Operations Evaluation" of project work must provide, for each activity, answers to the following questions:

1. Are adequate resources available?
2. Are they being properly applied?
3. Are they being effective?

A total program can be defined by combining the four functions of program, design, manufacture, and use with the four aspects of creation, assurance, resources development, and operations evaluation. This combination into a 4 x 4 matrix results in the sixteen activity groups illustrated by Figure 2.

The term "Quality" and "Reliability" have been omitted from the Activity Group titles. These words describe attributes of an equipment that result from adequate discipline in all sixteen activity groups. It is confusing when they are used by themselves to describe one activity group or one organizational unit, rather than a total program. However, they can be used without confusion as adjectives in phrases such as "Design Reliability Assurance Staff" or "Reliability Demonstration Test".

It should be noted that the 4 x 4 matrix of activity groups can be used to describe any engineering program. This is true whether the purpose is to achieve reliability, cost effectiveness, schedule control, or any other management purpose or equipment characteristic.

By our definition, a total reliability program covers all those activities that experience has shown are critical to obtaining reliable product operation. Similarly, a cost control program would cover all those activities that experience had shown were critical to cost control.

It is unfortunate that Government procurement agencies have been developing a series of specifications, each of which requires a management program aimed at a single attribute. For example, MIL-Q-9858 seeks to require a management system that will produce quality and MIL-R-27542 seeks to require a management system that will produce reliability. Other documents have been prepared to require value engineering, materiel management, and other worthy attributes.

It is the author's opinion that a single "Program Management Requirements Specification" should be prepared. Such a specification would require a management system covering the sixteen activity groups shown in Figure 2. It would require enough discipline to ensure that any attribute, such as reliability, quality, or value was achieved to the extent called for by the systems specification.

Critical Activities

To develop effective engineering disciplines, each Activity Group is broken down into more specific "Activities", then each Activity into even more specific "Activity Items", and then each Item into a group of "Checkpoints". For example, in the Boeing Aero-Space Division Reliability Program Plan, the first Activity in the Design Creation Activity Group is "Systems Engineering".

One Item in this activity is "Establishment of an Environmental Stresses Specification". Some of the Checkpoints for this Activity Item are:

1. Manufacturing Stresses
2. Transportation Stresses
3. Storage Stresses
4. Checkout Stresses

The reason that any checkpoint item, such as Manufacturing Stresses, is included in the reliability program is simple. It is, experience has shown, that failure to include such stresses in a system engineering environmental specification has caused failures in previous equipments.

Relation of Primary Functions to Failure Probability

Twelve years ago, Robert Lusser related stress and continuous strength variances to failure probabilities. Shortly afterwards, as a consultant to Redstone Arsenal, I extended Lusser's analysis to include discrete strength discrepancies. Today, we will restate some of this analysis in a manner that will provide a direct relationship between failure probabilities and engineering disciplines in the areas of programming, design, manufacturing, and use.

Figure 3 illustrates the ideal case of a product so manufactured that there is no strength variance, and so applied that there is no stress variance. For any value of "x", Pfx the probability of failure, is equal to Psx the probability that the stress will be greater than "x", multiplied by Pwx the probability that the weakness will be less than "x". Putting this into symbols, we write:

$$Pfx = Psx \times Pwx$$

For all values of x, either Psx or Pwx is zero. Therefore, for the ideal case, the probability of failure is always zero.

The Lusser concept of overlapping stress and strength probabilities is illustrated by Figure 4.

The effects of imperfect discipline in design, manufacture, and use are illustrated by Figure 5. Imperfect design discipline can result in stresses that exceed the limit assumed in a safety margin or failure rate calculation. For example, if an electronic circuit designer does not design against surges incidental to switching, these surges may produce voltages or currents in excess of the design intent. Similarly, imperfect disciplines in use can produce stresses that exceed the design intent. For example, shocks caused by dropping an equipment on a floor may far exceed the design intent. An outstanding example of use stress exceeding design limits was the subjecting of the 720 aircraft over Florida to negative accelerations of almost 4G.

The ways in which deficiencies in manufacturing disciplines reduce the intended strength are innumerable. The most common example is that of imperfect soldering reducing vibration strength of connections.

Figure 5 illustrates that lack of design or use discipline can cause failure of a perfectly manufactured item. Also, lack of manufacturing discipline can cause failure of a perfectly designed and used item. In between there are values of stress and strength for which failure is caused by combinations of design, manufacturing and use discipline deficiencies.

We will now express in symbolic form the difference between creation and assurance activities. We will use the symbol Pdcx to represent the probability that the design creation will produce a stress "x", and the symbol Pdax to express the probability that the design assurance function will allow such a stress to escape through the design review and testing system. Similarly, we will use the symbols Pucx and Puax for the probability of a use stress in excess of the design intent, and the symbols Pmcx and Pmax for the manufacturing strength. We can then write:

$$\underbrace{Pfx}_{\text{failure probability}} = \Big[\underbrace{(Pdcx \times Pdax) + (Pucx \times Puax)}_{\text{design} \qquad \text{use} \atop \text{stress probability}}\Big] \times \underbrace{(Pmcx \times Pmax)}_{\text{manufacture} \atop \text{strength probability}}$$

Relation of Programming to Failure Probability

It would be excessively expensive to apply every available discipline to every item in every program. Consequently, one of the purposes of program planning is to express management decisions on which disciplines will be applied to which items of equipment. These decisions are discussed in the "How Much Discipline" section of this paper.

At the risk of outraging our mathematical colleagues, we will use the symbol Ppc to represent the probability that a discipline will be omitted by the creators of a program plan, and the symbol Ppa to represent the probability that its omission will escape through the program assurance function. We can then write the non-mathematical symbolic expression:

$$\underset{\text{failure probability}}{Pf} = \underset{\substack{\text{program}\\ \text{creation X}\\ \text{assurance}\\ \text{probability}}}{(Ppc \;\; X \;\; Ppa)}$$

$$X \left[\underset{\substack{\text{design}\\ \text{creation X}\\ \text{assurance}\\ \text{probability}}}{(Pdc \;\; X \;\; Pda)} + \underset{\substack{\text{use}\\ \text{creation X}\\ \text{assurance}\\ \text{probability}}}{(Puc \;\; X \;\; Pua)}\right]$$

$$X \;\; \underset{\substack{\text{manufacture}\\ \text{creation X}\\ \text{assurance}\\ \text{probability}}}{(Pmc \;\; X \;\; Pma)}$$

The sole purpose of this symbolic expression is to illustrate that the probability of a particular mode of failure depends on the level of discipline in all eight Project Activity Groups.

Buyer or Seller Discipline

Practically all representatives of industry express opposition to a buyer-seller relationship in which the buyer tells the seller not only WHAT he wants, but HOW it is to be produced. Specifications in which the buyer describes how engineering is to be managed, or manufacturing is to be performed are considered as contrary to the ethics of the free enterprise system and therefore as un-American.

While pressing the case for intensified discipline, this author supports the industry viewpoint. The constraints imposed upon the seller should be to develop and adhere to his own self-imposed disciplines. The seller has a legitimate right to identify those Activities that experience has shown must be disciplined and he must develop the in-house ability to evaluate competitive management proposals. But, the seller should specify neither the method nor the organization required to achieve these disciplines.

In some cases, a seller may choose to impose upon himself disciplines that represent an industry standard. For example, in the case of welding of pressure vessels, a procedure developed through American Welding Society, may be chosen by a supplier as a standard for his own work. Even in the design area, a discipline such as for electronic parts selection, may be developed by an industrial group as an industry standard. However, adoption of such standard should be a voluntary decision made by the supplier.

The use of program plans in support of contracts provides the mechanics to establish necessary disciplines within the free enterprise system. The first step is for the buyer to identify those critical activities for which he requires the seller to have formal disciplines. The second step is to require the seller to define his own discipline for each critical activity. The third step is to negotiate a seller-prepared program plan for applying and funding each discipline. These plans must be flexible enough to permit the seller to improve on them without waiting for buyer approval.

Development of Discipline by Experience Retention

There is a real danger that any license to develop discipline will be abused by subjective prejudices. This danger exists even though the disciplines are developed by the seller's own functional executives. In the Boeing Aero-Space Division Reliability Program there is no place for such abuse. Our only basis for developing discipline is through our Failure Experience Retention System. This system is completely objective because it follows the highly reputable procedure of the "scientific method".

The technique of applying the scientific method to reliability technique has been discussed in several papers by the author. The "OBSERVE" step covers failures that occur in the research laboratory, during manufacturing test or in operational service. The "ANALYZE" step includes analysis of each failure in terms of human actions and what can be done by managers to so discipline these actions that failure will not recur.

The principal "HYPOTHESIS" is that if lessons from failure analyses are fed back into documents and future work is done in accordance with these documents, reliability will be achieved. Of course, it is recognized that documents do not achieve results by themselves. They must be used in the dedicated atmosphere described by the quotation from Dr. von Braun. The documents that we use for reliability experience retention are divided into the following three types:

1. Training and Motivation Texts
2. Directive Documents
3. Operations Evaluation Checklists

The first type provides for teaching and inspiring people, the second for directing them, and the third for objective self-checking and independent checking of compliance with the disciplines required by a project program plan.

The "TEST THE HYPOTHESIS" step in the scientific method consists of contracting for new equipment in a way that will ensure retrieval of the lessons that have been learned and fed back into the failure experience retention documents. It is our practice to develop, for each reliability critical activity, "Procedures" which are suitable for call-out in Project Reliability Program Plans and which require application of the necessary resources in technology, people, facilities and supporting suppliers.

How Much Discipline

The dominant purpose of any Project Reliability Program is to reduce the cost of mission accomplishment. In the case of manned space flight, the term "cost" includes a value placed upon the life of an astronaut and a value placed upon the national prestige that may be lost through unreliability.

The question of how much discipline to require on each project then becomes a systems engineering problem of trade-off between the cost of each discipline and the cost of unreliability that could be caused by omission of the discipline. We have attempted such cost studies at the level of a total system and down to the level of a single component, such as a connector.

For example, consider the discipline of "certification of the workers" who assemble transistors. The cost per transistor for the discipline might be $\$10^{-3}$, the probability of failure might be decreased from 10^{-6} to 10^{-8}, and the cost per failure might be $\$10^{-4}$. The cost per failure prevented then would be $1,000 and the savings per failure prevented would be $10,000. This would indicate that the cost effectiveness of this particular discipline for this particular part had an advantage of ten to one.

In the aerospace industry, we are still chronically short of reliability cost data. We lack data on the cost of applying each discipline, on the probability of failure being caused by its omission, and on the cost of each failure. A concerted national effort is required to get such data. Without it our reliability program management decisions are just educated guesses.

Conclusion

A study of history shows project fail, or even that nations fall, when discipline gives way to anarchy. In the case of aerospace engineering, we have the scientific knowledge to conquer space, and we have a growing body of reliability technology being generated by experience retention, but knowledge without discipline is not enough. To conquer unreliability, the partnership between industry and the National Aeronautics and Space Administration must do these things:

1. Ensure an adequate industry-wide Reliability Resources Development Program based on failure analyses, including failures predicted by theory or produced in research experiments.

2. Develop Contracting and Purchasing Methods that ensure application of these resources in a disciplined manner.

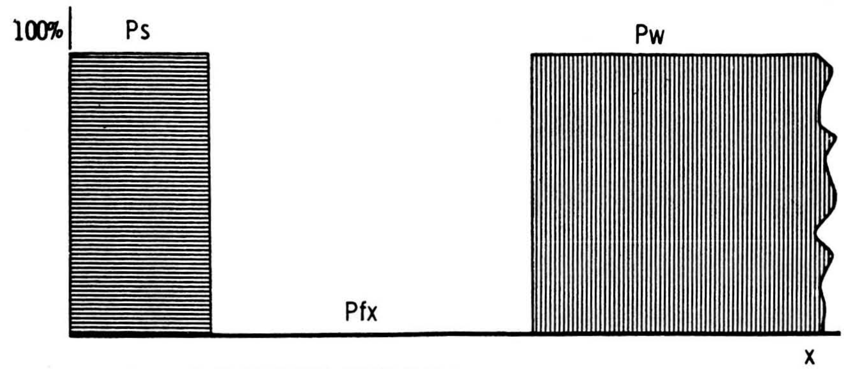
3. Develop Operations Evaluation Checklists and Procedures such that conscientious workers can check themselves and can be checked by others in a systematic, objective manner.

PROJECT ACTIVITY GROUPS			
1. PROGRAM	CREATION	5. PROGRAM	ASSURANCE
2. DESIGN	"	6. DESIGN	"
3. MANUFACTURE	"	7. MANUFACTURE	"
4. USE	"	8. USE	"

Figure 1 PROJECT RELIABILITY PROGRAM ACTIVITY GROUPS

	PROJECT		FUNCTIONAL	
	PRODUCT CREATION	PRODUCT ASSURANCE	RESOURCES DEVELOPMENT	OPERATIONS EVALUATION
PROGRAM	1. PC	5. PA	9. PRD	13. POE
DESIGN	2. DC	6. DA	10. DRD	14. DOE
MANUFACTURE	3. MC	7. MA	11. MRD	15. MOE
USE	4. UC	8. UA	12. URD	16. UOE

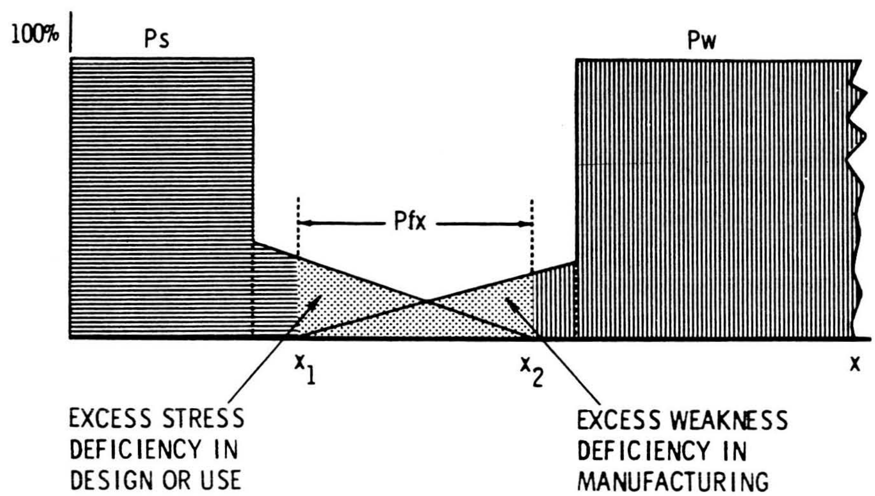
Figure 2 TOTAL RELIABILITY PROGRAM ACTIVITY GROUPS



Pfx = PROBABILITY OF FAILURE
Psx = PROBABILITY OF STRESS LEVEL GREATER THAN x

Pwx = PROBABILITY OF WEAKNESS, STRENGTH LESS THAN x
Pfx = Ps x Pw = 0

Figure 3 IDEAL SAFETY MARGIN



Pfx = Ps x Pw, FINITE IN REGION x_1 TO x_2

Figure 4 IMPERFECT SAFETYMARGIN

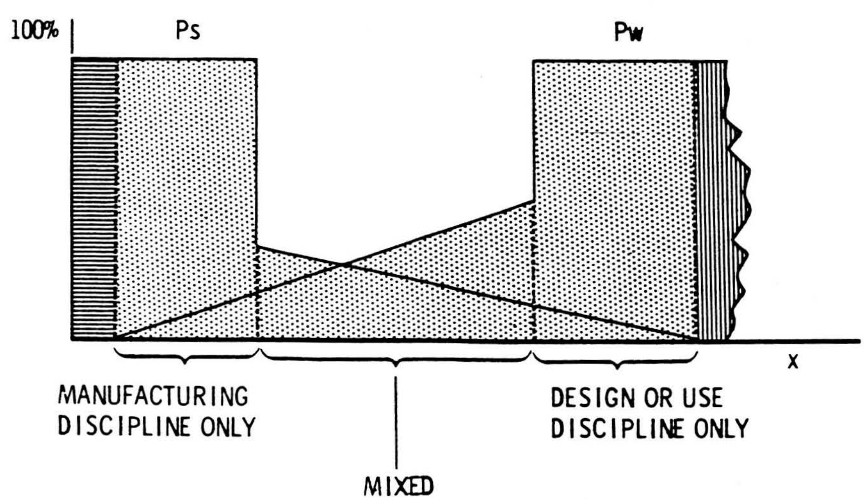


Figure 5 RESPONSIBILITY FOR FAILURE

RADIATION SHIELDING CONSIDERATIONS IN MANNED SPACECRAFT DESIGN

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INTRODUCTION

Space radiations have received considerable attention over the past few years--and have been cited, at various times, as a factor limiting or prohibiting manned space flight, or as being completely insignificant. Depending on the attendant conditions, any of these observations may have been correct.

A space radiation analysis requires (1) a model of the ambient radiation environment, (2) a model of the spacecraft or body which interacts with the environment and (3) the physical relationships which describe the interactions. A number of model environment and spacecraft data will be discussed. However, the purpose here is not to present new data, but rather to indicate the analytical procedures that are necessary to properly consider radiations in spacecraft design.

SPACE RADIATION ENVIRONMENT

A satellite orbiting within the magnetosphere of the earth will encounter protons and electrons moving in an oscillatory north-south motion and drifting in longitude around the earth. These particles comprise the Van Allen belts. The size, shape and variability of these belts have been well treated in the literature. The measurements made to date have shown a region of electrons and protons of the inner belt and a region mostly of electrons in the outer belt.

The number versus energy distribution of inner belt electrons or spectra at low altitudes as reported by members of the Van Allen group agrees well, in form, with estimates based on earlier measurements[1, 2, 3]. This spectrum, extrapolated to the peak flux altitude of the belt, is shown in Fig. 1. Also shown are the spectrum of electrons at the peak of the outer radiation belt* and the fission electron spectrum believed to be representative of the peak of the artificial radiation belt. A flux contour chart of the natural electrons at one longitude is shown in Fig. 2. It should be noted that, due to asymmetries of the magnetic field, this cross section is not constant at all longitudes.

In passing through matter, say the walls of a spacecraft, some of the energy of the electrons goes into the creation of gamma rays from the nucleus of the absorbing material. These bremsstrahlung gamma rays must also be considered as part of the radiation dosages that result from the Van Allen belt electrons. High energy protons of the inner Van Allen belt are another radiation constituent. Their energy spectrum and flux contours are shown in Fig. 3. These data are plotted in the B, L magnetic field coordinate system of McIlwain[4]. The electrons of the artifical radiation belt are also plotted in the same coordinates[5].

*As given by Dr. Van Allen in the ARS Space Flight Report to the nation.

High energy protons emitted during solar flares are another radiation constituent. The intensity, frequency duration and return periods for these events are not yet adequately known. It is known that the course of a flare event is dependent upon the intensity of the flare itself, the preconditioning of interplanetary space by the variable lower energy solar plasma and the geometry between the sun and the place of measurement. The presence of the geomagnetic field also influences the course of the event in the vicinity of the earth.

Therefore, flare events as measured near earth have shown differences in intensity by many orders of magnitude. As a result of the rather complex occurrence model and the poor statistics to date, the procedure for considering flare events has been to select one or more design events, selected from the most severe recorded, to represent the conditions to be expected. Three such design events are 23 February 1956 and two versions of the 10 May 1959 flare. The first derived from Foelsches plot, and the other two derived from Winckler's observations and from a later NASA version[6, 7].

The differential kinetic energy spectra for the events are shown below:

$$dN_1 = 2.563 \times 10^{-1}\, KE^{-1.2985}\, dE;\ 0.60 < E < 130 \text{ Mev}$$

$$dN_2 = 7.859 \times 10^{-1}\, KE^{-1.4460}\, dE;\ 130 < E < 550$$

$$dN_3 = 2.957 \times 10^{3}\, KE^{-2.5520}\, dE;\ 550 < E < 1600$$

$$dN_4 = 6.961 \times 10^{11}\, KE^{-5.040}\, dE;\ 1600 < E < 5000$$

$$dN_5 = 2.802 \times 10^{22}\, KE^{-7.850}\, dE;\ 5000 < E < 10,000$$

$$K = \sum_i \int dN_i = 5.0 \times 10^4 \text{ protons/cm}^2\text{-sec-ster} \quad (1)$$

$$dN = 9.39 \times 10^{9}\, E^{-4.8}\, dE \quad 20 < E < 10,000 \text{ Mev} \quad (2)$$

$$dN_1 = 6.268\, KE^{-2.07}\, dE;\ 5 < E < 60 \text{ Mev}$$

$$dN_2 = 1.3755\, KE^{-3.95}\, dE;\ 60 < E < 10,000 \text{ Mev}$$

$$K = \sum_i \int dN_i = 3.988 \times 10^{10} \text{ protons/cm}^2\text{-sec-ster} \quad (3)$$

Note that the second spectrum is of an instantaneous flux. It was assumed that this flux lasted for about 30 hours. The third spectrum is time integrated, and the first was assumed to be that of the peak flux--which decayed in intensity as t^{-2} (with t in hours).

The next constituent is cosmic radiation, consisting primarily of high energy protons and alpha particles, but also including significant amounts of nuclei of heavier elements. The equation of the cosmic radiation flux is given[8] as

$$N(>E) = 0.3\,(1 + E)^{-1.5};\ 5 \times 10^8 < E \text{ Mev}$$

Our knowledge of the space radiation environment has improved greatly in the last three years. In this time, there have been significant revisions in models of the intensity of the trapped radiation--with correspondingly significant changes in the associated hazard[9]. A number of solar flare proton occurrence models have also been presented over this period--and this is still a major area of uncertainty among different groups.

PRELIMINARY EVALUATION OF HAZARD

General Dosage Plots

Assuming the correctness of the environmental model, it remains to determine the extent to which these radiations must be considered in the design of a spacecraft. To do this, we must first qualify the environmental model with respect to the mission under consideration. To first size the problem, it is helpful to convert the environment from a distribution of particle fluxes to radiation dosages as a function of thicknesses of a standard absorber. We have done this, using a number of IBM programs described elsewhere[10, 11]. Plots of dose versus aluminum absorber obtained with the use of these programs are shown in Figs. 4 through 7.

The effect of orbital parameters on radiation dosages in the Van Allen belts is shown in Fig. 8. The effect of the geomagnetic field in screening out solar flare particles is shown in Fig. 9, for the 10 May 1959 solar flare. The proton energy cutoffs used in the preparation of these data were obtained from the solar plasma model of Obayashi and Hakura, as given in Ref. 7.

Initial Specific Evaluation

We can now make a first evaluation of the radiation hazard, using as an example a two-week equatorial earth orbit at 600-naut mi altitude. Table I shows radiation dosage schedule, within three thicknesses of aluminum, from the various constituents.

TABLE I

Preliminary Estimate of Radiation for Two-Week Equatorial Orbit at 600 Naut Mi

	Aluminum Shield Thickness (gm/cm²)		
Source	2	6	10
Van Allen Belt protons	223.0	119.6	85.4
Secondary neutrons	1.5	3.2	3.6
Van Allen Belt electrons	0.0	0.0	0.0
Secondary X-rays	12.7	8.8	7.6
Artificial belt electrons	31,233.0	0.0	0.0
Secondary X-rays	184.8	159.4	154.0
Cosmic rays	~1.0	~1.0	~1.0
Solar flare protons	0.0	0.0	0.0
	31,656.0	292.0	251.6

From this first analysis, it appears that the space radiation would be a significant factor on this mission. If we assume that our spacecraft structure would be at least 6 gm/cm² (~12 lb/ft²), then electrons would not be much of a hazard. If there were any parts of the craft which might be 2 gm/cm² or less, then electrons would be a very great hazard. We also see that by 6 gm/cm² the X-rays are the largest component of dosage. Furthermore, they have decreased only a few percentiles between 6 and 10 gm/cm², whereas the proton dosages have decreased almost 30%.

It would appear that a higher atomic number material should be used behind the aluminum. This would increase the absorption of X-rays. But an observation of this kind is only qualitative and could be misleading if used for design. The higher atomic number material might increase the dosages, if not preceded by the right amount of aluminum. Generally, an analysis of this type does not yield design information, but gives an indication that there may be a radiation hazard.

If this is indicated, then generally, for any given spacecraft weight, the design which surrounds the crew with the most uniform distribution of spacecraft materials will offer the best radiation protection. Therefore, if at this point in the preliminary design there are some tradeoffs possible between propellants and heat shield--or between a localized or more uniformly distributed heat shield--high mission dosages would favor the distributed heat shield as more efficient than the other two.

Initial Design Dosages

The next step is to determine the radiation dosages within the initial design. For this to be of any additional value over the first estimate, it is necessary to analyze the design in detail as to arrangement and composition of materials. Consider, for example, two adjacent equal-area sections of a spacecraft with net thicknesses of 0.2

and 1.0 gm/cm^2, respectively. The sum of the Van Allen belt proton doses (from Fig. 4) passing through each of these sections is about 17% greater than the dose through the average thickness of 0.6 gm/cm^2. This error depends upon the degree of averaging and upon the steepness of the dose versus absorber curve. It would thus be considerably larger for the May 1959 type of solar flare event than for either the protons of the Van Allen belt or the February 1956 type of solar flare.

In Ref. 10, radiation dosages within an Apollo command module were evaluated. The average vehicle thickness was 13.86 gm/cm^2. Using this value to approximate the shielding effectiveness of the spacecraft gives dosages that are considerably lower than determined using a multilayer, multisection analysis of the spacecraft. These factors are 5.41, 1.2 and 1.25, respectively, for the May flare event, the February flare event and the inner belt protons. Other reasons for providing a detailed geometric analysis of the spacecraft can be seen from Table II, for two early Apollo command modules[12].

TABLE II

Proton Dosages After Flare of 10 May 1959*

	Within L2C Command Module (rad)	Within M-1-1 Command Module (rad)
Neglecting CM equipment	1306	397
Including CM equipment	130	75

*Unprotected man--18,099 rad.

Although there are significant dosage differences between the two designs, the difference in a comparison made without evaluating the effects of inboard equipment would erroneously favor the second design. One should also not attempt to draw conclusions as to the types of design providing better radiation protection. This was shown when an evaluation of another command module, similar in design to the L2C (aft-re-entering cone), gave a radiation dose lower than the first two--namely 51 rad. This was later attributed to a more extensive ablator on the cone walls, together with a number of other factors, none of which pointed in advance to such a large decrease in dosage.

Another justification for the detailed geometric and composition analysis is that, in order for the results of the dosage determinations to provide inputs to design, it is necessary to know where the dosage "hot spots" are located. Therefore, the dosage contribution from each spacecraft region must be available from the calculation in the form of dosage distribution maps.

Radiobiological Factors

Another important factor, when dealing with manned spacecraft dosages, is that the human body is a complex irregular target with varying absorption properties, radio sensitivities and damage tolerances. The dosage values shown in the various figures have been entrance dosages. It is overconservative to consider these as whole body dosages, because this neglects self-shielding within the body--which reduces the dosage as the radiation proceeds through the body. One approach to estimating whole body radiation has been to assume a regular homogeneous shape--such as a water sphere--and determine the dose at a number of depths which can be used to evaluate total absorbed radiation dose.

Aside from the errors introduced by the model, the advantages of using a regular shape are negated by the fact that the body entrance doses coming from the asymmetrical irregular spacecraft are not isotropic. Using a more complex shape for the body model complicates the calculation considerably, but does not offer any great improvement unless the radiation anisotropy is included. However, any improvements in whole body dosage estimates are limited by the usefulness of the whole body dosage itself as the index of radiation damage.

The relative radiosensitivity and tolerance of different regions of the body have been considered in the recent development by NASA of a body model and schedule of allowable dosages to different regions of the model. While this model is not very refined, it represents a significant step toward better evaluation of the actual body damage. The introduction of a finite shape and a number of dosage check points complicates the dosage calculations considerably.

The dosage calculations using the body model may be set up using the body as an extension of the spacecraft, or vice versa--whichever gives the simplest geometry. We have no absolute measure of the geometric detail required, but it would appear that as much as a factor of 5 in improved accuracy would result from a 400 section by 8 layer spacecraft representation versus a simple average. Another factor of 2 error might result from using a single absorber, instead of the actual materials planned for the spacecraft. These calculations for a single combined spacecraft and body model might require four or five hours of IBM 7094 time for each dosage region. No one, as yet, has performed a parametric error analysis to determine the accuracy as a function of analysis detail.

Physical Factors

Implicit in the analysis has been the availability of computational procedures to determine the interaction between the radiation particles and matter--whether this be represented as a uniform spherical absorber or a spacecraft of complex shape and composition. These procedures, or programs developed from the physical equations, comprise the interaction model between the spacecraft and the environment models. The development and scope of the programs used are discussed elsewhere[10, 11]

Suffice it to say that, because of the broad energy distributions of each of the space radiation constituents and the significance of secondary radiations, these computational procedures must be capable of accepting the environment model in great detail and following interactions through a number of very small increments of absorber. As an example, consider the Van Allen belt electrons. If the inner belt were represented as a nonenergetic beam and if the passage through, say, 1 gm/cm^2 were determined--using only one or two depth sections--then the resulting bremsstrahlung dosage estimate might not be more accurate than a factor of ±20.

Operational Factors

The preliminary analysis may show that the hazard may be considerably reduced or eliminated by a modification of one or more of the operational factors. These include trajectory, mission duration and date. An obvious modification to reduce the dosages shown in Table I would be to lower the orbital altitude. Choosing an orbital altitude of 400 miles would reduce the dosages by a factor of almost 100, as can be seen in Fig. 8.

To substantiate this effect, it is necessary to calculate the dosage, using the complete mission time and the proper orbital elements. It was mentioned that the Van Allen belts are irregular in shape. The effect of the irregularities is that the trapped radiation environment is not uniform in any one orbit and may also vary considerably among successive orbits.

For an extreme example, take a 100-naut mi orbit injected at 0° latitude and 0° longitude, at an inclination of 40°. We found that no Van Allen belt protons were encountered until the middle of the seventh orbit. Then, in an interval of 2.7 minutes (northeast of Madagascar) about 17.5% of the total 12-hour orbital dose would be received. The remainder of the dose would be received over a period of 8.4 minutes, as the satellite passed over Mozambique and Madagascar. The significance of the injection point and--in turn-- the launch site is also apparent from this example.

In preparing the data used in Fig. 8, an attempt was made to smooth out these irregularities by considering 12 hours as the unit of exposure time. This approximation would be poor for the case just discussed, since each of the next four or five orbits would probably encounter significantly higher particle fluxes as the orbit passed through the South Atlantic anomaly. Unless the complete mission is used in the calculation, it is doubtful whether the radiation flux for the mission can be obtained more accurately than within a factor of two or three.

For the radiation constituents trapped in the geomagnetic field, these calculations would probably need about one hour of 7094 running time for every three days in orbit. It is desirable to preserve the dose (or flux) history in the calculations, so that the time and space locations of the high radiation regions will be known. In the case just mentioned, the short exposure periods might suggest the use of a garment-like radiation shield to be inflated with water during intervals within the belt and later drained back into a reservoir. Small orbital maneuvers to avoid the edge of the belt might also be considered.

Cosmic radiation intensity and solar flare frequency vary inversely and directly, respectively, with the solar activity cycle. If the results of the preliminary radiation analysis showed these to be significant contributors to the mission dosage, then changes in the launch date could be examined to determine whether this is a practical means of reducing the hazard. On a Mars or Venus mission, the total dose from cosmic rays could become comparable to the solar flare dose, as shown in Fig. 10. There might be little value in rescheduling the flight to a year of less solar activity, because the increase in dosage from cosmic radiation might more than offset the decrease in dosage from flare particles. But this cannot be determined without additional analysis.

PRELIMINARY DESIGN EFFORT

Following these preliminary analyses, "firmer" values of configuration and/or mission will evolve. A more detailed radiation analysis can then be performed and fed back into the design groups. At this point, some items may be "frozen"--for example, the mission profile--but some gross features of the configuration may still be modified. The more detailed analysis, which provides dosage distribution maps, will be useful in formulating these modifications.

We recently completed a preliminary design analysis for a LEM spacecraft, using the body model mentioned previously. Figure 11 shows the patterns used in this analysis. The results were as follows.

The average radiation dose from the NASA model of the May 1959 flare inside the LEM (preliminary design configuration) was 7456 rad.

The maximum eye dose was 2020 rad.

The maximum blood-forming organ dose was 133 rad.

The maximum skin dose was 2814 rad.

The maximum extremity dose was 9900 rad.

Note the nonuniformity of the dose and the fact that the maximum extremity dose was larger than the average entrance dose in the crew compartment.

Figure 12 shows the shield weights versus the probability of not exceeding the maximum allowable dosages for this spacecraft. The probabilities were derived from the environmental occurrence model. Shielding of each other by the two crewmen and shadow shielding by the moon were also considered in this analysis. Associated with this figure would be a set of dosage distribution charts, which locate the dosage "hot spots."

DESIGN EFFORT

Figure 12 and its supporting data are the results of analyses performed during the preliminary design phase. Figures 13a and 13b represent an attempt to trace out the procedure during the main engineering design effort which would follow. This was done for the LEM mission, starting from the point where a preliminary configuration has been evolved and using defined values of mission and payload. This may be considered as the beginning of the design effort.

The radiation analysis effort has three major inputs, as shown. Each of these requires improved or more detailed values as the design effort proceeds. Input data format is indicated next, leading into preprocessing of the input for the dosage calculations which utilize the environmental, the configuration and the body models--and the radiation interaction models. The adequacy of our existing input preprocessing techniques is shown by the shaded box code.

To support the dosage calculations, an experimental check on the accuracy of the interaction model is available from irradiated sample test panels. If the dosage allowables are not exceeded, the next step is an experimental verification, using more detailed spacecraft and body model sections. If the experimental checks do not support the analysis, then modifications to the spacecraft, body or interaction models are indicated in terms of the two-way connection and the analysis repeated with indicated modifications.

If the allowables are exceeded and the experimental check is satisfactory, then a dosage alleviation study would be made. This major substep is shown in Fig. 13b. Three categories of alleviation techniques are shown, together with examples of each. These techniques are then evaluated, considering dosage alleviation relative to penalties that may be imposed elsewhere in the system, as indicated.

Some of the techniques will be found impractical, or inefficient. One or more may offer considerable promise. The evaluation could require cooperation with many--if not all--of the project technical areas, with the government and with the command module contractor. For example, a possible alleviation technique may be removal of part of the command module shielding by the crewmen for use during the LEM portion of the mission.

Following the alleviation studies, the recommended technique (or techniques) can be implemented and the analysis procedure continued until the goal is reached.

SUMMARY

From the foregoing discussion, it can be seen that consideration of radiation shielding requirements is achieved "bootstrap" fashion in a series of feedback analyses. Inputs from various technical specialty areas, such as flight mechanics, structures and configuration design, are used to help define the radiation hazard. The radiation analysis, in turn, furnishes data to these specialty areas--and these are used to make modifications which will reduce the radiation dosages without adverse functional or operational effects.

Even a small amount of radiation shielding saved by the relocation of an already existing component is well worth the effort. In this regard, all the components and materials of the spacecraft may be thought of as part of the radiation shield "subsystem." This subsystem is then designed, along with the other subsystems, so that specific shielding mass is kept to a minimum. Furthermore, if shielding is required, then proper provision for its storage or mounting can be made.

This all implies that a high degree of systems integration capability must be available at the beginning of the preliminary design effort and that those performing that function need to recognize some of the problems and procedures of space radiation considerations.

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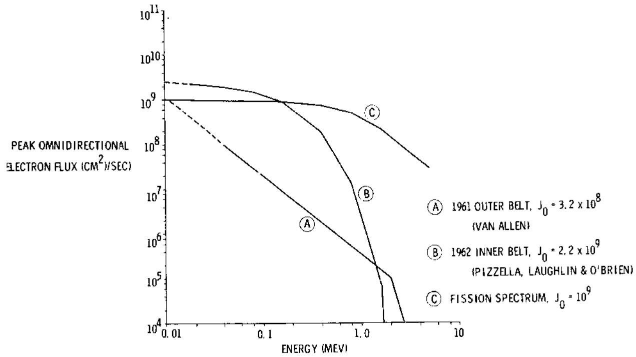

FIG. 1. INTEGRAL ELECTRON KINETIC ENERGY SPECTRA

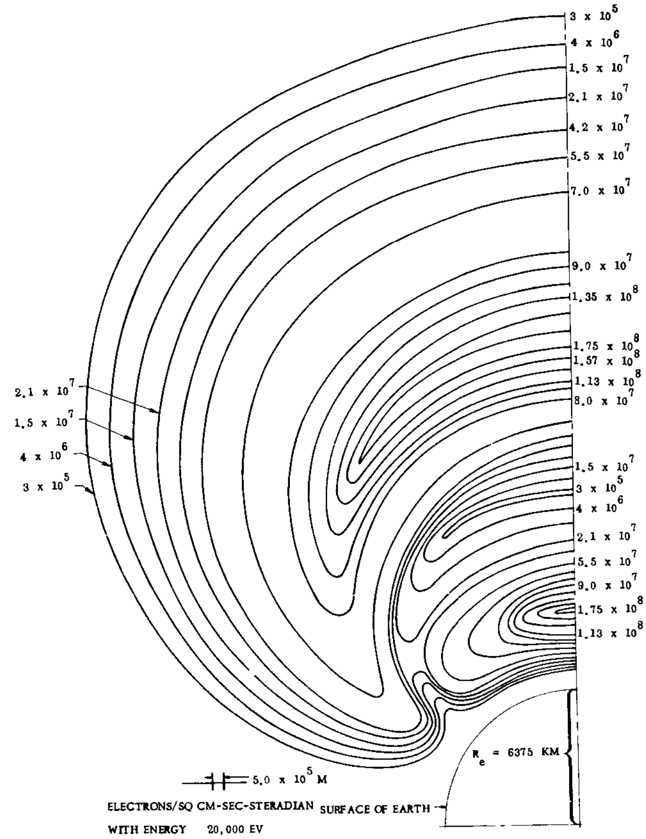

FIG. 2. FLUX OF ELECTRONS AT ONE LONGITUDE IN THE VAN ALLEN BELTS

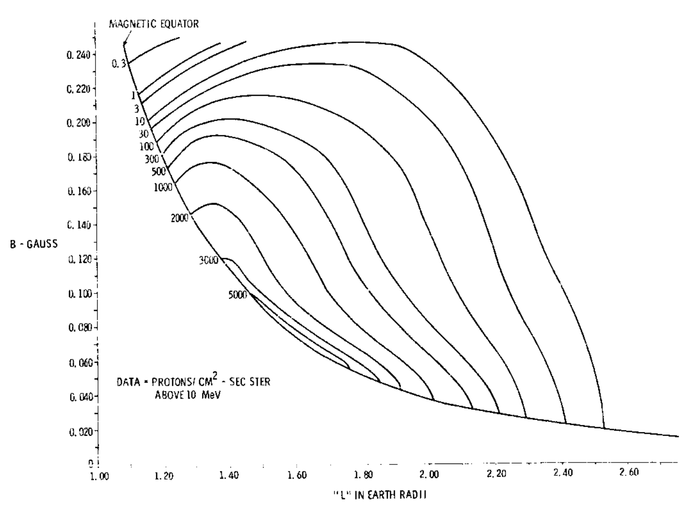

FIG. 3. VAN ALLEN BELT PROTONS IN B, L COORDINATES

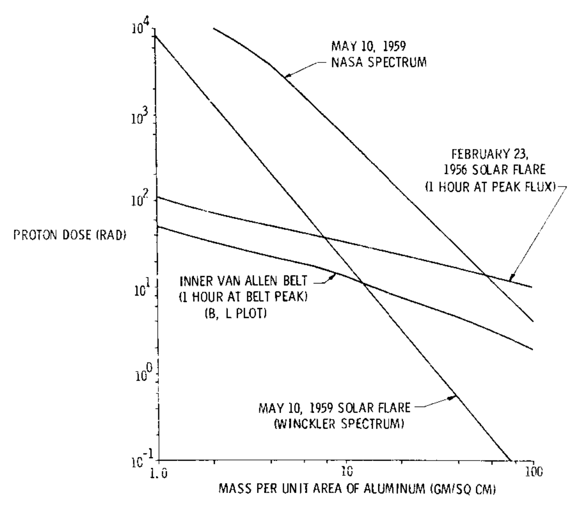

FIG. 4. DOSE-ABSORBER RELATIONSHIPS FOR PROTONS IN SPACE

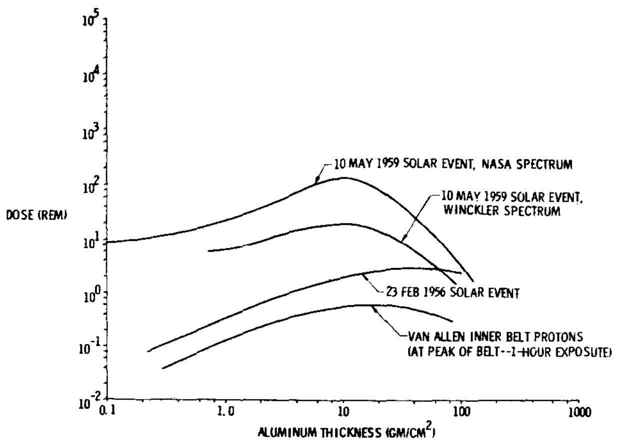

Fig. 5. Neutron Secondaries from Space Radiation Protons

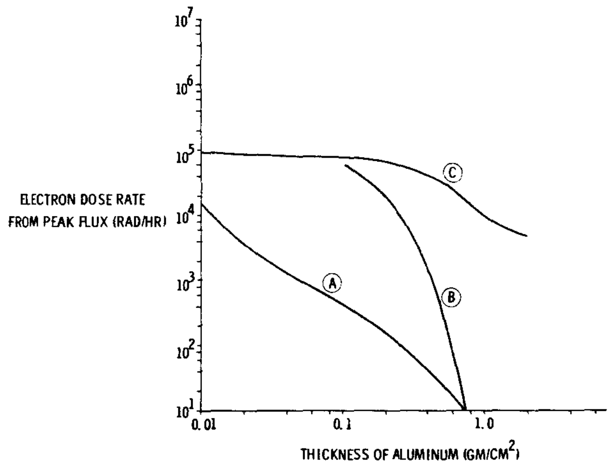

Fig. 6. Electron Dose Rates

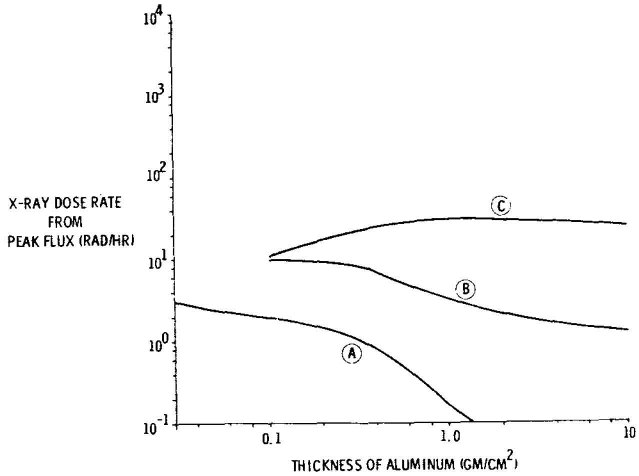

Fig. 7. X-Ray Dose Rates

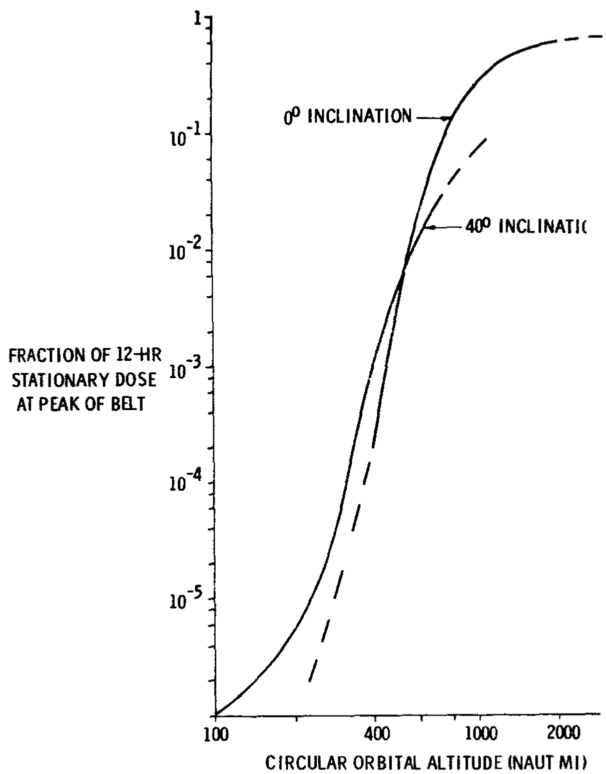

Fig. 8. Variation of Geomagnetically Trapped Radiation Dose with Orbital Altitude and Inclination

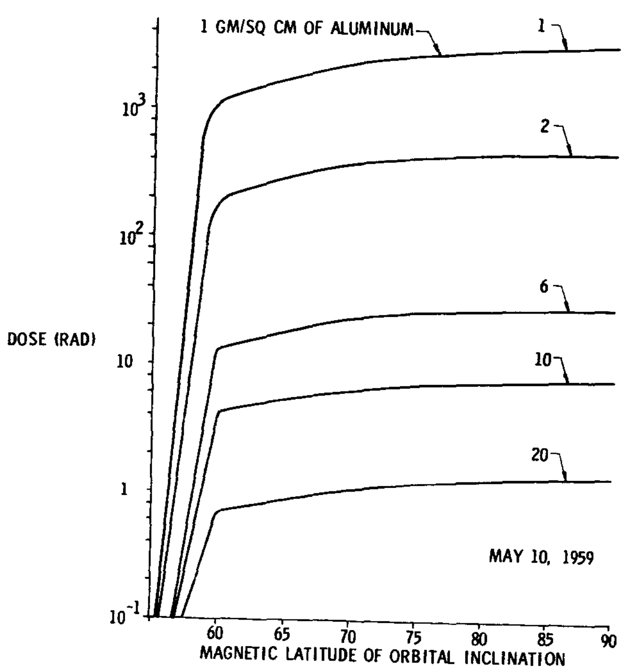

FIG. 9. SOLAR PROTON DOSAGES AS A FUNCTION OF ORBITAL INCLINATION AND ABSORBER THICKNESS

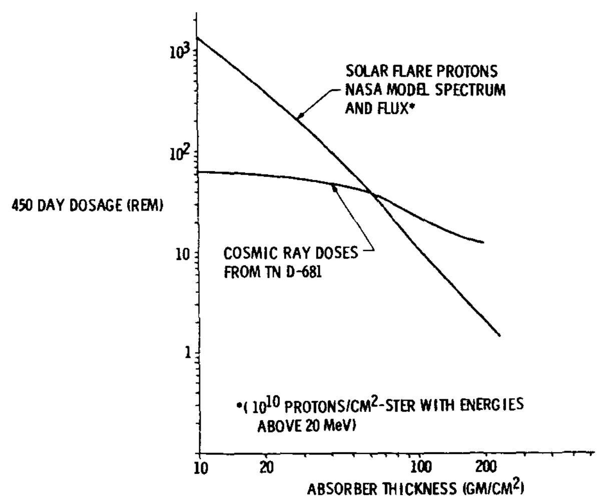

FIG. 10. 450 DAY MARS MISSION DOSAGES

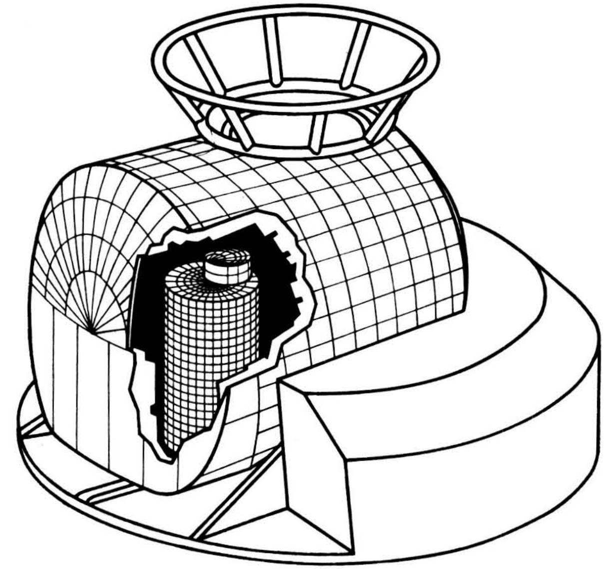

FIG. 11. SHIELDING CALCULATION PATTERNS

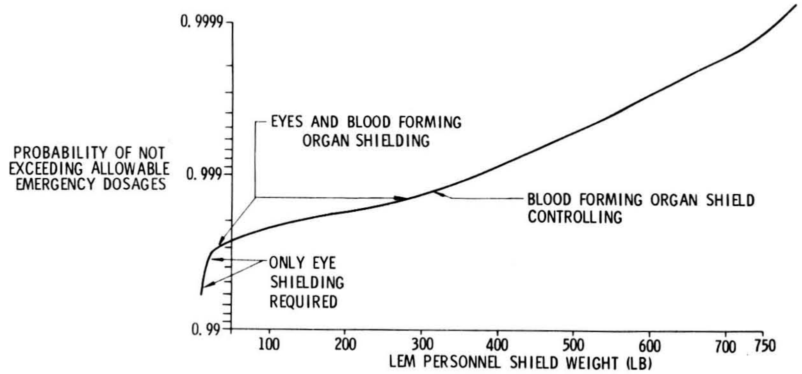

FIG. 12. PRELIMINARY EVALUATION OF LEM SHIELDING REQUIREMENTS

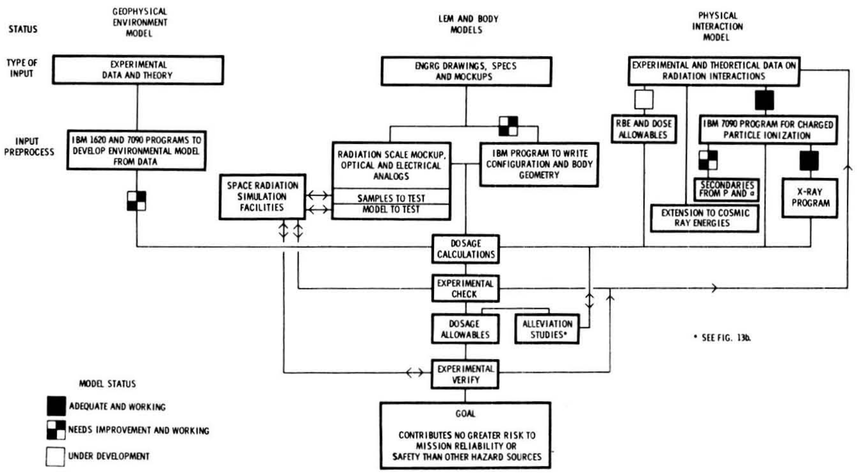

FIG. 13a. RADIATION ANALYSIS PROCEDURE

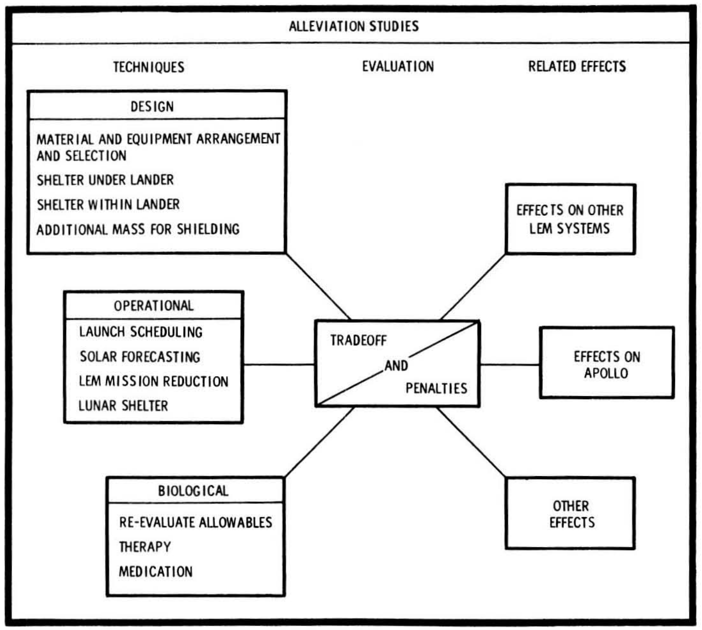

FIG. 13b. RADIATION ANYLYSIS PROCEDURE (CONTINUED)

N63-23226

FLIGHT SAFETY ASPECTS OF MANNED SPACE FLIGHT

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INTRODUCTION

During the past year, about 20 hours of flight experience in manned space operations was accumulated in Project Mercury. Although the actual flight time was quite short, the preflight preparations were extensive and considerable experience was acquired in the development and practical application of operational procedures intended to reduce the hazards of these flights.

The flights themselves also showed up several areas where the unique character of spacecraft operations requires special protective measures in the design of spacecraft systems and equipment.

The purpose of this paper is to describe some of the operational practices which have proven effective, and to call attention to design areas where special measures seem indicated to reduce hazards peculiar to space operations.

FLIGHT SAFETY PROCEDURES

The operational procedures that have proven valuable in efforts to augment flight safety are listed below:

Development Engineering Inspection, Factory Rollout Inspections, interface controls, Flight Safety Reviews, and environmental and operational mission simulations.

DEVELOPMENT ENGINEERING INSPECTIONS

The first problem encountered in trying to operate a spacecraft safely is the problem of getting it designed and built so that it can be operated safely. Spacecraft equipment must not only operate, it must also be checked to verify that is is operating or ready to operate. Systems must be planned from the start in order that the necessary checks can be made without incurring the hazards associated with breaking into plumbing and electrical circuits. Equipment requiring frequent replacement or servicing must be located and designed so that these operations can be performed without disconnecting or damaging other equipment. Electrical disconnects, particularly, must be in sight and within easy reach if the very serious hazards associated with bent or broken pins are to be avoided.

Because of the severe space and weight restrictions and the limited experience available at the time the design was laid down, the Mercury spacecraft has presented very serious problems in these areas. Successful operation has been achieved only at the expense of the most painstaking and time-consuming effort in reverification of disrupted circuits.

The procedure adopted in the Mercury program in an effort to inject some of these requirements derived from operational experience into the spacecraft hardware took the form of a series of Development Engineering Inspections conducted on each spacecraft as it neared completion by an inspection team made up largely of operational personnel and an Inspection Board chaired by the Operations Director. This approach proved quite effective in accomplishing such improvements as could be retrofitted or incorporated into the fabrication of subsequent articles, but it could not, of course, cure difficulties frozen into the basic configuration.

For programs beyond Mercury, requirements for checking and servicing equipment have been given greater consideration in the initial spacecraft layouts. In an effort to get the necessary details engineered into the hardware at an earlier stage than was done in the Mercury programs, the Development Engineering Inspection program is being extended by holding additional inspections, with somewhat smaller inspection teams, on each of the major subsystems during the subsystem development and qualification program.

FACTORY ROLLOUT INSPECTIONS

The next problem to be faced is the problem of assuring that the spacecraft, when it leaves the factory, is complete, contains only fully qualified parts, has been thoroughly checked out, and that all peculiarities observed at any time during the assembly and checkout process have been reported to the operational crew.

One solution to this problem that has demonstrated its effectiveness in the Mercury program has been the procedure set up by the Air Force and Aerospace Corp. for Factory Rollout Inspections on the Mercury-Atlas launch vehicles. The key feature of this program was the establishment of permanent technical teams of specialists assigned to each major subsystem of the launch vehicle. These teams reviewed in detail the status of each subsystem and its performance in the integrated systems tests at the factory prior to acceptance of the vehicle for shipment.

This concept of a Factory Rollout Inspection for the spacecraft as well as for the launch vehicle has been adopted in other MSC programs.

It has also been found desirable that inspections required during final assembly and test of the spacecraft at the factory be performed by inspectors from the operational group, and that engineering personnel from this group participate in the integrated systems and final factory acceptance tests of the completed spacecraft. This procedure is helpful in minimizing the amount of teardown and reinspection required at the launch site. More important from the standpoint of safety, this procedure acquaints the operational group with any possible symptoms of trouble that may assume significance during subsequent checkout tests at the launch site.

INTERFACE CONTROL

Another important requirement that became evident early in Mercury operations is the need for special measures to maintain effective control over the interfaces between major systems of the space vehicle. These systems are designed and built under the cognizance of different engineering teams in different parts of the country. Coordination as to details of interface equipment location or relocation is difficult. The structures involved are large and flexible. They are subjected to large loads and experience surprisingly large distortions, particularly during the atmospheric phase of flight. Under such conditions, inadequate physical clearances between various pieces of equipment and structure in the interface are an ever-present risk.

Any debris left in the interface can also become a hazard if it can wedge between tank domes and adapter walls in such a way as to produce local stress concentrations.

Electrical system changes are rather frequently required in the launch vehicle or spacecraft. Each proposed change in any system has to be examined for possible effects on other systems through the interface wiring connections. The risk that incorrect or obsolete wiring diagrams may be used for this purpose cannot be ignored.

To reduce these hazards in Mercury, it was necessary to create an official Interface Committee, reporting independently to the Operations Director. This Interface Committee was made up of engineers and inspectors selected from each of the organizations involved in the preparation of the space vehicle for flight. The committee was assigned responsibility for witnessing all launch site activities involving the interface and to verify that correct physical clearances existed, that equipment was properly secured, that the interface area was clear of debris, and that official wiring diagrams showed the exact configuration of the electrical circuits actually existing on the space vehicle on specified dates.

Although the Interface Committee approach has been effective in preventing difficulties in Mercury flights, additional measures are being taken in later programs in an effort to prevent some of the problems from getting as far as the final assembly of the vehicle at the launch site. No matter how effective these measures become, however, it is still not possible to eliminate the requirement for a final verification of the condition of the interface late in the launch preparations. In all probability the need will grow since future vehicles carry hypergolic liquid reaction-control fuels in this area. It is necessary to continue to insist, therefore, that adequate access and procedures for interface inspection are a safety requirement in all future space vehicles.

FLIGHT SAFETY REVIEWS

Preparations for manned space flight involve the coordinated effort of many people working in different locations and responsible for different systems and subsystems of the space vehicle. Changes to the hardware and changes in plans are frequent and unavoidable. Under these conditions there are substantial risks that mistakes will be made, that important symptoms of trouble will be unrecognized and unreported, or that someone somewhere will not get word of a supposedly innocuous change that is of vital significance to him.

The approach taken in the Mercury program to minimize these risks has been a particularly comprehensive series of Flight Safety Reviews conducted on each mission in the week preceding the launch. These reviews are conducted by a NASA Flight Safety Board which includes, among others, the Operations Director and the astronaut making the flight. Separate review meetings are held on the launch vehicle, the spacecraft, and finally the complete mission. The Air Force, in addition, holds its own in-house reviews of the launch vehicle.

The object of these meetings is to review with the engineers and inspectors directly responsible for the checkout of each space vehicle subsystem all information that exists on the prior history and current status of that subsystem. The discussions cover in detail all difficulties observed in checkout and all changes made as a result of these difficulties or as a result of the analysis of data from previous flights and ground test programs. The discussions also include verification of the qualification status of all equipment and of the useful life remaining in limited-life items. In the case of the launch vehicle they include consideration of any pertinent difficulties in the other programs where it is used.

The meetings are conducted in as an informal and leisurely manner as possible to encourage full and frank discussion of every potential trouble area. The basic philosophy governing the discussions is that a launch must not be undertaken as long as any observed difficulty remains unexplained or uncorrected.

These review meetings have been quite effective in concentrating attention on the detailed engineering problems of each vehicle. At the time of launch there has been no doubt in the mind of anyone involved as to the prior history and flight readiness status of all systems.

MISSION SIMULATIONS ON ENVIRONMENTAL SPACECRAFT

In all MSC programs an attempt is made to achieve the earliest possible flight date with hardware that advances the state-of-the-art. The problem is to decide just when each mission can be undertaken with an acceptable level of safety. The approach that MSC has taken has been to design our spacecraft systems with sufficient redundancy to absorb the expected number of random failures of parts without serious consequences, and to carry out a ground and unmanned flight-test program that will assure detection and correction of all "built-in" or "early development" sources of system failure. It can never be completely proved, however, that the test program has uncovered the last of these sources of failure. Some risk will always remain that a mode of failure associated with prolonged operation will not have been revealed.

To minimize this type of risk, it is important that selected phases of the ground test program be continued throughout the entire manned flight program.

One phase of the Mercury ground test program that has proven particularly valuable in disclosing problem areas of this sort has been a program in which the complete spacecraft has been subjected to a series of simulated flights in a simulated mission environment. In several instances, this program called attention to effects of the vacuum and thermal environment on spacecraft components that could have caused considerable difficulty in flight. This program has also been extremely useful in verifying the effectiveness and safety of the many changes that have been made to the spacecraft for various reasons.

A program of this type, run continuously throughout the life of the flight program, now appears to be an essential safety feature of any future manned space program.

MISSION SIMULATIONS DURING PRELAUNCH PREPARATIONS

A final feature of the Mercury program that deserves mention in connection with flight safety is the attention given during the prelaunch preparations to rehearsals and simulations designed to increase the proficiency of the pilot and ground controller team.

The most significant finding in the Mercury program to date has been the convincing evidence that the pilot can function effectively in the space environment. This demonstration that the space environment does not degrade pilot proficiency is extremely important to the safety of space flight operations. As a result of this finding, the flight procedures may be planned to take full advantage of the pilot's well-known capability for detection of rapidly developing malfunctions and for shutdown and switchover to backup systems or to alternate modes of operation. Many system malfunctions develop more gradually, however, and require study and analysis by system specialists on the ground before intelligent corrective action can be taken. Maximum safety in space flight can only be achieved by coordinated handling of problems by the pilot and the ground monitoring team.

To develop maximum proficiency in this joint effort, Mercury preflight preparations have included a comprehensive series of simulated flights with special simulation equipment and with the actual flight vehicle during the period just preceding each space mission. These simulations were made extremely realistic. They not only served to verify the feasibility of planned procedures and provide crew practice for the expected flight plan, but also included a wide range of emergencies deliberately introduced to show up areas where improved planning might be needed to eliminate all possibility of confusion or indecision. Preflight mission simulations of this type will continue to be an essential safety requirement for any future space mission.

FLIGHT SAFETY PROBLEM AREAS

Up to this point, a few of the preflight operational procedures used in the Mercury program that appear to have special merit in increasing the safety of space operations have been discussed. The Mercury flights also disclosed several areas where special attention to detail design seems indicated to cope with some of the unique characteristics of space flight. The most important of these areas are listed below:

Zero-g environment, structural deformation, heat removal and equipment temperature control, and fuel management.

ZERO-G ENVIRONMENT

Probably the most insidious hazard to reliability associated with the space environment is the possible effect of zero g on spacecraft equipment that is not completely free of debris. In orbit, every void in the spacecraft from the pressurized cabin itself down to the interior of a transistor or relay, including all gas and liquid tanks and lines, becomes a region where debris that would normally be held fairly securely in place by the earth's gravitational field now floats freely about under the influence of magnetic or electrostatic fields, fluid currents, or surface tension forces.

Under these conditions filters and screens become imperative in all liquid and gas systems to protect close tolerance valves, orifices, or impellers. Indeed, the most obvious manifestation of the phenomenon in Mercury was the stoppage of the unscreened cabin ventilating fan by debris in two of the early unmanned flights. This debris was present despite clean-room fabrication of the spacecraft, plus a very intensive effort to clear it of debris by repeatedly inverting and tumbling it.

Obvious protective measures also require complete elimination of exposed electrical contacts anywhere that sufficient debris could exist to short them. There may conceivably be still more subtle effects, however, where interactions between electrostatic and magnetic fields and accumulations of floating particles may have significant effects on equipment operation. Since these phenomena, if they exist, cannot be reproduced on earth, a great deal of imagination may be needed to visualize the nature of such problems and to determine the protective measure required. The brute-force solution of perfect cleanliness seems, from past experience, to be unattainable. In any event, it will become more difficult as we move toward micro-miniaturization where smaller and smaller particles of debris become capable of shorting out the more closely spaced electrical paths.

STRUCTURAL DEFORMATIONS

Because of the requirement for lightness, launch vehicle tanks, adapters, and the spacecraft itself are all susceptible to rather large distortions under load. During the atmospheric phase of flight, dynamic pressures approach 1,000 pounds per square foot at Mach numbers in the transonic speed range. Under these conditions fluctuating pressure distributions may produce rather severe buffeting loads and fluctuating wakes. Early in Mercury development flights several cases were encountered where structural reinforcement or redesign was required to prevent failure, or where increased clearances had to be provided to reduce the risk of interference or damage under load.

It is not feasible to duplicate the flight conditions at full scale with ground equipment, nor is it feasible to build up gradually to these flight conditions in a series of manned flights. Hence, each new space vehicle configuration will require at least one unmanned flight to provide verification of structural integrity under the conditions encountered in the atmospheric phase of flight.

Distortion and vibration that might not otherwise be harmful can be disastrous if they trigger limit-switches that are used to sense separation and initiate automatically subsequent steps in the flight sequence. The best protection from trouble of this sort is obtained by designing separation sensors to require travel beyond any possible structural deformation before actuation, and by requiring confirmation of separation by two sensors before activation of succeeding steps.

A final point that is worth mentioning because it has been too often overlooked is that the high rate of decrease of ambient static pressure that occurs during the launch phase of flight requires special attention to the venting of sea-level air trapped in adapters, shrouds, external equipment bays, and exposed equipment.

HEAT REMOVAL AND EQUIPMENT TEMPERATURE CONTROL

The problem of removing heat from equipment in spacecraft has given a great deal of trouble. Internal heat-generating equipment such as inverters requires special attention because the reduced density of the cabin oxygen atmosphere and the lack of natural convection under zero-g conditions inhibits convective heat transfer. Careful design is required to provide adequate forced convection or conductive heat paths to transfer the heat to points where it can be rejected to space by radiators or evaporative coolers. External equipment, notably the nozzles of the reaction-control system, has given trouble for the same reason. The vacuum environment eliminates all convective cooling of the nozzles. Special attention is required to prevent heat that is left in the nozzle walls after each pulse from leaking back up the propellant feed lines by conduction, with adverse effects on the propellant or the solenoid valves that control its flow. A very thorough and careful analysis of the heat flow and temperature conditions in all equipment is an important design requirement for future spacecraft.

FUEL MANAGEMENT

On the basis of flight operations to date, optimum management of reaction-control fuel is emerging as one of the most important requirements for successful space flight. It is of critical importance to safety because of the need to retain sufficient fuel for control during the retrofire and reentry maneuvers.

Mercury experience has indicated that direct manual control over the reaction-control thrust nozzles is liable to be quite wasteful of fuel. Indications are that the ultimate system for spacecraft may be a command-type control system where some form of autopilot fires precisely measured and timed pulses to produce commanded rate or attitude changes.

There is an obvious need for research on both the control power and response required in spacecraft and on the mechanization of control systems to meet these control requirements with minimum fuel use.

An associated urgent requirement for future spacecraft that has been brought out in Mercury experience is provision of more accurate and reliable indications to the pilot of fuel usage, both rate of use and quantity remaining.

Although experience in Mercury has been limited to management of attitude-control fuel, similar considerations can be expected to apply to management of velocity-changing fuel in rendezvous, orbit correction, and lunar landing maneuvers.

SUMMARY

In summary, Mercury experience has pointed out a number of places where special operational procedures or special attention to design details can produce increased safety in manned space flight. It has emphasized again the overriding importance of meticulous attention to detail in the design and operation of space flight hardware.

MISSION CONTROL FOR MANNED SPACE FLIGHT

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SYNOPSIS

The development of a complex vehicle requires the parallel development of a test and control organization to provide the support necessary to accomplish the test objectives and to qualify the vehicle for operational rather than research and development usage.

The manned space-flight program uses a pure research and development vehicle which is normally operating very close to the limit of its design envelope. In addition, the full spectrum of engineering techniques is applied to the development of a manned space vehicle.

This paper traces the development of the real-time controlling agency from its concept through a Mercury mission to Project Apollo. In the course of this paper the flight-control tasks are identified and developed commensurate with the time frame allowed.

AUTHOR

INTRODUCTION

Many papers have been written on the subject of operational plans and procedures for the Mercury program. It is the purpose of this paper to present a general survey of one particular aspect of operations, namely flight control. In this paper the growth of flight control for Project Mercury and the plans for the following programs, Gemini and Apollo, are discussed.

Until quite recently, research and development testing of aircraft usually followed the pattern of a fixed flight plan, with a predetermined set of tests followed by postflight analysis. The flight-envelope boundaries were approached slowly because inflight evaluation by the pilot and possible observers was limited by the available instrumentation and the ability to develop real-time solutions. The action in the event of emergencies was to return quickly to a more acceptable part of the flight envelope and, if necessary, abandon the aircraft.

More recently, the advent of air-ground data links has allowed a ground-based crew to monitor the test in progress, to modify the flight if necessary, and to recommend the most expeditious course of action to be taken when contingency situations arose.

The missile age brought with it the development of a ground-to-air data link by which information and commands could be sent from a monitoring ground crew or automated system to the vehicle in order to modify its flight plan.

The one significant point which was brought to bear in the manned space flight program was that the vehicle traveled almost to the extremes of its limit design envelope on every flight; and the need for a ground-based crew to monitor, evaluate, recommend and, if necessary, command became evident. The work that this ground crew accomplished is defined as flight control.

The fundamental aims of the flight-control crew are the safety of the astronaut and the successful completion of the mission. By far the largest task is the determination and recommendation for a course of action essentially in real time in the event of a contingency situation. The course of action may be an alternate mission, a change in mode of operation, or premature termination of the mission. In Project Mercury, the definition of real time varies from a matter of seconds to about an hour, depending on the phase of the mission.

The following transcript of the MA-5 mission illustrates the rapidity of the evaluation and decision process of the flight control team. High thruster activity indicated an out of orbit mode in the automatic stabilization and control system; and if this anomaly continued, the reaction control system fuel supply would probably have been depleted prior to the end of the third orbit. The flight control team at Hawaii identified this condition, and the decision actually to terminate the mission occurred during this 12-second period. This discussion took place between the California Spacecraft Communicator and Systems Monitor and the Flight Director at the Mercury Control Center.

CAL	CAL thruster activity on 69 and 70.
CAPE	Roger.
CAPE SYSTEMS	Did he say activity or no activity?
CAL	Activity on 69 and 70.
CAPE	You've got 12 seconds.
CAPE	Go ahead with retrofire.
CAL	Thruster activity on roll at CAL.
CAL	5, 4, 3, 2, 1, fire.
CAPE	Roger.
CAL	Retros fired.

CAL Fire 180813.

CAPE Roger.

CAL We have confirmation of retros 1, 2, and 3 fired.

CAPE Roger.

CAPE Roger, understand.

CAPE Did you copy, Recovery?

CAPE Standby for time.

RECOVERY Roger, standing by.

MISSION CONTROL

The objective of mission control is to increase the probability of mission success and crew safety. Any deviation from a nominal mission requires that a decision be made, and this decision may either increase the chance for mission success or jeopardize the overall mission objective and, thereby, affect the life of the space crew members. In order to augment the analysis and decision-making capability, every concept, function, procedure, and system must be designed and implemented with mission success as the primary objective.

The mission control organization for Project Mercury was implemented to provide centralized control of remote sites capable of data exchange and command control. The organization, exclusive of technical support personnel, consisted of 16 flight controllers in the Mercury Control Center and 42 flight controllers at the remote sites. This organization, shown in figure 1, was responsible for the detailed conduct of the mission from vehicle lift-off to landing.

Flight control is the portion of mission control pertaining primarily to the aspects of flight dynamics, vehicle systems operation, and spacecraft crew performance and can be defined as the integration of the spacecraft crew and the ground personnel necessary to accomplish manned space flight.

Flight control consists of five phase-oriented tasks. They are defined as follows:

1. The preflight-preparation task represents the largest single part of the mission-control function. It consists of the development of the operational concepts, the determination of facility and personnel requirements, and the detailed flight-operations training. Included are the preparation of mission documentation and operating procedures, the preparation of mission logic and associated computer programs, and the training of operating personnel.

2. The mission-control task consists of supervision and coordination of mission ground support, the command control of unmanned vehicles, and the direction of the overall mission.

3. Spacecraft-systems and crew-performance analysis enhances mission success and crew safety by supplementing the crew in analyzing the telemetered and voice data to determine systems status.

4. Flight profile analysis consists of monitoring the flight program, development of alternate profiles when contingencies occur, the coordination of changes to the mission plan in real time, and the determination of go-no-go status for subsequent phases as based on data from systems, flight dynamics, and crew performance.

5. Postmission analysis is a detailed review of the mission operations, vehicle system performance, and the performance of the spacecraft crew.

During the initial phases of Project Mercury, flight controllers were obtained on a part-time basis from the engineering organizations within the Space Task Group (now Manned Spacecraft Center), and a relatively small group accomplished the premission preparation tasks and performed the postmission analysis. However, with the advent of the actual Mercury missions, it was recognized that a full-time organization was required to perform this function.

The ability to control a mission is a function of the preflight preparation, the experience levels of the flight control team, and the quality and type of data obtained from the network in real time. All aspects of the mission are reviewed, and the operational concepts are developed and compared with the test objectives. The systems are studied on the basis of their operational function and utilization, and failure modes are identified. As a result of this premission analysis, mission profile deficiencies are identified. The results of these studies are documented in three primary handbooks:

1. Mission Rules
2. Vehicle system schematics
3. Trajectory working papers

Each document is designed to allow rapid reaction to a contingency and to provide accurate real-time decisions.

As a result of this documentation, it becomes obvious that some automation is required, primarily in the area of trajectory analysis. A real-time digital computing system is provided for this purpose.

A major purpose of the documentation is the formulation of contingency plans and the criteria and methods for implementing these plans. This formulation, in fact, forms the largest part of the real-time flight control problem.

The Mission Rules are premission guidelines for decision making and contain most of the contingencies that can occur during a mission. For certain time-critical cases, the Mission Rules include the detailed procedures necessary to implement this decision.

Another primary document is the Flight Control Handbook which primarily contains the current spacecraft system schematics. These schematics, as typified by figure 2, define a total system operation and contain: (1) a power source, (2) components and controls, (3) displays for the astronaut, and (4) displays used by the flight controllers.

If a malfunction is indicated by telemetry or by astronaut readout, the probable cause or failure can normally be isolated through one schematic, and the necessary action, if any, can be taken.

The third major document utilized during the mission is the trajectory working papers, which contain most of the launch abort trajectories, the reentry trajectories, and the nominal orbital trajectories. The logic utilized to develop this document is essentially the same as the logic utilized in developing the computer programs for a specific mission. The tracking information obtained from the network is processed by the computers; then it is presented to the flight dynamics personnel via plotboards and digital displays. A flight dynamics console is shown in figure 3.

TRAINING

Personnel training is accomplished in the following three phases:

1. Individual
2. Team training
3. Network training

The individual training is accomplished by briefings, detailed systems and operational studies, preparation of mission documentation, and observation of mission operations. Flight controllers participated in the preliminary Mercury flights, both as individuals and later as teams, to build up their capability to perform their eventual task.

The team and network training is necessary to develop the decision capability of the flight control organization. The team assignments are usually made 3 months prior to a mission, and, after the systems updating briefings, the teams begin an intensive period of training in the spacecraft procedures trainer and site console trainers. During this phase, the flight controllers are trained in both the command-communicator and systems-engineer positions. A third flight controller will fly the spacecraft procedures trainer, which provides both telemetry and communications inputs to the console trainers. These exercises primarily consist of normal and contingency procedures associated with portions of the actual flight plan. When the team training is completed and the required confidence level is attained, the third phase of training begins. This phase, in preparation for a live mission, normally commences 2 weeks before the mission. Simulation exercises have been previously taped and transmitted to the network sites. A simulated mission normally begins at 4 hours before lift-off with a vehicle and network countdown. This simulation affords a checkout of the prelaunch procedures and the test sequencing. At simulated lift-off, the procedures trainer provides closed-loop telemetry and voice data to the Mercury Control Center operating positions.

The procedures trainer, shown in figure 4, is normally manned by the mission or backup astronaut for these tests. This action provides for the indispensible integration of the spacecraft crew and ground personnel and establishes the required confidence between these personnel. Trajectory data are provided from taped data to the switching and distribution area and eventually to the flight dynamics plotboards. Whenever possible, the ground systems are used in the same manner as they would be for an actual mission. This procedure develops confidence in the systems and is particularly important in the case of the real-time computer complex. At the time corresponding to the loss of communications at Cape Canaveral, the simulation becomes open-loop. The network sites at acquisition transmit radar information from prepared paper tape through the teletype lines to the computing center. The telemetry data are played into the remote site displays, and the flight controllers evaluate and respond to the data. Figure 5 shows a remote site console group.

The network training exercises all aspects of flight control and aids in perfecting the Mission Rules and operational procedures. The final objective of these exercises is the development of confidence in the readiness of the network equipment and flight controller personnel prior to initiating the terminal countdown.

The increased complexity of the Gemini and Apollo missions is partly a result of the mission profiles, for example, the larger incremental velocity capabilities of the space vehicles and the subsequent capability to change the orbital parameters. This complexity results in a significantly enlarged tracking and computation requirement. In addition, for Project Gemini, both the Gemini spacecraft and the Agena target vehicle have this incremental velocity capability. The command control of this capability in the Agena systems by ground personnel is essential to mission success. The basic tasks of flight control are essentially unchanged; however, the expansion of the systems to include the in-orbit maneuvering capability required greater automation of the orbit-determination and command-generation function.

The complexity associated with the mission profiles will require greater data exchange, both from the spacecraft to the ground and between the ground sites themselves. The bulk of the telemetry data will be assimilated by a number of highly specialized remote site personnel. In addition, significant amounts of these data will be automatically transmitted to the Integrated Mission Control Center (IMCC).

INTEGRATED MISSION CONTROL CENTER

The Integrated Mission Control Center facility, shown in figure 6, is to be implemented in Houston, Tex. This new facility will provide the centralized control capability necessary for the conduct of the Gemini and Apollo missions. The facility includes dual mission operations control rooms capable of various combinations of simultaneous real-time missions, simulation exercises, or system checkouts. At the present time, it is not planned to conduct two missions simultaneously. The real-time computing complex will be utilized at IMCC to assist in providing better real-time decision capability for both the vehicle systems and trajectory analysis. Basically, the IMCC will be staffed and operated the same as the Mercury Control Center; however, the mission analysis capability within the IMCC will be enhanced by the development of mission support specialists. This procedure will enable most flight control personnel to concentrate on a specific system or mission phase. At the network remote sites, the flight control teams are essentially the same as those for Mercury missions with the addition of a systems monitor for the Agena vehicle or the

Apollo Lunar Excursion Module and the S-IVB launch-vehicle stage.

The IMCC mission operations room will direct the facilities and organizations necessary for accomplishment of the mission. These facilities and organizations include the following:

1. Internal and external computing facilities
2. Communications facilities
3. Network stations and flight control teams
4. Launch facilities
5. Spacecraft and flight crew

Personnel within the mission operations control room represent the primary decision-making group associated with the mission. The Operations Director is responsible for the overall performance of the mission. He has delegated much of this authority from lift-off through landing to the Flight Director. The Flight Director is supported by a staff of approximately 15 personnel, as indicated on the organizational chart in figure 7.

GEMINI

Rather than develop a description of each position in the mission operations control room, significant portions of the Gemini mission which detail flight control operations for both manned spacecraft and unmanned flight vehicles are presented. The Flight Director will again direct the mission operations from the lift-off, through powered flight, to recovery of the spacecraft and crew. The Flight Director will continue to receive inputs concerning the performance of the spacecraft and its systems, the launch vehicle, and the flight crew from flight control personnel; and on the basis of their analysis, and by correlation with the detailed mission rules, he will direct continuation or abort of the mission.

The basic objective of the Gemini mission is to place a target vehicle in orbit and follow it with a manned vehicle which is to rendezvous with the target vehicle with a minimum of fuel usage and the shortest time. This mission adds several tasks to mission planning.

First, tne countdown of both vehicles must be conducted in the same time period; second, the launch time and launch azimuth of the second vehicle are extremely cirtical if the objectives are to be met.

There is about a 20-minute period each day when launch can occur into the correct orbital plane and with an acceptable phase difference. This time period corresponds to a plane error of 0.4° and a phase difference of 70°. Of course, larger errors can be accommodated if the velocity capability of the Agena is used.

The launch phase remains the most critical period of the flight. However, a redundant guidance system with automatic and manual switchover has been provided in the Gemini-Titan configuration. In addition, ground monitoring will determine requirements for switchover for malfunctions which occur at a slower rate. The Gemini spacecraft has a manual abort system unlike the present Mercury abort system which is automatic. This manual system results in a much more reliable sequence system but requires very close monitoring, with short decision and response times both in the spacecraft and on the ground. A Ground Flight Controller will assist the astronaut in identifying valid abort requirements.

Network Flight Controllers will continue to monitor the normal progress of the mission and to determine the course of action in the case of abnormalities.

These abnormalities may be associated with trajectory deviations, in which case velocity corrections must be determined and transmitted to the remote sites and thence to the spacecraft. They may also be associated with spacecraft systems malfunctions, in which case corrective actions must be determined or an alternate flight plan, which may result in early termination of the mission, must be provided.

Preplanned mission documentation and computer programs will again be used to assist in determining the course of action in the case of contingency situations. One point which should be mentioned is that, in the case of Gemini, the crew has a much greater onboard capability than before. The computer controlled guidance and navigation system together with the propulsion system provides the crew with the capability to complete a normal mission without further trajectory assistance once the initial ephemeris of the two vehicles has been properly established by the ground system.

The personnel at the ground stations will assist the astronauts in vehicle checkout prior to reentry. They will also update the onboard computer with the latest information for reentry and insert the correct retrofire times. The reentry of the Gemini vehicle will be similar to that of the Mercury spacecraft. The main difference from Mercury is the capability to modulate lift during the reentry. The lift modulation combined with the paraglider landing system may require additional flight-control support. At the present time, studies are being made of the utilization of a radar controlled, ground controlled intercept type of approach for assisting the crew in the landing phase.

During the reentry phase, the IMCC will direct the recovery force to the proper landing area and aid them in recovery of the flight crew. During this period, flight control will continue to maintain contact with the crew and support them as required. After the mission, the flight controllers, both at the IMCC and remote sites, will be required to undergo a detailed postmission analysis in order to determine actions and problem areas for future missions. Although the Gemini mission with its two vehicles, one manned and one unmanned, will require detailed analysis of a large quantity of information, the flight control team, with its training and aided by automatic aids, will be able to evaluate this information, reach a decision, implement this decision, and take action to support the flight crew and assure the successful completion of the mission.

APOLLO

The early Apollo missions will be conducted in a manner similar to that of the Gemini missions. However, as the missions increase in both frequency and complexity, the manpower requirements necessary to support these missions will become excessive. Test programs are now underway at the Bermuda site to provide computer data processing of spacecraft data. The site data processing will be combined with wide-band data lines to transmit the tracking, telemetry, and voice data to the IMCC for evaluation and decision. The ability to provide a highly skilled flight-control team with near continuous full bandwidth data is the ultimate objective of flight control.

The Apollo Lunar Mission, and all subsequent Manned Space Vehicles, will continue to require the support of a ground flight control team. Whenever a vehicle is probing the envelope of ultimate performance and expanding the bounds of known space, Flight Control will aid in providing the added margin toward mission success.

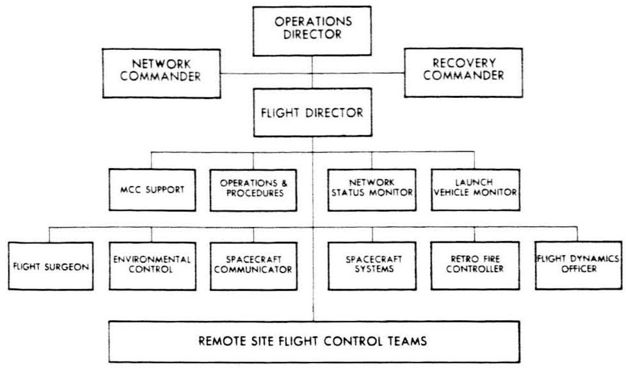

Figure 1.- Mercury Control Center organization

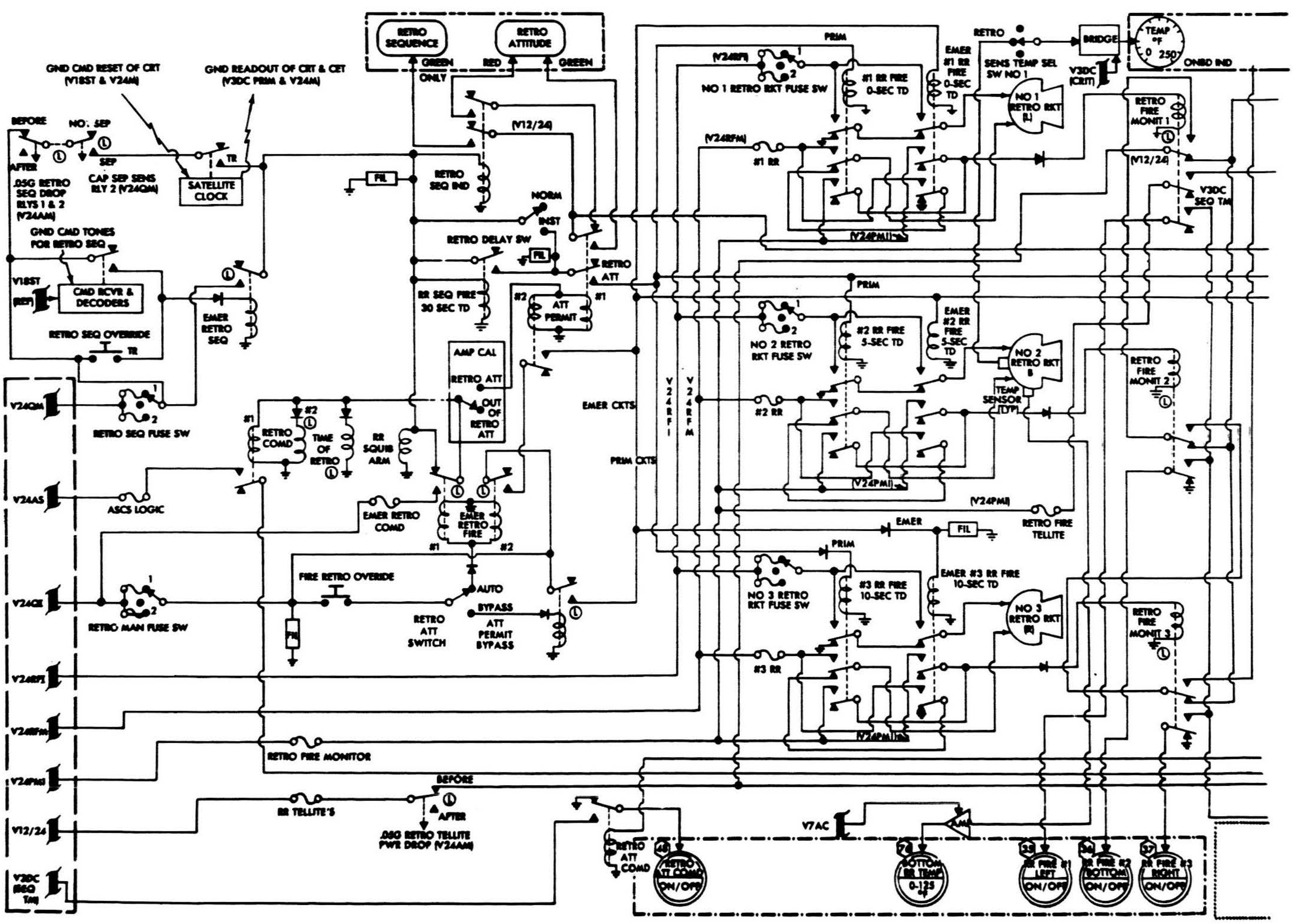

Figure 2.- Spacecraft schematic diagram

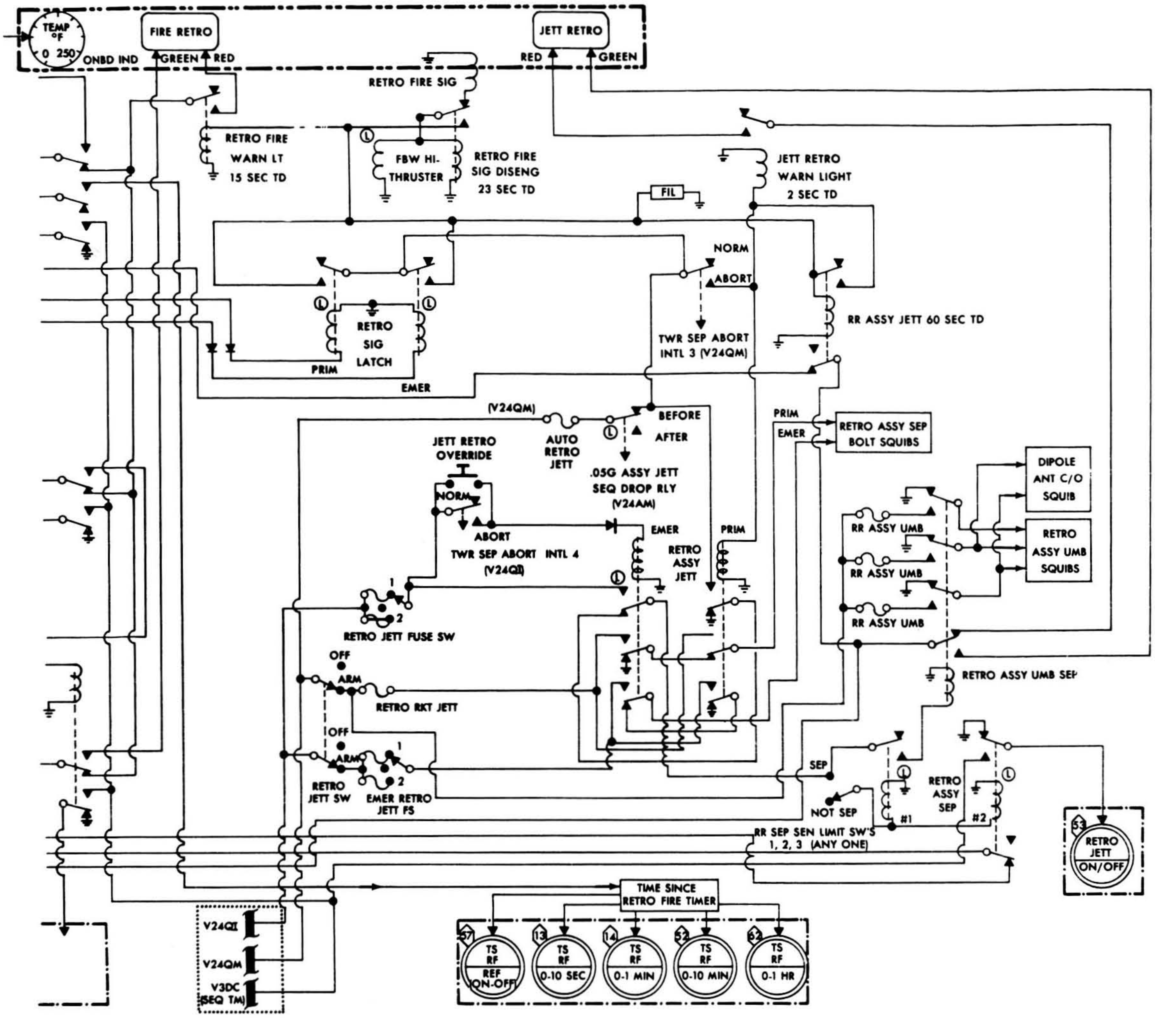

Figure 2.- Concluded

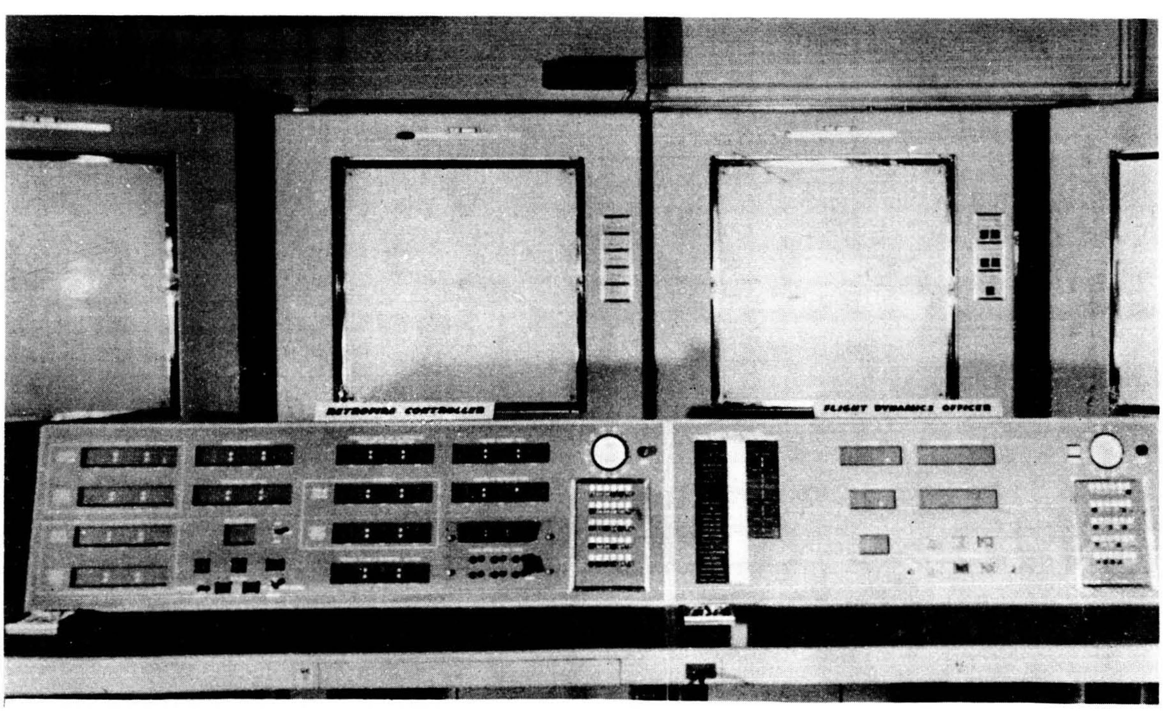

Figure 3.- Flight dynamics console

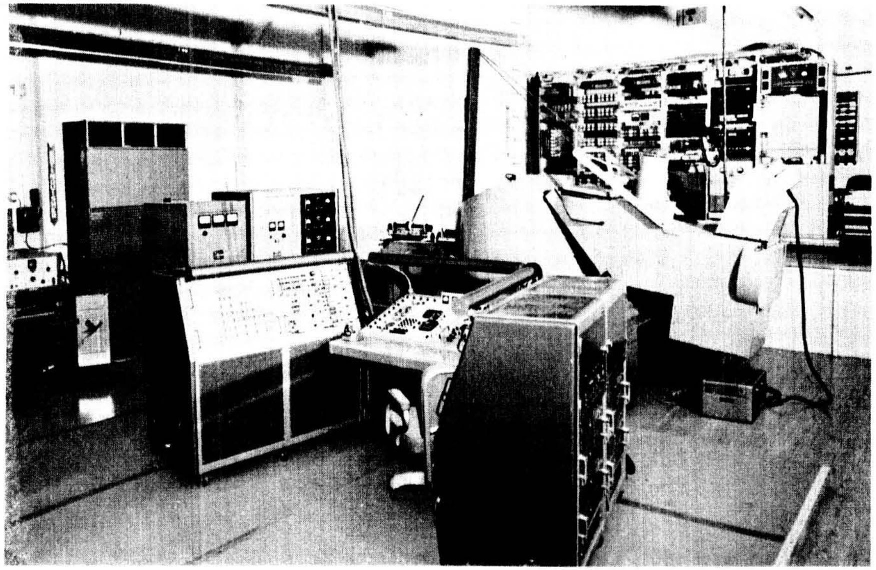

Figure 4.- Mercury procedures trainer

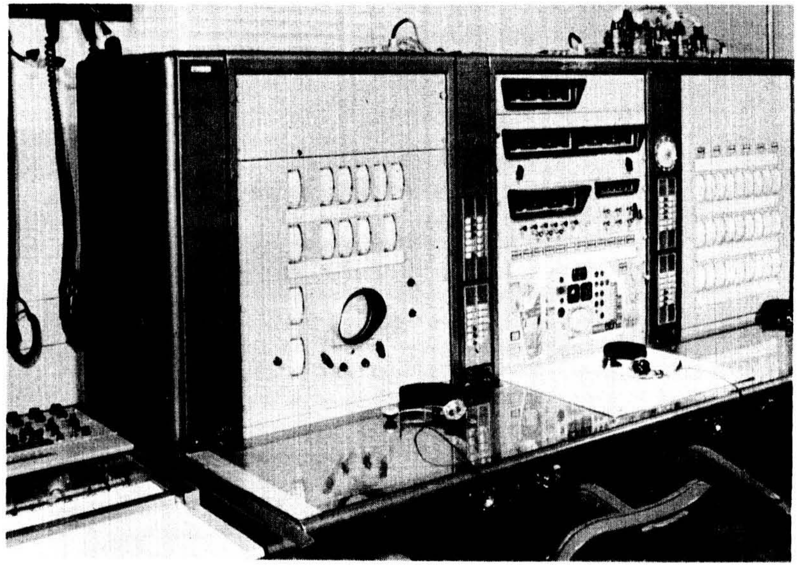

Figure 5.- Remote site console group

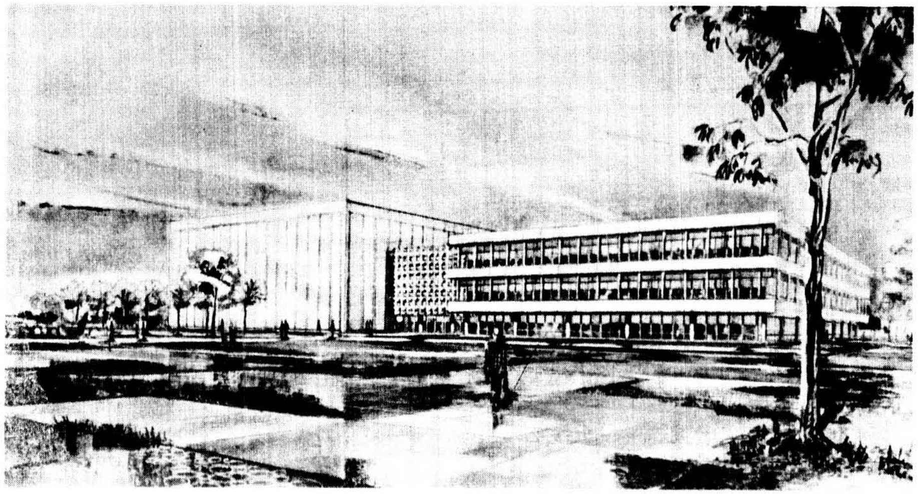

Figure 6.- Exterior view of IMCC

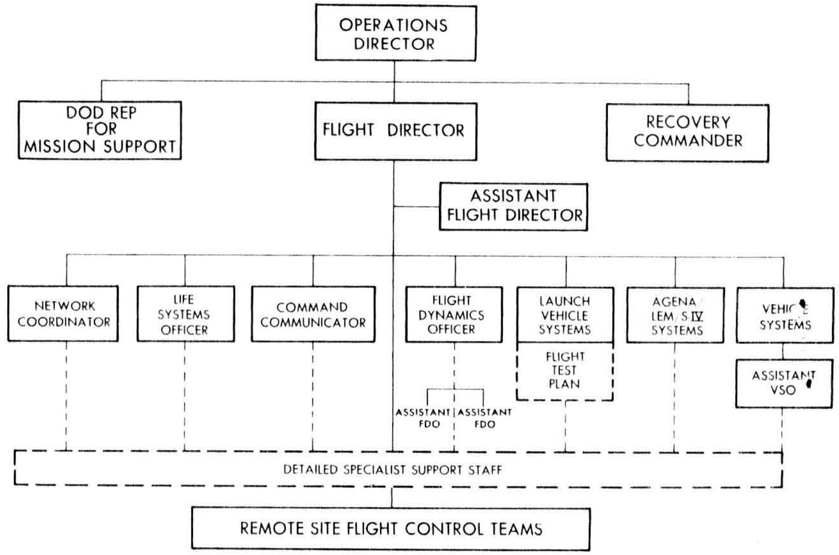

Figure 7.- IMCC organization

SPACECRAFT TRACKING AND DATA ACQUISITION

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The purpose of this paper is to present a summary review of spacecraft tracking and data acquisition. The subjects to be covered include (1) the reasons for tracking, (2) the restrictions imposed on tracking by nature, (3) the merits of various types of tracking as a function of altitude and coverage, (4) the accuracy needed for position determination, (5) factors involved in tracking data acquisition, and (6) possible changes in current technology to better meet the requirements of tomorrow.

Introduction

One of the purposes of tracking a spacecraft is to establish the initial conditions for the equations of motion. These initial conditions provide a set of trajectories bounded by a volume of finite radius. The magnitude of this radius is a function of the prediction accuracy. In the limit, if the tracking equipment were perfect, if we had precise knowledge of the physical constants, etc., this radius would be zero, and hence the error between the actual spacecraft trajectory and the predicted, or computed trajectory, would also be zero.

Spacecraft location is useful (1) in correlating an observed event with the position of occurrence or measurement; (2) in determining a better knowledge of the forces acting on a space vehicle from the discrepancies between the spacecraft location and the predicted location; (3) in evaluating and determining corrective maneuvers necessary to place the spacecraft on the desired trajectory; (4) in determining where ground antennas should be oriented to receive information from the spacecraft, and/or to relay commands to the spacecraft.

Let us now examine some of the sources of error and their magnitude in trajectory prediction. The major errors contributing to the inaccuracies in prediction are:

1. Random and Bias Errors. These include noise, calibration errors, antenna sag, atmospheric refraction, multipath, and uncertainty in physical constants.
2. Uncertainties in Spacecraft Environment. This includes atmospheric drag, and uncertainties in ground station location with respect to a reference inertial coordinate system.
3. Imperfect Trajectory Selection. This is due to the non-optimum weighting coefficients, the use of linearity assumptions which may not be appropriate, and the truncation of iteration procedures.

Let us now see how some of these sources of error affect the technique used in determining the trajectory (doppler, range) and what are the lower bounds on these errors.

Uncertainties in Speed of Light

The effect on doppler and range as a result of the uncertainty in the speed of light can be understood from the following relations by:

$$V_r \approx c \frac{f_r - f_t}{f_t}$$

where

V_r = radial velocity of spacecraft $\left(\frac{dR}{dt}\right)$

c = velocity of light

f_r = observed frequency

f_t = transmitted frequency

From this relationship it is seen that a fractional error, or uncertainty, in c produces the same fractional error in V_r. For an uncertainty in c of 1 part in 10^6, the uncertainty in the doppler velocity is also 1 part in 10^6. As an example, for a spacecraft radial velocity of 25,000 feet per second, an uncertainty in the velocity of light of 1 part in 10^6 would produce a doppler velocity error of 0.025 feet per second which, as we shall see, is quite small with respect to other doppler errors. On the other hand, an uncertainty of 1 part in 10^6 for the velocity of light produces a range error of 10^{-6} R, which can be significant for very long ranges. [See Fig. 1.] This is, of course, due to the proportionality relationship between range and the speed of light. It follows then, that uncertainty in the speed of light is relatively unimportant to doppler data but can be important to range data in deep space.

Uncertainties in Station Location

The trajectory prediction procedure is essentially an attempt to predict the position of a spacecraft in an appropriate inertial coordinate system. This is done by relating the spacecraft to known points in the coordinate system (location of tracking antennas). Any error, or uncertainty in the position of these reference points will, therefore, degrade the prediction accuracy.

A few extremely accurate surveys to specify the positions of points on the earth's surface have been made with accuracies of 1 part in 10^6. This implies that one can not expect to relate station position to a fixed reference more accurate than about 120 feet, or 0.02 nautical miles.

The range data accuracy, for low elevation angles, can be expected to vary between ±0.02 nm due to the uncertainty in the station location. As we shall see later, the major components of doppler data error may arise from station location uncertainties, particularly at low spacecraft altitudes.

Uncertainties in Atmospheric Density

Uncertainties in atmospheric density evaluation prevent accurate determination of the instantaneous orbital parameters, and hence produce errors in predicted position. Suppose that the spacecraft's position and velocity vectors were precisely known at a particular time and that the future positions were deduced by solving the equations of motion. If the equations of motion utilized an atmospheric drag force different from the actual drag force, then errors in position prediction would result. For a satellite in a 155-nm circular orbit, and an uncertainty in the drag force of 20 per cent (e.g., a 20-percent uncertainty in atmospheric density), tracking must be accomplished at least every two orbits (three hours) to keep the position prediction accurate to 0.1 nm.

From the above, it is seen that the unknowns in nature, such as the variation in the atmosphere, the uncertainty in the speed of light, and a lack of knowledge of the shape of the earth restrict the accuracy to which a spacecraft's position can be predicted.

The Merits of Various Types of Tracking as a Function of Altitude and Coverage

The accuracy with which the ephemeris of a satellite must be determined is a function of the mission of the satellite. A navigational satellite may require very accurate knowledge of position while some communication satellites may have much less stringent accuracy requirements. The tracking equipment necessary for tracking as well as the amount of tracking time then becomes a function of the mission of any particular satellite.

Simulation studies have been conducted at WDL in an attempt to determine the relative merits of different methods of satellite tracking. Comparisons of range and range-rate tracking over a variety of altitudes have been conducted. A chart indicating equivalent tracking capability is shown in Figure 2. This figure indicates the relative accuracy with which range or range-rate measurements must be known to determine ephemerides of similar accuracy. The errors considered are random errors and are not specified as to their source. The error in measurement is thus a lumped error of all possible sources. No study was made as to the apportioning of this error. This figure indicates that range-rate measurements provide the most sensitive measuring technique for low-altitude satellites, while poorer quality ranging systems may be used for higher orbital cases to obtain tracking accuracies that would require very sensitive range-rate systems.

Although Figure 2 shows the relative sensitivity of range and range-rate tracking, it is interesting to note an expected tracking performance given a realistic tracking situation. Figures 3(a) and (b) show the tracking error which might be expected using range-rate and angle tracking for a low-altitude satellite for various times after tracking has been completed. These figures point out that additional data from either additional equipment or additional tracking contacts tend to improve tracking accuracy. The linear nature of the simulator allows the error resulting from tracking with additional equipments to be estimated. If an equipment has 1-ft/sec range-rate accuracy with an 0.33-mrad angular capability, the expected error would be one-third the minimum error in these two figures. Available equipments listed in Table I may be fitted into this chart using imagination and linear approximation. This particular simulation uses a known atmosphere. If a 20-percent atmosphere uncertainty causes an ephemeris error of 0.1 nm in 3 hours, it becomes apparent that refined equipments do not improve the prediction capability.

Even though the source of tracking errors has not been detailed, it is interesting to note the effects of station off-set upon range-rate measurements. A station at 40° latitude was offset 0.02 nm in both latitude and longitude. Figure 4 shows the resulting error in the range-rate measurements. This simulation run indicates the high sensitivity of range-rate measurements to station offset and implies the difficulty in obtaining data that is actually as accurate as the measuring capability of the tracking equipment. The use of range-rate measurements as an effective means of locating a station is also indicated.

The effects of coverage may be seen in Figure 5. The addition of data from additional tracking contacts gives rise to a large reduction in the expected tracking error. For this particular simulation the tracking data was processed in three separate groups, each one containing the tracking data from several tracking sites. The sudden change in accuracy is caused by the sudden addition of data. The tracking summary for this run is found in Table II. The actual tracking was done at Canaveral, Carnarvon, and Hawaii. A complete picture of the effects of coverage on tracking accuracy is a complex one. Tracking accuracy is dependent both upon the number of data points as well as some geometric quality factor. The nature of a pass, high or low elevation, as well as the spacing between tracking contacts influences the obtainable tracking accuracy. Figure 5 indicates that a point is reached where additional data becomes less influential in accuracy improvement.

The tracking errors caused by atmospheric variations are greatest for low-altitude satellites. The probability of a low-altitude satellite being above the tracking horizon for a given tracking site is much less than that for a high-orbital-altitude satellite and, as a result, there is much less opportunity for a station to track those satellites needing the most frequent up-dating. The tracking coverage for low-altitude satellites may be increased by the use of a satellite-to-satellite-to-ground tracking technique,* a tracking satellite at high altitudes results in longer periods of contact with the ground station. The increase in coverage is readily apparent by this technique but the effects of equipment error have not as yet been evaluated by our simulation runs.

As tracking accuracy is improved by additional coverage, so it is improved by additional types of measurements. Figure 6 compares the smoothed tracking accuracy which may be obtained by tracking with 1.0-mrad angular accuracy; 1.0 mrad angles and 0.5 nm range; and 1.0 mrad angle, 0.5 nm range, and 1.0 ft/sec range-rate. Figure 2 indicates that at 50,000 nm a range-rate system of 0.05ft/sec sensitivity would be required to substantially improve accuracy. This is apparent in Figure 6 as the additional range-rate measurements cause only a slight improvement in the accuracy of tracking.

These simulation runs stress the fact that additional tracking information will increase

*D. B. Davis, "Satellite Assisted Tracking," 9th Annual Aerospace and Navigational Electronics Conference, Baltimore; 1962.

Table I

TRACKING SYSTEM CHARACTERISTICS

	AN/FPS-16	AN/FPQ-6	VERLORT (AFMTC Mod II)	IIR	Range-Range Rate	JPL PN Ranging	Azusa MKII	MISTRAM
Range	500 nm	32,000 nm	2000 nm	32,000 nm	Lunar	Lunar	1000 nm	1000 nm
Range Accuracy	15 ft	15 ft	50 ft	±12 ft	±45 ft	±90 ft	±10 ft	1 ft
Range Resolution	3 ft/bit	6 ft/bit	3 ft/bit	6 ft	13.3 ft	15 ft	1 ft	0.03 ft
Angle Accuracy	0.1 mil rms	0.05 mil rms	2 mils rms	0.1 mil rms	0.1°	0.02°	0.01 mils	$20\,(10^{-6})$
Az. Tracking Rate	40°/sec.	28°/sec.	20°/sec.	26°/sec.		2°/sec.	52°/sec.	
El. Tracking Rate	30°/sec.	28°/sec.	20°/sec.	26°/sec.		2°/sec.	23°/sec.	

tracking accuracy but at some altitudes one type of measurement may require only off-the-shelf techniques while the other may be pushing the "state-of-the-art."

The information gained from the tracking simulation runs show

1. Expected tracking accuracy at 125 nm given a specified tracking situation
2. A means of extracting these curves to actual equipment by linear extrapolator and a range-rate, range equivalent plot
3. An indication of the effects of additional tracking data by both extended coverage and supplementary measurements
4. Statements of the limiting effects of nature.

It is apparent from the charts that tracking accuracy may usually be improved through increasing measuring and coverage capabilities. Cost trade-offs are thus possible comparing equipment developments, additional ground stations for increased coverage or the use of a high-altitude satellite for increased tracking coverage. No attempt at these cost trade-offs is made in this paper but directions for further tracking improvement are indicated.

Table II

	No. of Data Points	No. of Tracking Observations
First Revolution	172	3
First and Second Revolution	457	6
First, Second, and Third Revolution	675	9

Spacecraft Acquisition

Up to now we have been speaking about spacecraft position prediction. Let us now discuss the subject of acquiring the spacecraft signals that are to be used in computing its position.

Spacecraft acquisition is a twofold process of locating signals from the spacecraft in both position and frequency at a given time. Accordingly, the acquisition process can be divided into two steps: locating the spacecraft in space, and locking onto its signals in frequency. The order of these two operations depends on the strength of the signal to be acquired as well as the relative uncertainties in these quantities. If the signal strength is sufficiently high, the acquisition antenna can be omnidirectional (or at least can cover the entire search area within its beamwidth) so that the system has to acquire only in frequency. Likewise, the uncertainty in frequency may be sufficiently small compared to the receiver bandwidth that only a scan in position is required.

Usually the first task is to sector scan and track the spacecraft, especially on the first trackable pass. This need arises from the uncertainties in orbital parameters upon launch. The orbital parameters contributing greatest to the need for sector scanning are (1) the deviation in spacecraft bearing relative to the desired bearing at orbit injection, and (2) the deviation in orbit period from nominal period. The first defines the azimuth sector, and the second defines the deviation in expected time of arrival.

The main area of interest for initial acquisition exists near the horizon plane of the tracking station. Acquisition can be accomplished by sector scanning the antenna in azimuth only, thereby setting up a "fence" in the sky at the radio horizon. The height of the "fence" is equal to the antenna elevation beamwidth, and the width of the "fence" or sector must be at least equal to the uncertainty in azimuth bearing. For any given section width, the antenna scanning rate must be such that (1) the spacecraft appears within the antenna beamwidth at least once, and (2) the spacecraft remains within the antenna beamwidth sufficiently long so as to permit the detection and acquisition circuits to perform properly.

The basic elements of the space-time acquisition are illustrated in Figure 7. The spacecraft breaks radio horizon at an azimuth and time such that the antenna may need to perform a space search.

The spacecraft's rate-of-change of elevation angle varies with elevation angle, altitude, and position relative to the tracking site. Since the rate of change of elevation angle is smaller for low elevation angles, a space search at the lowest possible elevation angle is desirable.

The uncertainty in orbit inclination is a function of the booster propulsion and guidance subsystem tolerances. These tolerances may cause deviations of greater than 1° in inclination. The resulting deviation in azimuth varies with maximum elevation angle and is much larger for low passes than for overhead passes.

Uncertainty in the orbital period may be as great as five minutes in the first orbital pass after launch. This uncertainty depends mainly on whether injection occurs at such a point that orbital tracking data can be obtained immediately after injection. The scan of the tracking antenna must be performed during the complete uncertainty interval around the nominal time of acquisition. After the first orbit, the period is usually known sufficiently well that the search time becomes much smaller.

It is important to note that periods of satellites at low altitudes may vary from pass to pass because of changes in atmospheric density. For this reason the uncertainty in period never reaches zero, and predictions of orbital parameters cannot be made with certainty more than about a day in advance for satellites with apogees less than 200 miles altitude. At higher altitudes, however, long-term prediction of ephemerides is possible and practical to implement.

The process of acquiring in position and time can be improved in at least three ways, all of which are concerned with reducing uncertainties in azimuth and time into view. The most obvious way to reduce these uncertainties is to improve the quality of tracking data, particularly during launch and the first few hours of tracking. Tracking systems are constantly being improved at the various launch ranges. Hence, this error source is being reduced.

Reducing booster and guidance errors during launch and injection also tends to reduce bearing and time uncertainties in the tracking data. Regardless of any improvement in tracking capability, the anticipated improvement in booster performance will tend to decrease uncertainties in spacecraft coordinates in the immediate post-injection phase of operations.

A third way to reduce the errors in acquisition data is to improve the speed of the trajectory computation process, mainly by speeding the transfer of data from tracking devices to the computers. Replacement of low-speed teletype data transmission by high-speed lines and such data-handling equipment as Kineplex or Dataphone can decrease the time required for processing tracking data by an order of magnitude. Likewise, the use of satellites to relay tracking data can reduce even more the time needed for data transmission in worldwide tracking networks.

Frequency-Time Acquisition

While the antenna beam is traversing the satellite signal, the tracking receiver must find the signal frequency and indicate a "hit." This may be done by tuning the receiver across the search band, or by having numerous (channeled) receivers stagger-tuned. The width of the search band which must be covered depends upon the maximum doppler shift, spacecraft and ground transmitter instabilities, ground receiver instabilities, and the ground receiver's ability to predict the doppler shift. If a tunable receiver is used to find the frequency, then it must tune over the search band and indicate a "hit" while the signal remains in the antenna beam. Figure 8 gives the defining equation for receiver tuning rate and bandwidth.

Additional problems occur in achieving phase-lock of receivers and in synchronizing tracking signals. The frequency sweep rate of the voltage-controlled-oscillator in the phase-locked loop is an inverse function of the loop filter bandwidth. The rate of change of signal frequency depends on the rate of change of doppler shift and on the transmitter and receiver oscillator instabilities. Hence, the loop filter bandwidth must be chosen to maximize sweep rate and to allow for deviations in the signal frequency while minimizing the receiver noise bandwidth.

If the transmitted signal is phase modulated, the receiver usually acquires the carrier first with a narrow-bandwidth filter channel and then switches to wider bandwidths to lock on to the modulation. In many tracking systems only the carrier is of interest (for range-rate data), and the modulation is used for telemetry. However, the modulation may be sidetones or pseudo-random noise (PN) sequences for ranging. Here, the usual procedure in acquisition is for the ground station to generate a signal (sidetone or PN sequence) like that from the spacecraft, and then to lock it to the received signal by shifting its modulation generator in phase until transmitted and received signals lock in the spacecraft. The generator at the ground station is left free-running to remain in lock with the spacecraft signal, and the amount of shift in phase needed to do this is proportional to range. This locking procedure can take several minutes for long code sequences at lunar distances.

The frequency-acquisition problem is being attacked by improving oscillator stabilities on the ground and in the spacecraft, and by the use of adaptive filters in the receivers. The latter item allows wider bandwidths for initial acquisition or re-acquisition and narrower bandwidths for data reception after acquisition is accomplished.

This foregoing discussion on acquisition applies both to radio and to optical tracking. Optical techniques can result in narrower beamwidths and greater angular accuracies than are achievable with radio tracking systems. Also, laser techniques will allow wider bandwidths for shorter pulses and more precise ranging.

Another feature which may change the methods of spacecraft acquisition will be the presence of men as part of the equipment in advanced spacecraft. In this case the spacecraft may have the capability of acquiring the ground station instead of the present method of operation. Such an approach might be favored, particularly if the spacecraft has on-board devices for position measurement independent of the ground stations. Man can also serve as an adaptive element in the acquisition process.

Areas for Improvement in Tracking

Tracking accuracy may be improved by improving equipment accuracy, increasing coverage or improving the knowledge of nature and the natural constants. The limits imposed by the knowledge of the speed of light restrict improvement in range tracking by direct range measurements. The use of lasers or phased arrays seem to offer improved angular accuracies while lasers offer an opportunity to use much higher frequencies and obtain the resultant improvement in range-rate measurements. Lasers also allow shorter base lines to be used for interferometer types of tracking approaches. Coverage may be increased by the use of more and more tracking sites or through the use of satellites serving as tracking sites.

These improvements are all possible if the need is found to exist.

FIG. 1 Range Error Due To Speed Of Light Uncertainty As A Function Of Range

$\left[\Delta R = \frac{\Delta c}{c} R = 10^{-6} R\right]$

Uncertainty in Speed of Light (c) = 1 Part in 10^6

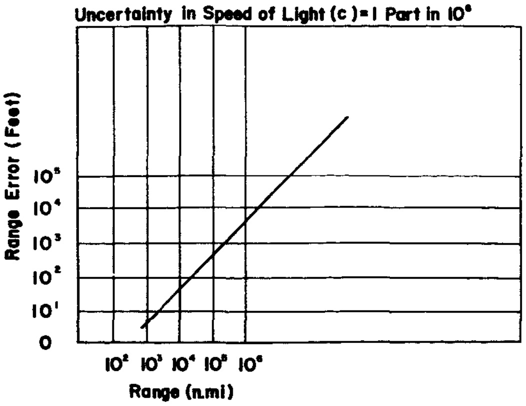


FIG. 2 Comparison of Range and Range-Rate Tracking

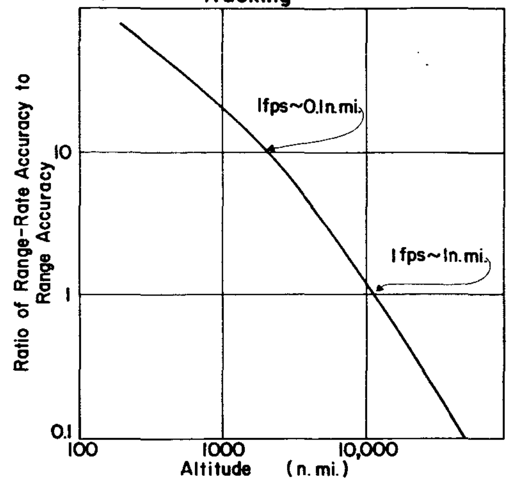


FIG. 3A
Observational Pass Data — 2 Passes

∇ (Start, Duration, Maximum Elevation)
∇ (61, 5.5, 90°)
∇ (91, 5.5, 90°)

Satellite Altitude: 125 n.mi

Gaussian Tracking Data Inaccuracies

(σ_S (nmi), $\sigma_{\dot{S}}$ (fps), σ_A (mrad), σ_E (mrad)

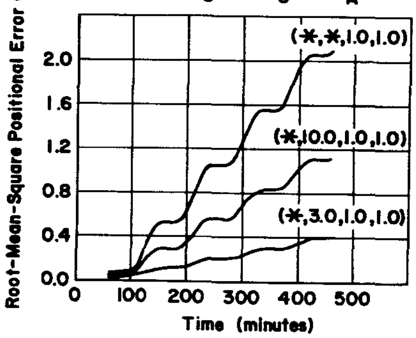


FIG. 3B
Observational Pass Data — 3 Passes

∇ (Start, Duration, Maximum Elevation)
∇ (24, 5.5, 90°)
∇ (38, 5.5, 90°)
∇ (61, 5.5, 90°)

Satellite Altitude: 125 n.mi.

Gaussian Tracking Data Inaccuracies

(σ_S (nmi), $\sigma_{\dot{S}}$ (fps), σ_A (mrad), σ_E (mrad)

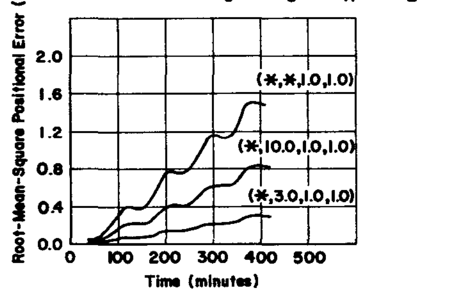


FIG. 4
Peak Root-Mean-Square Doppler Tracking Data Distortion

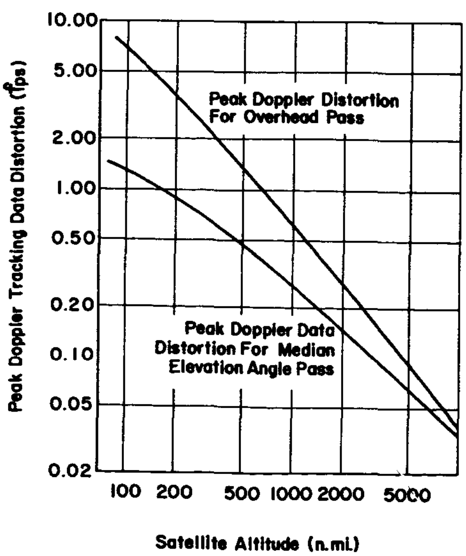


FIG. 5 Error In Predicted Position

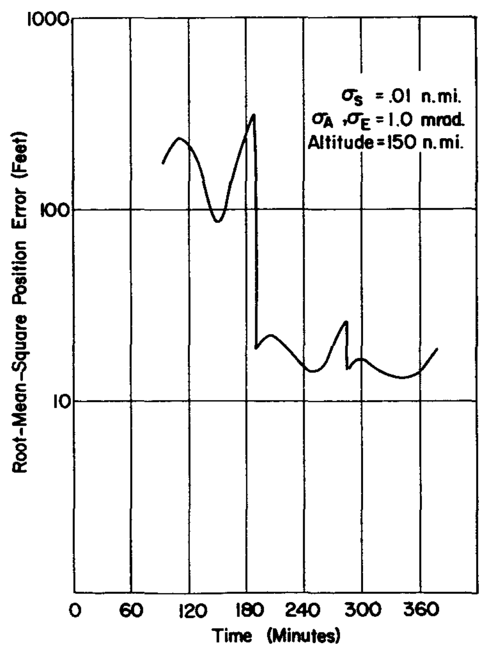
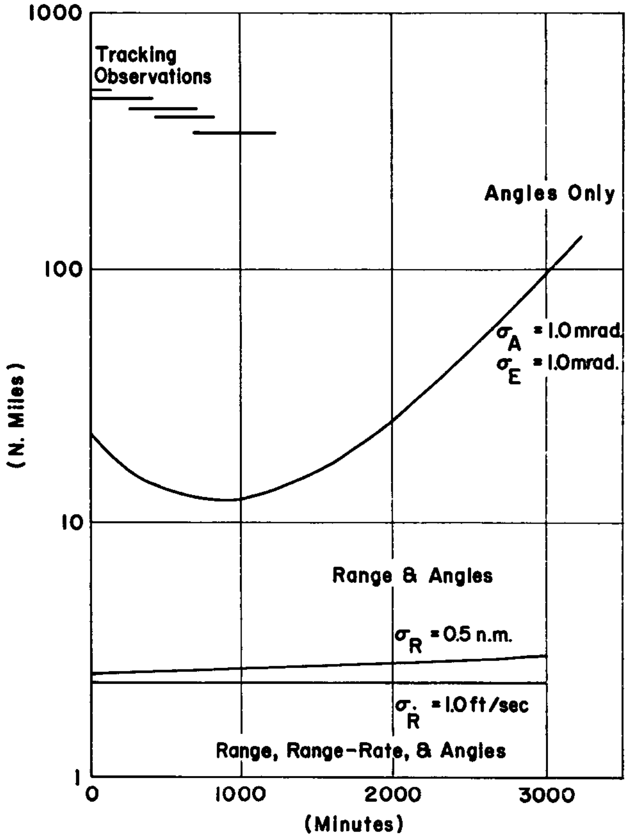


FIG.6 Tracking Errors for 50,000n. mi. Satellite



Basic Elements of Space-Time Acquisition

FIG. 7

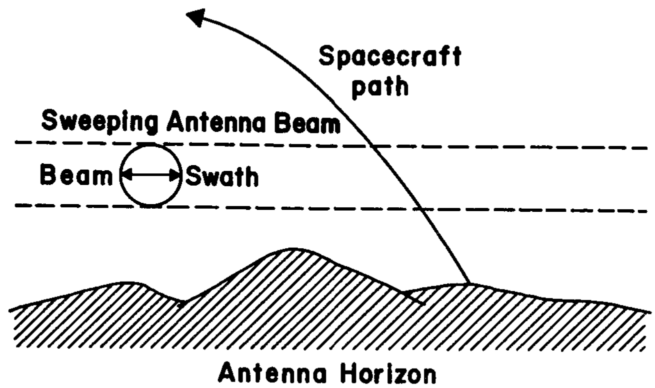
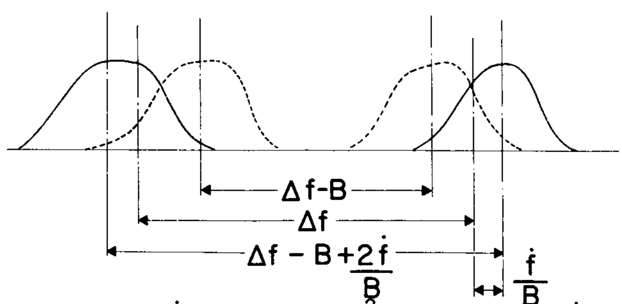


FIG.8

Receiver Tuning Rate and Bandwidth



$$\dot{f} \geq \frac{1}{T}\left(\Delta f - B + 2\frac{\dot{f}}{B}\right) \text{ or } \dot{f} \geq \frac{B\Delta f - B^2}{BT-2} \text{ provided that } \frac{\dot{f}}{B} \leq B$$

T = Time for antenna beam (3 db) to scan past signal (sec.)

Δf = Frequency uncertainty or search bandwidth (cps)

$\dot{f}$ = Tuning search rate (cps/sec.)

B = Acquisition bandwidth (cps)

GEMINI LAUNCH ESCAPE

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INTRODUCTION

The Gemini launch escape modes have been tailored to the design and dynamic characteristics of the launch vehicle and spacecraft. Before the pilot's role is discussed with respect to the operation of the escape system, a brief review of the spacecraft and the launch vehicle configuration will show why it has been possible and desirable to include a flight crew in the abort decision loop.

GEMINI LAUNCH SEQUENCE SYSTEM

The Gemini spacecraft incorporates a manual sequence system to control the major spacecraft sequences and system operations. This concept is in contrast to the Mercury spacecraft design where unmanned orbital flights required an automatic sequencing system involving numerous timers and interlocks.

The launch vehicle for Gemini is the standard Titan II modified to attain increased mission reliability and pilot safety. As shown in figure 1, redundant guidance, autopilot, and first stage hydraulic control systems have been provided. A redundant hydraulic control system is standard on most modern aircraft. This Gemini backup system can be triggered manually or automatically.

A primary area of concern in manned launch vehicles is dynamic behavior in the event of guidance or hydraulic malfunctions in the high dynamic-pressure flight regime. Aerodynamic instability will cause the Gemini launch vehicle to diverge to breakup attitudes within 1 second when control failure occurs at maximum dynamic pressure. Because of the need for immediate switching under this condition, the backup guidance and control system is automatically triggered. Automatic switching is accomplished by abnormal rate gyro signals, full-engine gimbal position, and low-hydraulic pressure. The attitude rate switching level for the first stage is 3.5 degrees per second in pitch and yaw and 20 degrees per second in roll. During second-stage flight, where no aerodynamic divergence is involved, the levels are opened up to 10 degrees per second in pitch and yaw. Manual switchover will be initiated if guidance or control malfunctions cause a slow divergence which is sensed by the pilot or by ground tracking. It is evident therefore that design action has been taken to reduce the possibility of catastrophe due to guidance or control failures.

Based on a thorough failure analysis by Martin-Marietta Corp., Aerojet, Aerospace Corp., and the National Aeronautics and Space Administration, additional malfunction sensors are incorporated. The parameters which are sensed in the launch vehicle and displayed in the spacecraft are as follows:

1. Fuel-tank pressure (Stages I and II)
2. Oxidizer-tank pressure (Stages I and II)
3. Engine-chamber pressure (Stages I and II)
4. Primary-guidance or control failure
5. Excessive rates
6. Staging signal

Stage tank pressures are critical because of structural and pump suction head requirements. Figure 2 shows various pressure time histories for the Stage I fuel tank. The two top curves indicate normal excursion envelopes of gas ullage pressure. The fuel tank is pressurized by fuel-rich exhaust products from the turbo pump. The structural threshold curve starts out at ambient pressure at liftoff, increases to about 3 psi above ambient pressure at 60 seconds (maximum dynamic pressure) and increases beyond 90 seconds because of tank stresses due to increasing axial acceleration. Pump suction head limits in terms of tank ullage pressure are shown. At periods greater than 70 seconds, pump head requirements are more critical than the structural requirements. Superimposed on the plot shown in figure 2 are tank pressure histories resulting from two kinds of malfunctions: a broken autogenous pressurizing line and an ullage leak of 2 square inches. From the pilot's standpoint, the critical time for these types of failures is near liftoff when ullage volume is small and ullage pressure decreases rapidly. Tank pressure requirements as a function of time are very nonlinear. This type of parameter can best be monitored manually. If an automatic malfunction sensor were designed to follow these pressure curves, it would be nonlinear and complex.

Figure 3 shows the launch vehicle displays in the spacecraft. The tank pressures are displayed on vertical meters with two indicators per tank. These meters have two secondary uses. If the left set of needles is rapidly driven upward full scale, it is an indication that the primary guidance power system has failed. When all four of the Stage I tank pressure indicators peg out, it is an indication of physical staging. The numbers in the center of the Stage I display are time markings which indicate the structural limits of the tanks as a function of time. The crosshatched area in the lower portion of the figure indicates to the pilot the pump head pressure requirement. The small triangle-shaped symbol in the Stage II display indicates the minimum pressure for Stage II engine start.

Engine chamber-pressure lights above the tank pressure indicators are activated if either Stage I or Stage II chamber pressures drop below 65 percent. Consequently, the Stage II engine light is on during Stage I flight and is extinguished at staging. One might rationlize that physical acceleration cues would negate the need for an engine status indication. However, partial losses in thrust are not always immediately detectable; consequently, a visual indication of chamber pressure is desirable. The Titan II staging sequence involves a simple fire-in-the-hole technique where the second stage is ignited before separation from Stage I and before Stage I thrust has completely decayed. This procedure eliminates the need for ullage rockets.

The staging light is illuminated by the staging signal and is extinguished 1 second later when physical separation occurs.

The rate light is triggered by signals from the rate gyros in the launch vehicle. The rate thresholds are the same as those which trigger the backup guidance system. The launch vehicle gyro outputs are filtered to reduce their response to the launch vehicle natural bending modes. This discrete indication of launch vehicle overrate can be cross-checked with rates which are measured in the spacecraft and displayed on analog needles superimposed on the eight-ball attitude indicator.

The guidance light illuminates when the backup guidance and control system is selected.

The digital timer located near the launch vehicle displays is very useful in correlating launch vehicle events.

Figure 4 shows the launch acceleration and dynamic pressure time histories. Also indicated are the three Gemini launch escape modes. Ejection seats are used from the launch pad to an altitude of 70,000 feet. Ejection seats are feasible for Gemini because of the low order pressure wave which originates from burning hypergolic propellants as in Titan II. Above 70,000 feet, the spacecraft drag has reduced sufficiently to permit separation of the spacecraft by salvo fire of the retro-rockets. For this escape mode, the top section of the adapter is retained and the resulting configuration (fig. 5) is aerodynamically stable, small end forward. The adapter section is separated at apogee of the escape trajectory. After staging, when dynamic pressure is negligible, the escape mode involves shutting down the booster and separating with the translational rendezvous propulsion system.

The escape hatches and the ejection seats are triggered by the actuation of either pilot's D-ring located on the forward portion of the seat. The launch vehicle is shut down and the retro-rockets are fired by a control on the left console. The maneuver rockets are fired by a translational control handle located just above the pilot's left knee. Figure 6 shows the location of these controls in the spacecraft.

LAUNCH SIMULATION PROBLEM

A launch simulation was conducted to evaluate the launch vehicle displays and to confirm that the crew could assess the status of the launch vehicle and take abort action if required. The simulator chosen for this study was the moving base aerospace flight simulator at Ling-Temco-Vought, Inc. A photograph of the simulator is shown in figure 7.

Three degrees of angular freedom were available in the moving base cockpit with adequate displacement and washout capabilities to simulate small perturbations of the normal vehicle accelerations. In addition, a pitch rotation of ±100 degrees from the horizontal permitted a partial simulation of the direction and magnitude (up to 1 g) of axial accelerations. The cockpit motions were accomplished with hydraulic servo mechanisms driven by analog signals. The cockpit was rotated in pitch to 57 degrees from the horizontal for the launch position. At liftoff the cockpit was rotated from 57 degrees to 75 degrees, producing the sensation of lift-off thrust to the pilot. The cockpit then continued to rotate up to 90 degrees, representing the first few seconds of acceleration after launch. For abrupt changes in axial acceleration, such as staging and loss of thrust, the cockpit was rotated rapidly downward.

The combination of engine and aerodynamic noise was simulated by a high fidelity speaker system located in the astrodome surrounding the cockpit. The noise spectrum was reproduced from an actual cockpit recording obtained during a Mercury-Atlas launch. The maximum intensity level inside the closed cockpit near the pilot's head was 104 decibels which occurred at maximum dynamic pressure. Corresponding noise deviations were programed as applicable for each simulated malfunction.

The simulation program involved 51 malfunction runs representing nine major types of malfunctions which are as follows:

1. Partial loss of thrust - one engine (Stage I)
2. Total loss of thrust - one engine (Stage I)
3. Total loss of thrust - both engines (Stage I)
4. Staging failures
5. Tank (fuel and oxidizer) pressure losses
6. Roll malfunction (Stage I)
7. Direct-current power failure
8. Instrument malfunction
9. Display-light failure

The selected malfunctions were based on failure analysis data for the Titan II launch vehicle. Personnel from NASA and Ling-Temco-Vought, Inc., established the number of runs for each type of malfunction, and the time that the malfunctions were to begin. The selection was based on the criticality of the malfunctions with respect to anticipated pilot difficulty in detecting and evaluating the cues and the response time for taking corrective action. Normal launch vehicle runs were interspersed throughout the simulation. The most difficult malfunction runs were selected for use in the simulation regardless of their probability of occurrence in actual flight.

The NASA astronauts who participated in the simulation were given only 1 day of indoctrination. Each pilot was scheduled for approximately 75 runs, 65 runs having malfunctions and 10 being normal.

Each of the 51 malfunction runs was presented to the pilots at least once, and the 14 most difficult runs were presented twice to each pilot. The runs were randomly distributed so that the pilots had no way of knowing which problem would be presented next. They were aware of the general nature of the possible malfunctions, but they were not aware of the time during flight at which the malfunctions were programed.

A digital computer controlled simulator attitudes, vibration, and the magnetic tapes. With the exception of pilot action which completed the runs, the simulator was operated in an open loop configuration. In addition to the launch-vehicle-related displays, the cockpit contained a D-ring ejection seat handle, launch vehicle shutdown and spacecraft abort handle, and a secondary guidance switch. The pilot's control response to each run was recorded, as was his verbal assessment of each run.

It became readily apparent to the pilots that the most critical malfunctions were engine failures or tank pressure losses immediately after liftoff or immediately after staging. The critical engine failures were readily detectable through redundant cues, including decrease in sound level, decrease in acceleration, and illumination of the combustion chamber pressure light. Pilot reaction time to this failure was as low as four-tenths of a second. Reaction time requirements varied from approximately 1 second to $2\frac{1}{2}$ minutes, depending upon the type and time of malfunction. Several of the malfunctions, such as sensor failures and gradual tank pressure losses were noncritical and required no abort action. For the majority of the failure modes, there were multiple cues such as is the case with engine failure. Tank pressure losses were sensed by redundant transducers driven by redundant power sources and presented on redundant meters. For tank pressure failures which occurred after the first 5 seconds, the rate of decay was relatively slow. The pilots were able to let several pressure failures decay parallel to and just above the structural limit or were able to wait until the pressure dropped to within 1 psi of the structural margin before taking abort action.

With only 1 day of familiarization and with partially developed displays, the pilots were able to analyze and react correctly to the critical malfunctions. Extreme monitoring accuracy should be possible after instrument development has been completed and the pilots have received the intensive familiarization and training which will precede a manned Gemini flight.

A manual abort system will provide added operational flexibility by enabling the flight crew to choose an abort time which may reduce the possibility of aborts at high dynamic pressure; to choose optimum abort times compatible with contingency recovery areas; and to reduce the probability and the risk involved with an inadvertent abort.

There is good analogy here to aircraft operations where the pilot by using his flight instruments, engine displays, and physical cues is able to assess accurately the validity and seriousness of various warning or malfunction indications.

SUMMARY

In summary, Gemini mission reliability and crew safety have been enhanced by incorporating a redundant guidance and control system and a flexible launch vehicle monitoring system.

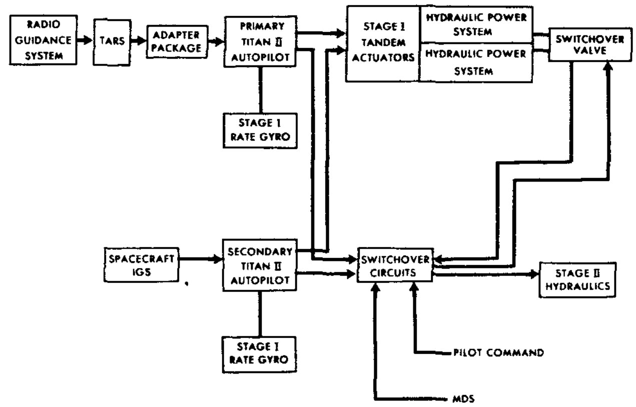

Figure 1.- Flight control and guidance system

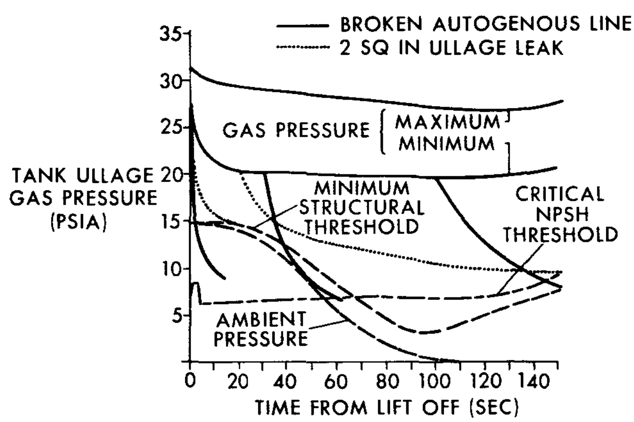

Figure 2.- Stage I fuel tank pressures

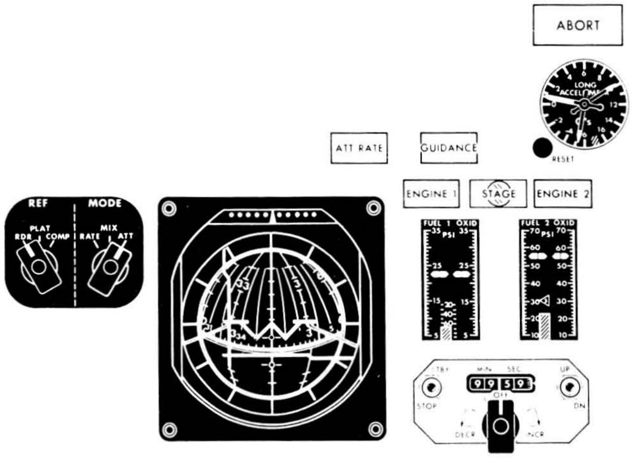

Figure 3.- Booster monitoring displays

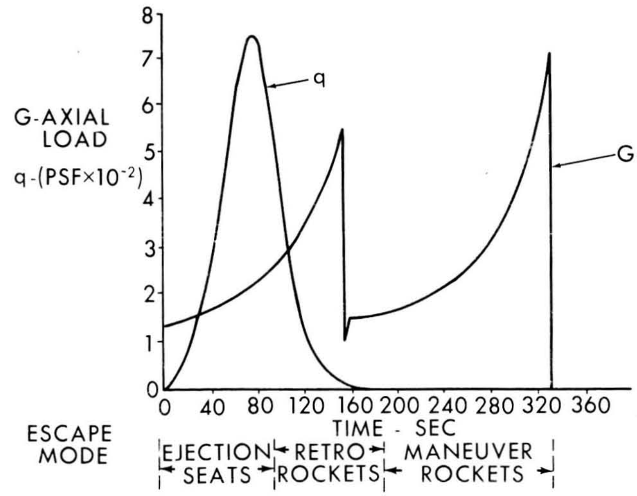

Figure 4.- Gemini launch parameters

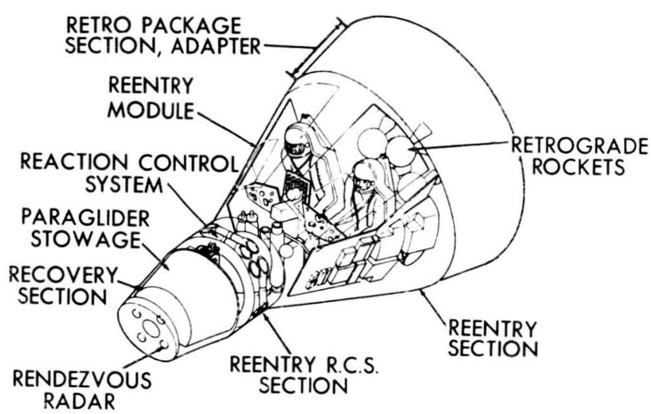

Figure 5.- Gemini retro-escape configuration

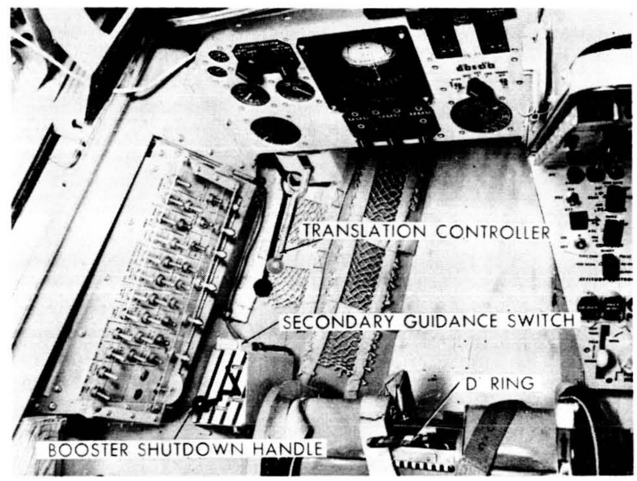

Figure 6.- Escape controls

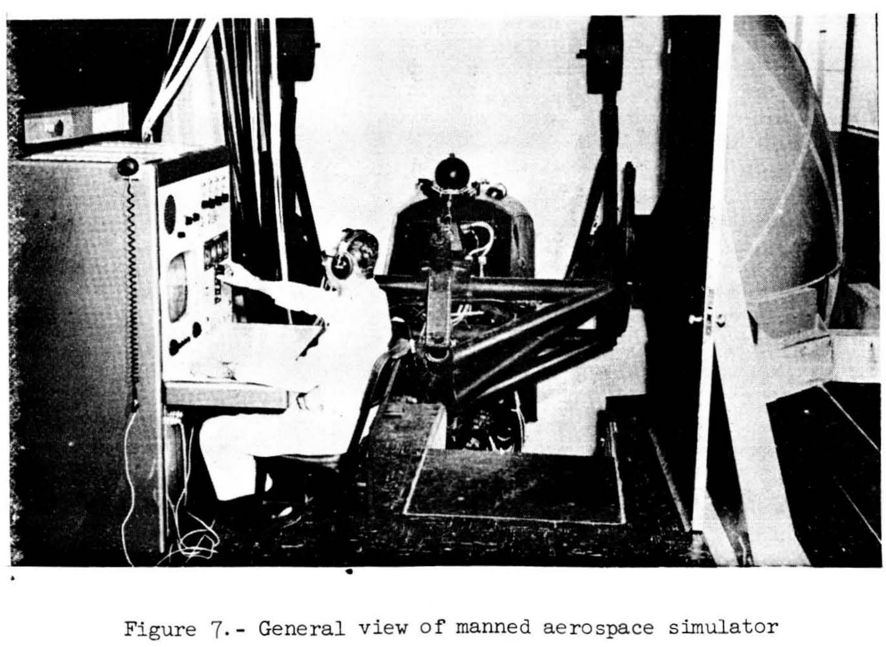

Figure 7.- General view of manned aerospace simulator

A MANUAL ABORT TECHNIQUE FOR THE MIDCOURSE REGION OF A LUNAR MISSION

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Introduction

Although manned space flights require a very high probability of mission success they must have an even higher probability of crew survival. This emphasis on crew survival leads to requirements for abort techniques and subsystems that enable safe return from all phases of the mission in any event other than a catastrophy.

The phases of space flight in which emergencies requiring abort are most likely to occur depend on the mission. Since for a lunar mission, the vehicle will be in the midcourse region for over 75 percent of the flight time, it is essential to study the problem of abort during this phase.

The object of the paper is to describe an abort technique which will require a minimum of on-board equipment and to illustrate the performance of the scheme by presenting the results of a digital simulation of the system.

Discussion of the Problem

Consider a manned spacecraft on a typical lunar mission encountering an emergency that necessitates an immediate return to Earth. Let us assume that the emergency was caused by a failure in both the communications link and the on-board computer, so that all of the necessary calculations for the abort must be made either on board without the computer, or else prior to the emergency. Such an abort problem can be considered in three parts.

First, the state vector at the time of the abort maneuver must be predicted.

Second, a safe return trajectory must be determined within the existing constraints, that is, landing site restrictions and available change in velocity.

Third, the vehicle must be oriented properly and the abort maneuver performed so that the desired return trajectory is achieved.

One method of predicting the state vector after an emergency has developed is to make a series of celestial observations and process them in some fashion that will best estimate the current position and velocity. These results can then be updated to the abort time. Several such schemes exist but generally they either require a complex computer to process the results, or they are so simplified that they have marginal accuracy and cannot be used near the moon. Furthermore, sufficient time may not exist to make the necessary celestial observations to determine the state vector accurately. Thus, consideration must be given to the idea of calculating the state vector of the vehicle in advance to minimize the pilot's task in the event of an emergency. If the state vector is predicted for some time in the future, this information gained prior to the emergency can be utilized to give a reasonably accurate prediction. This leads naturally to restricting the number of points on the trajectory from which aborts will be considered. These points are termed abort way stations, and should occur frequently enough to handle all the possible aborts.

If the best estimate of the state vector is routinely computed on board for the next one or two way stations, in the event of computer failure the state vector would be known at those future points and sufficient time would exist for abort calculations and vehicle maneuvering. The abort calculations would be based on trajectories tabulated before the flight for each of the way stations. Since these trajectories are computed before the flight, they can represent the exact physical situation with the inclusion of the earth's oblateness and the perturbing bodies such as the moon, sun, etc. The scheme proposed in this paper uses precalculated charts computed in this fashion.

Once the abort velocity increment is computed, the vehicle must be oriented in the proper direction and the abort executed. Errors in thrusting during the abort maneuver can cause large deviations in the return trajectory and must, therefore, be corrected with subsequent maneuvers (cf. ref. 1). The present study assumes that the midcourse navigation system is not operable so an alternate method of computing subsequent corrections is required.

If an inertial measurement unit were used to monitor the abort maneuver, thrusting errors during the abort could be measured and corrected immediately afterwards with a vernier engine that can be controlled more accurately. Since any good quality inertial measurement unit can measure the magnitude of the velocity increments more accurately than they can be applied, the required vernier correction is known quite accurately immediately after the abort maneuver has been executed. It will be shown that the residual errors from all the sources considered are small enough to allow a safe entry. Before these errors are examined, a description of the proposed system and the mathematical model will be given.

Description of the Proposed System

The proposed system consists of precomputed charts and an inertial measurement unit with an optical sighting device which can be used to aline two of the axes in the desired directions. It is assumed that the primary navigation system employs an estimation process similar to the one described by McLean[2], and that the system is programmed to give, for preselected ranges, the time to go and the velocity components. In the event of a failure this system provides estimates of these quantities at two future positions.

Abort Charts

The abort charts consist of velocity hodographs plotted for each reference trajectory. Typical hodograph plots are shown in figure 1. The radial velocity is plotted along the abscissa and the horizontal velocity is plotted along the ordinate. The dashed line represents the velocity history of the reference trajectory, and points corresponding to specific ranges are marked. The solid lines represent all of the achievable in-plane safe return trajectories at the respective ranges shown. For example, at the range of 100,000 kilometers, the vehicle would have the velocity represented by the solid circle on the dashed line if it were on the reference trajectory. The vector joining that point with any point on the solid line would represent the abort velocity increment at that range. The further the abort velocity increment extends to the left along the return velocity curve, the shorter will be the return flight time. Since the fuel on board the vehicle will be limited, the curves are terminated on the left. The terminal point on the right of each solid line represents the minimum practical abort velocity increment required to return the vehicle to a safe entry. If some velocity on the solid line results in a return to one of the desired landing sites, the velocity is termed a "landing site point" and the corresponding point on the reference trajectory is an abort way station.

The difference between the maximum and minimum flight time can be several days; therefore, only the points toward the left end of the curves should be of interest if return time were an important consideration. Thus, abort way stations can be located at ranges where the landing site point occurs at the left portion of the curve. If three landing sites are considered, abort way stations can be found with nearly minimum flight times which are separated by time intervals of four to five hours along the reference trajectory[3].

Restricting the aborts to in-plane maneuvers limits the number of solutions that return to a particular landing latitude; however, it also reduces the number of charts necessary to describe all possible aborts. Furthermore, in-plane aborts result in a constant entry track which is desirable for ground tracking during entry. Thus, this paper is limited to in-plane abort maneuvers; however, the proposed technique can easily be extended to consider out-of-plane aborts.

When an emergency of the type considered arises, the pilot quickly determines which abort way station he is approaching and selects the proper charts; for example, the one shown in figure 2. As in figure 1, the solid curve represents all of the achievable safe in-plane aborts for a vehicle at a given range. In general, however, the vehicle will not be on the reference trajectory and the pilot will use the best estimate of his radial and horizontal components of velocity as given by the computer before the failure.

The pilot will choose the landing site he prefers, taking return flight time into consideration, and calculate the abort velocity increment by merely subtracting as indicated by the two equations

$$\left.\begin{aligned} \Delta V_R &= V_{R_{Hodograph}} - V_{R_{Estimated}} \\ \Delta V_H &= V_{H_{Hodograph}} - V_{H_{Estimated}} \end{aligned}\right\} \quad (1)$$

The hodograph velocities for various specified landing sites can be given in tabular form to any desired accuracy as illustrated by the table in figure 2.

After the abort velocity increment is computed the vehicle is maneuvered to the proper attitude so that the abort can be executed at the time the vehicle reaches the way station.

Inertial Measurement Unit (IMU)

Since the thrust vector is predominately radial, a relatively simple optical device can be used to aline the vehicle and the inertial measurement unit. Figure 3 illustrates how the abort maneuver would be accomplished. The angle θ is the thrusting angle and is computed from:

$$\theta = \tan^{-1} \frac{\Delta V_H}{\Delta V_R} \quad (2)$$

The IMU would be alined with one axis pointing toward the center of the earth and a second axis lying in the orbital plane. The orbital plane can be established by some reference star or by observing the earth's track in the star background. The third axis is normal to the orbital plane and for the planar aborts the velocity in this direction would be held to zero. Since the IMU can be alined quite accurately, errors in thrusting during the abort maneuver would be measured and corrected subsequently with the vernier engines.

Assumptions

The analyses in this paper are based upon the following assumptions:

1. That the velocity corrections, including the abort maneuvers, are impulsive.

2. That the actual trajectory of the space vehicle can be accurately described in terms of the linear perturbations about a precalculated reference trajectory.

3. That the errors considered are random and gaussian and can be represented accurately with second-order statistics.

Performance Criteria

In evaluating the performance of the abort system some criteria must be stated. Since the prime objective after an abort is to return to earth within the entry corridor, the deviations from the center of the corridor have been established as a measure of the performance of the system. It is convenient to express the center of the corridor in terms of the conic vacuum perigee, and the deviations from the center of the corridor as deviations in perigee altitude. Thus a satisfactory entry was assumed to result if the perigee of the return trajectory was 6430.00 ±30 km.

To evaluate the performance of the proposed system the covariance matrix of the errors in the state vector was determined after the vernier correction and the miss at perigee caused by these errors were examined.

The errors that are considered are:

1. Errors in the knowledge of position and velocity at the time of the abort

2. Errors in thrusting during the abort maneuver

3. Errors in making the vernier correction

4. Error caused by delay in accomplishing the complete velocity increment

Position and Velocity Errors

The errors in the knowledge of position and velocity at the time of the abort are functions primarily of the navigation system. In this study a navigation system similar to that described by McLean[2] was assumed. The covariance matrix of errors in estimation resulting from this system was obtained at each of the abort way stations along the reference trajectory. For each way station a reference abort trajectory was found and a transition matrix along this new reference trajectory was used to update the covariance matrix of errors in estimation from the abort time to the time of the vernier correction. A prediction matrix was then utilized to compute the rms miss at perigee resulting from these errors.

Thrusting Errors

The thrusting errors considered were errors in aiming and in cutoff. As mentioned earlier, abort thrusting errors are measured with the IMU and subsequently a vernier correction is made. If the vernier correction were made at the same time as the abort, the only errors resulting from the abort maneuver would be measurement errors and vernier thrusting errors. This is illustrated by the velocity diagram shown in figure 4. The true vernier correction is by definition the abort thrusting error. However, because of the measurement errors, the indicated vernier correction is slightly different. Furthermore, the actual vernier correction will be different from the indicated vernier correction because of vernier thrusting errors. The total error is then the vector sum of the measurement error and vernier thrusting error. Since the measurement errors are small and analogous to vernier thrusting errors, in this analysis they are considered as part of the vernier thrusting errors. The intended vernier correction is the indicated error resulting from the abort maneuver, thus the statistics of the vernier correction are described by the covariance matrix of abort errors. These statistics were used in the matrix analysis illustrated by Battin[4] to determine the covariance matrix of errors in vernier thrusting. The prediction matrix was then utilized to compute the miss at perigee due to errors in vernier thrusting.

If the indicated vernier correction is not applied at the same instant as the abort maneuver, a third error will result since the on-board computer is not available for updating the indicated vernier correction to the actual time of its implementation. The covariance matrix of errors due to the delay was also computed, and the miss at perigee due to this error was obtained using the prediction matrix.

Numerical Example

To illustrate the performance of the abort technique the results of a digital simulation of the system will be given. The simulation model is described below.

Trajectory

The reference trajectory that was selected is entirely ballistic and lies approximately in the moon's orbital plane. Injection occurs at a perigee of 6450.0 km on February 21, 1966, with a velocity of 11.015858 km/sec. The vehicle passes ahead of the moon and reaches perilune at a lunar altitude of 185.0 km after a flight time of 70.68 hours. The moon's gravitational attraction rotates the direction of flight so that the vehicle returns to earth and enters the atmosphere in the center of the entry corridor with a flight-path angle of -5.5°. The total flight time from injection to the return reference perigee is 144.88 hours.

Observation Schedule

The primary navigation system assumed was similar to that described by McLean[2]. This particular example uses 45 observations with three velocity corrections on the outbound leg and 35 observations and two velocity corrections on the return leg. The observation sequence prior to the first velocity correction is shown in detail in figure 5. Similar sequences were followed preceding each subsequent correction.

Abort Way Stations

For in-plane aborts the band of latitudes from which a landing site can be selected is restricted and depends on several characteristics of the reference trajectory (cf. ref. 3). These are:

1. Inclination of the reference orbit

2. The entry range capability of the vehicle

3. The abort velocity increment capability of the vehicle

The ability of the system to achieve a safe entry will be independent of the first two parameters. Representative values were chosen for the example below, where the entry range is taken as approximately 3000 nautical miles and the inclination of the translunar trajectory is approximately 28.6°. The maximum abort velocity considered in the example was 2.0 km/sec.

Abort way stations were selected at nearly equal time intervals along the reference trajectory and the landing sites that resulted from small variations of abort velocity were determined. It was found that if three landing sites were selected, near minimum return time trajectories could be found at all the way stations selected. A summary of the abort way stations, their spacing, their corresponding landing sites, and the velocity increment necessary to achieve the landing site are given in table I.

Although for this study it was more expedient to select the way stations and then determine the landing sites, in general this procedure would be reversed. That is, desired landing sites would be selected and then the abort way stations would be determined.

Emergency Stations

It is not necessary to examine every point on the reference trajectory to evaluate the abort performance. It is only necessary to examine aborts from those points known to give "poorer" results. These less satisfactory cases will occur at points where the covariance matrix of errors in estimating the state vector is greatest. This generally occurs before a sequence of observations. Figure 6 shows an enlarged portion of the trajectory with the observation schedule and abort way stations. The solid circles numbered 1 through 4 represent the first four way stations and the stars represent emergencies at the "critical" points. For example, the first case to examine occurs after injection and before the first observation. Any emergency occurring after injection and before the first way station was approached would result in an abort at the first way station, and the uncertainty in the estimation of the injection errors would be the major source of error in the perigee miss. The covariance matrix of injection errors updated to the first abort represents the errors of estimation at that point. If an emergency developed very near the first abort way station, sufficient time would not exist to perform calculations and maneuver the vehicle; therefore the abort would have to be made at the second way station if a desired landing area were to be reached. However, none of the scheduled observations could be made, and thus the error matrix at injection must be updated to the second abort way station to determine the error in estimation of the state vector. In this study it was assumed that 15 minutes is required to prepare for an abort maneuver; thus if the emergency occurred within 15 minutes of an abort way station, the abort could not be made until the following station.

If the emergency were to occur after the first observation, an abort would be made at the second abort way station and the subsequent observations shown could not be made. Thus the error matrix would be that obtained after the first observation updated to the second way station. If the miss at perigee due to error in the estimated state vector were small for the case just mentioned, emergencies after subsequent observations would certainly result in correspondingly smaller misses because each observation would further reduce the error in the estimated state.

Another type of "critical" point would occur if an emergency arose at the time of the first velocity correction. Such an emergency would result in an abort at way station 4 since there would be sufficient time to make the necessary computations.

This same idea is followed throughout the remaining leg of the trajectory. Table II defines the emergency points that were considered and the way station where the abort was initiated for that emergency.

Results

The rms errors in perigee miss due to each of the errors considered were computed for each of the emergencies discussed and the results are presented in table III.

The rms perigee miss due to the errors in the knowledge of position and velocity at the time of the abort is tabulated in column 2. As expected the results are poorest before any observations are taken. Indeed, point 2, which is the worst condition, represents the effect of an emergency after injection that could not be corrected until the second way station. An abort at the same way station with only one observation, emergency 3, results in a great reduction in the miss at perigee. This illustrates the importance of the observation schedule in relation to the abort way stations.

If the abort maneuver were executed and no subsequent corrections were made, extremely large perigee errors would result from the small errors in thrusting (cf. ref. 1). Consequently, a vernier correction would be necessary. The statistics of the vernier correction depend on the statistics of the abort errors. The covariance matrix of errors resulting from the abort maneuver was computed with the assumption of an rms error in aiming of 0.5° and an rms error in cutoff of 1.0 percent. These assumptions result in vernier corrections of about 20 to 25 m/sec rms.

The end-point errors are not directly affected by the errors in the abort maneuver but only by those in applying the vernier correction. The assumption for the vernier errors was 0.5° rms in aiming and 0.2 m/sec rms in cutoff. Also, an error arises from the delay in applying the vernier correction since the on-board computer is presumed not available for updating the indicated vernier correction to the actual time of its implementation. The effects on terminal conditions due to these two sources of error are given in columns 3 and 4 of table III. The vernier thrusting delay time was taken as 15 minutes. Because the trajectory sensitivity coefficients generally increase as the time-to-go increases, the miss due to vernier thrusting errors is larger for the later aborts. The miss due to delay in applying the vernier is a function of the position on the trajectory as well as the delay time. Near the earth, where the curvature of the trajectory is greater, the effect of the delay is largest.

The total miss at perigee resulting from all three sources is tabulated in column 5. It is interesting to note that the error resulting from an abort at the last way station considered is only 10 km. This way station is well within the sphere of influence of the moon. Aborts on the outbound leg of the trajectory at ranges beyond the last way station are of little value since the return time without an abort is only slightly longer.

The 3 sigma miss at perigee due to all of the errors considered is within the allowable error of

±30 km for all but emergency condition 2. This error could be reduced to an acceptable value if the observation schedule were altered so that a better estimate of position and velocity could be obtained prior to the first way station.

Concluding Remarks

A manual abort system consisting of precomputed charts and an inertial measurement unit has been described. The system was evaluated by simulation on the digital computer and the results were given.

The results of this study show that the use of precalculated charts to aid in the computation of the abort velocity increment is feasible. The incorporation of abort way stations makes possible the location of desired landing sites on the charts. The error in estimating the state vector at the abort way stations is dependent on the observation schedule used; thus the location of the way stations should be considered in choosing the observation schedule.

Since the abort and vernier velocity corrections can be measured to a higher degree of accuracy than they can be made, the performance of the system is limited by the aiming accuracy of the IMU and the cutoff characteristics of the vernier engine. A substantial improvement in the system would be realized for ranges of 200,000 km or more if these were improved.

In defining the landing site, a constant entry range was assumed. Actually, the vehicles being considered for the lunar mission have some aerodynamic capability and can achieve a fairly large "footprint." Since the errors in making the abort will cause deviations from the reference, the landing site could not, in general, be achieved with a constant entry range. Thus, the aerodynamic capability of the vehicle would be required to reach the desired landing site.

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TABLE I.- DATA CONCERNING THE WAY STATIONS

Way station	Range, km	Time, injection to perigee, hr	Abort velocity increment, km/sec	Landing site
1	40,000	29.30	1.71	1
2	90,000	32.44	1.63	2
3	125,000	33.19	1.78	1
4	155,000	49.72	1.73	3
5	180,000	55.69	1.81	2
6	205,000	63.59	1.81	1
7	230,000	73.75	1.74	3
8	250,000	79.91	1.76	2
9	270,000	87.55	1.74	1
10	290,000	97.84	1.65	3
11	308,000	97.58	1.88	3
12	325,000	106.62	1.90	2
13	340,000	111.66	1.84	1
14	355,000	121.61	1.74	3

Landing sites:

1. Coast of India, East of the city of Cuttack.
2. Hawaiian Islands, near city of Honolulu.
3. Puerto Rico, near city of San Juan.

TABLE II.- DEFINITION OF THE EMERGENCY CONDITIONS

Emergency condition	Time from injection, hr	Range, km	Number of observations	Way station
1	0	6,500	0	1
2	2	39,200	0	2
3	2.5	46,000	1 Earth	2
4	6.5	90,500	8 Earth	3
5	12.0	136,000	9 E, 8 M	4
6	14.5	153,000	9 E, 8 M	5
7	18.5	180,000	10 E, 8 M	6
8	22.5	202,000	15 E, 8 M	7
9	27.5	230,000	19 E, 10 M	8
10	34.5	262,000	19 E, 10 M	9
11	35.5	270,000	19 E, 10 M	10
12	40.0	290,000	19 E, 10 M	11
13	44.5	308,000	19 E, 10 M	12
14	49.0	325,000	19 E, 10 M	13
15	53.0	339,000	19 E, 10 M	14

TABLE III.- ROOT-MEAN-SQUARE PERIGEE MISS RESULTING FROM THE ERRORS CONSIDERED

Emergency condition	RMS perigee miss, in kilometers, due to errors in -			Total rms miss, km
	Knowledge of state vector	Vernier thrusting	Delay in thrusting	
1	6.1	1.2	6.5	9.1
2	21.0	2.3	2.8	21.3
3	5.8	2.3	2.8	6.8
4	3.0	3.3	1.8	4.9
5	3.5	4.0	1.4	5.5
6	5.0	4.8	1.2	7.0
7	6.6	5.4	1.2	8.6
8	4.0	5.8	1.1	7.1
9	2.2	6.4	1.3	6.8
10	1.8	6.7	1.3	7.1
11	2.0	6.9	1.4	7.3
12	2.4	8.2	2.2	8.8
13	2.7	8.7	2.7	9.5
14	3.0	8.9	3.1	9.9
15	3.5	9.0	2.6	10.0

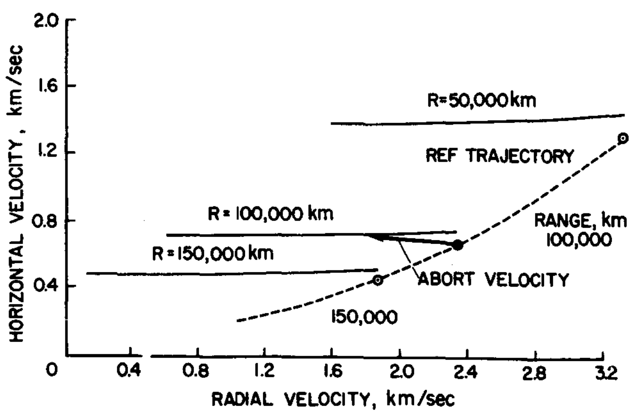

Figure 1. - Typical abort hodograph charts.

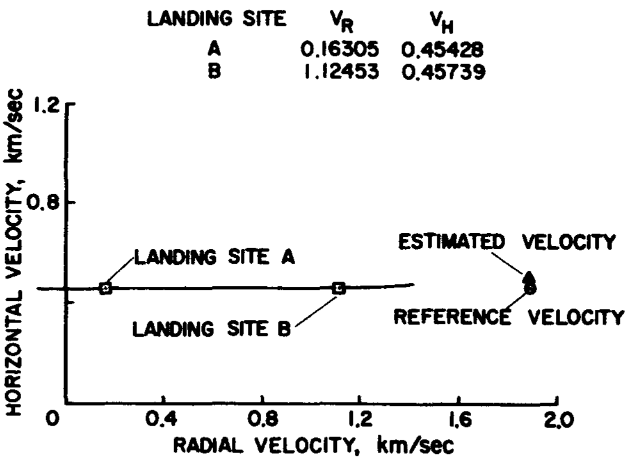

Figure 2. - Abort chart for way station 4.

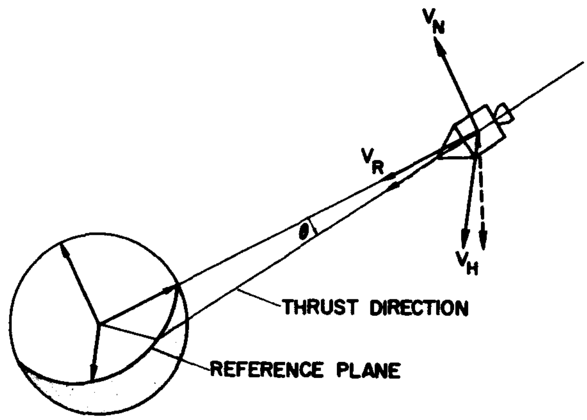

Figure 3. - Alining the IMU and the thrust vector for an abort.

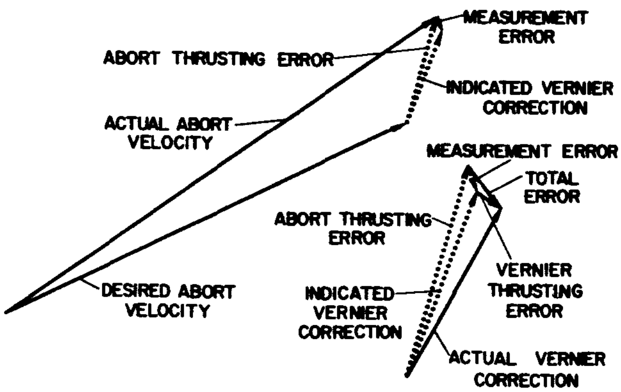

Figure 4. - Velocity diagram.

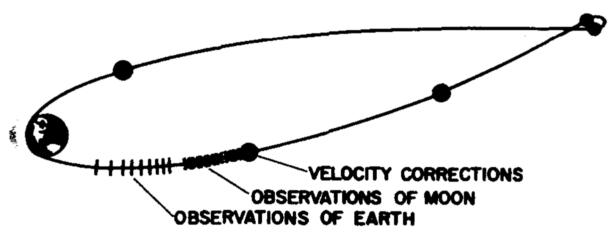

Figure 5. - Lunar trajectory observation schedule.

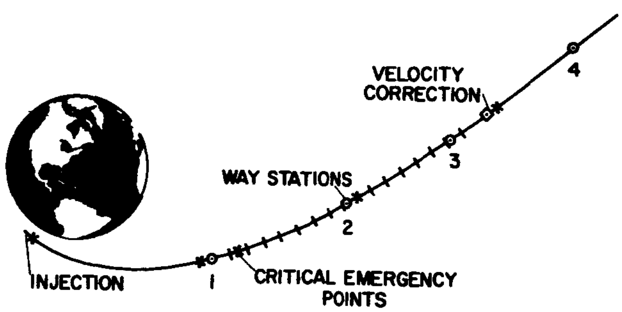

Figure 6. - Location of critical emergency points.

SPACE STATIONS AND THE NATIONAL SPACE PROGRAM --- A CHALLENGE

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The combination of two great organizations into a single national AIAA, the title of this conference --- "Manned Space Flight" --- and the theme of today's session --- essentially, "Where do we go from here?" --- reflect an understanding of space as a single national effort, our awareness of the fundamental importance of man in this effort, and our concern as to the proper evolution of the national space program.

Against this background, it is most appropriate to discuss space station activities as an element of our national space program. Specifically, I plan to emphasize certain aspects of the space station program which, in my judgment, are not adequately considered. Further, I will suggest that such aspects pose special challenges and that we all --- individually and collectively --- have a responsibility to meet these challenges.

Perspective

The first aspect is Perspective. Our grasp of Perspective is, at best, an imperfect thing. Without Perspective, it is possible to encourage programs of modest improvement with resultant demands on our national resources and, without realizing it, take a series of small steps at the expense of accomplishing really fundamental space milestones and realizing national objectives.

Each person in a policy or decision-making role --- not only in Government but in the private sectors of the economy as well --- should have developed in his own mind the fundamental milestones and objectives which constitute our national space program. Only by testing each new program --- such as a space station --- against this picture can an effective judgment be made which is responsive to national space program needs rather than the parochial interests of a given organization or industrial element.

The picture, for example might be divided into the fundamentals of two types of mission --- manned and unmanned; three types of mission regimes --- earth orbit, lunar and deep space; and four types of supporting technological effort:

Transportation Systems --- the ability to put things up and move around in space.

Reentry Technology --- the ability to return.

Operational Technology --- the ability to operate effectively in space.

Bioastronautics --- the science of man's well-being in space.

From a Perspective such as this, it is possible to develop the fundamentals, milestones, sequence of events and their interrelationship for our national space program.

The point is that regardless of whether this or some similar technique is used, the decision maker --- Government or industry --- should have a master picture of this type that identifies the evolving nature of our national space program and its mainstreams of activities. This Perspective, reflecting a national program stripped of its trivia and reduced to fundamentals, can make an effective contribution to the key management decisions of resource application and major program starts.

Another example of Perspective lies in the manner in which we emphasize the various aspects of our activities. Surely, a viable national space program should have ever-changing shadings, peaks and valleys, tones and overtones to be a total and meaningful picture. If we lose our Perspective, it is easy to accede to the momentum of large organizations and channeled thinking. This can continue to carry us into areas which originally held great promise, but where --- in fact --- we should be reducing our effort and redirecting it to new regimes of interest.

For example, the next five years will see several hundred satellites in earth orbit. In that same period, there might be thirty or so in the cislunar regime and less than a score in deep space. The unknown, however, is pretty much in direct proportion to distance away from our planet. Maybe the momentum of our early earth orbital activities is carrying us forward at the expense of our newer deep space regimes of interest. We should ask ourselves "Does a station program affect earth orbital plans?" "Was the total impact of an operation such as Mariner II of greater significance --- technically and nontechnically --- than the last several corresponding earth orbital operations?" If the answers are in the affirmative, then a reassessment of emphasis is in order to equate activity with opportunity.

Here, the Perspective that we need is to recognize the changing currents as well as the fundamental patterns of our national space program; to learn to slow down activities in certain areas and expand them in others as time progresses.

In the area of Perspective then, the challenge is twofold. First, we must discipline ourselves to go through the often agonizing process of developing a fundamental, objective picture of our national space activities. And second, this picture must be utilized when passing judgment on proposed new missions, hardware programs, or lines of endeavor. In this manner we can avoid doing "more of the same" or doing something "just a little better" but rather take our resources and apply them to the fundamentals that will yield the greatest progress as a function of time and investment.

Applications

The subject of space station Applications holds several areas of special interest. One of these is in the area of activity patterns.

Utilizing the national Perspective previously described, it appears from analyses that Applications fall into a pattern of two categories --- civil and defense --- and four mission regimes: Test, Military, Orbital Launch and Applied Use. Of particular note are the common engineering and scientific test interests in both the civil and defense categories in the early phases of a space station program. On the other hand, hardware evolutions into military and orbital launch missions suggest more distinctive second generation Applications roles peculiar to each category.

Studies indicate that even 4 - 6 man space station projects may cost as much as two billion dollars. It is clear that duplication of effort would be expensive. Therefore, the interests of the national space program could best be served by a single initial project responsive to both civil and defense test needs, for technology wears no label.

The challenge then is to conceive a management arrangement, hardware concept, and coordination rapport which effectively implements and efficiently discharges the initial project in an evolving station program.

Examination of the full potential for space station Applications represents a second area of interest. The challenge here lies with each of us to think of new, constructive, and intelligent uses for space stations; uses which can stand the "should be done vs. could be done" test. The Perspective that we spoke of above --- the national Perspective --- is fundamental to this task. The key is cross fertilization --- rather than compartmentation --- of thinking. Should one, for example, continue to think in terms of hypervelocity reentry materials and aerodynamic tests which employ scale models, large launch vehicles, and cost tens of millions for single data points? Could a space station be employed to launch various size models at different angles of attack and velocity into an instrumentated range for data points? In another vein, could an earth orbital space station program be configured --- in timing, management, hardware --- so as to efficiently provide a backup lunar reconnaissance capability?

The point is this: it would appear that unexplored possibilities exist for space station Applications. Examination is required so that those having merit can be brought to light and included in the formative stages of program definition. In addition, Applications thinking of this type might indicate areas in our national space program where activities could be curtailed in favor of more effective concepts. It is just as important --- and often more difficult --- to know when to stop things as well as start them.

The challenge, then, in the area of Applications is twofold. The first recognizes the national character of our space program and the common civil and defense test needs in the early phases of a space station activity. It suggests that a program concept be formulated which makes maximum utilization of existing --- and distinctive --- capabilities and capacities while at the same time being fully responsive to the specific missions and objectives of the agencies involved.

The second challenge is to penetrate the subject of space station Applications deeper than we have heretofore. This will more factually determine the characteristics of the program and hardware. Also, it might contribute to the objectives of our unmanned satellite programs and indeed, in some cases, provide a better data return.

Evolution

One evolutionary possibility starts with a relatively modest station and evolves into one or more advanced operational concepts when their requirements are defined and the impact of the "g" vs. no "g" problem is established by actual space test. Maximum utilization of existing launch vehicles and spacecraft is made so that only the station itself represents a fundamentally new hardware item.

In this concept, the basic unit might be a 4 - 6 man/20,000 pound/1 year minimum life article aimed at satisfying initial civil and defense test needs. Titan III and Saturns I and IB would be candidate launch vehicles.

From this point the program could evolve into advanced projects utilizing one or more of the original stations as the basic "bricks" of the ultimate "building." These might take the form of large wheeled stations if gravity is required; large "molecular" designs if size but not "g" is the fundamental criteria; an operational complex of several small stations operating at different orbit altitudes and inclinations; or any combination of these second generation systems. The merit of an Evolution of this type is principally one of efficiency. It could be designed and immediately employed to satisfy known test needs and grow to meet foreseeable applications. The programing could digest major decisions --- such as "g" vs. no "g" --- without costly backtracking or delays. Indeed, its flexibility would also appear to give the concept a backup potential for supporting existing projects.

An even more minimal first generation concept is based on a smaller unit - typically a 2 - 3 man station sizing. In some versions the stations might be a new hardware item with modular concept growth while in other arrangements existing systems might be modified for a prototype station capability. The principal merit of the concept is claimed to be an earlier availability. However, there is currently no indication that concepts of this type are particularly attractive when entire program Evolution is considered. The initial costs appear significant --- as high as a billion dollars --- while so-called earlier availability is bought at the expense of longer Evolution. There is no indication at this time that the station program could evolve directly from this minimal concept to most postulated applications without still needing an intermediate capability.

The challenge in the area of Evolution is straightforward. It is simply to define a program which:

(a) Makes maximum utilization of our present hardware systems.

(b) Reflects long range Evolution rather than short-term expediency.

(c) Is flexible enough to absorb major decisions and back up existing programs.

(d) Is consistent with some reasonable demands on Resources.

Resources

Offhand, I don't recall any major space program which has attained all of its objectives and has run significantly under estimated costs. On the other hand, we are all familiar with unfortunate examples of projects which imposed greater demands on Resources than originally anticipated. The challenge here is to come up with a space station program which has the Perspective that we spoke of earlier, is dynamically Evolutionary, and yet imposes reasonable demands on our Resources.

The "reasonable resource" criteria might be satisfied by the "single Phase I/two step evolution/existing launch vehicle and spacecraft" program previously mentioned. But in keeping with our original objective of touching on aspects of station programing not ordinarily discussed, the earliest Resource problem facing a space station program is adequate cost estimating.

It is difficult to meet cost targets once a program has been initiated. But even more discouraging is the wide variation of cost estimates furnished for decision making on program starts. How do you reconcile launch cost estimates for a certain next generation vehicle that change by a 4:1 factor, depending on the source, or judge cost estimates for the same program from competing organizations that range from $1.7 billion to $16 billion?

Cost analyses are not mysterious. They require a good, honest effort with plenty of digging, heavily garnished with judgment. Whether it is called Value Engineering, Systems Analysis, Cost Effectiveness or just plain Cost Analysis, the endeavor must take its rightful place in Government and industrial organizational structures like any other skill --- aerodynamics, propulsion materials or design --- and held to equivalent accountability. To discount the problem fails to recognize the fundamental fact that one of the few things that can hold us back in space is a disenchanted Congress and public who authorize resources to get a job done and who are then faced with an agonizing reappraisal calling for either project termination or hundreds of millions more to meet original objectives.

Having raised the problem, it's a fair question to ask what might be done about it. The solution has several elements, including organizational, as noted above. But one facet lies, in my judgment, in the area of making increasing use of the modest amount of historical cost data being accumulated by our multibillion dollar national space effort.

In the aircraft business, parameters and patterns of dollars/pound of airframe; subsystems costs; costs as a function of payload, range and velocity; etc. were plotted, studied and known to a degree which permitted realistic estimates of new programs on the one hand and exposed inept projections on the other.

What is past is prologue, at least if tempered by judgment. We are in the Year Seven of the Age of Space. A limited amount of cost experience is available for factoring into our analyses. To fail to consider this source of practical experience and judge its meaning on its own merits fails in my opinion to recognize the changing market place and decision arena.

Decision criteria-wise, we are paying about \$110,000/lb. for a group of four manned systems. Even more interesting is the fact that the cost spread is less than 15%. What would you think of a presentation that projected space station program costs at 1/5 or 1/10 this pattern without --- at least as an absolute minimum --- explaining why the costs differed so markedly from current experience?

In summary, a Resource challenge is to make the changes --- in organizational structure and mental attitude --- that are required to produce more factual and accurate cost data to go with program recommendations.

Management

If a manned space station program represents one of the fundamentals of our total national space program, it then follows that the program (made up of several projects) is very important and will undoubtedly be large and demanding of time and effort. The control of programs such as this with gestation periods in terms of years, which cost hundreds of millions of dollars and which involve thousands of people, presents one of the most formidable Management tasks ever attempted. The programs are complex to say the least. With the best Management we will have difficulty in realizing our goals. Without good Management we can be assured that those goals will not be met.

Many subscribe to the integrated task force or team approach made up of a single body of persons, physically located together in one area, representing the several disciplines involved and responsible to one person alone who, in turn, has full authority and commands the resources allocated for the project. But whether this or some similarly modern Management concept is selected, it is apparent that a "business as usual" philosophy will not suffice. Space success is not a "business as usual" proposition. Rather, it poses special challenges which demand a different approach.

One criteria that might be used to check on the effectiveness of any proposed space station Management would be the identification of the one person who is completely responsible for all facets of the program; who has full responsibility and decision-making authority inside of resource allocations; and who is on the job full time on that project and that project alone. If he is not high in the organization; if he does not have responsibility for all aspects of the program (launch vehicles, payloads, et cetera); if he is not given full line authority with the resources placed at his disposal rather than controlled by a parallel organization; and if he is not on the job full time to the exclusion of all other distractions and responsibilities, then it is suggested that the Management structure may not be adequate to administer a national manned space station program.

If a single station project is initiated to satisfy initial civil and defense test needs, a new dimension of coordination and cooperation will be imposed on industry and Government. The challenge is to propose Management concepts which are responsive to the national rather than institutional characteristics of the project and which have clear lines of authority and a centralized point of control and responsibility.

Decision Making

Perspective teaches us that our manned space station program is not a project, but a program made up of a series of projects. It is not a discreet element, but a stream of effort. Consequently, a great number of programing decisions are involved through a decade --- decisions as to when to start and what the end product should be ten years from now, and decisions that we will not know the answers to until we are part way through the program. Consequently, we can visualize the need for a flexible, evolving space station program which lends itself to the introduction of major decisions without disruption and waste motion.

Decisions facing us during the evolution of the program include possible lunar reconnaissance; gravity versus no gravity; size and characteristics of the station when used as an orbital launch facility for both lunar shuttle systems and our initial manned deep space flight --- possibly a Mars or Venus orbiter. The questions arise: "Is it possible to conceive a flexible evolving manned space station program, with the key decisions identified?" "Is it possible to have a program capable of responding efficiently and rapidly to whichever the way a decision goes so that we aren't caught off base or have to backtrack?"

The answer to these questions, in my judgment, is yes. And the challenge to the Government organizations and industry elements that plan to participate in the program is to lay out a simple, straightforward, evolving manned space station program which will provide the flexibility required by the decisions makers to accommodate the unanticipated and unusual as well as the expected and the known.

Support

Discussion of the aspect of Support as the last item belies its importance. It has several dimensions --- national, organizational, and individual --- which merit comment.

First, in terms of national support, the 7.6 billion dollar FY 1964 space budget request recently submitted to Congress is indicative of the current Administration's aggressive support of our national space program. The 40% increase over last year's budget is heavily oriented to manned space flight with much of the technology and actual hardware directly applicable to a space station program. As a matter of fact, two-thirds of an initial station project's hardware --- launch vehicles and ferry spacecraft --- are currently under development.

The second dimension --- organizational support --- concerns the prototype station itself. While a procurement program has not yet been released, important groundwork is being laid. Currently, more than $4 million per annum is being invested in industry contracts alone for space station studies and subsystem hardware development. Several times this amount is being spent on in-house activities. The awareness of NASA and DOD as to the importance of space stations and the support that they deserve is underscored by this effort.

The third dimension of support is individual. It applies to each of us. Congressional support is required for the expanding space budget with its emphasis on manned space flight and eventually a space station program. Scientific support is needed in order to capitalize on a station project's test and research potential. And public understanding and support of a multimillion dollar space station project is essential.

Each of us in the aerospace field has a responsibility to contribute to this effort, to stimulate support in all of these areas, to communicate with his co-workers and the public. We cannot be content to "let the other fellow do it" and effectively discharge that responsibility.

The challenge, then, to each and everyone of us, is to support the Administration in its expanding space program, and to communicate with --- and broaden the base of --- public and scientific understanding and support of our national space station program.

Conclusion

Only infrequently does history give us an opportunity to formulate and implement an activity ---such as this multiproject space station program ---whose impact might well take its place with that of nuclear energy as being one of the fundamental milestones in the accomplishments of man. But such an opportunity poses a challenge, facets of which have been discussed. The effectiveness with which we respond to the challenge will dictate, to a great degree, the success that we will have with our national space station program.

SPACE AGE TRANSPORTATION SYSTEMS

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INTRODUCTION

Within this session on future space systems, the other papers and speakers will deal largely with missions and activities to be conducted in space. This paper will discuss briefly launch vehicle concepts, or space transportation system elements, which seem compatible with the nature of space travel assumed for the 1970's and 1980's. We will not dwell on prospects for initiation of space projects which would create such traffic demands, but will concentrate on vehicle concepts potentially available for the assumed missions. It should be kept in mind, however, that the flow from demand (requirement) to capability is not a one-way stream. The traffic market will be greatly influenced by the capacity and efficiency of available transports.

Near future space activities can be grouped into three categories: Earth orbital, lunar, and planetary. It can be assumed that orbital and lunar traffic will be first to achieve high levels on a sustained basis, while planetary missions are still on an intermittent and exploratory basis. Therefore, we will concentrate for this discussion on orbital and lunar traffic - - under somewhat matured operational conditions.

Transportation is of course required for initial placement of men and equipment into orbit and on the lunar surface. However, crew rotation and sustained logistic support will create the major traffic demands. Some typical estimates for support of space stations and lunar bases of varied sizes are shown in Fig. 1. These data do not include indirect items such as service and maintenance crew requirements in orbit for support of lunar and planetary missions.

A composite projection of cargo and passenger traffic rates for the coming decades is shown in Fig. 2. This projection was compiled recently by Mr. H. H. Koelle, Director of Future Projects for the Marshall Center, taking into consideration mission desires, logistic support estimates, and tempered with expected budget constraints.

We can normally expect some degree of freedom in payload size per trip for cargo transportation, which may influence a choice toward larger vehicles, resulting in lower flight frequencies. Passenger transportation, on the other hand, will require a certain frequency of trips which is to some extent independent of available vehicle sizes. Crew rotation will be necessary at staggered intervals; unpredicted needs for personnel and for special equipment will occur at orbital and lunar stations. In this discussion, we will therefore give first attention to passenger transportation, involving vehicle concepts and operational modes approaching a "shuttle" or "ferry" mode of operation.

LAUNCH VEHICLE CONCEPTS

The launch vehicles and stages to be developed under the SATURN and APOLLO programs will provide an initial capability for orbital and lunar logistic support. In looking toward this future operational environment, we must ask: Under what conditions will these vehicles become obsolete, and what types of vehicle concepts appear promising as their replacements?

The major areas of improvement we will seek in "second generation" vehicle concepts will be in operational characteristics and economic efficiency. Some of these objectives are illustrated in Fig. 3., along with alternate paths available.

REUSABLE LAUNCH VEHICLES

Studies over a period of time have led us to believe that higher levels of operational reliability and economy are achievable with reusable vehicles, particularly for some classes of vehicles and operations. That is, the "airplane" approach to vehicle design, development, and operation will get us closer to our objectives than will the ballistic missile approach. We will discuss the application of the reusable vehicle concept for Earth to orbit transport, which seems nearly certain to be adopted at some time in the future. The extension of this concept to other elements of the lunar transportation system will be discussed as one of the major alternatives currently under investigation.

ORBITAL PASSENGER FERRY VEHICLES

Considering future transportation of passengers between Earth and orbit, and cargo traffic at high frequencies, vehicle studies conducted over the past several years have converged toward highly reusable, airplane-type launch vehicles. These have normally fallen into two classes:

1. Rocket-boosted vehicles, designed for vertical take-off (VTO) and winged fly-back and landing.

2. Horizontal take-off (HTO) vehicles, equipped with advanced air-breathing propulsion, designed to operate from runways and for aerodynamic flight during early boost.

In our recent studies of the rocket airplane concept, new constraints and emphasis have been introduced, with results differing somewhat from earlier studies. First emphasis has been placed on reliability and passenger safety. Beyond this, special attention has been given to passenger accommodation, because of a belief that space transportation for this period must not be limited to astronauts. Operations in space will involve persons of varied talents, age, and physical condition. As a part of this emphasis, design constraints have been imposed to limit flight accelerations (boost and re-entry) to 2-3 g's, and to 4 g's under abort conditions. This factor, plus an approach to initial boost, has led us to depart from the conventional marriage of rocket boost and VTO. This work indicates the rocket booster in the HTO mode to be a good candidate, and is being investigated in current studies.

When designing HTO vehicles for operation from runways, the rocket vehicle size is invariably limited by runway weight limitations. In recent studies, this problem has been alleviated by an approach which is not new, but seems applicable. The vehicle lift-off weight is kept down by keeping the major undercarriage and the initial boost propulsion equipment on the ground, in the form of a ground accelerator sled. Under these conditions, HTO is competitive with VTO, if not superior from a performance standpoint. One comparison of VTO with HTO in this form is shown in Fig. 4, i.e., indicating the sum of drag and gravity losses to be a smaller fraction of stage ideal velocity in the HTO case.

Admittedly, these analyses were not approached from a completely unbiased standpoint. It was felt that the HTO mode of operation is preferable unless the necessary penalties are too great. When designed to the "g" constraints for non-astronaut passengers and approached from the ground accelerator standpoint, the studies indicate no major penalties, and perhaps an advantage for the HTO mode. Among the reasons for preference of the HTO mode for a passenger transport are the following:

1. Passenger and payload preparation is simplified.

2. Preparation of launch vehicles for flight is simplified.

3. Boost flight more nearly approaches past flight experience.

4. HTO mode is more adaptable to "g" constraints and passenger abort under possible emergency conditions.

HTO ROCKET AIRPLANE CONCEPTS

These considerations have led to vehicle concepts as shown in Figs. 5 and 6. These vehicles are designed for transportation of 10 passengers (plus crew and limited cargo). Both consist of LOX/RP first stages and LOX/Hydrogen second stages, giving lift-off gross weights in 1-1.5 million lb range. A limited amount of storable propulsion is provided in the payload stage for final adjustments in orbital plane and altitude, for rendezvous maneuvers, and for de-orbiting.

Of the two approaches to full recovery and reusability, that typified by Fig. 5 seems to be the goal toward which we should work, i.e., where passenger provisions are integral with the reusable second stage. This minimizes the number of vehicles or stages to undergo recovery and flight preparation processes. An interim configuration employing a separate payload vehicle in combination with an expendable second stage may prove advisable as a first step before achieving complete reusability. With our present knowledge of orbital recovery, the price for orbital recovery will be high in terms of inert weight, development effort, and vehicle size for a given passenger complement. In any of the configurations, however, lifting recovery and the "low acceleration" ride will be maintained for the passenger vehicle.

GROUND ACCELERATOR CONCEPTS

Two methods of propulsion are currently under consideration for the ground accelerator. One is a conventional liquid rocket sled (Fig. 7) with the exception that the vehicle engines are used for boost, with the sled only supplying propellants to replenish the vehicle tanks. This approach has the advantage that all boost propulsion is operating from beginning of the ground run, allowing verification prior to release and lift-off.

The second approach (Fig. 8), not yet studied in as much depth, envisions a "linear steam turbine" which employs turbine blades along the track. A greater impulse is attainable from the steam in this manner than by exhausting through rocket nozzles.

MISSION PROFILES

A typical flight profile is shown in Fig. 9. The ground acceleration to 400-600 fps can be accomplished within 5,000 ft or less. The sled has the ability to decelerate (water brake) the vehicle-sled combination in the event that all systems are not confirmed for lift-off.

A fairly steep climb-out follows lift-off, followed in turn by a nose-over toward a flight path angle of about 20 degrees at first stage burn-out. The "saw-tooth" appearance of the flight acceleration curve results from a first approach, using incremental shut-down of rocket engines to control "g" levels. Further study is expected to show advantage in engine throttling to maintain a near-constant "g" level.

After aerodynamic deceleration from cut-off speeds of approximately 6,000 fps, first stage turn-around occurs a few hundred miles down-range, and return flight to the launch or adjacent landing site is accomplished with subsonic turbo-jet or turbo-fan engines.

The orbital passenger vehicle, in some versions integral with the second stage, re-enters from orbit, achieving the desired landing site by choice of time for retro firing, in combination with maneuvers executed during re-entry. Operation as a glider is planned, unless propulsion proves to be necessary for "go-around" capability during landing approach.

BOOST PROPULSION

System studies and supporting research on several variations of this basic vehicle concept are now active within NASA, the Air Force, and Industry, with primary differences being in boost propulsion. The spectrum of boost propulsion choices available to us, as shown in Fig. 10, ranges from pure rocket to pure air-breathing engines. Marshall work to date has concentrated more on the "rocket" end of the spectrum, complementing work at other agencies on advanced air-breathing systems. As this work proceeds, however, there are indications that some combination of the two propulsion modes may be superior to either in a "pure" form.

The potential of air augmentation for rocket engines is being investigated as a "mixture" of the two propulsion modes. This approach has possibilities for noise reduction, as well as impulse improvements with minimum inert weight addition. Its potential should be best for an HTO vehicle of this type, with its tendency toward longer flight duration in the atmosphere, and having considerable velocity at flight initiation. The size and shape of necessary shrouding and its compatibility with the basic vehicle are being investigated in current studies.

Initial vehicle studies were based on use of rocket engines currently under development; however, several variations in engine hardware are also being considered in current studies. The first step is to explore possible use of F-1 and J-2 components in arrangements more adaptable to the vehicle concept, including multiple chambers feeding into a single large nozzle. Rocket engines of new design based on high chamber pressure are being investigated for comparison with the earlier rockets. The most attractive of these will then be compared with the conventional air-breathing propulsion concepts.

NEAR FUTURE OBJECTIVES

In view of the probable need for a national decision within the next few years as to vehicle concept and development program, our near future efforts are directed toward:

1. Complementary vehicle studies, covering major candidates in sufficient depth to allow a valid national choice at the appropriate time.

2. Continuing systems analyses by which candidate concepts can be narrowed and the remaining concepts optimized.

3. Supporting research in pacing technologies.

ORBITAL CARGO CARRIER VEHICLES

The development of SATURN V in the 1960's and the expected development of NOVA in the 1970's for manned planetary missions provide a tremendous capability for transportation of large payloads to orbit. The question for the later time periods then becomes one of operating efficiency and cost effectiveness. Current NOVA studies include efficiency of operation and probable obsolescence rate among the prime factors in selection of candidate vehicles. This has led to strong consideration of advanced engines, and to consideration of recovery/re-use in varying degrees.

Competitive vehicle concepts are being studied for comparison with the SATURN V under operational conditions assumed for the 1970's. The first of these studies is an investigation of advanced versions of the SATURN V, including possible conversion of the stages for recovery and re-use. A SATURN V equipped for winged recovery of the first stage is shown in Figs. 11 and 12, with the latter giving an indication of stage size relative to current large aircraft.

Studies of competitive concepts of new design are planned in the near future for comparison with SATURN V and modified versions thereof. This can then give us the best indication of the profitable life-time for the SATURN V, and can give direction for supporting research on approaches which seem most likely as its eventual replacement.

LUNAR TRANSPORTATION

The requirements and transportation systems for the early exploration of the lunar surface have been discussed in descriptions of the APOLLO program and its extensions. If we assume that the lunar activities will be expanded to a sizeable lunar base, the need for more economical and efficient means of transporting people and cargo becomes more pronounced. We can look for this improvement in several ways including larger launch vehicles such as NOVA, introduction of nuclear propulsion into portions of the mission profile, reuse of transportation system elements, or possible combinations of these methods.

We cannot at this time establish firm requirements for a lunar base because of the unknowns of the lunar environment and base functions. We can make estimates from what is known and make parametric analyses to determine the effect of these assumptions on the logistic requirements. When examining the logistic requirements for a large lunar base (approximately 50 men) we can categorize these payload requirements as: Initial base facilities, personnel delivery, life support equipment, base maintenance and improvement, and return vehicles. We then try to tailor our transportation systems to satisfy these requirements. A large portion of these payloads are one-way trip requirements, but one of the most critical demands is the rotation of personnel. Looking at these two requirements separately, one may decide to develop two separate transport systems; however, it is desirable from the standpoint of vehicle inventory to develop one system that can satisfy both requirements. Such a system would likely be preceded by several evolutionary steps.

The first improvement from the present chemical expendable systems is the use of a nuclear escape stage with the SATURN V vehicle, employing an operational stage resulting from the RIFT/NERVA program. This system will increase the lunar capability of the SATURN V by about fifty percent. This is still an expendable system, with the nuclear stage being disposed of, most likely, by injection into a solar orbit after delivering the payload to escape velocity.

The return of personnel with this system will be the same as the all chemical systems, i.e., minimum propulsion (for lunar escape) and Earth reentry at parabolic speeds. As the number of personnel at the base increases it will no longer be practical to train and/or condition all personnel to withstand this environment. It becomes necessary, therefore, to reduce this environment by applying propulsion during the return phase. The performance capability of the nuclear systems, and the desire, for economic reasons, to recover and reuse these systems leads us to the consideration of the reusable nuclear ferry concept.

REUSABLE NUCLEAR FERRY

The nuclear ferry system would have a mission profile as shown in Fig. 13 and the mission would consist of the following operations. When used in combination with the previously described Earth-to-orbit passenger ferry, acceleration levels are held to about 2-3 g's for the lunar round-trip.

1. Lunar cargo, and propellants for cislunar propulsion are transported to Earth orbit.

2. The ferry is fueled and loaded with cargo in orbit.

3. Out-bound passengers board the serviced ferry after being transported to Earth orbit by the reusable orbital ferry vehicle.

4. Upon arrival at lunar orbit, out-bound passengers and cargo are transferred to a chemical shuttle vehicle which operates between lunar surface and orbit, and return passengers are transferred to the nuclear ferry.

5. The nuclear ferry returns to Earth orbit and is prepared for subsequent flights.

6. Returning passengers are transported to Earth in returning reusable orbital ferry vehicles.

A typical conceptual design of the reusable nuclear ferry and lunar shuttle vehicle is shown in Fig. 14. The ferry weighs approximately 550,000 pounds at ignition including 300,000 pounds of propellant. It can deliver up to 22 people and 55,000 pounds of useful cargo and propellant for the shuttle descent to the lunar surface. The shuttle is a reusable system having a LOX/H_2 propulsion system of the Pratt and Whitney RL-10 type.

POSSIBLE SYSTEM EVOLUTION

An example of how such a system might evolve with time is shown in Fig. 15. The shuttle, while shown as being available in the late 1970's with the nuclear ferry, is not limited by the current state-of-the-art, but is shown as required for this system. It could be available sooner if desired for another mode of operation.

CRITICAL CONSIDERATIONS

This system of transportation is not cheap from the standpoint of performance, technology development, or development cost. Payload must be sacrificed to provide propellant for the ferry return trip, and new and greater demands are placed on nuclear engine development and orbital maintenance and service operations.

The most critical item for this mode of operation is the development of a nuclear propulsion system which has the capability to meet the demands of the mission. The design criteria of such a system as compared to the RIFT/NERVA system is as follows: Three times the thrust; three times the power level; four to five times the burning life time; seven to eight times the starts; twice the engine efficiency, in terms of pounds thrust to pounds of engine weight. Nuclear engine concepts which offer promise in this regard are currently under investigation.

While the system described has many desirable features, more must be learned about the problems to be encountered and the economical attractiveness of such a system before we can say this is the approach to be pursued for the future lunar transportation systems.

Studies are now being performed to better answer some of the more critical problems and include the following:

1. Nuclear propulsion parametric studies for a number of space missions to determine the requirements, development problems, feasibility, and availability.

2. Design studies of the nuclear ferry for operation with expendable and reusable SATURN V class vehicles and NOVA. This includes radiation environment, maintenance and degree of reusability, micro-meteorite protection for long exposure time, cargo packaging, crew compartment and abort requirements, etc.

3. Operational analysis studies to determine how much a system might integrate into the total space transportation requirements and evolution. This includes the comparison of the merits of this system with other candidate systems that might be available in the same time period.

4. Orbital operations studies to determine the requirements for servicing and launching of nuclear propelled orbit launch vehicles.

SUMMARY and CONCLUSION

Space transportation systems of different types are being investigated for support of manned operations in Earth orbit and on the lunar surface at the levels expected in the 1970's and beyond. Among the alternatives, the concept of reusable space vehicles, operating as ferry or shuttle transports, seems to approach most nearly the characteristics of successful Earth surface transportation systems.

Application of this concept, particularly for passenger transportation, has been discussed for transport between Earth surface and orbit, between Earth orbit and lunar orbit, and for operation between the lunar surface and lunar orbit.

It seems likely at this time that an airplane-type, reusable ferry vehicle will be developed within the foreseeable future for transport of passengers between Earth and orbit. Efforts are now active within NASA, Air Force, and Industry to define alternatives in boost propulsion and operating modes to allow sound decisions and choices prior to commitment to a large program.

Application of the concept to lunar transportation, in the form of reusable nuclear ferry vehicles and lunar shuttle vehicles, appears at this point as one of the promising alternatives. A better understanding of the operations, vehicle designs, and its potential benefits is now being developed to allow comparisons with competing modes, and with the required technological developments.

While these concepts are not presented as the solution to space flight needs of the future, it is our hope that some probable future trends in space transportation have been portrayed in this discussion, and has stimulated your interest. It is also evident from the discussion that necessary decisions on the part of planners of future vehicle development programs will not be arrived at easily.

*The opinions expressed in this paper are those of the authors and not necessarily those of the National Aeronautics and Space Administration.

ACKNOWLEDGEMENT

Studies of future space transportation systems are at present being conducted by aerospace firms under NASA/MSFC sponsorship. Much of the material used in this paper is based on current studies with the Boeing Company, Ling-Temco Vought, The Lockheed Corporation, and North American Aviation.

FIG. 1

CREW ROTATION & LOGISTIC SUPPORT REQUIREMENTS (TYPICAL ESTIMATES)

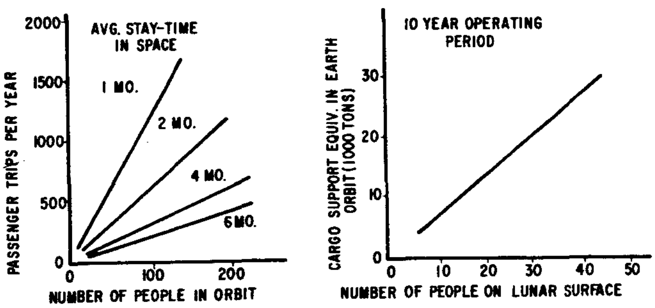


FIG. 2

PROJECTED U.S. SPACE TRAFFIC

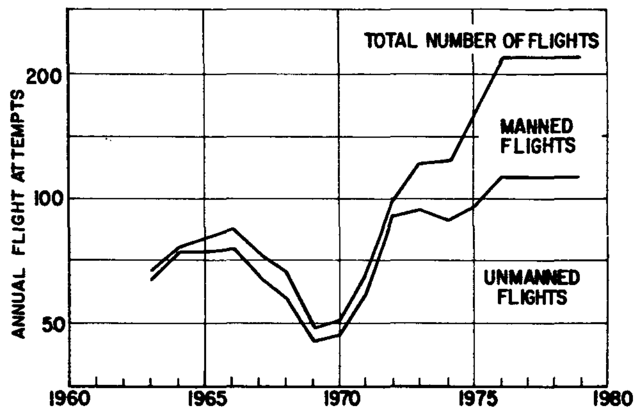


FIG. 3

ALTERNATE PATHS TO RELIABILITY & ECONOMY

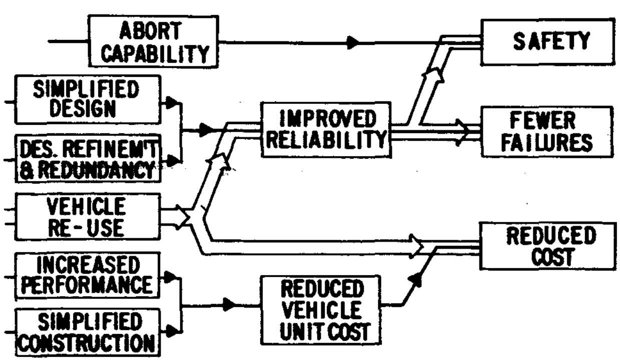


FIG. 4

COMPARISON OF FLIGHT LOSSES

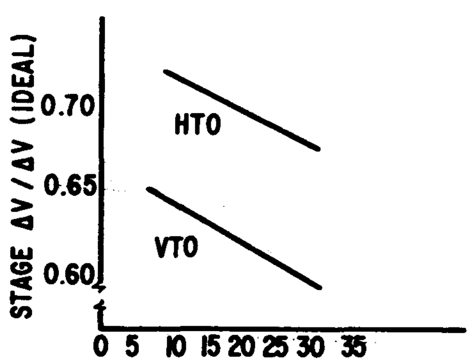


FIG. 5

VEHICLE CONCEPT

REUSABLE ORBITAL PASSENGER FERRY

(INTEGRAL PAYLOAD STAGE)

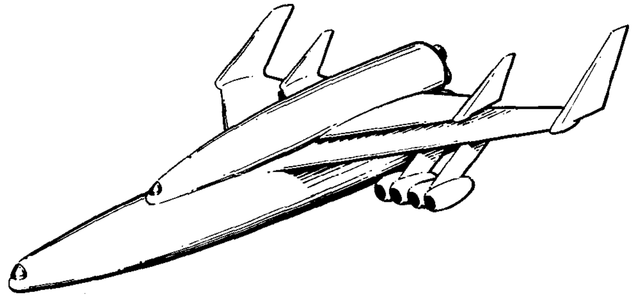


FIG. 6

VEHICLE CONCEPT

REUSABLE ORBITAL PASSENGER FERRY

(SEPARATE PAYLOAD STAGE)

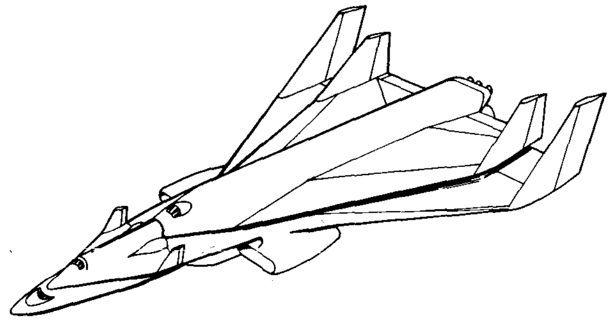


FIG. 7

GROUND ACCELERATOR CONCEPTS

PROPELLANT SLED

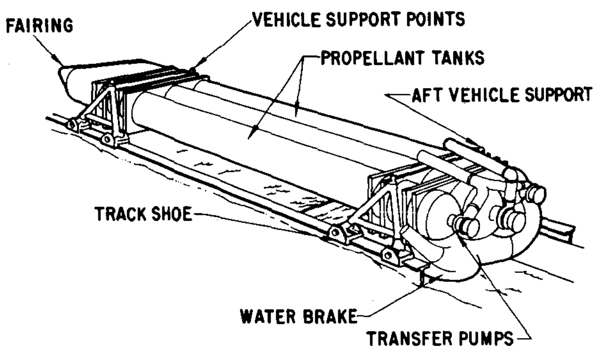


FIG. 8

GROUND ACCELERATOR CONCEPTS

LINEAR TURBINE

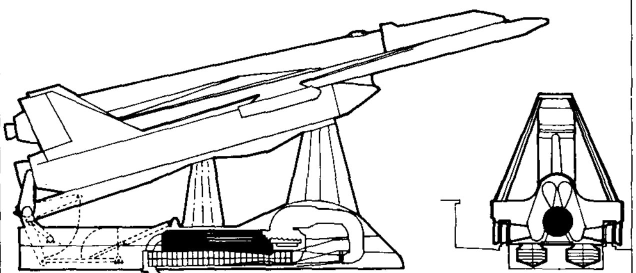


FIG. 9

TYPICAL BOOST TRAJECTORY

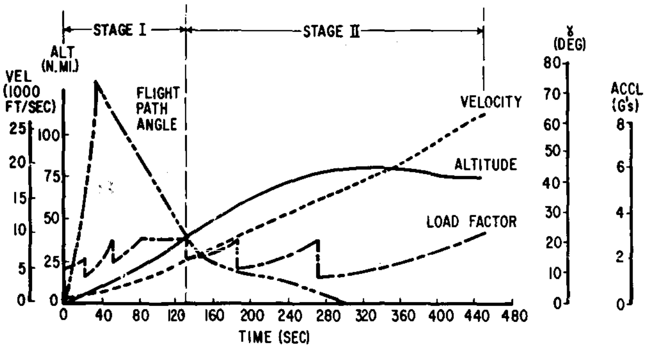


FIG. 10

PERFORMANCE / WEIGHT TRADE-OFF TRENDS FOR CANDIDATE BOOSTER ENGINES

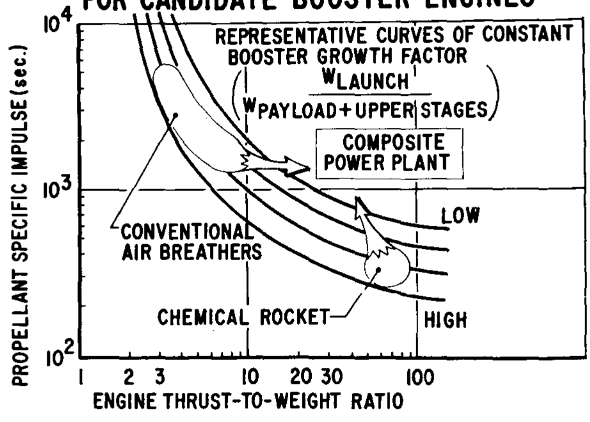


FIG. 11

MODIFIED SATURN V
WITH REUSABLE S-IC STAGE

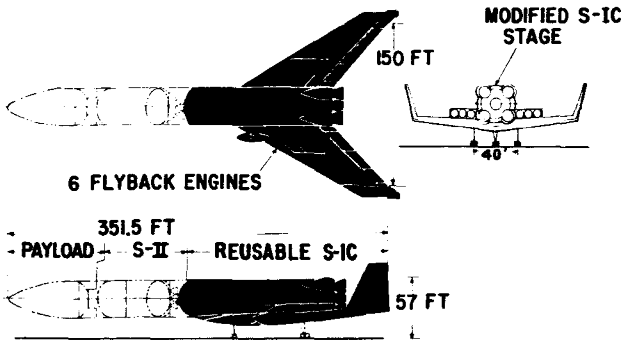


FIG. 12

SIZE COMPARISON–REUSABLE SATURN V STAGE

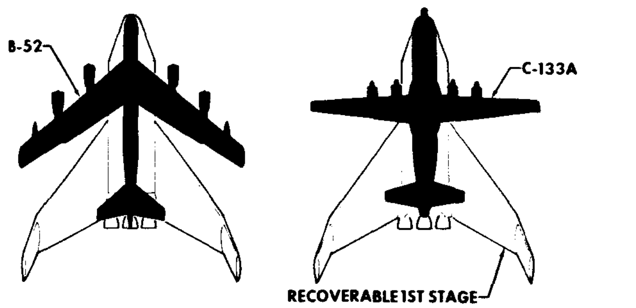


FIG. 13

REUSABLE NUCLEAR FERRY MISSION PROFILE

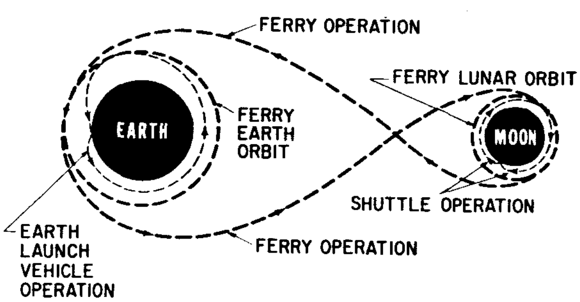


FIG. 14

CONCEPTUAL VEHICLE DESIGNS
REUSABLE NUCLEAR FERRY AND LUNAR SHUTTLE

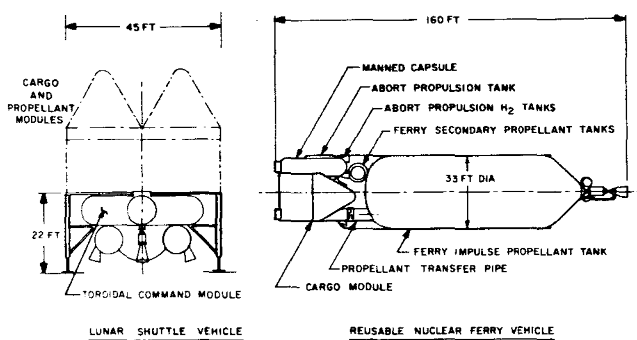
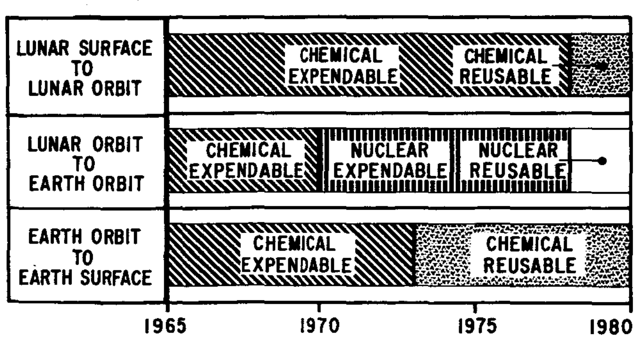


FIG. 15

POSSIBLE EVOLUTION OF LUNAR TRANSPORTATION SYSTEMS



MANNED FLIGHT TO MARS AND VENUS IN THE 70's ~~Summary~~ Report

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Abstract 23232

Velocity requirements associated with various flight paths to Mars and Venus and the resulting propulsion and re-entry weights are combined and varied with other missions system weights which are a function of mission goals, trip time and environmental protection. From these weight trade-off analyses are evolved spacecraft design concepts and total mass requirements on Earth orbit to accomplish the various missions. Mass requirements so established are compared with launch rocket capabilities.

System requirements covered in the analyses, and briefly described in the paper, to determine total mass requirements on Earth orbit are:

(1) Propulsion systems for Earth orbit departure.

(2) Earth entry systems for different missions.

(3) Life support systems.

(4) Navigation and control systems.

(5) Reconnaissance and scientific instrumentation for gathering data during course of missions, and requirements for storage and/or readout.

(6) Space power supply systems.

(7) Vehicle design concepts.

AUTHOR

Introduction

This paper is a summary of the results of a study contract recently completed with the Future Projects Office of NASA's Marshall Space Flight Center, Huntsville, Alabama. The many people participating in the study, whose efforts are represented here, will be identified to enable the reader to contact the investigator directly should his deeper interest make this desirable.

Few references will be listed here as a complete list of references is included in the final report[1] of the above study.

Even though an iterative process is used to establish economical missions, a process in which departure velocity, return velocity, life support and shielding weight (and for stop-over missions planet arrival and departure velocities) must be traded off, the discussion will begin with representative interplanetary flight paths. Included in the same section will be mission requirements, in terms of mass on Earth orbit, as determined from the systems considerations.

Mission Analysis and Requirements

The study described in this section was conducted by R. V. Ragsac, with inputs from R. R. Titus and Z. A. Taulbee, and with specialized inputs from others to be described later.

Flyby Trajectories

Two quite different classes of trajectories have been found for Mars trips which differ from each other in terms of Earth orbit mass requirement, time of flight, and sensitivity to the time in which the trip is taken. A representative of the high energy, or "hot trip", class of trajectories is shown in Fig. 1 for departure from Earth 31 October 1970. In this class of trajectories the vehicle dips inside of Earth's orbit, crosses Earth's orbit on its outward trip to Mars, passes Mars very close to the aphelion of the transfer ellipse, again crosses Earth's orbit, and finally arrives back at Earth approximately a year and a half after the starting date. Because in this trajectory arrival at Mars occurs when the vehicle is at or very close to the aphelion of the transfer trajectory, the energy of this transfer trajectory is greatly influenced by the position of Mars on its own rather highly eccentric orbit at time of vehicle arrival. Arrival for the trajectory shown in Fig. 1 occurs near its perihelion; therefore the trip shown represents a favorable transfer time for this class of trajectory. Later missions, for which arrival occurs with Mars further from the sun, involve higher energy transfer orbits, and therefore increases mass requirements on Earth orbit, as will be shown later.

The Martian perturbation of the interplanetary trajectory was based upon the nearest approach to the surface of Mars being one tenth of the radius of Mars, i.e., the radius of nearest approach was taken as 1.1 times the radius of Mars, as shown in the center of Fig. 1 by $r_p = 1.1$. $V_\infty = 0.10$ means that at nearest approach to Mars the hyperbolic excess speed with respect to Mars is 0.19EMOS.

The other main class of trajectories includes the low energy, or cool, type of trips shown in Fig. 2. It will be noted that although these trajectories do not carry the vehicle inside of Earth's orbit, they do reach out considerably beyond the orbit of Mars. By so doing the energy of the transfer ellipse, and therefore the mass requirements on Earth orbit, are not nearly as sensitive to the distance of Mars from the sun at time of arrival as had been noted for high energy, or hot, class of trajectories. This class, however, does involve greater trip durations as indicated in the figure.

It has been found that for both classes of trips total mission requirements, as expressed in terms of necessary mass on Earth orbit, increase as the nearest approach to Mars is decreased. However the effect on the trip time is not the same with a variation in nearest approach distance for the two classes. For the high energy class a reduction of the nearest approach distance increases the total trip time, whereas for the low energy class reducing the nearest approach distance decreases the total trip time.

(NASA Contract NAS8-5024)

A representative Venus nonstop flyby trajectory is shown in Fig. 3, which is a pictorial display of a one year, non-stop round trip, leaving in the conjunction period of 1972. It is possible to obtain such a plot for all possible trips of interest in any conjunction year along with the corresponding trajectory information of departure and arrival at Earth and of planetary passage. A systematic analysis of this type of transfer trajectory[2] yields a set of velocity contour charts on which lines of constant planetary passage distances are drawn. It has been found for the Venus flyby trips that a reduction of the nearest planetary approach distance results in a reduction of both the mass requirement on Earth orbit and also the total trip time.

A very fruitful class of interplanetary missions possessing great potential for planetary reconnaissance is the multi-planet flyby or interplanetary grand tour. For these missions the spacecraft passes both Mars and Venus on the same trip. The necessary planet-vehicle trajectory alignment for each type of grand tour is repetitive every 2,338 days (6.4 years), however the mission requirements will not be the same for every opportunity due to the eccentricity of Mars' orbit. It has been found that both planets may be passed for essentially the same requirements as a Mars non-stop trip, measured in terms of both necessary mass on Earth orbit and total trip time. Figures 4 and 5 present trajectory plots for two types of grand tours along with the dates of the several events. The trajectories of the interplanetary grand tours were extensively analyzed so that a scaled plot of each leg could be drawn to include the effects of planetary bending at the time of passage.

The entire study of both the flyby and the stopover missions, to be discussed later, utilized realistic joined conic trajectory data[3] which include the effects of terminal planetary perturbations, orbital eccentricities and inclinations.

Flyby Mission Requirements

It was pointed out earlier that for the high energy class of trajectories (those in which arrival at Mars occurs at or near aphelion of the transfer ellipse) mission requirements are quite sensitive to the distance between the Sun and Mars at time of arrival, and therefore quite sensitive to the time period in which the trip is made. This sensitivity is clearly shown in Fig. 6. It can be seen that for a Mars light-side flyby, employing a chemical propulsion system to escape from Earth orbit, both the required Earth departure speed and the mass requirement on Earth orbit increase greatly as the trip is delayed from 1971 to 1975. The chemical system employed here was a 430 sec I_{sp} hydrogen-oxygen system. It can also be seen that the minimum mass requirement does not necessarily coincide with the minimum departure speed requirement. This is especially true for the 1975 trip.

These trips, along with many others, are also presented in Fig. 7 which shows Earth orbit mass requirements plotted now versus total trip time. The mission requirements just discussed are those for the group of curves labeled Mars high-energy light-side, $r_p = 1$. For all of the trips shown in Fig. 7 the chemical Earth escape propulsion system mentioned above was assumed and the return capsule was assumed to be a modified Apollo command module with a solid propellant retro rocket (250 sec I_{sp}) to decelerate the vehicle from its approach speed down to the parabolic design re-entry speed of the Apollo, after which the command module ablative heat shield absorbs the final re-entry heat pulse.

As noted, considerable saving in both mission time (approximately 2 months) and mass requirement on Earth orbit can be achieved if the high-energy dark side passage is employed instead of the previously discussed high-energy light side passage. This kind of a passage however, would entail a great reduction in thedata which could be accumulated at Mars by photographic means.

As mentioned earlier, the low energy trips shown by the three curves on the extreme right of the figure are not nearly as sensitive to the year of the trip as are the high energy trips, and in fact, the minimum requirement for a 1975 low energy flyby is somewhat less than for 1971 and 1973.

On the left side of the diagram are shown the mission requirements for Venus flyby trips occurring in three different years. It is noted that the mission requirements for all three of these trips are nearly the same in both mass required on Earth orbit and in total trip time, which is about one year. The variation is small because of the low eccentricities of the orbits of both Earth and Venus, and the small difference which does occur is due mostly to the change in the alignment between the node of the orbital planes of Earth and Venus and the major axis of the transfer ellipse. It is noted further that the mass required on Earth orbit for the 1970 Venus trip is down almost to the capability of a single C-5 launch.

Requirements for three grand tour trips are shown, two-planet flyby trips for 1970 and 1972 and a three-planet flyby trip for 1970. It is noted that the mission requirements for both the Mars-Venus 1970 trip and for the Venus-Mars-Venus 1970 trip are both lower than the minimum value for the Mars trips alone, and that even the mission requirement for the 1972 grand tour is lower than for any of the high-energy lightside Mars trips and requires less total trip time than any of the trips taking the vehicle by only Mars.

In an attempt to save total mission mass requirements, re-entry systems other than a rocket retro system were investigated. These systems will be shown later but the savings available by employing one of these systems, a drag brake system, can be seen by comparing the mission requirements shown in Fig. 8 with those of Fig. 7. The curves of Fig. 8 are not the same shape as those in Fig. 7 because the trajectories themselves have been modified to take full advantage of the drag brake re-entry system.

Figs. 9 and 10 show comparable data for an escape propulsion system employing a modified Nerva nuclear engine and the retro and drag brake Earth re-entry systems respectively. Two things seem to stand out above all others in these two figures. One is that if a nuclear escape propulsion system is used the total mass requirement on Earth orbit for the Venus flyby missions is down to the launch capability of a single C-5, whether a retro Earth re-entry system or the drag brake system is used. The other important point is that for the high energy Mars trips the increase in mass requirements for the later years is much less (especially for the drag brake re-entry system) than it was for systems employing the oxygen-hydrogen chemical propulsion system.

A direct comparison between the Mars light-side mission mass requirements for vehicle systems employing a chemical escape propulsion system, and either a retro or drag brake re-entry system, with those for a vehicle system employing a nuclear escape system and a drag brake re-entry system is shown in Fig. 11. Here again it is shown that the increase in mission requirements as the trip is delayed in the 70's is reduced for a drag brake re-entry system and is brought nearly to zero if both a nuclear escape propulsion system and a drag brake re-entry system are employed.

Fig. 12 shows a similar comparison for Venus lightside flyby trips for departures in 1970, 1972 and 1974. It is noted that for the 1970 trip approximately 275,000 lb are required on Earth orbit if a chemical escape system and a retro rocket re-entry system are employed, and that if a drag brake system is substituted for the rocket retro re-entry system, approximately 20,000 lb can be saved on Earth orbit. If a nuclear rocket escape propulsion system is used and the drag brake re-entry system is retained, the mass required on Earth orbit can be reduced to approximately 160,000 lb, well within the launch capability of a single C-5. In fact, the mission requirements for all three years shown are below the orbital launch capability of a single C-5 if the nuclear escape propulsion system and the drag brake re-entry system are employed.

Orbiting Stopover Missions

A sample mission profile for a 1971 Mars 10 day stopover using a chemical escape propulsion system and a drag brake re-entry system is shown in Fig. 13. The stopover time of 10 days selected for both Mars and Venus missions was believed to be of sufficient length to perform all the experiments and observations necessary from the capture orbit.

System requirements in terms of mass on Earth orbit and total trip time are shown in Fig. 14 for Mars missions and in Figs. 15 and 16 for Venus missions. These system requirements were obtained by an optimizing method[4] which establishes a locus of minimum mass-on-Earth-orbit points by systematically analyzing the parametric effects of in-bound and out-bound leg duration, total trip time and departure and arrival velocities at both Earth and the destination planet. This integrated approach, although requiring numerous calculations of the total interplanetary system mass, eliminates arbitrary selection of an interplanetary trip from the multitude which are available.

The importance of the nuclear escape stage for the stopover missions is clearly shown in Fig. 14. It is seen here that, as was the case for the Mars flyby missions, mass requirements increase much more slowly as the trip is delayed within the 1970 decade than for chemical escape propulsion systems.

Fig. 15, which shows the effect of trip selection on Venus 1972 10-day stopover mass requirements, indicates the various kinds of trips which must be explored if a minimum is to be found. It is noted in this figure that for both chemical and nuclear escape propulsion systems minimum mass requirements are for the curve labeled II-B. By observing the small insert in the lower part of the figure it is evident that this means approximately 1/3 of the total trip time should be spent on the outbound segment and approximately 2/3 on the return segment of the trip.

Because of the low eccentricities of the orbits of both Earth and Venus, mission requirements for years other than 1972 will be rather similar to those shown, as was the case for the flyby missions. However, because the orbits of Venus and Earth are not exactly co-planar, two general departure regions are available, each distinguished by the resultant range of total trip time.

In Fig. 16 this effect is shown by the two curves for a given Earth escape stage. All possible departure and arrival regions were studied and only those two yielding lower system mass and reasonable trip time are presented. The two curves may be made to join together smoothly by applying a midcourse plane change which would cause the two departure regions to unite. This technique was not used, however, because the one departure region corresponding to the curves labeled II-B in Fig. 15 yields a distinctly lower mass minimum and a shorter mission duration. Only the trips of optimum trip segment distribution are shown in Fig. 14.

It is noted that the mass requirements for both 1973 Mars stopover missions and 1972 Venus stopover missions are approximately one million pounds in Earth orbit if a nuclear escape propulsion system and a drag brake Earth re-entry system are used.

The effect of orbit stay times greater than ten days was briefly investigated for Mars 1973 trips. The increase in transportation system mass on Earth orbit was found to be almost linear with stopover time, with the nuclear system again being the least affected. Staying over 60 days instead of ten, approximately doubles the mass requirement for systems employing a chemical Earth orbit escape stage, whereas only a 50% increase was found for vehicle systems employing a nuclear escape stage.

Martian Orbit Lifetimes & Perturbations - Because the altitude of the parking orbit must be great enough to give the orbiting vehicle sufficient drag life to carry out its mission objective, drag lifetime was computed for orbits around Mars as a function of pericenter altitude for two different orbit eccentricities and for two different models of the Martian atmosphere, the Yanow model[5] and an extreme model[6] which was designed to be the maximum density profile for the Martian atmosphere. This model is fashioned after Earth's profile with the largest gradient of slopes that seems plausible. The results, as computed by L. F. Koehler, are shown in Fig. 17.

Because of the effect of the orientation of the capture orbit on the return propulsion requirements at the time of departure, the study also included an investigation of Martian orbit perturbations as caused by the non-spherical shape of Mars. The perturbation factors were obtained from observed perturbation data on Phobos and Deimos. Results are shown in Figs. 18 and 19.

Phobos Stopover Missions - Mars' largest and closest satellite, Phobos, appears to offer certain advantages as a stopover station for the interplanetary mission. To identify the major requirements for such a mission a preliminary investigation of a particular set of Mars orbital excursions was conducted by M. G. Ross. The mission was assumed to occur during the opposition of 5 Dec. 1975. As was the case for the stopover missions at a lower altitude above Mars, it was assumed that energy dissipation at Mars would be by aerodynamic braking which would put the vehicle on an eccentric orbit whose apocenter was at the altitude of the desired circular orbit. Then, just enough propulsion to circularize the orbit would be employed.

It was found that for the Phobos stopover mission, if the circularizing maneuver is combined with the plane change maneuver to put the vehicle in the orbital plane of Phobos, total required mass on Earth orbit is 1,191,560 lb as compared with 1,094,500 lb for the orbit of 500 km altitude. Thus, a mass increase of less than 9% makes possible direct contact with material of the Martian system, and at the same time probes are still sent down to the surface of Mars for acquiring atmospheric and surface information.

Orbit Launch Vehicles and Operations

This part of the study[1] was conducted by J. F. McLaughlin and is meant to show the relationship between mission requirements and planned vehicle capability. Both chemical and nuclear specific orbital propulsion systems and stages were examined. Fig. 20 shows the net payload injected out to some mission speed vs the orbital stage impulse propellant required to achieve this speed which is shown as hyperbolic excess speed in terms of Earth mean orbital speed or EMOS. Curves are shown for values of EMOS from zero (just barely escape) all the way to the hyperbolic excess speed of 0.30. The diagonal dash lines show the mass required on Earth orbit which for any curve is the sum of the net injection payload shown, the impulse propellant, and the stage inert weight.

Since the abscissa is orbit escape stage impulse propellant (in this case oxygen and hydrogen), chemical stages of known propellant capacity as well as burn time indices can easily be added to the diagram to make it more usable. Because the diagonal lines show mass required on Earth orbit, launching rockets of known capability can be conveniently shown at the proper diagonal location.

For a given propellant value the net injection payload can easily be found for different values of hyperbolic excess speed. Conversely, for a given value of injection payload, impulse propellant required can be found as a function of required hyperbolic excess speed, and is increasing rapidly with increasing speed at the high speed and high propellant corner of the diagram due to very large gravity losses for these massive stages which employ a single J-2 engine.

Since mission trajectories can be classified in terms of the required hyperbolic excess speed and the net injection payload, requirements can be determined for a given mission objective. Areas could be added to this diagram bounded by requisite speed and payload mass which would then make it possible on one diagram to relate mass requirements on Earth orbit for various missions with orbital launch stages carrying the required amount of propellant and also with launching rocket capability.

These mission requirement areas have been added to Fig. 20 which results in Fig. 21. It can be seen that although an S-IVB stage fully loaded on Earth orbit can achieve the 1970 and 1972/74 Venus flyby missions, the mass on Earth orbit is beyond the capability of a single C-5 launching, but seems to be rather well matched for two C-5's with orbital rendezvous and propellant transfer.

The S-II has possible applicability as an orbit launch vehicle making flyby missions possible to both Mars and Venus. It appears feasible to orbit an S-II fully loaded with liquid hydrogen with one Saturn C-5 and then to rendezvous LOX tankers and payload vehicles as needed. The mission requirements shown in Fig. 21 are for a retro rocket re-entry system and, as mentioned earlier, requirements could be reduced if a drag brake system is employed.

Similar graphs were made for Earth orbit escape stages employing nuclear propulsion systems of various power levels. Fig. 22 shows the results for a 1500 mw engine. The minimum payloads and hyperbolic excess speeds making up the envelope boundaries are the same as those in Fig. 21 and the same constants apply. It is immediately noted that an orbital stage carrying 110,000 to 120,000 lb of liquid hydrogen can accomplish both the 1970 and the 1972/74 Venus flyby missions. Moreover, the total mass required on Earth orbit can be handled with a single C-5 launching. If the rendezvous of two C-5's is utilized, the Mars low and high energy trips and the 1972 grand tour are available. For these, however, the propellant capacity of the orbit escape stage should be approximately doubled, the exact capacity depending on the mission or missions chosen.

A comparison of the various nuclear escape stages which were examined is summarized in Fig. 23 which shows the injection payload vs impulse propellant of the orbit escape stage for two values of hyperbolic excess speed, 0.15 EMOS for the upper family of four curves and 0.30 EMOS for the lower family of four curves. It is important to note that for propellant values slightly over 100,000 lb, the approximate requirement for the early Venus flyby missions, a performance gain is realized by increasing the power level from 1000 mw to about 1500 mw for both families of curves.

For these early Venus flyby missions the 1500 mw value seems to be rather well matched with mission requirements, orbital escape propellant capacity of 110,000 - 120,000 lb and the orbital launch capability of a single C-5. However, for slightly more ambitious missions a considerable, further gain is realized by increasing the power to 4000 mw, which of course also decreases the operating time of the nuclear propulsion stage. One hour operating time limit lines are shown on the graph for the families of curves representing both speed values. Fig. 23 also shows that for the injection payload masses associated with flyby missions, there is very little, if any, to be gained by going to power levels above 4000 mw and in fact, there is a net reduction in performance by going above 4000 mw in the lower mission requirement regions indicating that the gain in performance afforded by the higher power level is more than offset by the increased mass of the propulsion system.

Earth Entry Systems

The Earth and planetary entry phase of the study was conducted by R. R. Titus with assistance from J. H. Chin. The study of entry into Earth's atmosphere is primarily concerned with entry velocity. The general effect of the high entry velocity associated with Mars and Venus trips, as compared with lunar flights, is to greatly reduce or eliminate entry corridors and to introduce aerodynamic heating phenomena which were of minor significance for the lower entry velocities.

This study is an outgrowth of past entry problems; beginning with missile cones and continuing with entry of the Discoverer capsules, the Mercury program, and the presently proposed Apollo missions. Each, in turn, has presented new and increasingly difficult problems. These require major restrictions and design limitations compatible with prescribed human tolerances. For the Mercury program, deceleration levels and aerodynamic heating pulses to which the crew is subjected are predictable by theory and are substantiated experimentally.

In the Apollo program entry at parabolic velocity tends to subject the crew to higher deceleration forces and it also introduces an additional aerodynamic heating phenomenon, viz. that of radiation from the gases behind the bow shock. For the return from interplanetary flight, where entry is at hyperbolic velocity, the entry corridor is small or nonexistent for the ballistic case, and lifting entry is required. Also, the heating definition now includes the very involved phenomenon of nonequilibrium radiation heating which may be the major contributor, and the deceleration forces must be closely controlled by entry trajectory selection.

Four basic concepts of entry systems believed to be operationally available in the early 1970 time period were considered. These are: (1) an Apollo type capsule, (2) an Apollo capsule with exospheric rocket braking to Apollo design speeds, (3) a conical shaped vehicle using a small nose radius to minimize the stagnation heat rates, (4) a high altitude drag brake system. Fig. 24 shows a comparison of these four entry systems in terms of their total mass as a function of entry velocity. Although the Apollo configuration is very good for the re-entry speeds associated with lunar return, the heat shield mass increases very rapidly with increasing re-entry speeds in order to survive the increased convective and radiative heat flux. Convective heat flux increases approximately as the third power of the velocity and is inversely proportional to the square root of the nose radius of the vehicle. The equilibrium radiative heat flux may increase as much as the 12th or 14th power of the velocity and is directly proportional to the nose radius.

For the higher entry speeds entry system mass can be reduced from that shown for the Apollo type capsule by adding a retro propulsion stage to break an Apollo type vehicle down to its design entry speed. A propulsion stage mass will increase exponentially with the required velocity reduction. Total mass, of course, is dependent upon the specific impulse of the propulsion system and the structural factors used.

If an ablative vehicle is designed with a nose radius to minimize the total effect of the convective, equilibrium radiation, and nonequilibrium radiation heating rates, the entry system mass can be reduced still further as shown by the curve for the optimized capsule in Fig. 24.

The fourth curve is for a vehicle design consisting of the Apollo capsule fitted with a large light-weight drag brake to utilize the upper atmosphere for deceleration while greatly reducing the stagnation heat flux due to the low densities. A propulsion system is included to provide, or supplement, the lift vector required for a single pass entry.

Since it was found in the study that a saving in entry system mass is amplified in terms of mass requirements in Earth orbit by a factor of 20 to 40, a reduction in the entry system mass becomes very important.

Re-entry Trajectory Mode

Re-entry of the drag brake system will consist of four major phases as shown in Fig. 25. Phase I is the near Earth hyperbolic approach trajectory and defines the entry velocity; magnitude and flight path angle. Phase II incorporates a ballistic entry and terminates when the local flight path angle becomes zero. This point is here defined as pull-out. Phase III begins at pull-out when the negative lift propulsion system is started to provide a single pass entry. During this phase additional atmospheric braking to parabolic speed occurs at a high altitude. A decreasing thrust could be used in order to maintain constant altitude. However, for this study a constant thrust was assumed instead, which means that initially thrust is insufficient to maintain altitude and the vehicle climbs. Near the end of this phase when parabolic speed is reached, the thrust is greater than that required for maintaining altitude and a negative flight path angle is generated.

The propulsion system was assumed to have a specific impulse of 300 sec. The thrust level and pull-out altitude can be chosen such that the flight path angle and speed match the Apollo type re-entry. At the end of Phase III, the drag brake, with its negative lift rocket system, is jettisoned. Phase IV is the final descent and recovery.

Stagnation Temperatures

Since the uncertainty of the total heat flux to the stagnation region is larger than the variation of flux from stagnation point to drag brake perimeter, a uniform total heat flux to the frontal surface of the drag brake was assumed.

The heat flux from the wake to the rear surface of the drag brake, expected to be much smaller in magnitude, was neglected. Transient conduction heat transfer analysis indicates that the skin attains approximately the radiation equilibrium temperature by radiating the incident, time-dependent, heat fluxes equally from both surfaces. Consequently, the maximum temperature is approximately the initial radiation equilibrium temperature.

Stagnation temperatures for the drag brake re-entry system were computed on this basis and are shown in Fig. 26 as a function of pull-out altitude and entry velocity. Although lower temperatures are experienced for the higher pull-out altitude, greater drag brake area and negative propulsive lift are required. However, for the lower pull-out altitudes the increased temperatures require some penalty in terms of mass per unit area of the drag brake system. Fig. 27 combines these effects and shows total entry system mass as a function of entry velocity and pull-out altitude. A 10-g limit line is shown on the figure.

Life Support Systems

This phase of the study effort, which was based on a three man crew for missions of 365 and 600 days, was under the direction of R. S. Thomas with assistance from eleven colleagues. A two-gas cabin atmosphere was recommended and assumed for the study, 180 torr O_2 and 180 torr N_2. Atmospheric water, urine, and carbon dioxide regeneration by physico-chemical means is preferred over an open or biological regeneration system. Metabolic supplies are calculated on the basis of an average metabolism of 2,820 K calories per day. Radiation shielding weights are based on a 0.001 probability of receiving 200 rads to the blood-forming organs from all sources of radiation throughout the mission.

Meteoroid shielding is based on Whipple's 1962 distribution and Summers' hypervelocity penetration equation. For the rotating (gravity providing) vehicle configurations, gravitational acceleration of 0.3 to 0.85 g's was assumed provided. Thermal control is based upon space radiators and circulating liquid coolants. A flow diagram for the recommended physico-chemical, nearly closed, life support system is shown in Fig. 28.

Greatest weight saving for regeneration systems is in the recovery of atmospheric water. The recovery of atmospheric water presents maximum saving for minimum effort and will doubtless be used in missions of more than a few days duration. The next greatest saving is in the recovery of water from urine. The problem here is somewhat more formidable although a number of solutions appear to be feasible. Generally, it is desirable to use a high temperature distillation to destroy bacteria and low temperature distillations to produce less ammonia. The system presently favored at LMSC is the vapor-compression distillation method. The recovered water is electrolyzed to provide oxygen and hydrogen for carbon dioxide reduction.

The smallest saving comes from the recovery of oxygen from carbon dioxide. The problems here are an order of magnitude greater than those of urine distillation. No complete system has really been developed even in the laboratory. It is noted in the block diagram that waste is accumulated in four different places. Leakage, as shown on the diagram, will be a function of cabin pressure. As mentioned, the partial pressure of oxygen was assumed to be 180 torr, to provide an alveolar partial pressure (the partial pressure in that part of the lungs containing gases effectively in equilibrium with the arterial blood) of 100 torr. It was further assumed that to avoid decompression sickness the cabin pressure should not exceed 360 torr if it contains substantial nitrogen. The 180 torr O_2 plus 180 torr N_2 was finally selected over lower pressures to maximize time for action, minimize fire and explosion hazards, and to minimize blower fan power.

Ionizing Radiation

The main radiation problem confronting manned interplanetary flight is that of solar flare protons. The unshielded dose from the largest of these events has been estimated at from 1000 to 8000 rads.

The proton producing flares can be roughly grouped into two categories, relativistic (R) and non-relativistic (N-R). The relativistic flares produce protons with energies of several hundred mev and higher. There have been seven to nine of these in the last 21 years. The non-relativistic flares occur about once a month during a solar maximum and about once a year during a solar minimum. Maxima (and minima) are about 11 years apart, with the next maximum to occur about 1969. The relativistic flares show some indication of occurring between maxima and minima. The duration of observable proton producing flares is from 10 to 100 hours.

A question still unsettled is the variation of dosage with radial distance from the sun. The model that seems to best account for the observed phenomena would indicate about an inverse 2.5 power diminution. For conservatism the inverse cube power was assumed to determine the Venus flux and the inverse square power to determine the Mars flux with extrapolation from Earth fluxes. Based upon this variation of flux with distance from the sun, Fig. 29 was plotted which shows aluminum shielding requirements versus the probability of receiving more than 200 rads aggregate to blood forming organs. Shielding for both Venus type missions of one year and Mars type missions of 600 days duration occurring during solar maximum and during solar minimum, are shown. A basic aluminum shielding area density of 56 grams/sq cm was assumed, with a variation in both directions depending upon mission requirements, to provide a maximum acute dose from relativistic solar flares of 25 rads and a 0.001 probability of exceeding an aggregate dose from all radiation of 200 rads.

A summary of the weight and power levels for the complete life support and crew protection systems for a three man crew associated with representative Venus and Mars trip durations is shown in Table 1.

Guidance and Control

For the study of guidance and control techniques, conducted by J. A. Carson, the interplanetary mission was divided into three phases: (1) injection, (2) midcourse and (3) planet approach, with the midcourse and planet approach phases repeated on the return leg of the journey. Guidance measurements for both the midcourse and planet approach phases are based on triangulation using optical angles and time. At least three angles at the time of measurement are required to establish a position fix. A minimum of two successive position fixes establish the required corrective impulse.

For the midcourse phase, angles are measured between pairs of celestial bodies, at least one of which must be a planet or the sun. As the target planet is approached the method is changed to one in which the angle subtended by the planet disk and the angles between two stars and the planet center are measured. Inertial guidance is assumed during the launch and the injection phase and during corrective thrust periods. The selected guidance system includes star trackers, a field scanning telescope, an accurate time reference, inertial platform with platform mounted accelerometers, attitude control sensors, servos and a computer.

Functions which should be delegated to the human navigator are manual sextant observations and assistance to automatic measuring devices by initiating navigation fix sequences and avoiding readings based on false targets. He may also make instrument substitutions and minor repairs and perform in-flight calibration of instruments using comparative checks or autocollimation techniques.

The angle measurement techniques for both midcourse position fix and planet approach fix are shown in Fig. 30. Angles between three selected pairs of celestial bodies are used, and in the vicinity of the planet, range is determined by stadiametric means. Perturbation techniques are utilized, which implies that the approximate position, velocity and time are known. This restriction seems reasonable because very large deviations from the nominal trajectory would probably be catastrophic.

A list of optical guidance and control components, their characteristics, and expected 1970 accuracy, is shown in Table 2. The estimated 1970 weight, volume, and power requirements for the trajectory determination system, attitude control system, computer system, and command link equipment are shown in Table 3.

Reconnaissance and Scientific Instrumentation

A feasibility technique study, performed by E. I. Curtis, of high resolution optical reconnaissance from a manned Mars flyby vehicle was based upon considerations of a previous study[7] which dealt with reconnaissance and scientific data collection requirements for both planetary manned flyby and orbiting missions. The special case of extremely high-resolution optics is the most productive and yet the most demanding reconnaissance function to be performed and has considerable design problems introduced by the dynamics associated with the vehicle and the mission. A major goal of the flyby mission is to provide landing site detail of sufficient refinement to implement future precision manned landings. In order to synthesize a simple system capable of meeting stated performance requirements; focal length, film resolution, stability requirements, light conditions, required film speed (ASA) ratings, varying frame rates, and data collection and transmission rates were explored.

An information and criteria flow diagram indicating man's role in the system is shown in Fig. 31. Reconnaissance and instrumentation functions are established by mission goals brought forth by both the questions which remain unanswered after flights of unmanned vehicles and research programs, and by the future plans for manned landings. The various functions to be performed and the environmental data from prior research, as well as limitations imposed by the vehicle dynamics, will define the sensors to be used which in turn will establish the various techniques and parameters.

The probes indicated in the block diagram, for which a large weight allowance was made in establishing mission requirements, would be released from the manned flyby vehicle to gather atmospheric and surface data. Information from the probe instruments will be read through the probe data link to the on-board data storage and processing center which is also receiving data from the on-board sensors. In the vicinity of the target planet, data will be accumulated at such a high rate that considerable attention to the data processing function is necessary if saturation of the data storage systems and the ground link are to be avoided. Here, man plans a very important role in making decisions regarding the storage and transmission of data as well as in the fields of target selection and determination of techniques to be employed.

A mission model is used to establish flight dynamics, geometry, data gathering time, and the resultant tentative reconnaissance and scientific instrumentation is shown in Fig. 32. The highest resolution sensor is the 120 inch focal length camera shown as item 1 using standard film. An electrostatic camera is shown as an alternate in the event that the radiation environment precludes the use of standard photographic film.

It is assumed that for either system a maximum resolution, at nearest planetary approach, will be 1.5 meters. However, the product of resolution and swath width will not be as great for the electrostatic camera as for the standard camera and film system.

Other instrumentation are listed in the order of decreasing resolution, with alternate systems, and information from alternate systems, bordered by dashed lines. The information from these on-board systems and that from the landed and/or orbiting probes, believed to require 16 TM channels plus a TV channel, result in the data quantity shown on the right side of the diagram. It appears that to transmit over a 36 day period the data from the reconnaissance and instrumentation equipment shown, for the mission model in which at least 3 x 10" bits of information are gathered during the 105 minute period in the vicinity of the planet, a transmitter power of 36 watts and bandwidth of 36 KC will be required for the 36 day readout time. For the remainder of the trip, with only the on-board scientific instrumentation and the continuous real time status reporting channel (shown at the bottom of the diagram) operating, transmitter power and bandwidth can be reduced to 6 watts and 6 KC respectively.

The weights, volumes, and power requirements for the on-board reconnaissance and instrumentation system, as well as for the probe instrument systems, is shown in Table 4. About three kilowatts of the indicated power requirements are for the coherent, side-looking radar. This power demand should not prove to be a great problem, however, because the total energy required (less than 6 kikowatt-hours) could be carried in the form of primary batteries for about 100 pounds or less.

Power Systems

Power system studies for early manned interplanetary missions, conducted by F. V. Bischof with inputs from G. E. Rich and C. M. Bennett, indicate that two types of space power systems are feasible. These are the nuclear and the solar dynamic conversion systems. The SNAP 2 nuclear reactor-turbo-generator system and the solar concentrator-turbo-generator system appear capable of producing the indicated power requirement of 5 to 6 KW and can be installed on either a rotating or a zero-g configuration. Fig. 33 shows a possible arrangement of a modified SNAP 2 installed at the end of a long separating boom of a rotating configuration.

For solar dynamic power systems, both rubidium cycles and biphenyl cycles are under consideration. Although the higher operating temperature of the rubidium system results in a considerably smaller radiator than that for the byphenyl system, total system weights are nearly identical. This is because the low operating temperature of the biphenyl system permits the use of light weight materials, such as aluminum and magnesium, for fabrication of the power unit and radiator components. Fig. 34 shows a typical solar generator heat balance for the rubidum cycle system with an output of 6 KW.

Vehicle Concepts

Several spacecraft configurations were generated and evaluated for early manned interplanetary flyby missions by J. Zoszak. Spacecraft were configured so that for minimum missions they could be launched with their three-man crew by a single booster to a low altitude orbit, and checked out on this parking orbit prior to injection into the requisite interplanetary transfer ellipse. A modified Apollo command module was used as one of the major components in the spacecraft and as the final Earth re-entry body.

Of the configurations examined the rotating (gravity) spacecraft received the major emphasis primarily because of the lack of knowledge of the effect on the human body of very extended periods of weightlessness. A solar powered version of the rotating spacecraft which was studied is shown in Fig. 35. The command module, which is basically a modified Apollo command module, serves as the launch vehicle for the crew, as the command and control center for the spacecraft during the long journey, and as the Earth re-entry body. The service module serves as the crew's living and recreation center, and contains the primary life support equipment.

These modules are connected to the central hub section by the extended rigid spoke structure. The command module has either a retro-propulsion or a drag brake system for providing the initial deceleration during Earth re-entry. The total mass of this combination of command module and its Earth braking system is equal to the mass of the service module. The central hub section has the midcourse propulsion unit attached at one end, the power supply unit at the other, and has the solar flare shelter mounted inside.

The spacecraft is folded into the compact assembly on the launch booster as shown in Fig. 36. The spokes are in their retracted condition to fit within the height of the payload envelope. At launch the crew is in the command module to which is attached an emergency escape rocket to pull the command module away in the event of an abort situation. In a normal launch this rocket is jettisoned after first stage burnout.

On orbit, the following erecting procedures are carried out: (1) forward section of the adapter is jettisoned, (2) command module is rotated 180° and is secured to the end of one spoke. This can be accomplished by either a cable system or by the attitude control system on the command module, (3) the service module rotates 90° and is secured to the other spoke. The cable system can also be used to move the spokes and module into position before the power supply is erected and checked out. All subsystems in the spacecraft are then checked out on orbit before proceding with the interplanetary phase.

The vehicle then has the configuration shown in the middle of Fig. 37. If the condition and operation of all necessary systems are found to be satisfactory, plans continue for orbital launch of the spacecraft when it reaches the proper point in its orbit on a subsequent pass. For this launching, the spacecraft is oriented relative to the velocity vector and the orbital launch booster system is operated to inject the spacecraft into the programmed heliocentric orbit. When the required velocity is reached the booster is shut down and jettisoned. The spokes are then fully extended and locked in place, the solar power unit is extended and oriented, and the spacecraft is either spun up to the desired angular velocity at this time or after sufficient sightings are taken to determine whether an initial velocity correction is required.

During Earth launch and orbit escape phases the probes shown on the diagram can be mounted within the lower adapter, between the hub and the injection booster. During the interplanetary phase these probes would be rotated as shown to avoid impingement of the plume of the midcourse rocket. Prior to launching the probes, the spacecraft would be despun and oriented relative to the destination planet.

A spacecraft weight summary is presented in Table 5 for three of the configurations investigated. These are, from left to right in the table: (1) the rotating configuration employing a solar dynamic power supply, which was just discussed, (2) a zero-g configuration employing a solar dynamic power supply, and (3) a zero-g configuration employing a nuclear power supply separated from the crew by the spent orbit escape stage which is retained. The retention of this stage accounts for the greater midcourse propulsion mass as shown in the table. It is noted that although the life support mass is less for the lower duration Venus trips, this reduction is approximately offset by the increase in solar flare shelter mass because of the requisite closer proximity to the sun.

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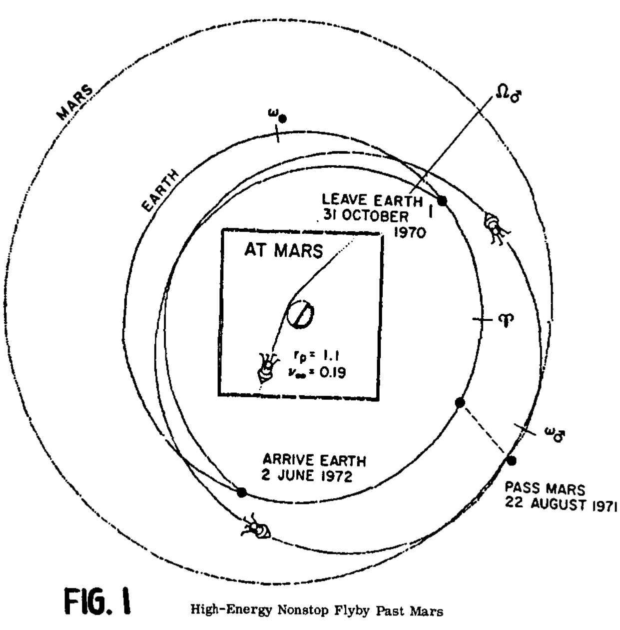


FIG. 1 High-Energy Nonstop Flyby Past Mars

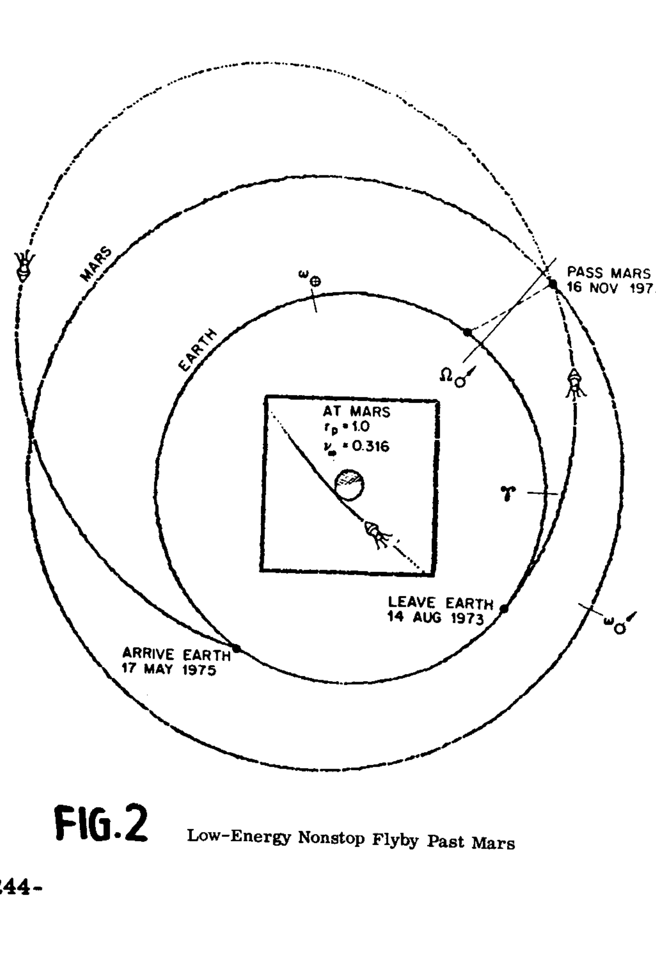


FIG. 2 Low-Energy Nonstop Flyby Past Mars

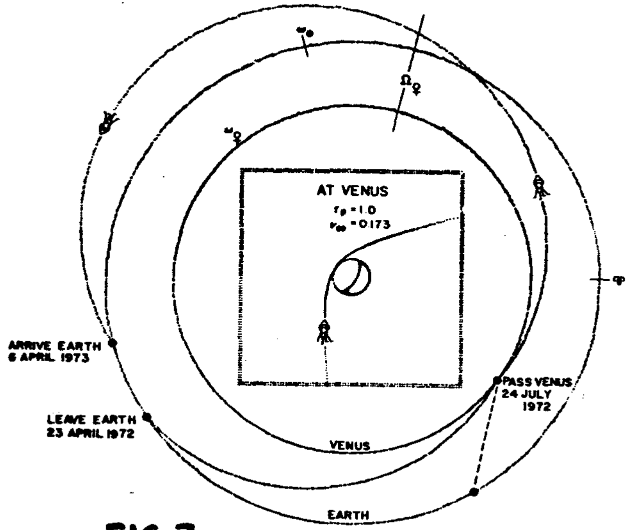


FIG.3 Nonstop Flyby Past Venus

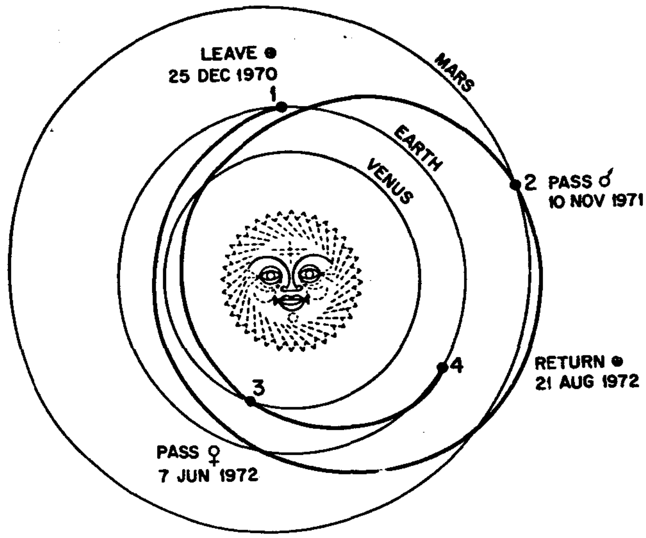


FIG. 4 Two-Planet Flyby 1970–1972

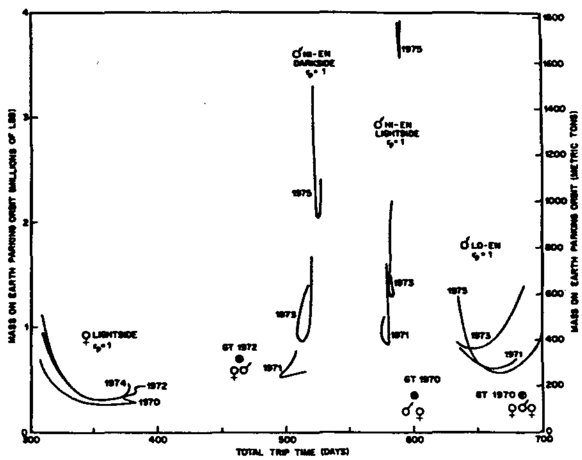


FIG. 7

Flyby Mission Requirements for Various Years (Chemical Escape – Retro Earth Entry)

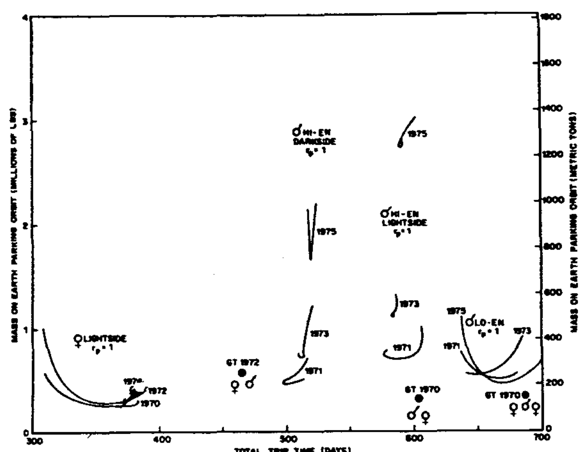


FIG.8

Flyby Mission Requirements for Various Years (Chemical Escape – Drag-Brake Earth Entry)

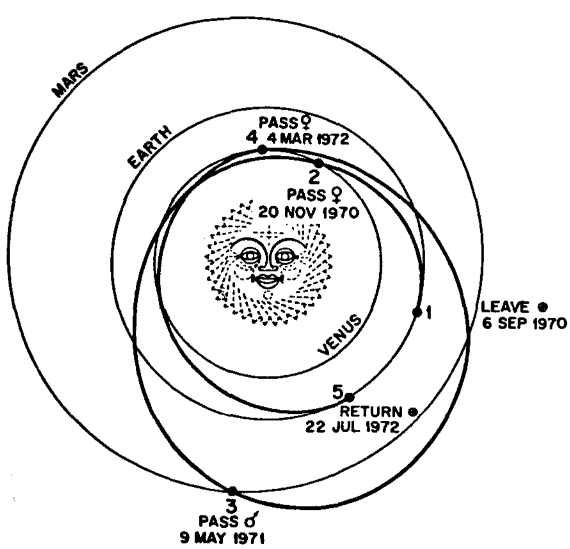


FIG. 5 Three-Planet Flyby 1970–1972

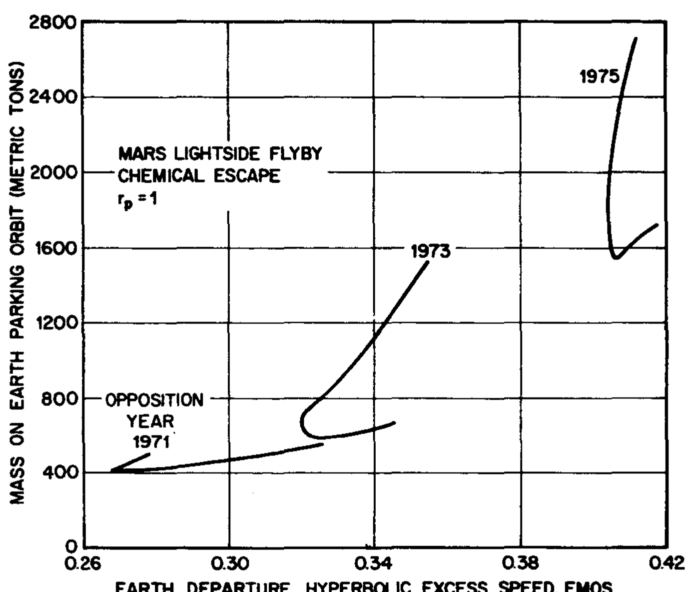


FIG. 6 Trip Selection Criteria

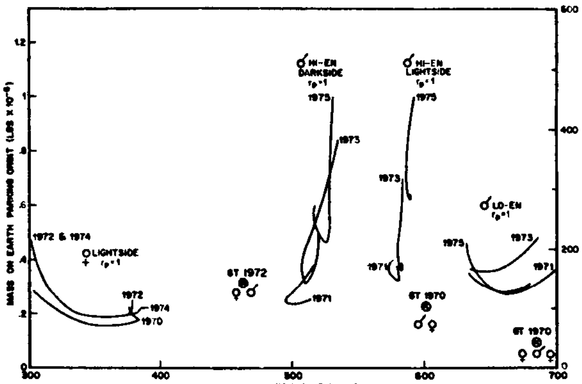


FIG.9

Flyby Mission Requirements for Various Years (Nuclear Escape – Retro Earth Entry)

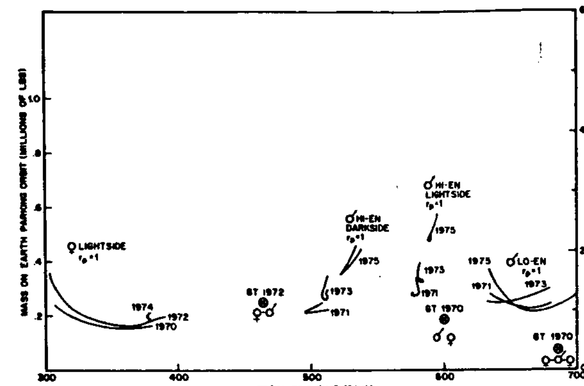


FIG.10

Flyby Mission Requirements for Various Years (Nuclear Escape – Drag-Brake Earth Entry)

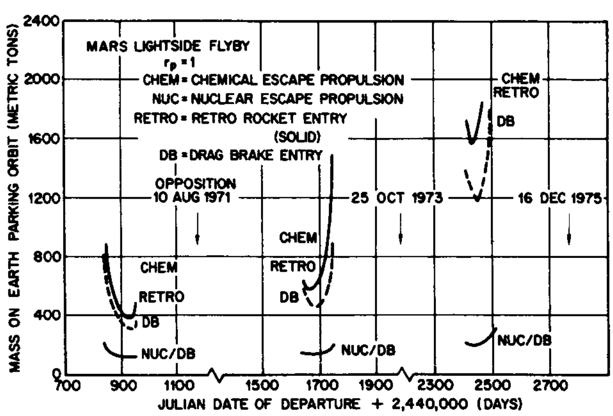
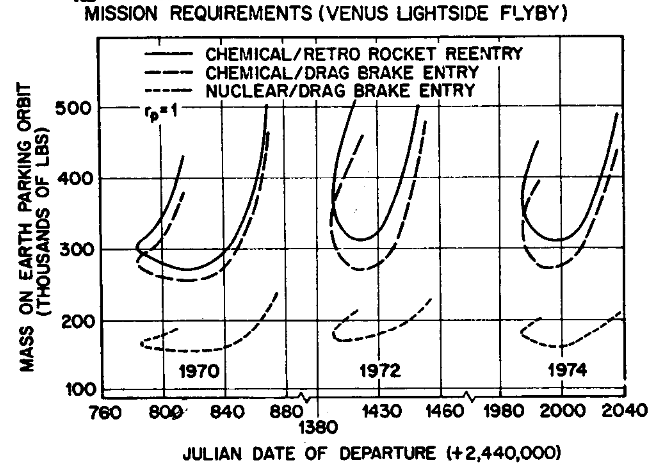


FIG.11

Mass Reduction Using Drag Brake Entry

FIG.12 EFFECT OF ESCAPE & ENTRY SYSTEMS ON MISSION REQUIREMENTS (VENUS LIGHTSIDE FLYBY)



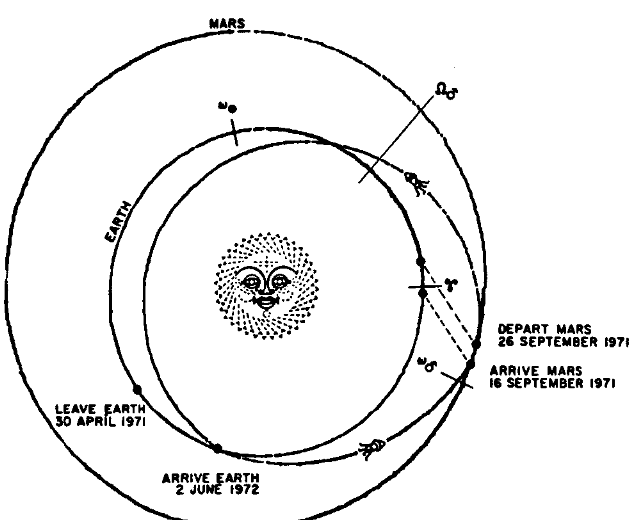


FIG.13

Mars Capture Mission with Ten-Day Stopover

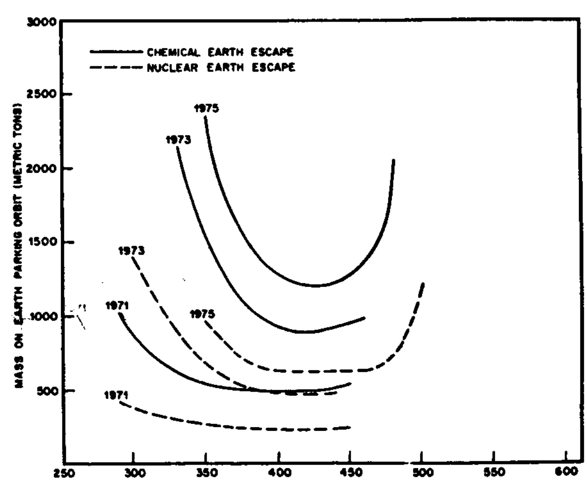


FIG.14

Mission Requirements for Various Oppositions – Mars Ten-Day Stopover (Chemical Escape – Drag-Brake Earth Entry) (Nuclear Escape – Drag-Brake Earth Entry)

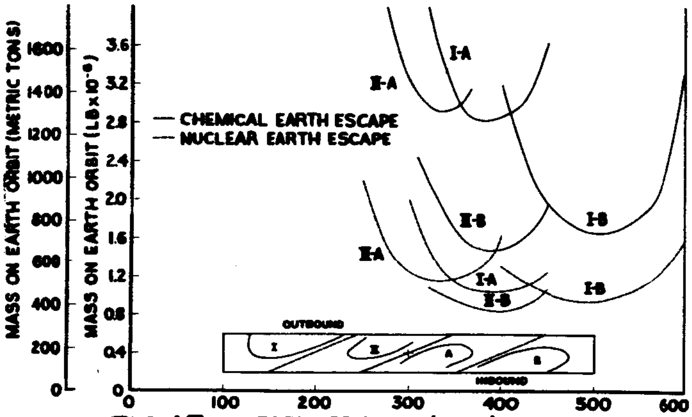


FIG. 15

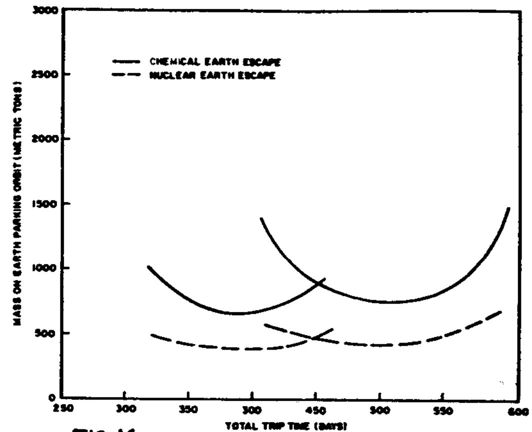
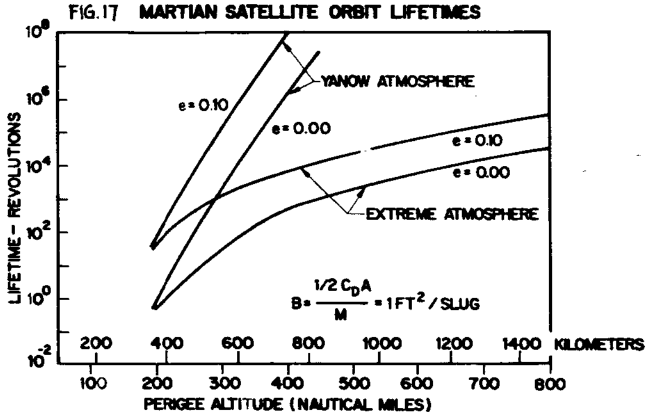


FIG. 16
Mission Requirements for 1972 Conjunction, Venus Ten-Day Stopover (Chemical Escape – Drag-Brake Earth Entry) (Nuclear Escape – Drag-Brake Earth Entry)

FIG. 17 MARTIAN SATELLITE ORBIT LIFETIMES



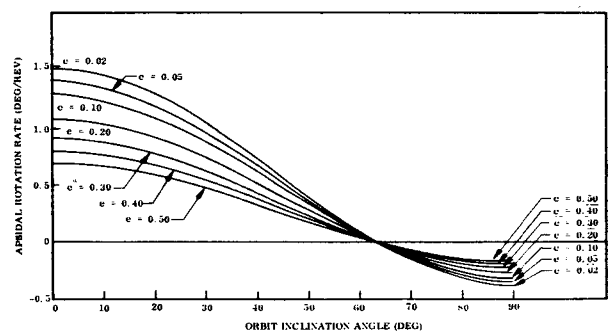


FIG. 18
Rotation Rate of Line-of-Apsides for Elliptic Orbits Around Mars, 300 nm Periapsis Altitude

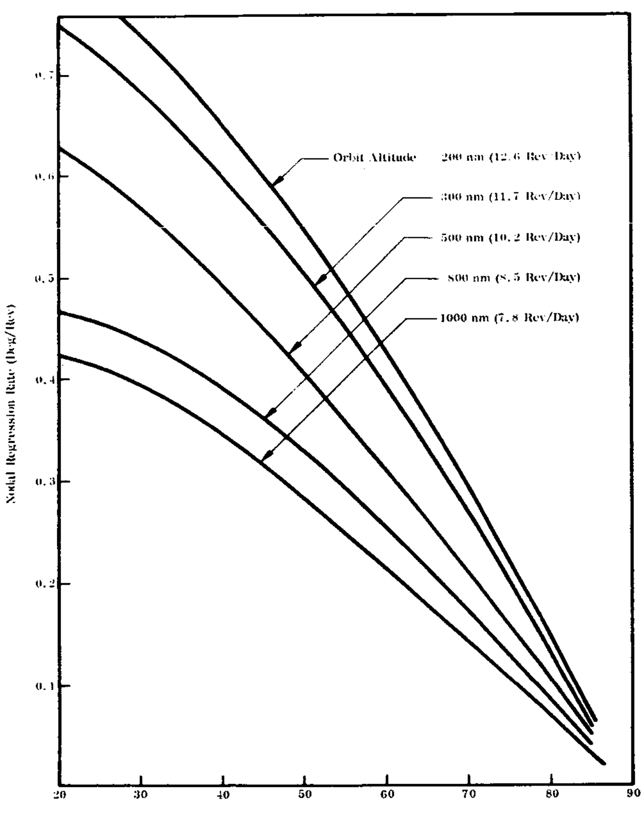


FIG. 19 Nodal Regression Rate for Circular Orbits Around Mars

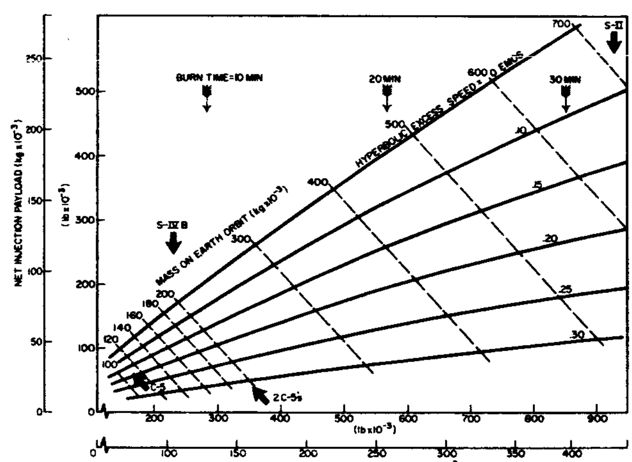


FIG. 20 Performance of Orbit Launch Booster Using Single J-2 Engine

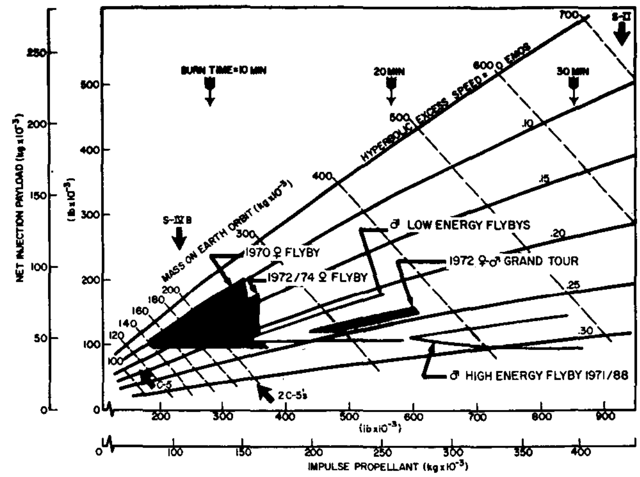


FIG. 21 Performance of Orbit Launch Booster Using Single J-2 Engine

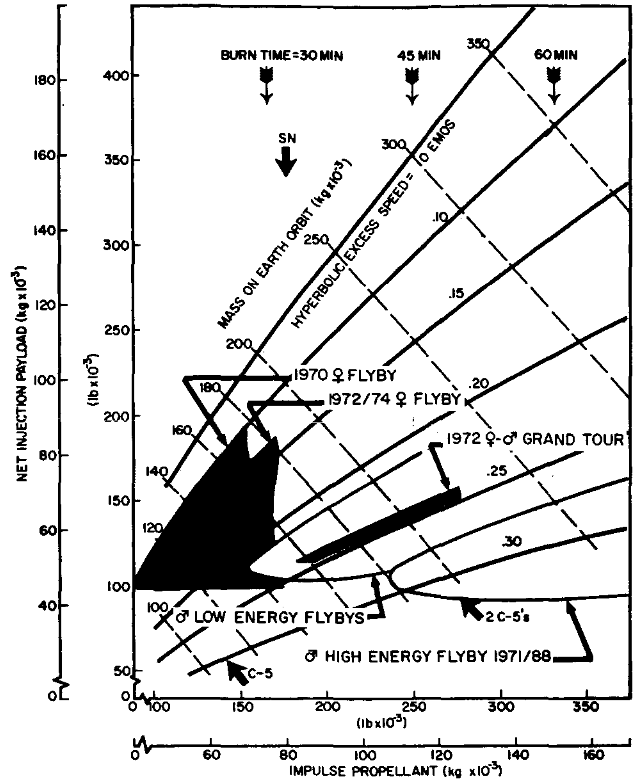


FIG. 22 Performance of Orbit Launch Booster Using 1500 Mw Engine

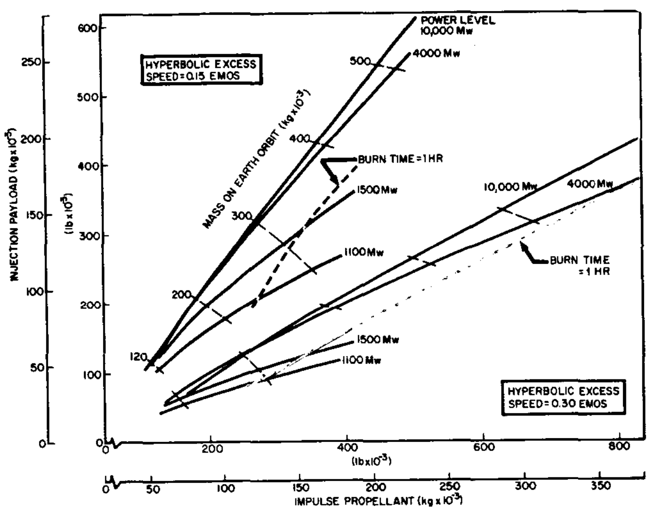


FIG. 23 Comparison of Nuclear Orbit Launch Vehicle Performance

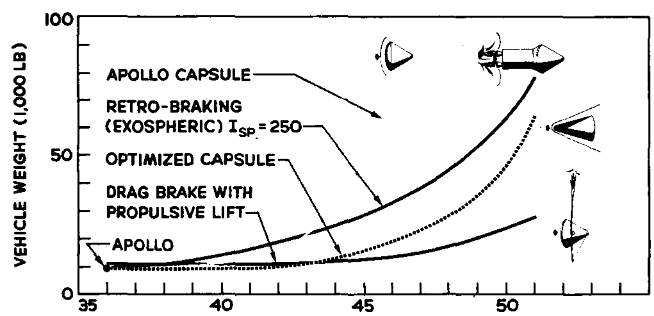


FIG. 24 ENTRY VELOCITY (1,000 FT/SEC)

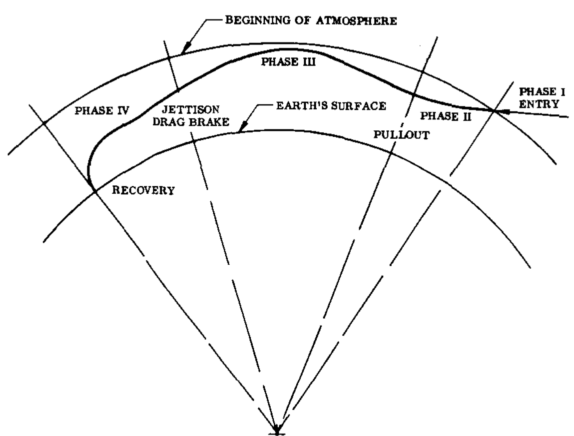


FIG. 25
RE-ENTRY Trajectory Phasing

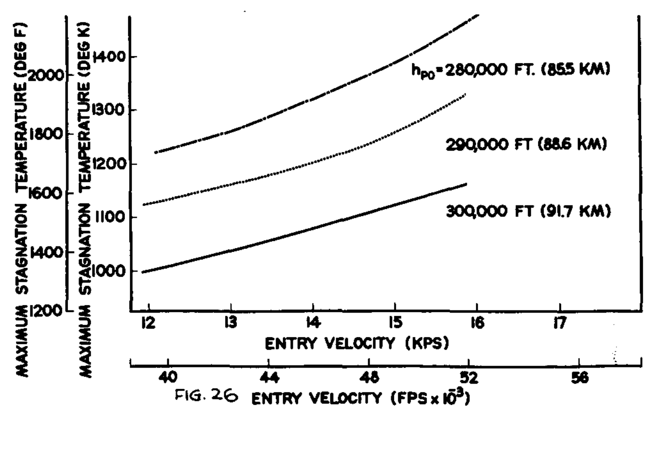


FIG. 26 ENTRY VELOCITY (FPS x 10^3)

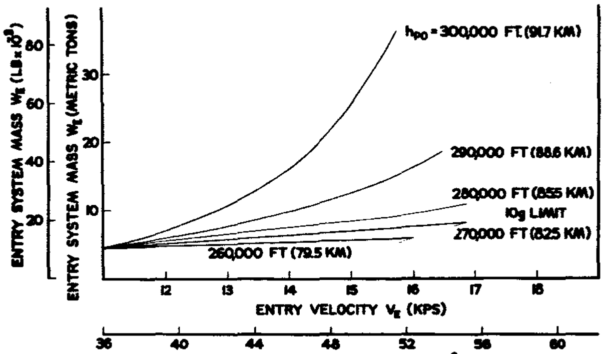
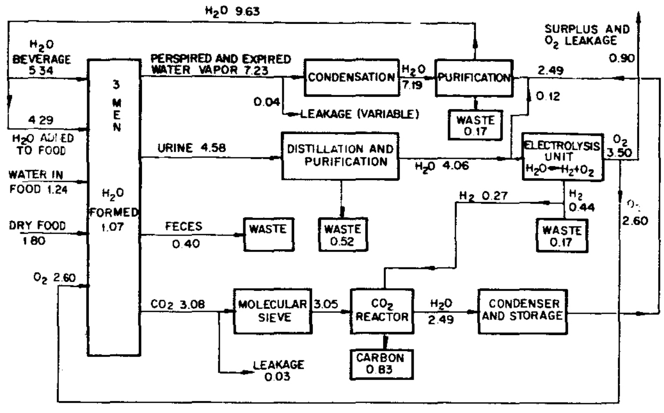


FIG. 27



ALL QUANTITIES IN kg PER DAY

FIG. 28 Nearly Closed Life Support System

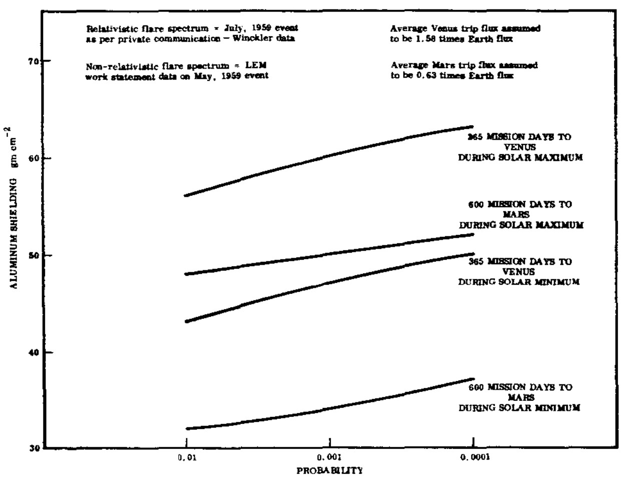


FIG. 29 Probability of Receiving > 200 Rads Aggregate to Blood-Forming Organs from Solar Flare Protons

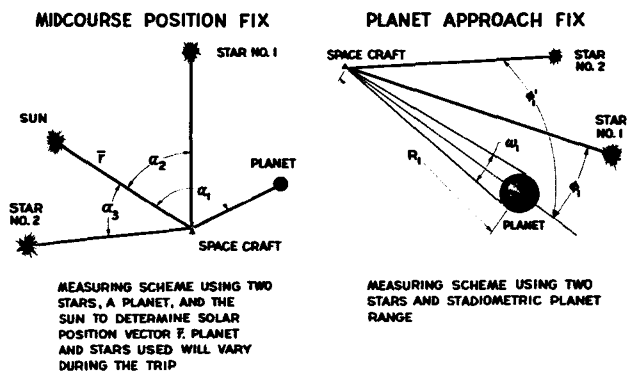


FIG. 30 Navigation Optical Sighting Geometry

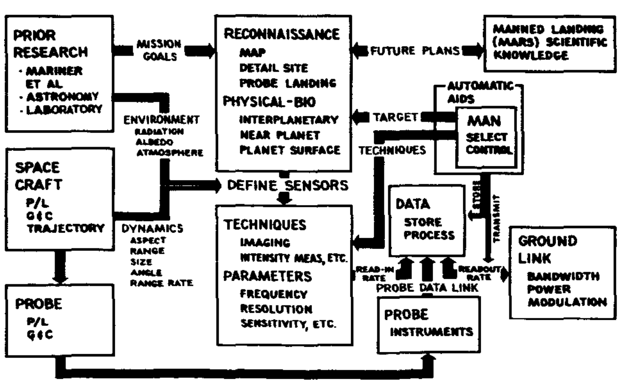


FIG. 31 The Manned Flyby Reconnaissance Role

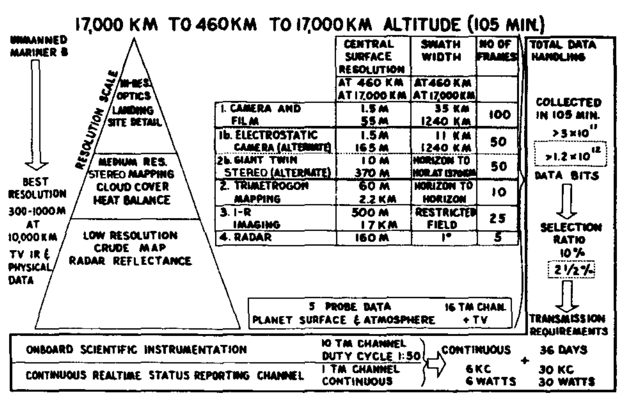


FIG. 32 Payload Performance Capability Manned Mars Flyby Reconnaissance

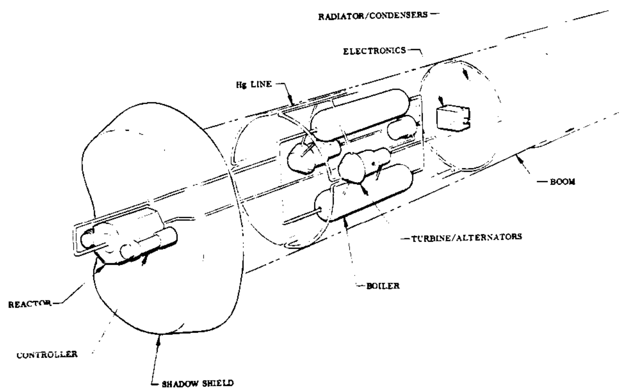


FIG. 33 Nuclear Power Plant – Rotating Vehicle Configuration

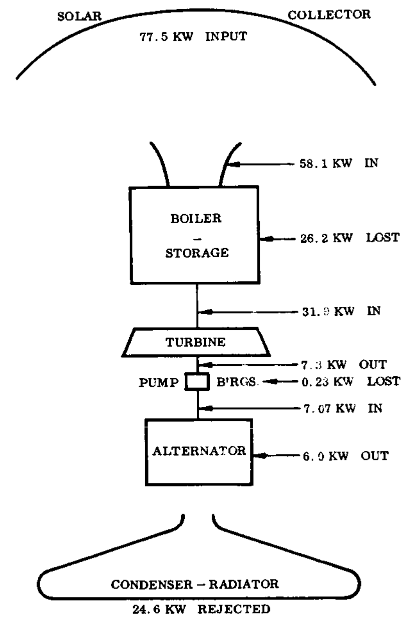


FIG. 34 Rubidium 6-kw Solar Turbogenerator

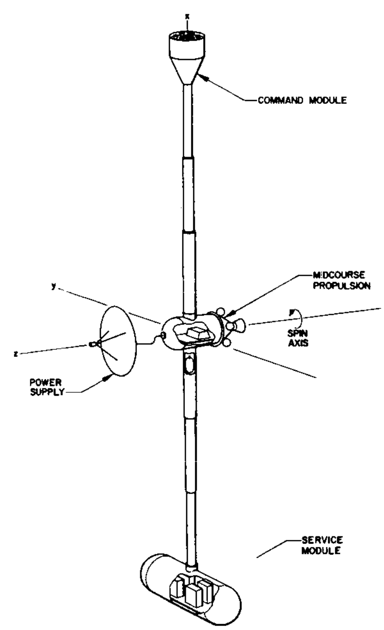


FIG. 35 Rotating Solar Powered Manned Interplanetary Spacecraft

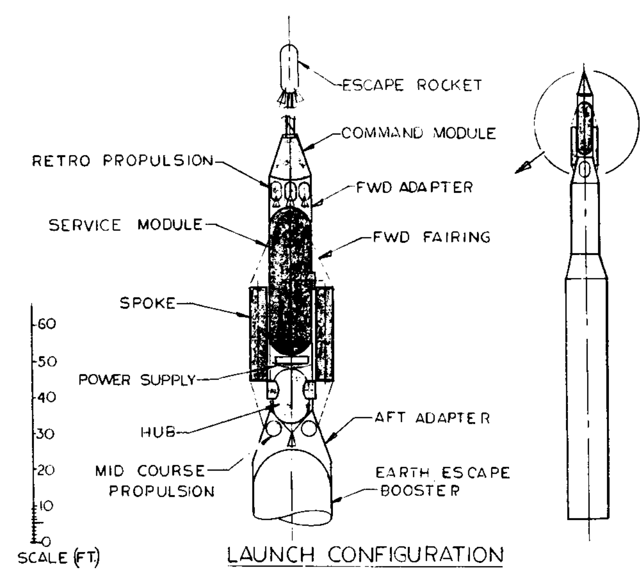


FIG. 36 Launch Configuration

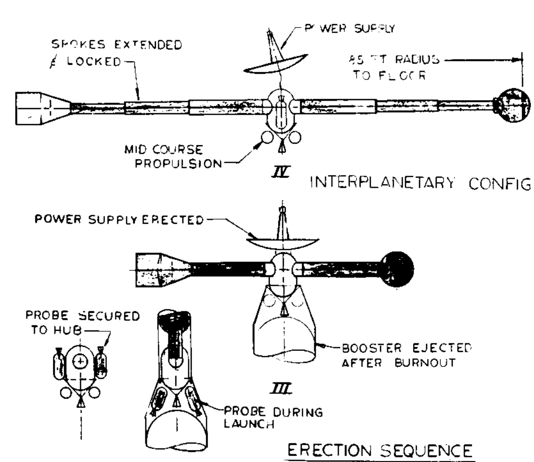


FIG. 37 Erection Sequence

TABLE-1 MASS AND POWER SUMMARY FOR LIFE SUPPORT AND CREW PROTECTION SYSTEM FOR THREE MEN (kg)

	365 Days		600 Days	
	Open	Closed	Open	Closed
Gas Supply	2,330	1,230	3,670	1,850
CO_2 Removal	90	–	90	–
Contamination Control	30	30	50	50
Food and Water	5,810	2,250	9,590	3,500
Crew Support Equipment	70	70	70	70
Temperature and Humidity Control	435	435	435	435
Regeneration Equipment	–	300	–	350
Sterilization	10	10	10	10
Sound and Vibration Control	10	10	10	10
Pressure Suits	100	100	100	100
Utility Water	10,000	200	16,000	200
Radiation Shielding	8,000*	8,000*	6,000*	6,000*
TOTAL	26,885	12,635	36,025	12,575
Power Level (Average)	2.6 kw	4.7 kw	3.1 kw	5.1 kw
Power Level (Peak)	3.9 kw	6.1 kw	4.6 kw	7.6 kw

*Assuming 365-day mission to Venus and 600-day mission to Mars

TABLE 2 OPTICAL INSTRUMENT CHARACTERISTICS AND PERFORMANCE

Guidance & Control Optical Components	Image Detection	Output	Accuracy 1970	Instrument Application
Star Trackers Precision Gimbal Servos	Image Dissection Tube, or Vidicon	Gimbal Position	3 – 5 sec	Midcourse position and align inertial components
Field Scan Telescope Locking Gimbals	Vidicon or Mosaic	Image Scan Signal Analysis	10 sec	Planet approach position
Sun Tracker Locking Gimbals	Masked Photo Electric Cells	Angle Position Signal to Attitude Servo	30 – 50 sec	Vehicle coarse attitude references
Star & Planet Tracker, Locking Gimbals	Photo Multiplier or Image Dissection Tube	Angle Position Signal to Attitude Servo	30 – 50 sec	Vehicle coarse attitude references
Telescope-Camera (Manual oper'n.)	Photo Plates* or Electrostatic Film	Manual Measure of Solar System Bodies Positions on Star Field	1 sec or better	Manually operated midcourse position
Space Sextant (Semi-automatic)	Observer's Eye	Prism Angular Position Pick-off	3 – 5 sec	

*Requires special protection from radiation fogging.

GUIDANCE AND CONTROL PHYSICAL SYSTEM CHARACTERISTICS SUMMARY

TABLE 3

(Estimated for 1970 State-of-the-Art)

	WEIGHT (kg)	VOLUME (m^3)	POWER (w)
TRAJECTORY DETERMINATION SYSTEM	65	0.14	200
Inertial Instruments, Optical Sensors, Atomic Resonance Time Reference and Associated Electronics			
ATTITUDE CONTROL SYSTEM	23	0.05	25
Inertial, Optical and Associated Electronics			
COMPUTER SYSTEM	32	0.10	150
Memory, Logic Elements and Input/Output			
COMMAND LINK RECEIVER AND TRANSLATION EQUIPMENT	2	7×10^{-3}	5
TOTALS	122	0.30	380

Notes: Environmental control equipment, attitude torquers, and power conversion equipment is not included.

Computer memory and clock are continuous; other items operated intermittently.

MANNED FLYBY POTENTIAL PAYLOAD RECONNAISSANCE & SCIENTIFIC INSTRUMENTATION

TABLE-4

	WEIGHT (lb)	VOLUME (ft^3)	POWER (kw)
ON-BOARD RECONNAISSANCE SYSTEM Detail-Site Camera (120 in. F.L.) Stereo Mapper, IR Imaging, Radar + Processors, Displays, etc.	750—1250	31—37	3.3 (105 Min) (Mainly Radar)
ON-BOARD SCIENTIFIC INSTRUMENTS Interplanetary & Planetary Measure — Fields, Particles, Air (Growth from Mariner-Centaur Experiments + Display/Analysis Aids for Manned Direction	450	17	0.2
ON-BOARD TOTAL	1200—1700	48—54	3.5 kw (peak)
PROBE INSTRUMENT SYSTEM TV, Weather, Gas, Radiation, Soil, Biological Tests + Beacon, Command Rcvr, Data Storage, Converters, Transmitter	210	8 cu ft	0.050 kw

TABLE 5 SPACECRAFT WEIGHT SUMMARY (KILOGRAMS)

	Rotating Configuration		Zero-g Configuration			
			Solar Power		Nuclear Power	
	♀	♂	♀	♂	♀	♂
Emergency Escape Rocket	4,535	4,535	4,535	4,535	4,535	4,535
Command Module	9,525	9,525	9,525	9,525	9,525	9,525
Service Module	2,950	2,950	2,270	2,720	2,270	2,720
Life Support	4,990	7,260	4,990	7,260	4,990	7,260
Power Supply	680	680	680	680	680	680
Hub and Spokes	3,630	3,630	–	–	–	–
Solar Flare Shelter	7,940	6,080	7,940	6,080	7,940	6,080
Midcourse Propulsion	7,875	8,010	7,210	7,300	9,840	9,980
Adapters and Fairings	1,725	1,730	1,790	1,880	1,810	1,905
Shield (Nuclear Power Supply)	–	–	–	–	2,220	2,220
Miscellaneous	1,360	1,360	1,135	1,135	1,135	1,135
TOTAL	45,210	45,760	40,075	41,115	44,945	46,040

♀ Venus flyby trip time, 330 days

♂ Mars flyby trip time, 600 days

MAN'S UTILITY IN MILITARY SPACE MISSIONS

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Introduction

I have been asked to discuss with you this morning, "Man's Utility in Military Space Missions," a very controversial subject with great emotional content. It is interesting to reflect on the fact that nearly all early thoughts on space travel dealt with manned systems and in particular, manned military systems. As our total national program emerges, it is surprising to see that the only serious manned space activities are directed towards exploration with the real utility of man in military space systems poorly defined and poorly supported.

The total national program is developing at an extremely rapid pace, with the rate of build-up now well over $1 billion a year per year, exceeding already that involved with the erection of the ballistic missile program several years ago. This great rate of advance is having a profound influence on the entire fabric of the country and to no ones surprise is creating all sorts of problems.

I will try to make the following points:

First, the advance of the total national space program has been amazing and as a result we now have a rather firm grasp on many aspects of space technology. The direction that the national program is taking is surprising to many who felt at the outset that it would be dominated by military programs.

Second, although many possible military missions in space have been identified by many studies, only Reconnaissance/Surveillance has achieved any real recognition and is being developed seriously.

Third, the so-called "space gap" has been closed in most areas of space activity except in the field of manned space developments. This remaining gap is being closed through the massive lunar program, Apollo, designed to put man on the moon and return him before the end of the decade.

Fourth, rationale for man in military space is usually based on doctrinal or other inadequate arguments. It is very costly to have man as a part of a space vehicle system and therefore if he is to be included, his contribution must be an important one. Possible good uses of man in military weapon systems is in the role of operator of varied sensors in a sophisticated, large space station for total surveillance and as an important subsystem in advanced intercept or inspection missions.

Fifth, the building block programs of Titan III, Blue Gemini and X-20 are extremely important and bear on the development of man's ultimate pay-off in space weapon systems.

Sixth, the great cost of placing payloads into orbit is such that it limits most of today's important manned military space possibilities. The frequency of military launches will probably permit economical recovery of booster stages. The two stage winged, fully recoverable, space launcher is an important possibility for the future and it should be included in the building block program.

Seventh, finally the military space program is floundering badly and needs a firm goal to focus its research and development objectives.

The Developing National Space Program

I think that all of us have been amazed by the rapidity with which our national space program has been developing during the past two years. I also believe that many of us are amazed at the direction this program is taking. Certainly the studies made in the late 1940's and early 1950's, mostly by the RAND Corporation, correctly predicted that the state of the art would support the orbiting of large payloads, correctly predicted that the impact of this on the mind of man would be startling, but did not predict that the major task of the national program would be going to explore the moon before a full exploration was made of the military impact of space operations.

Prior to Sputnik I, the idea of space operations for any purpose was given little acceptance at any level of the government or by the country at large. Except for a few enthusiasts who usually overstated space objectives, there was little real interest in funding the very expensive efforts involved. With 20-20 hindsight many people now write or speak of how brilliantly they predicted space activities and how they were a small voice lost in the wilderness. In point of fact, there was little real support in money or concept. There was the marginal IGY project Vanguard set up with inadequate funding and a booster designed to escape using the military ballistic missile boosters that could really do the job; there was a program in the Air Force, Weapon System 117L dealing with Reconnaissance Satellites, again with only token financial support, and the small voice of the Army team at Huntsville under Dr. Von Braun calling for support of space activities, again with no real support in the Army or elsewhere.

This small effort would have produced practically no results by this time had it not been for the shattering effect of the Russian space successes. The first shock came after Sputnik I in October 1957, and the second after the first manned orbital flight in the spring of 1961. These two events have shaken the confidence of the country to its foundations, called for new super management in the Department of Defense, new alignments of industry, great expansion of research efforts in the universities, a great emphasis on education to the higher levels of science and engineering science, called for the creation of non-profit organizations to manage massive weapons systems, created the large operating NASA out of the old NACA, and resulted in the creation of many new scientific advisory boards to all echelons of the military, the Congress, and to the President himself.

The present National Space Program was erected in two massive steps. After Sputnik I, the Air Force was given the green light and proper funding, after considerable dalliance with the old ARPA organization, to get on with Weapon System 117L. This resulted in the identification of the three unmanned programs, Discoverer, Samos, and Midas. The Army was directed to manage the synchronous active communication satellite "Advent" and the Navy was given "Transit" the navigational satellite. NASA was put in business with the first scientific unmanned satellite, mostly involving the explorer series and was handed the old Air Force program MIS, a program to put man in orbit at the earliest possible moment. This, after being absorbed by NASA, became Project Mercury. The Air Force pinned its hopes for more sophisticated manned space operation on project Dyna Soar, a hypersonic winged vehicle whose objectives were many things to many people. The main factor in all this was that space was now a good word and activities were started across a wide spectrum of possibilities with military, scientific experiment and exploration moving forward abreast with about equal emphasis.

This situation was abruptly changed with the first manned orbital success of the Russians in 1961. Here the second great acceleration took place, but this time a major divergence in objective became obvious. The major motivation of the new expanding program was to be "prestige" and the lunar program Apollo was identified. This focused the entire NASA space effort and has permitted the identification and funding of many sub programs in support of Apollo. In the meantime, many proposed military space programs have faded away as a result of studies involving mission requirements and cost effectiveness to a point where the total military space program is poorly defined and poorly supported.

Today then as we see the great Apollo objectives bringing into focus a massive effort in space exploration and scientific experiment under the direction of NASA, the military space mission becomes more vague and poorly defined making it harder to support research and development in these areas and to keep alive those programs that have been started.

Military Missions

Mr. John Rubel, Assistant Secretary of Defense, in a speech before the Aerospace Luncheon Club of Washington, D. C. in October 1962, made the point that of the total Defense Department expenditures allocated to space activities, about half were programmed for clearly identifiable systems with very promising and immediate military pay-offs, and the other half was in support of "building blocks" being developed to provide the option to do the military missions of the future that could not be clearly identified today. He also pointed out that in spite of much study by all concerned, there had been really very little new in the way of mission concept and as of today there were only a few credible military potentials for space. These included reconnaissance, communication and navigation. Beyond this were the building blocks, the largest single item being Titan III being developed in spite of the fact that there is no mission assigned to it as a hedge against future possibilities. Today we should include the Blue Gemini and Dyna Soar or X-20 as other major elements of the building block program.

The expenditures on the clearly identifiable space missions has been quite stable over the past few years except for those involving warning and inspection. The building blocks, however, have had and will have a difficult road due to their nebulous position with respect to identifiable missions. There seems to be much difference of opinion on these programs within the Department of Defense itself, particularly with respect to the two manned programs, Blue Gemini and Dyna Soar. One year they must be defended as programs designed to develop state of the art and flight experience, to study military potentials of man in space, and to yield important research information with no reference to immediate military mission. Several years later the climate will have changed and these programs must be defended on immediate military capabilities. The project people on the building block programs must stand loose and defend in the popular vein of the moment. In most cases, the delay in getting the policy word down through the various staffs takes about two years, and as the policy seems to change on two year cycles, it inevitably finds the project people 180 degrees out of phase with the policy makers.

The military space missions that appear today as probable or possible can be listed below in descending order of probability.

1. Support of Ground Activities - Communication, Navigation, Weather
2. Reconnaissance/Surveillance
3. Inspection/Interception
4. Defense of U.S. Space Activities
5. Defense against Ballistic Missiles
6. Offensive Weapon Delivery

Of this list, only the first two are completely credible today. The first, communication, navigation, and weather are space capabilities that improve the operational characteristics of our present ground based systems. The second, the total surveillance job, is the military space mission that is receiving real support and in some aspects, adequate funding. Inspection and interception is considered important by most, but in spite of this, the backing of this interest doesn't include a serious allocation of money. Defense against threats to our own massive space objectives is a mission quite similar to inspection and interception, but it has broader implications.

Defense against ballistic missiles has always been one of the great hopes for space oriented systems. Unfortunately, in the studies made to date there has been little confidence developed that today's state of the art will support a realistic solution to this difficult problem. It continues to be studied in the hope that someone will develop some new concept that will lead us to an acceptable solution.

Delivery of offensive weapons from space systems has always been the number one hope by all military space enthusiasts. Strategic bombing is the blue ribbon military job today and any new system that can be proven credible has little money problems to contend with, e.g., Atlas - Titan - MM - Polaris, etc. Unfortunately, no one has come up with a good concept of how this might be made a more effective delivery system than the ICBM.

We have then only one active military mission that has universal credence (Surveillance) and a great deal of money is being spent in these areas. Beyond that, however, the military is groping around for better state of the art, better concepts and better argument to develop military space missions further. Beyond this then are the three building block programs: Titan III, Blue Gemini, and Dyna Soar. This plus much applied research adds up to a total military space program of close to $1.5 billion per year. Mr. Rubel thinks this may be too much. He can only be proven wrong by better concepts and above all, by better arguments.

The Space Gap

The massive build-up in the national space program following the shock of Sputnik I in October 1959, is now bearing fruit in a massive series of launchings by both the NASA and the Air Force. This involves the use of standard launch vehicles and space craft, such as NASA's Explorer, Mariner, Echo, Relay, Tiros, OSO, and Mercury, and the Air Force's unmanned programs using Thor and Atlas Agena boosters. Both organizations have launched countless sounding rockets and research experiments as a by-product of many other booster launches. The USAF demonstrated the first recovery of a payload from orbit in its Discoverer program; the NASA has identified and investigated the nature of the Van Allen belts with its Explorer and the interplanetary phenomena of Solar proton propagation and Solar winds with their Mariner. Mariner II made one of the most spectacular flights in its recent transit of Venus this past December. Spectacular demonstrations have been made of intercontinental communication when European TV programs were transmitted through Telestar and Relay and Tiros has been a great success in acquiring good weather information.

The unmanned program has been very successful and each day brings more data and a better understanding of many aspects of space. There is little question that we are no longer behind Russia in unmanned space operations and the so-called space gap has been more than closed in this area.

Although Project Mercury has been a spectacular success in our manned space flight capability, it is still considerably short of the Russians. This gap cannot be closed before the Gemini program and many feel that it won't be really closed before the Apollo lunar program. In any case, there is little doubt that the Russians are much more interested in manned space activities than we have been and without question they are serious in their interest in manned military space possibilities.

Man in Military Space Missions

The part of the building block or future capability programs that is most hotly debated is the real utility of man in space military activities. On the one hand, man is a remarkable piece of machinery having very fine sensors and a built-in computer that is largely self-programming. He can adapt to new situations rapidly, perform certain maintenance and repair operations, add to the redundancy of control systems, and perform major tasks in sophisticated recovery operations -- all this for only 180 pounds. Unfortunately, man is a very fragile piece of hardware and must be protected from the harsh environment outside the atmosphere. He must be provided with life support elements such as oxygen, food, and waste disposal, and made comfortable and happy. He mustn't be made to withstand high or low extremes of temperature, too much acceleration in launch or recovery operations, and must be provided with abort capabilities throughout the flight envelope. All of this will add up to several thousand pounds over an unmanned system. As of today, the cost of launching payloads into orbit comes to approximately $1000 per pound; it is very expensive to include man in a space system. To include him, the system must be designed to make use of his very special talents, otherwise he'll just be along to carry the flag.

As of today, it has been impossible to make a convincing case for a manned military space mission, yet many feel instinctively that man will ultimately be involved in ways that cannot be clearly defined today. This situation has called for the building block programs already referred to - Titan III, Blue Gemini and X-20 aimed at an exploration of man's capability to contribute actively in military space systems. The ardor with which one defends these programs today seems to a function of various parochial interests, a situation that keeps these expensive programs in dire peril.

The inclusion of man in operational space systems cannot be defended on simple doctrinal grounds or simply for his maintenance or redundancy contributions. Rather a very careful analysis must be made to prove that including him will provide either unique capabilities unattainable in unmanned systems or low cost effectiveness in relation to particular unmanned systems.

As an example, it is very difficult to prove that a manned reconnaissance system has either a unique capability or low cost effectiveness over unmanned systems. This is particularly true if the reconnaissance job is done more or less the way it is being done or could be done with unmanned reconnaissance satellites. One is not confident that a man in an orbiting vehicle with, say, one camera can provide enough advantages over the single camera in an unmanned satellite to pay his way. On the other hand, it is quite possible to find some credence in a system that combines many different sensors into one payload with man present to select various sensor modes and to perform maintenance and repair operations. This capability would be maximized in a space station oriented to various activities concerning total surveillance, and possibly command and control. The crew of this station would have to be ferried to and from the space station in some sort of transfer service. Man's contribution would be very real in such a system, and its cost-effectiveness would be good.

The space station, although very costly to place into orbit in the first place, nevertheless has an effective advantage in that the very expensive sensors involved are neither thrown away or reentered. The space station can be considered a permanent station and the various sensors maintained and resupplied through ferry mission. All of the cost advantage of the space station would be dissipated however if the costs of ferrying men and material to the station were not minimized. Here the large cost of $1000 per pound is a staggering hurdle, and we must have relief from such costly operations. The best way that is known today for making an order of magnitude change in this cost is by recovery of launch vehicle stages for re-use. One can postulate that the ferry and resupply of a surveillance space station will be on frequent enough intervals to make such a recovery economically sound.

One of the most attractive possibilities for full recovery is the winged two stage airplane type launcher in the now familiar B-52/X-15 mode. The first stage would incorporate an air breathing propulsion system, perhaps a turbo ramjet, and be capable of cruising out to achieve a desired orbital plane and to accelerate to a launching Mach number and altitude. The second stage would have the capability of achieving orbital velocities with adequate payloads. The second stage could be propelled by high pressure rockets or supersonic burning ramjets. Both stages would be piloted, fully recoverable and reuseable. Although this type of launcher would be expensive to develop, the operational costs would be a great deal less than for a non-recoverable system, assuming that ferry missions were required on frequent intervals.

A pre-strike total surveillance space station then could maximize man's capabilities and result in a military space system that would have unique capabilities and good cost effectiveness. Man would be required in this space station to perform operational duties and he would have to be rotated on reasonable intervals. If the recoverable launcher system just discussed were developed to perform the ferry mission, man would also be involved in piloting both stages to perform launching, rendezvous, docking, transfer, and recovery functions.

Another very important role for man will be on inspection, interception, and negation missions. Here man's very real capacities for sizing up situations on the spot and taking action, both offensive or defensive, will be invaluable. He will be very effective in the rendezvous maneuver, can take proper action to identify the character of an unknown satellite, can decide what action is necessary as far as negation is concerned, can communicate information to the ground, receive instructions, feint, spoof, and perform other tactical maneuvers. It is hard to imagine this job being done without him.

The man in an inspection and negating satellite would also be most efficiently launched by the two stage aircraft type launcher as this will provide the fastest reaction time, the capability of acquiring any orbit at the earliest moment, the capability of launching covertly, and the lowest cost due to recoverable characteristics of the system. The space vehicle probably should be a lifting reentry type vehicle making it possible to maneuver on reentry to return to a secure base with the shortest possible wait in orbit and land accurately without the requirement of turning out large recovery teams.

The surveillance manned space station and manned vehicles to perform inspection and interception missions then are two strong possibilities for the future. They should be supported by launchers that have military capabilities of low cost, rapid reaction time, and simplicity. The Titan III is an important potential for this system as is the even more interesting two stage aircraft launcher to break through the $1000 per pound cost barrier. The building block program supports these capabilities and they must be pressed forward. We should add to them a space station capability and the two stage aircraft type launcher. If this could be identified as the manned military mission in space, it would focus the rambling and largely unsupported program of the Air Force, as Apollo is doing for the NASA.

The Blue Gemini and the Dyna Soar

Although we may be looking ahead to space stations, manned space interceptors, and new flexible and recoverable launchers, today we are hotly engaged in defending important elements of this future program. The most controversial are the Blue Gemini and Dyna Soar or X-20 programs.

The Blue Gemini is a development from the NASA's Gemini program laid on about a year ago to support the total Apollo mission through conducting manned space flight experiments in the interval between the end of the Mercury program and the beginning of Apollo flights. It was really identified as a program to provide "flying time" for the NASA team that would eventually have to face up to Saturn and Apollo. It was felt that Mercury was too small and too critical on its Atlas (D) booster for any further development. Gemini, making use of Titan II, could be launched more reliably, carry more payload, and perform some experiments such as rendezvous and docking. Recovery was to be a little more sophisticated than Mercury in that it was to be a trimmable ballistic configuration with a hypersonic $(L/D)_{max}$ of about three tenths (0.3). It is also to use a Regallo sail wing to enable it to perform horizontal landings on skids. The Air Force, also interested in acquiring "flying time" has identified Blue Gemini and hopes this program will help it along to the next step, the Dyna Soar or X-20.

The most controversial program of all is Dyna Soar and this comes about in large part due to its past history. Dyna Soar was a boost glide bomber in 1956, a reconnaissance system in 1957, an advanced hypersonic flight experiment in 1957, an orbital weapons system in 1958, a suborbital aerodynamic experiment on Titan I in early 1960, on Titan II in late 1960, and a fully orbital space vehicle on Titan III in 1961. It has been massaged by everyone including continual review by the USAF Scientific Advisory Board, by the DSB, by PSAC, by RAND, by NASA, and by WSEG. It is a miracle that it still survives.

The crux of Dyna Soar is its lifting reentry capability with relatively high hypersonic L/D, and as a consequence a large lateral maneuver range. As a by-product it will have the capability to perform horizontal landings without auxiliary aides and incorporates an efficient radiation cooled structure. The great emotion over the program involves the debate over the value of all this in light of the fact that there is a severe weight penalty involved.

In 1960 Dyna Soar was identified as a program to achieve suborbital velocities (about 17,000 ft/sec) using Titan I as a launcher; later that year Titan II was incorporated as the booster giving Dyna Soar a maximum velocity of 22,000 ft/sec. All this time many felt that the Air Force should get on with space experiments at once and the heavy winged glider should be replaced by a lower L/D and more efficient structural shape that could be orbited by Titan II. Many still think so and even though now Dyna Soar is perched on top of Titan III with plenty of capability to give it orbital velocities with good payloads, the critics still feel Dyna Soar is inefficient and the large lateral maneuvering on reentry and simpler landing capabilities are not important enough to justify the cost. Most of these people think that Blue Gemini will do all that was originally programmed for the Dyna Soar.

There are those of us who violently oppose this view and are fighting to keep the Dyna Soar in the program. We feel the lateral maneuver capability has great military potential and compromises to lower the L/D are leading us in the wrong direction. Dyna Soar is an important program and I only hope it can be kept alive. Most of NASA, the Air Force, and a large portion of DOD agrees with this.

A Focus for USAF Space

In summary then, manned military space missions have not received enthusiastic support during the past few years and its posture with respect to the total national space program continues to decay. This is due to a lack of a credible first line mission, the development of highly sophisticated manned space vehicles by the NASA, the parochial arguments for the actual configuration of the space vehicle, the poorly thought out doctrinal arguments for man in military space that rightly receive little credence by the administration, the placement of the program that exists in the so-called building block part of the military space plan, and, of course, the very large costs involved.

The military space program continues to progress without any proper focus. The only real mission it has today is in a sophisticated unmanned system and this is, of course, in the heavy classified areas so that its progress and accomplishments are unknown to most. The manned military space capabilities could be maximized in the surveillance space station, in the manned interceptor, and in the fully recoverable launcher. The Department of Defense would do well to get on with these systems as they have real military importance and would focus the otherwise rambling military space R & D efforts.

In the meantime, the projects that we have going, the Titan III, Blue Gemini, and Dyna Soar are extremely important and must be continued. If these drop out of the program, manned military space systems are dead for the next decade and the military problems this could create might be very large.

SPACE MATERIALS FOR THE FUTURE

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With increasing complexity and time durations for space missions, some new properties of materials are becoming important. At the very low gas densities encountered in space, bearings, sliprings, gear trains and other devices requiring motion between surfaces show increased tendency to cold weld and normal lubricating techniques cannot be used. Work on cohesive properties of materials at low pressures are described and possible methods of overcoming cold welding problems are discussed. Recent developments in superconductors are reviewed and applications for these materials in space vehicles are discussed. A superconducting material which should make attainment of fields of 100 kilogauss practical is described.

AUTHOR

The materials problems of our national space program are the most diversified and specialized which have been encountered in connection with any single technical objective in this country. The purpose of this paper is to review the nature of these problems and to discuss the general approach necessary to solve them. In addition, two important research areas will be discussed in more detail. One of these - the operation of mechanical devices in space - is growing increasingly important and may seriously handicap our space effort if it is not more fully investigated. The other area concerns superconductors, a relatively new class of materials which will almost surely find use in the space program and therefore deserve careful evaluation for this application.

In the early days of our space program, the general principles of chemical rockets were available to us and our efforts were primarily concerned with scaling up so that vehicles could be propelled into space. The question of coming back to earth was a more serious one at the time because there was no good method available for withstanding the high temperatures of re-entry. An intensive materials research program was initiated which overshadowed all other space materials work. Eventually, several solutions were found to this problem and, while there remains much to be done in the field of re-entry, the interest in the materials for space use has taken on far broader aspects in the last several years.

Table I outlines some of the major materials requirements in our space program. This list is by no means complete but will serve to illustrate some of the difficulties we are facing. In reviewing this outline, it is interesting to see the wide diversity of materials requirements. It is clear that a broad range of talent is going to be necessary to solve these problems. In most cases, the demands being made on the materials are pressing them to the very limit of their performance capabilities. Compounding this situation, difficult combinations of material properties are demanded. For example, in the case of future chemical propulsion systems, nozzles must not only withstand a temperature very near the highest melting point known but they must provide erosion and corrosion resistance while they do so. A thermionic converter powered by nuclear fission demands a cathode which is capable of operation at around 1500°C, but it must also at this temperature have

TABLE I

SOME OF THE MAJOR MATERIAL REQUIREMENTS OF THE SPACE PROGRAM

Device	Materials Application	Requirements
Propulsion		
Chemical	Propellant (Solid)	High heat of reaction per unit weight but sufficient stability to permit mixing and storage without premature reaction.
	Nozzles (Liquid and Solid)	Resistance to temperatures in excess of 3500°K, also erosion and corrosion resistance. Minimum weight desired.
	Storage Tanks (Liquids)	High strength-weight ratio at cryogenic temperatures.
		Cryogenic insulation capable of acceleration without damage, having minimum space and weight requirement and capable of permitting storage of cryogenic fluid for periods as long as a year.
Ion Engines	Ionizer	High temperature stability, compatibility with material to be ionized, porous structure.
	Accelerating Electrodes	Low sputtering rate, insolubility in ionizer material, no reactivity with ionized material, expansion coefficient similar to ionizer.
Nuclear Fission	Fuel Elements	High neutron efficiency, dimensional stability, strength at high temperature, shock resistance, good thermal conductivity, minimum weight.
	Nozzles	High temperature stability, compatibility with propulsion gas, erosion resistance.
Thermonuclear Engine	Magnet	Superconductor capable of generating fields of 100 kilogauss or more.
	Insulation	Combination of insulation suitable for minimizing heat transfer between magnet at cryogenic temperature and plasma at millions of degrees.

TABLE I
(Continued)
-2-

SOME OF THE MAJOR MATERIAL REQUIREMENTS OF THE SPACE PROGRAM

Device	Materials Application	Requirements
Protective Devices		
Re-entry	Nose Cone	Stability at temperatures of 3000°K or more, low rate material loss, minimum weight, maximum insulating value for protection within nose cone.
Shielding	Shield Material	Capability of stopping high speed atomic particles, minimum weight and volume.
	Magnet	Superconductor to generate fields up to 50 kilogauss.
Temperature Control	Vehicle Surface	Controlled ratio, absorptivity for visible radiation to emissivity in infrared radiation. Must resist changes due to combination of ultraviolet radiation, particle radiation or exposure to vacuum.
Aerospace Structures		
	Hot Frame Structures	Low creep and sustained strength at 1650°K, future extension to 2800°K, oxidation resistance.
	Control Surfaces	Low warpage and erosion at re-entry temperature up to 2200°K, oxidation resistance.
	Thermal Barriers	Low heat transfer to cold structures, low emissivity of multiple radiation shields, light weight and oxidation resistance, low thermal diffusivity.
	Lift Surfaces	Self-cooling by ablation or porous structures for transpiration cooling.
Energy Conversion		
Photovoltaic Converters	Energy Conversion Material	Generation high EMF from visible radiation, availability in large thin sheets, resistance to radiation damage.

TABLE I
(Continued)
-3-

SOME OF THE MAJOR MATERIAL REQUIREMENTS OF THE SPACE PROGRAM

Device	Materials Application	Requirements
	Surface Coating	High electrical conductivity, optical properties to control temperature and minimize loss of useful radiation, protection against particle radiation.
Thermoelectric Converters	Junction Materials	Generation of high EMF per unit temperature difference across junction, high electrical conductivity, low thermal conductivity, suitability for high temperature service.
Thermionic Converters	Cathode	High temperature stability, chemical stability, high electron emission, low evaporation rate, compatibility with alkali vapors.
	Anode	Temperature resistance, dimensional stability, chemical stability, low work function.
	Insulating Spacers	High electrical resistance, good temperature stability, chemical stability.
MHD Converters	Magnet	High field superconductor.
	Electrodes	High temperature stability, erosion and corrosion resistance.
	Containment Structure	High temperature stability, erosion and corrosion resistance.
Nuclear Fission	Fuel Elements	High neutron efficiency, strength at high temperature, shock resistance, good thermal conductivity, minimum weight.
Thermonuclear Converter	Magnets	Superconductivity capable of shaped fields up to 100 kilogauss at 10°K, efficient thermal insulation.
	Containment Vessels	Low outgassing and sputtering for 10^{-9} torr or lower service, minimum secondary electron emission.

TABLE I
(Continued)
-4-

SOME OF THE MAJOR MATERIAL REQUIREMENTS OF THE SPACE PROGRAM

Device	Materials Application	Requirements
	Ion Sources	High ion yields, low contamination of ion beam by sputtering or outgassing.
	Thermal Insulation	Light weight capable of conserving high temperature of plasma, highly resistant to radiation damage.
Mechanical Devices		
Bearings, Gears, Sliprings, and other moving parts	Lubricants	Low vapor pressure, resistance to radiation damage, means of distribution, suitable performance in absence of oxygen.
	Low Friction Surfaces	Resistance to cold welding and wear.

strength, dimensional stability, a proper work function and ideally provide a suitable sheath for a fuel element.

It is clear from these examples why more and more attention is being focussed on composite materials. In order to get the range of properties demanded for space devices, it will be more frequently necessary to resort to coatings of porous material impregnated with a second material and structures which consist of an aggregate of materials bonded together. Present techniques for construction of nozzles and nose cones have relied heavily on use of composite materials. More and more examples of this technique will follow. One such possibility being investigated at the present time is in connection with some of the very highly energetic fuels being considered for solid propellants. While these materials show much promise of increased specific impulse of engines, they are often so active that it is difficult or impossible to handle them for manufacture of propellant grains or to store them after compounding. The use of protective coatings for the individual particles of these materials is being investigated and there is hope that these coatings though protective will have fuel value so that little energy loss will result in their use.

Because of the complexity and extreme performance demand for space materials, it is going to be increasingly important to bring materials people and design people into closer contact. There is little chance of solving most of the problems in Table I by ordering catalog items from a material supplier. The communication between these people cannot be limited to specifications written by a design engineer for delivery by a materials engineer in an arms length deal. Most of these problems will best be solved by the device engineer and the material engineer educating each other sufficiently so that they can jointly arrive at some compromises to solve the problem - a compromise which neither would be likely to comprehend alone. While there are a great many capable materials people in the organizations concerned with device development, there is obviously far more of this specialized talent in the materials industry. We will make more progress on our materials development through increasing alliances between the organizations specializing in devices and those who are expert in materials.

Mechanical Devices in Space

Most of the requirements listed in Table I are so unique that the solution of one requirement will offer little help on the others. There is, however, one area which will have an exceedingly broad application and which must be well understood if we are to be successful. This is the performance of mechanical devices in space.

Mechanical devices involving bearings, gears, and sliding parts have become so routine in our earthbound thinking that there has been a tendency to overlook possible problems with these devices in space. For terrestrial use, we have developed systems of lubrication and bearing design which let us quite accurately design for varying loads, temperatures and surface speeds. It is only when we get to the extremes of these conditions that we have diffuclty. Unhappily, there are some important differences between the environment of space and that on earth which have an important effect on the behavior of mechanical devices. In the first place, there is essentially no atmosphere in space. Figure 1 gives an indication of the great reduction of residual gas which occurs as one goes into space[1]. This chart is expressed in terms of the relative density of gas molecules in space as compared with the atmosphere on earth. One does not need to go very far away from the earth before the gas density reaches only a minute fraction of that which we have here. At 1500 miles away from earth, the molecular density of the atmosphere is only about 10^{-17} times that on earth. This is equivalent to a gas pressure of around 10^{-14} torr.

Under these conditions, if a surface by one means or another becomes clean of its protective oxide layers there is little opportunity for reoxidation. Figure 2 gives some idea of what these relationships are like[2]. It will be seen that even 500 miles from the earth (around 10^{-11} torr) a surface once clean of its oxide will stay in that condition for a period of perhaps an hour before being recontaminated. When one goes further into space, surfaces once clean will remain clean for very long times.

Clean surfaces of metal can easily be obtained where there is rubbing of one surface on another so that in such devices as gear trains and bearings, in all likelihood the surface will soon be worn clean if no steps are taken to prevent this.

When very clean surfaces are pushed against one another, even with modest pressures, there is a considerable tendency for welding between the surfaces. Figure 3 shows a summary of some work which was done at National Research Corporation to see what the bonding tendencies might be like[3]. These experiments consisted of breaking a sample of metal at very low pressure and then pushing the surfaces back together again. Under these conditions, the surfaces contained no contaminant layer provided the pressures surrounding them are low and the exposure times are of short duration.

Following the breaking of a specimen and brief exposure, the two surfaces are pushed back together with a force just great enough to cause plastic deformation (around 90,000 lbs/sq. in. for steel and 60,000 lbs/sq. in. for copper). Then the joint is put under tension again to see what force is necessary to break the bond. These results are compared in Figure 3 with the strength of the virgin metal. It will be seen from this curve that in all cases, even at low temperatures, there is a decided sticking tendency which in similar experiments conducted in the presence of air are negligible. It is interesting to note that increasing the pressure of gas in the system from the 10^{-9} range used in these experiments up to 10^{-4} torr results in a reduction in the strength of the weld area. Where surfaces of metal rub on each other, the cold welding tendency becomes even more serious than in these static tests and bonding occurs at much lower temperatures and pressures.

Our normal solution to this problem of cold welding between moving parts is to provide lubrication for the surface. In space, this problem is made much more serious not only because of the increase of cohesive tendencies but also because the lubricants, which we customarily use are for the most part sufficiently volatile so that they disappear into the space environment in a very short time.

It is difficult to make a comparison of evaporation rates for various lubricants because these materials are almost always mixtures with components having a wide range of vapor pressures. However, Figure 4 will serve to give an idea of the magnitude of this problem of lubricant loss. The lower bar graph shows the rate at which a common lubricant will disappear if used in space[4]. In order to replenish this loss, it would be necessary to supply 3 or 4 grams per month of lubricant for each square centimeter of bearing surface, if bearing could be held at 55°F. However, in space with the absence of convection cooling, bearings will almost always run well above this temperature and if the common lubricant illustrated in Figure 4 were to run at 200°F, the loss per square centimeter would be around 100 grams or about 1/4 pound per month.

One way to minimize this lubricant loss is the use of protective housings around bearings and other moving devices in such a way that a somewhat elevated pressure is maintained around the surfaces to reduce loss of lubricants. The disadvantage of this approach is that it calls for substantial addition of weight to a mechanism in the form of shields and housings. The shielding approach does not eliminate loss but only reduces it. If really tight enclosures are used, then new problems are encountered in the form of seals for shafts. In missions of long duration, it is desirable to find another technique.

A more fundamental solution is to search for materials which will have very low vapor pressure and will, therefore, evaporate more slowly than conventional lubricants. There are many materials of this type being studied. Several are shown in Figure 4 and compared with evaporation rate for the conventional lubricant[5]. However, even with these materials, there is an appreciable loss as bearing temperature approaches 200°F. In evaluating these lubricants, weight loss alone is not a sound basis for consideration. The light fractions of the materials evaporate first and the residue continually increases in viscosity, even though fresh lubricant is added. Lubricants designed for high bearing load are generally dependent on additives for their performance. These materials have the characteristic of attaching themselves to the bearing surfaces and insuring that no direct metal to metal contact is possible. These additives usually have a very different vapor pressure from the main body of the lubricant. They are either stripped or concentrated in the process of evaporation but in either, they become ineffective. Professor George S. Reichenbach[6] of M.I.T. has recently observed another phenomenon concerning additives which needs careful study. He has found that many of the common additives do not function properly in the absence of air. It appears that the presence of oxygen is vital to the chemistry by which these additives attach themselves to bearing surfaces so that without air, they are no longer useful.

In addition to usual liquid lubricants, other techniques are under consideration. Solid lubricants which usually have low vapor pressures are being studied and a number of these offer promise from the standpoint of stability in vacuum[7,9]. Here the problem is to find a way to distribute the materials to insure that surface is continually covered with no bare spots where cold welding can occur. Soft metals,[8,9], for example gold, are being considered as lubricants. Here again, while vaporization is not a problem, the question of film maintenance continues to be of concern.

Some problems present a particular challenge. For instance, there is frequent need for moving electrical contacts in such devices as sliprings and commutators. Here we normally run such devices without lubrication in order to prevent interference with electrical signals and because of the surrounding atmosphere the surfaces remain sufficiently contaminated to avoid cold welding. Figure 5 gives an example of the behavior of a device of this type at a pressure of approximately 10^{-9} torr. In this case, a slipring potentiometer being tested at National Research Corporation was rotated at 2 cycles per minute. In the beginning and for a period of 5 minutes, the device delivered a very clean signal identical to the input. After 8 minutes of operation, some noise could be seen. After 15 minutes of operation, the device was stopped and then after a short interval, started again. The signal at that point was totally unrecognizable compared with the input as can be seen from Figure 5.

Work at Lockheed[10] offers some hope for a solution to this type of problem by providing a source of oil vapor in the immediate vicinity of the contact point so that at any given time, a small amount of oil is adsorbed on the surface and prevents direct contact. It is also possible that by control of loading on the contact surfaces, some improvement can be made. Devices involving sliding electrical contacts are extremely important in space devices and a concerted effort is necessary to find an adequate solution to this problem.

In studying performance of moving parts for space use, it is extremely important to do the best possible job of simulating space conditions, particularly the reduced pressures encountered in space. A great deal of work has been done at pressures in the 10^{-5} to 10^{-6} torr range. It is doubtful how valid such tests are. If bare surfaces of metal are exposed in such tests, they may survive through contamination from the residual atmosphere in the test chamber. Operation at a pressure of at least 10^{-9} to 10^{-10} torr is desirable and such equipment is now readily available.

Some of the lubricants which have been chosen, based on laboratory simulation of space environment, have now been evaluated in tests which were actually conducted in space. There has been little discussion of the results but recently Henry Frankel of NASA[11] reported that so far tests of this kind have given very discouraging results which he attributes to the fact that laboratory evaluation on earth was not done at sufficiently low pressures and did not include a full simulation of the space environment.

More knowledge about design of mechanical devices capable of operation in the space environment is necessary and this field should receive more attention. In addition to short range studies armed at quick answers for specific applications, it is desirable that a more basic and general program be undertaken in the hope that more thorough understanding of frictional processes may lead to new concepts for overcoming wear and cold welding problems in space.

Use of Superconductors in Space

A number of propulsion devices now under consideration are dependent upon the ability to generate large magnetic fields of considerable volume. In addition, if it were possible to maintain large magnetic fields in spacecraft, there are other applications which would become interesting. Some of these are summarized in Table II, which also gives some idea of the field strengths required. In addition to these items shown in Table II, it is almost certain that if designers had at their command the use of large fields, uses would be found in instrumentation. To date, the use of high fields in space devices has had relatively little attention. Fields up to 5 or 6 kilogauss were practical in the form of permanent magnets but beyond this, power and cooling requirements were so great as to be wholly impractical for magnets of a conventional nature. For example, a 100 kilogauss solenoid of conventional design having a 20 cubic inch volume would require a 2000 kilowatt power supply and cooling system equivalent to 500 gallons per minute of water. It is difficult to see how such a device could be put in space, although a field of this strength in a much bigger volume is necessary if thermonuclear devices are to be used in space. During the last few years, new developments in the field of superconductors have made the maintenance of high fields in space possible. The performance of these materials offers so many new possibilities that it warrants some detailed discussion.

Superconductors are a class of materials which are capable of carrying electrical currents with no resistance loss at all, but certain conditions must be met in order to make this situation possible. One of these is the condition that the temperature be below some particular value. Figure 6 shows a typical curve obtained when one plots the resistance of a superconductor against temperature. The resistance of the material falls off in the usual way as temperature is reduced until a so-called "critical temperature" is reached. At this point, the resistance drops off sharply and even the most careful measurements indicate complete absence of resistance at temperatures below this point. If a temperature is selected which is well below the critical temperature and an increasing external magnetic field is applied to a superconductor, the material loses its superconducting properties at a particular field strength. This point is known as the "critical field" for the material and increases as temperature is reduced. Of course, the real interest in superconductors comes in connection with their current-carrying capability. Figure 7 shows the type of curve obtained in this case. Under these conditions, the current-carrying capability as a superconductor decreases regularly as the applied field increases until some ultimate critical field is reached, at which point no further superconducting current is carried.

These general characteristics of superconducting materials have been known for a long time[12] and there has been speculation from time to time that if proper materials could be found, it would be possible to build electrical systems in which there would be no resistance at all. Under such conditions, it would be fully practical to build magnets in which huge fields could be maintained with very little power requirement.

Dr. Stanley Autler[13] of the M.I.T. Lincoln Laboratory in 1960 built a small superconducting solenoid of this type which served to stir up a great deal of interest in the field and to cause people to review the kinds of materials which might be useful for making superconducting solenoids. Following that development, a number of materials began to appear. J.E. Kunzler[14] of the Bell Telephone Laboratories reported a method of producing a superconducting wire with a core of the brittle intermetallic compound, Nb_3Sn. Shortly after this, Kunzler[15] and simultaneously Berlincourt[16] of Atomics International, reported the use of niobium-zirconium alloy in the form of wire. Since that time, many small solenoids of field strengths up to 60 kilogauss have been built from this alloy and at the present time, solenoids with volumes of several cubic feet at 50 kilogauss are being constructed from this material.

Niobium-tin, which in many ways has better magnetic properties, has been studied less than niobium-zirconium because it has the disadvantage that the superconducting core of the Kunzler wire is extremely brittle and the wire cannot be wound without serious damage to the properties. It is necessary in using this material to wind it into solenoids and then heat treat the entire assembly. This offers serious difficulty in terms of expansion coefficients and selection of structural materials and insulations as the heat treating temperature for the material is around 1000° C.

Figure 8 shows the characteristics of these materials and also shows the properties of a new superconductor which is in the last stages of development at National Research Corporation. In each case in this figure, the properties are reported as current density for the active areas of the conductors. For niobium-zirconium, this is the entire cross section[17]. For the Kunzler-type niobium-tin, this is about one-quarter of the whole cross section in the form of a core of Nb_3Sn inside of a niobium sheath[14,18,19]. For the National Research Corporation material, it consists of a thin film of Nb_3Sn on either side of a niobium strip. It can be seen from these curves that the niobium-tin compound provides higher current densities in the active areas than the niobium-zirconium alloys and, even more important, permits much higher fields. The National Research Corporation superconducting Nb_3Sn has the advantage that it can be wound into solenoids after final heat treatment without any damage, thus removing the handling problems which have limited the use of the Kunzler wire.

Figure 9 shows the actual current carried by the National Research Corporation superconducting ribbon. At 20 kilogauss, a ribbon 0.0011" thick and 0.0625" wide is capable of carrying nearly 100 amps. The same ribbon at 100 kilogauss can carry nearly 20 amps. These currents are being carried by very thin films of Nb_3Sn on either face of the ribbon.

While superconducting solenoids have been built, there are many problems still to be investigated. The current densities shown in Figure 9 are considerably reduced when these materials are used in winding solenoids. The superconductors carry less and less current as solenoids become larger. The cause for this reduction and the possibility of eliminating it are not fully understood.

At the present time, there are no adequate inspection methods for the conductors and performance is still rather variable. However, in spite of these limitations, solenoids are already being built and sold by a number of companies.

In connection with the use of superconducting solenoids in space, several important principles have already been demonstrated. Superconducting solenoids have been operated in such a way that after a solenoid is energized by means of a bank of storage batteries, it can be shorted out by means of a superconducting link and put into a persistent operation, which no longer required any power supply. Under these conditions, the currents circulate in the superconducting solenoids with no resistance loss at all and the field associated with this circulation of current is maintained steadily. The one requirement in such a situation is that the material be maintained at a low temperature continuously in order to permit this operation. In general, the temperatures that are best suited for these superconductors are around 4°K, although in the case of niobium-tin, it might be practical to operate as high as 10°K.

Some time ago, maintaining temperatures in the 4° to 10°K range for appreciable lengths of time would have been a serious problem. However, as a byproduct of interest in long-term storage of cryogenic fuels, new low temperature insulations have been developed which make it look feasible to maintain such systems[20]. Several superinsulations are available which should make it practical to design cryogenic storage systems to maintain a superconducting solenoid at operating temperatures in a space vehicle for as long as a year[21].

Combining the capability for maintaining low temperatures and the promise of the superconductors for use in solenoids, it would appear entirely practical today to think in terms of using magnetic fields of up to 50 kilogauss in space. Within the next year, it undoubtedly will be possible to consider volumes of as much as a cubic foot in such applications and possibly within a year, the capability for field strength will increase from 50 to 100 kilogauss.

Summary

The materials performance in space is so closely related to device performance that a much closer cooperation between materials people and device people is going to be necessary to make real progress. Standard materials are not going to solve many of today's problems. More and more, the use of combinations of materials such as coatings or composites will be necessary. The solutions to these problems are generally going to be very specific and of little broad use.

One field of investigation where there is a broad application is in the area of materials for use in bearings, sliprings and other devices where surfaces are required to move on one another and/or in space. Some progress has been made in the field but much more knowledge is necessary in order to insure that the wide range of devices which will be required to operate in space environment can perform satisfactorily for missions of long duration.

The field of superconducting materials (which so far has found no application in space technology) will, within the next few years, become important. Using these materials, it appears entirely practical to build solenoids capable of reaching fields of as much as 100 kilogauss. Using present technology for maintaining low temperatures over extended periods, it should be entirely practical to transport solenoids of this type into space for use over long periods of time.

The use of such solenoids should be helpful in speeding the development of some of the new propulsive devices under consideration and will also probably find application in other space devices.

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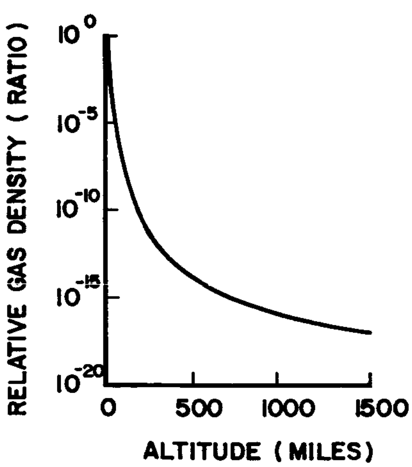
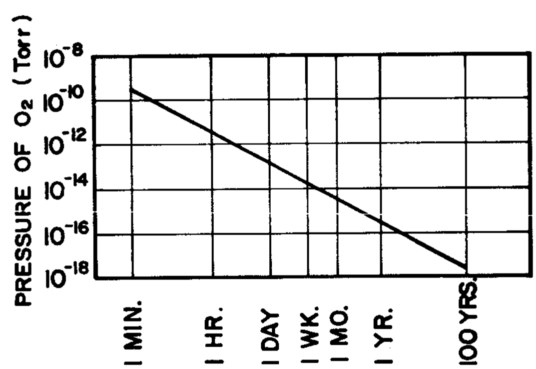
TABLE II

APPLICATIONS SUPERCONDUCTING SOLENOIDS FOR SPACE PROGRAM

	Field Strength Kilogauss	Use
Magnetohydrodynamic Converter	50 to 100	Conversion thermal energy to electrical energy
Shielding	20 to 50	Protection spacecraft from particle radiation
Thermonuclear Power	100	Generation electrical energy from nuclear fusion
Re-entry Communications	20 to 50	Deflection plasma to permit communications
Mass Spectrometers	20 to 50	Analysis of high velocity ions



FIGURE, I RELATIVE ATMOSPHERIC DENSITY AT VARIOUS ALTITUDES ABOVE THE EARTH
(ADAPTED FROM SPACE MATERIALS HANDBOOK)



FIGURE, 2 TIME FOR FORMATION OF A MONOLAYER OF OXYGEN ATOMS ON A METAL IF ALL IMPINGING MOLECULES STICK. (BASED ON 10^{14} ATOMS PER Cm^2 IN A MONOLAYER)

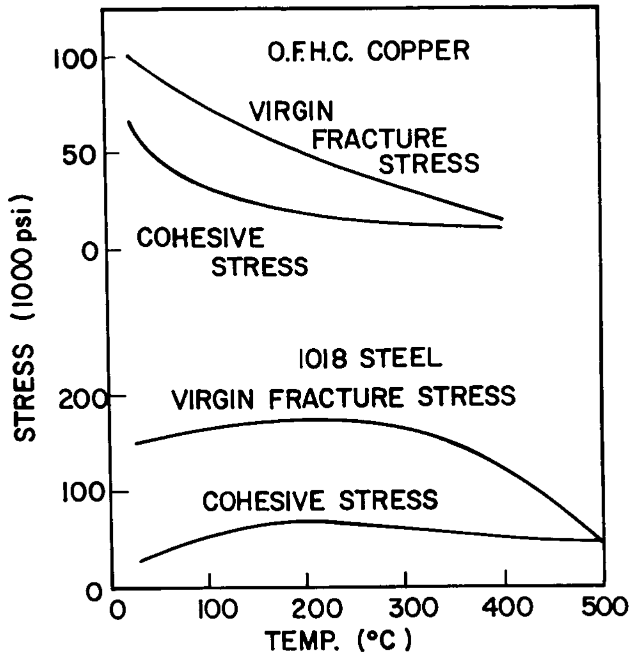


Fig. No. 3 - MAXIMUM OBSERVED COHESIVE STRESS COMPARED TO VIRGIN FRACTURE STRESS FOR COPPER AND STEEL

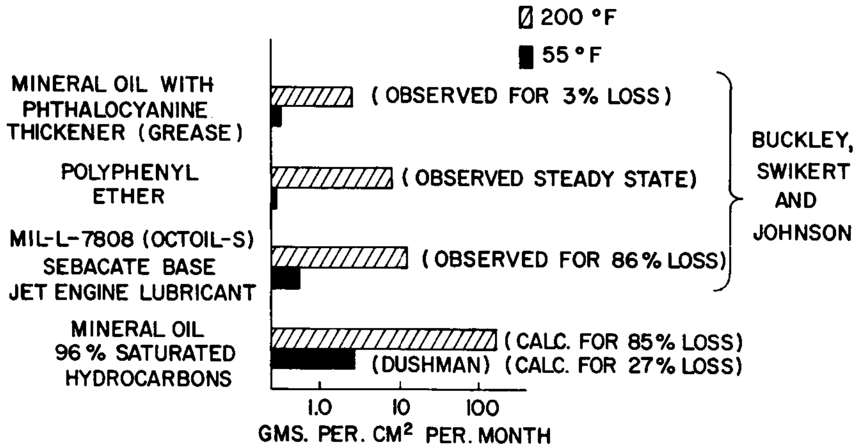
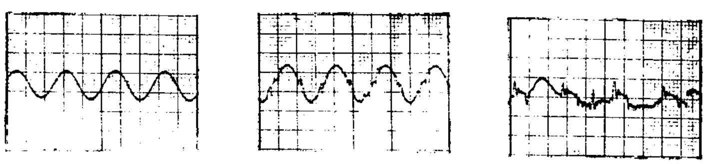


FIGURE 4 EVAPORATION RATES FOR VARIOUS OILS AND GREASES IN VACUUM

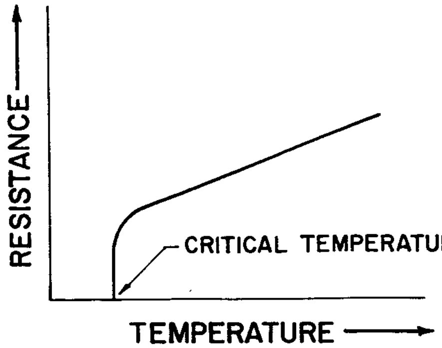


AT THE START

AFTER 8 MINUTES

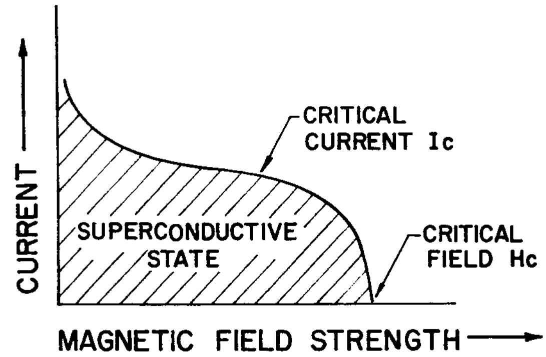
AFTER STOPPING AT 15 MINUTES AND RE-STARTING

FIGURE 5 DEVELOPMENT OF NOISE IN A SLIP RING POTENTIOMETER OSCILLATING AT 2 C.P.S. IN A VACUUM OF 10^{-9} Torr



TYPICAL RELATIONSHIP RESISTANCE vs TEMPERATURE FOR SUPERCONDUCTORS

FIG. 6



TYPICAL BEHAVIOR CURRENT IN SUPERCONDUCTORS

FIG. 7

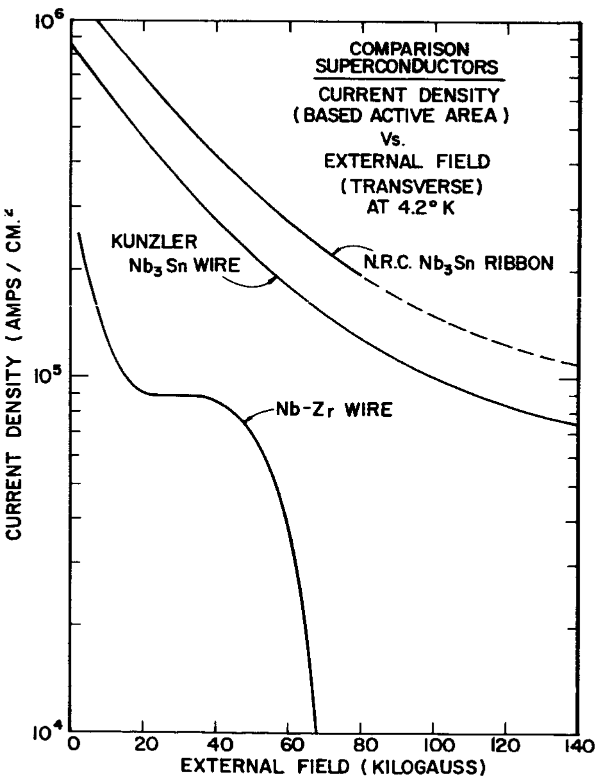


FIG. 8

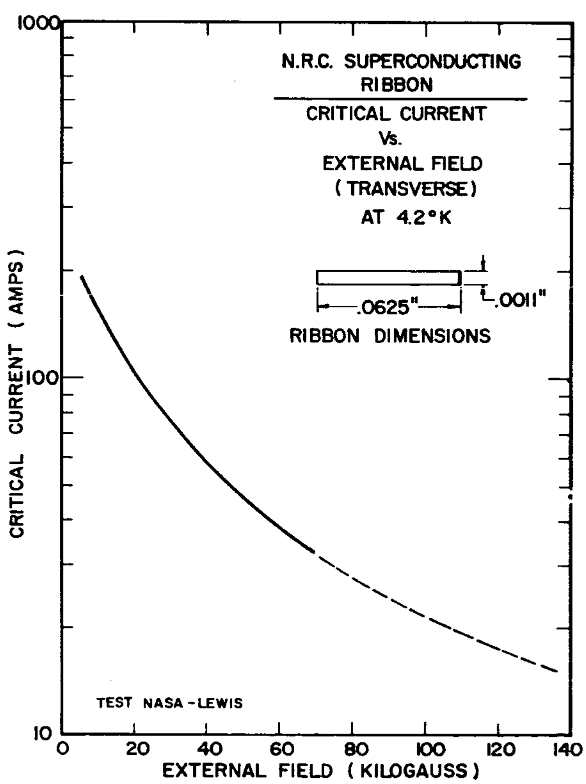


FIG. 9

RESEARCH REQUIREMENTS: Life Sciences Research Needs

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A little over two years ago, a paper voicing optimism for an increasingly accelerated tempo in the pursuit of necessary programs in bioastronautics was presented in this city (Ref. 1). This paper predicted that the practical problems of maintaining man in space would receive greater emphasis through the use of biological packages in satellites and probes. These would create a means for the direct observation of the effects of physical stressors in space on viable systems and organisms. Although the Russians apparently had placed much greater emphasis upon the biomedical aspects of space flight than we in the U. S. had, it was then predicted that with our increased enthusiasm for bioastronautics, our programs would begin to match those of the Russians in sophistication and numbers.

Unfortunately, these predictions have failed to materialize. The only notable efforts in the study of viable organisms in space have been those studies conducted on the Astronauts themselves. It appears then that practically all of the basic problems in bioastronautics are still with us. In many cases little has been accomplished in their resolution.

Two well-defined areas of interest have meanwhile developed in the realm of space biology. One area, that of exobiology enjoys popularity with the academic biologists who are mainly concerned with the realm of space as a potential source of extraterrestrial biological specimens. This concept encompasses the possibility of finding life or the precursors of life in space and on planetary bodies other than the earth. Investigations in exobiology may possibly lead to an increased knowledge of the origins of life. It is certain that man's fund of knowledge will be far advanced through the study of exobiology.

The other area of interst in space biology is that maintained by experienced aerospace medical personnel, many of whom have spent a professional life time in practical aerospace biomedical research. These men accept the credo that man is engaging in space flight; therefore, it is incumbent on them to see that the astronaut makes his flight safely and can perform useful work in a normal environment while in space.

In the higher councils of the Nation, the exobiologists have at present the predominant speaking voice for biologists in general. Their discussions imply that in their opinion the primary reason for space flight is that of the opportunity to investigate space for possible detection of extraterrestrial life. In most cases the practical aerospace medical scientist would beg the issue as to what was the primary reason for putting man into space. In concert with the apparent Russian viewpoint, he feels that the practical bioastronautic problems are those of immediate moment, and that when these applied biomedical problems are solved and man is capable of living in a space station or of traveling to other planets in a space ship, then the spaceman can safely and capably apply himself to the studies as defined by those interested in the exobiological concept. The key to the mastery of space, is still man himself.

As one of the latter group, I am heartened by recent developments indicating an increasingly accelerated program of practical projects related to man's sojourn in space. The biosatellite program under the aegis of Dr. Orr Reynolds has been reinvigorated and an organized program has been established at the Ames Research Center. This facility has the responsibility for the development of six biosatellites. It appears that a productive biosatellite program may come into being despite many initial setbacks. The recently established Office of Biotechnology and Human Research directed by Dr. E. B. Konecci, is organizing an extensive program to expedite the resolution of existing biomedical problems. The office of Dr. C. H. Roadman continues to function as the space medical office with direct progressional medical interest in the astronauts and their problems. NASA appears to have a well-rounded organization for the maintenance of a strong space medical program. It appears then that we are again in a position to predict an increase in the tempo of onslaught on space biomedical problems. Hopefully this will result in a speeding up of programs such as Apollo, which will greatly extend space flight. Thus, when man is safely and comfortably esconsced in his space station or space ship capable of extended flights, he can devote his time to exploration for the possible detection of life in space. He will also be able to study physical phenomena as seen more clearly in space itself and from these observations perhaps gain an understanding of the origins of life. Deo Gratia!

What then remains to be done in bioastronautics research before man can spend a reasonable time in productive performance and observation aboard a manned space system? The concept of the various physical stressors that will complicate man's existence in space remains the same. Man will be projected from a homeostatic environment to one totally lacking in his environmental necessities with some redundant agressors added to the picture. This simplifies the stating of his requirements. He needs everything! The fundamental problems remain the same; the question is, what has been accomplished and what remains to be done?

To more clearly assess the questions posed above, we can outline the physical stressors in space which must be mitigated. Additionally, certain miscellaneous requirements must be added. The outline is as follows:

A. Physical Stressors Inherent in Space

1. Total lack of a liveable atmosphere.
2. Extreme temperature variations.
3. Presence of ionizing radiations.
4. Presence of meteoroids.
5. Absence of terrestrial gravity.
6. Absence of terrestrial magnetic force.

B. Physical Stressors Generated in the Space Ship

1. Acceleration stresses on escape, reentry and abort of vehicle.
2. Noise.
3. Vibration.
4. Ultrasonics.

C. Stressors Generated by Man Himself

1. Toxic contamination.
2. Waste disposal requirement.
3. Sustenance requirements.
4. Maintainance of health and body tone in the space man.

D. Behavioral Stressor Mechanisms

1. Psychological effects of space flight on man.
2. Effect of mixed stressors.
3. Biological monitoring - psychophysiological correlation.
4. Training requirements.

It might be stated at this point, that certain of the above stressor mechanisms can be satisfactorily studied in laboratories or in simulators on the earth's surface. Other studies would be further validated if they could be checked out in a biosatellite or a space station devoted at least in part to biologic experimentation. A few problem areas cannot be simulated on earth and for their resolution, the studies must be done in a biosatellite or space station.

In consideration of our first listed stressor mechanisms, that of a total lack of an atmosphere in space, we are faced with the ongoing problem of life support system development. The ultimate in the development of a life support system will be one in which the direst life sustaining requirements such as breathing, oxygen, absorption of carbon dioxide and other noxious gases, water reclamation and perhaps recovery of edible substances will all be recycled through a continuously repetitive process of biochemical breakdown and resynthesis. With such a system, extended lunar and planetary explorations are within reason. Many closed cycle systems are being laboratory tested at present. These include the algae and other plant photosynthetic methods, chemical photosynthesis, electrolysis and photolysis, and other laboratory methods that show promise.

Present research efforts directed to the Gemini and Apollo flights involve the partially or semi-closed environmental systems. In these systems, superperoxides and ozonides are used to remove the carbon dioxide and water vapor from the cabin air and oxygen is released to recondition the gaseous environment. A system of this nature will continue to be suitable for orbiting flights where return can be effected at will, or for replenishable space stations. Noxious gases can be controlled by filters or chemical absorption. Humidity control in all systems would undoubtedly be a condensation and collection system. Condensation and collection of water would also provide a potable water source.

The physico-chemical means for life support in use now may possibly become the back up for the biological (plant or algae) systems which are in the experimental stage. For long term multi-man missions, the biological life support system shows promise. The space farm concept is one which should be thoroughly investigated. The maintainance of plant or algae life for oxygen generation, waste products and gas disposal, food for man and beast should not lack serious consideration.

The partial pressures of the environmental gases in the space structure are still a matter for study. At present 100% oxygen concentration at 5 psi pressure is considered adequate for the manned compartment of the orbiting space craft. This oxygen pressure is physiologically sound insofar as the oxygen requirement, negative potential for oxygen toxicity, and possible aeroembolism are concerned. There is however some concern as to the suitability of this pressure where micrometeroid penetration might occur. Further investigation in the physiological rationale of gas combinations including the requirement for nitrogen or other inert gas is required.

Temperature variants as a stressor mechanism are under continuous study. The need for control is basically in the random heating either from the outside solar source or from heat generation within the space craft. The present method of heat control is by dissipation of waste heat through radiation to space, either by liquid, vapor of absorption cycles. Future space craft will be equipped with chemical refrigeration systems. Space craft have been insulated to maintain a physiologically suitable internal environment. The existing problem area in thermal variations in space flight are those of environmental control systems, rather than bioastronautic research.

Undoubtedly the most hazardous stressor in space for the unprotected Astronaut is that of ionizing radiation either cosmic or from solar flares. The variations of concentration and specific types of ionizing radiation in space are of such a nature that it is hard to establish standards for protection. The effect of ionizing radiation upon man is not only the result of the intensity, but also of the prolongation of exposure. Solar flares may bring the level of radiation to lethal proportions, but in addition continuous exposure to radiation at a sublethal level may bring about radiation illness of an incapacitating nature. Prediction of solar flares, new and novel shielding, radiation protective clothing and pre and post exposure pharmaceutical treatment for radiation sickness are all in the present study category. Cosmic radiation as stated before is probably the most important biological problem in space flight.

The presence of meteoroids in space introduces a possibility of casualty production during long-term space station occupancy. Experiments done by firing hypervelocity metallic and silicon particles through the wall of a prototype space craft containing rats by the Astronautics Division of the Chance Vought Corp. has demonstrated the possibility of animal damage by blast, burn, intense light or wounding by spalled material. The possibility of explosive decompression also exists in this type of an episode. The insulated construction of the wall of a space station should give reasonable protection from internal explosion by the penetration of the micrometeoroids. Penetration by larger meteoroids could only be handled by gas tight bulkhead compartmentation of the ship. Further work should be done to investigate the partial pressures of gases comprising the internal gaseous environment of the space ship to reduce to a minimum the possibility of the explosive effect and the associated intense light resulting from meteoroid penetration. The micrometeroid protection of the man outside the space ship in a space suit should also be investigated.

The absence of terrestrial gravity in the space platform or in space flight introduces the biological problem of weightlessness. While some of the biological fraternity have been lulled by the apparent lack of subjective symptoms in the successful orbital excursions of the Astronauts and Cosmonauts, Soviet scientists and some of our own Biologists express fear that there is a possibility of disturbance in cellular metabolism in long-term flights.

The Astronautics Division of Chance Vought has had a proposal outstanding for more than two years consisting of a special biological payload for a satellite to study possible physiological changes in cells exposed to weightlessness during their entire life span. A statement has been attributed to Russian Bioastronautics specialists, (2), who said that possibly after a weeks' duration in space there may be a breaking down of molecular or cellular distribution in the human body. The statement also includes the observation, that the more short lived the cell, the earlier it could be expected to breakdown due to the effects of weightlessness. The question of the effect of long-term weightlessness certainly should be considered in space station design. Studies are needed to clarify the possibility of the occurrence of muscular atrophy, decalcification of bony structure with the inevitable result of calculus formation, interference to cardiovascular function with possible thrombotic formation and the possibility of a generalized disturbance of cell physiology in space flights of more than a week.

The gross effect upon the labrynth has as yet not been satisfactorily resolved. This is a problem of great significance according to the Russian expert Yazdovsky and it is hard to believe that space station design, in the absence of further basic knowledge would envision anything other than the rotating platform. As stated in the beginning of this paper, NASA-Ames is engaged in the design of six biological payloads for satellite exposure. These will be used predominantly for the study of the effects of long-term weightlessness at the cellular level. Cells of plant and animal origin will be investigated for mitosis and growth, in bacteria, fungae, higher plants and lower animals. Enzyme activity, membrane permeability, cyclosis, behavior of biological fluids and other cell functions can be investigated in an orbiting satellite to determine the effects of long-term weightlessness at the individual cell level. Vought Astronautics had done some studies on protoplasmic streaming in the amoeba while it was in a weightless state in the Keplerian trajectory. A conceptual design for microscopic observation of cells in space was as stated before proposed by Vought Astronautics more than two years ago. The activities of the biological satellite group at Ames Laboratory, if carried to a satisfactory conclusion could shed some light upon the biological effects of long-term weightlessness. It is hoped that this will be in time to influence the future design of space stations. If the space station is in use before this problem is thoroughly investigated, then periodic examinations of the men occupying the space station will be necessary to determine the possible effects of this stressor on their health.

If it is necessary to design the future space station in the form of a rotating wheel for the application of some gravitational force upon the crew, another complicating syndrome will beset the men. This is called the Coriolis factor. Coriolis acceleration is generated when the man is in motion perpendicular to the axis of rotation of the space station. Vertigo and disorientation are pronounced when the man attempts to move to, or away from, the center of rotation. If the rotating wheel is the design of the future space station, continued laboratory work on the Coriolis effect is necessary.

The magnetic field of the earth exists from the fact that the earth acts as a gigantic bar magnet. The maximum height of the magnetic flux field of the earth at the equator is about 15,000 miles, hence all mankind is continuously exposed to this magnetic force. Certain biological experiments have been conducted on animals in generated magnetic fields of significant intensities. J. M. and M. F. Bernothy (Ref. 3) have exposed rats to magnetic field strengths of 4200 Gauss, observing changes in development, mortality rate, activity, body temperature, appearance, food consumption, fertility and in hematology. The question may be asked that if increased magnitudes of magnetic flux can cause these changes, what would a greatly reduced or absent magnetic field bring about in men exposed to space flight?

In considering the generated stresses that are inherent in the space ship design, we include those of acceleration, noise, vibration and ultrasonics. Gross acceleration has been demonstrated as not hazardous either in normal ascent or reentry when the man is properly positioned so that the applied acceleration is transverse through the long axis of the body. The escape trajectory has fortunately never been required and crash deceleration has as yet not been encountered. Continuous studies on restraints and crash harnesses are being conducted for the protection of the Astronaut. Supine couch configurations and other seating structures are being designed for greater comfort, performance and protection.

The noise in the Mercury capsule in orbit has been recorded as a continuous 87 decibels. This is slightly higher than the maximum of 80 decibels approved for continuous exposure to men in space craft. In a space station the men will not be protected with a space suit helmet and radio phone cups as were the pilots in the Mercury flights, hence further study in noise abatement and ear protection is necessary for space stations.

The limits of vibration upon man are incompletely delineated. Practically all of the human studies in vibration have been done on vibrating platforms which vibrate in the longitudinal or transverse plane. The effects of combinations or vectored vibration modes on man are not too well understood. Low frequency vibrations can vary from mild incapacitation due to annoyance, to permanent physical damage. Below 2 cps the body of man responds as a unit mass. Above 2 cps the various organs and soft tissues have a resonant frequency. Random vibrational modes may cause damage to specific organs. Studies of the effect of vibration on man as a debilitating stressor both in performance and as a frank casualty producer are being conducted in various laboratories. The need at present is a device to expose the subject to vibrations in all planes of reference.

Ultrasonic vibrations may be a potential liability to man in long-term space station occupancy. Ultrasonic exposure results in compression waves passing through the tissues of those exposed with the generation of internal heat as the most manifest symptom. The long-term effects of ultrasonic vibrations both upon the physical aspects of the human body and upon the performance quotient of the individual should be more thoroughly understood before long-term space station occupancy becomes routine.

The physical stresses upon man in a closed compartment generated by himself or his requirements are those of his own toxic emanations and production of waste. His need for water and sustenance create additional stressful circumstances. The ultimate in the environmental containment of man in space would of course be the closed cycle environmental system where waste material would be recycled and converted into food and water for continuous reuse. While this is the ultimate in conceptual design for environmental control systems, the present space system dictates the absorption of toxic emanations by chemical means, the collection of waste water by condensation and purification, the handling of waste material by chemical treatment and storage. The present research in food is in dehydration, packaging, palatibility and reduction in human waste production. For long-term space flights the use of algae for food is being considered as well as biochemical procedures that utilize carbon and hydrogen for the possible conversion to sugars. Suitable plants and animals for use in space platforms as living sources of food are not impractical and are being investigated.

Visual requirements in space comprise a problem area that needs further investigation in bioastronautics research. The lack of reference points and inability to perceive depth will be a serious detriment to a man moving freely in a space suit external to the space craft or to individuals attempting to assemble material in space. The difficulties of simulating space on the earth for these visual studies are great but not insurmountable. An existing space station would of course be ideal for space vision studies.

Instrumentation for space vehicles has brought about a need for reevaluation of the advanced techniques currently proposed for aircraft. The aircraft display which uses a geocentric vertical reference system with the vehicle attitude related to the earth's horizon would lack the sophistication required in space flight. Horizon reference is not necessary for control in space, where there are five different reference possibilities for vehicle attitude. Research in three dimensional display systems is in progress and eventually a single three dimensional display may be able to handle all data required for maneuvering in space.

The concern of the behaviorial sciences over confinement and the selection of Astronauts has been reduced somewhat with the extended cruises of the nuclear submarines in which 60 days of continuous underwater cruising has become commonplace. The exceptional behavior and performance of the Astronauts has also allayed fears of aberrant behavior patterns. A recent report, however by Zubeck, Welch and Saunders (4) describes a decrease in alpha frequencies of the electroencephalograph observed in a fourteen day confinement where the subject was exposed to continuous white noise and unpatterned light. The ECG records were still abnormal one week after the test and long lasting motivational losses were observed. Apparently their findings indicated to them that prolonged periods of perceptual deprivation can produce some long lasting disorganization of brain activity. Studies in behavioral patterns in long-term confinement should continue for psychophysiological information.

Simulation of space flight in all its psychophysiological aspects will be an ever expanding field of endeavour as the full requirement becomes obvious to those in operational control of Astronaut training. There will be no back seat checkout in space flight and the potential Astronaut will have to be fully prepared as a Pilot on his first flight. Present simulators are efficient in the simulation of flight parameters; however, they do not create a realistic operational situation. Motivational control is desirable since the operator performs in a known no risk situation. Because of this, his simulator performance in many ways does not replicate the operational equivalent of the man. An approximation of operational space flight may be achieved if in some way the man can be motivated to do well. The possibility of an unpleasant experience as a result of poor performance may be a means of achieving this. The calibration of motivation may be achieved through psychophysiological correlates.

The investigation of physiological correlates to psychological responses and performance abilities in stressful situations may be an economical way to detect and predict the effects of work and of the environmental restrictions placed on the Astronaut during an actual space mission. The combination of biological monitoring and data handling which make such detection and prediction possible must be investigated.

A means of assessing the effects upon man of long-term exposure to an unforgiving environment must be established to assure safety in the space station. What does the overall stress of continuous residence in space do to the decision making ability of a man, what personality changes will occur and what will his general effectiveness be under these circumstances?

In the development of psychophysiological correlation and information acquisition systems, new and more sophisticated biological monitoring equipment will be necessary. In a recent request for a proposal from NASA, Edwards Flight Research Center for a design study of a psychophysiological information acquisition, processing and control system, stress was placed on the fact that biological monitoring was still in the basic medical measurement category. The suggested parameters of monitoring were heart, lung, vascular, brain, metabolic, thermal, neuromuscular, autonomic conditioning and behavioral patterns. These were to be correlated to the dynamic parameters of the machine and the environmental parameters of the living space. Needless to say, this is a formidable, yet necessary task of information acquisition correlation and display.

In conclusion, I would like to remind you that this resume of requirements of bioastronautic research is necessarily a broad brush treatment. Fundamentally as stated before, everything that has been accomplished can be further examined for sophistication and improvement. Beyond this there is a tremendous requirement for bioastronautics research in the areas that have only been philosophized and conjectured upon. This paper concentrates upon the psychophysiological and biomedical problems of space flight as research areas for the present and future. We have not discussed the future plans and hopes of exobiological research and exploration. Life on the planets, the relationship of biological systems to time and space, the precursors of life and the possibility of understanding the origins of life we leave to the future of man in space. Whether or not the above knowledge is the primary reason for space flight does not concern us. We feel the primary problem in astronautics today is to assist in the resolution of the biological problems involving man in his space station of the future so that he may function effectively, efficiently and safely during his entire mission. When this is an accomplished fact, scientists may become Astronauts and make their observations first hand. I would like to repeat, that in our opinion, the key to the mastery of space, is man himself.

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FUTURE PROBLEMS IN RE-ENTRY PHYSICS

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It is a dangerous task to attempt to discuss the future research in any field. To write about such a subject we must see the problems clearly enough to discuss them and, this being the case, this work ought properly to be called the research of today. Thus, this paper shall be restricted to discussing research problems which we can see today which we will have to face in the immediate future if we are to vigorously pursue the exploration of interplanetary space with manned and unmanned vehicles.

The area which has been singled out for discussion here is the field of re-entry physics, a name which has been adopted to denote the basic phenomena involved in interactions between a body or vehicle and a planetary atmosphere. We shall consider only our knowledge of the basic phenomena involved in such interactions and what these processes mean to the vehicle, mostly through the heating problem. Of course, there are many other aspects to a planetary entry which do not fall into the area of re-entry physics, such as lift and drag, guidance and control, stability, communications and landing. In most situations one or more of these aspects will dominate a mission profile or a vehicle design, at least in an indirect way. In general, a vehicle design calls for specific performance which, in turn, determines the re-entry maneuver and, consequently, the environment. Our knowledge of re-entry physics allows us to transform such performance specifications into vehicle design data.

Thus, to look at the re-entry physics problems of future space missions we must look at what these missions might be and what the parameters and the limits of the performance will be on such missions. Having specified the performance requirements, we can assess the aerophysical problems.

Manned re-entries from lunar missions at speeds approaching escape velocity are with us today in the Apollo program. Some day there may also be a requirement for a re-entry of unmanned vehicles from lunar missions, possibly for returning valuable evidence from a lunar expedition without the restrictions imposed by a manned system. Likewise, for the exploration of interplanetary space it is safe to assume that both manned and unmanned missions will be of interest. One needs to differentiate between manned and unmanned missions in any discussion such as this because the limitations which must be placed on manned vehicle performance due to physiological restraints result in considerable differences in the re-entry problems. It is generally accepted that the maximum deceleration of a manned vehicle must be kept below 10 times the acceleration of gravity, unless a breakthrough is made in the technology of passenger restraint and/or our understanding of human anatomy. Thus, manned re-entry will always be made at relatively shallow re-entry angles to minimize decelerations.

The transit times for a minimum energy, Hohman-type, one way transfer from Earth to Venus and Mars are about 150 and 250 days, respectively. By the use of about a factor of 2 of excess velocity, these times can be reduced to 25 and 50 days, respectively, as shown in Fig. 1. Of course, this is a very gross simplification of a very complex problem, given for illustrative purposes. In actual practice there are small launch windows for Venus or Mars missions and the size of the launch window can also be increased by use of excess velocity. Shorter transit times will effect savings in the life support system weight as well as the radiation shielding requirement for manned vehicles. In addition, the logistic advantages of reducing the time required for exploratory ventures and removing some of their scheduling restrictions will surely legislate the use of these higher characteristic velocities at a time in the future when advanced propulsion systems have minimized the added launch penalty. Because any velocity imparted to a vehicle may be dissipated in planetary entry, high re-entry velocities, both for Earth and other planets, are of obvious future interest.

Re-entry technology today is based on a broad research foundation developed in support of missile and satellite programs. The heating problem of all re-entry vehicles designed to date has been dominated by convective heating considerations with a significant but secondary role being played by radiative heat flux. This is the case for the Apollo re-entry vehicle, where the radiative heating accounts for approximately 30% of the total heat input. The relative magnitude of these two sources of heating are manageable to some degree by designing the proper combination of nose radius and deceleration altitude, i.e., ballistic coefficient W/C_DA, but the extremely strong dependence of radiation intensity on entry velocity will result in high speed re-entry being radiation dominated.

Our only re-entry experience in the velocity range above escape speed where radiation should be the dominant transport mechanism is with meteors which have been observed to enter the Earth's atmosphere at velocities between 37,000 and 238,000 ft/sec. The direct analogy of this phenomenon to the high speed re-entry problem has led the aerodynamicist to invading the field of meteor dynamics, which previously had been the domain of the astrophysicist.

Applying our knowledge of entry phenomena, Riddell and Winkler [1] have shown that only very small (micron size) and very large (100 ft diameter) objects can survive the interaction with the atmosphere throughout the above meteor velocity range. The very small objects have such a low ballistic parameter, W/C_DA, that they decelerate at such a high altitude that they can reradiate their heat at a reasonable temperature. The large bodies have such a high value of W/C_DA that they do not decelerate before impacting on the Earth's surface and consequently do not dissipate their energy. In the intermediate size range only objects with velocities at the lower end of the meteor entry velocity range can be expected to survive re-entry. This is consistent with meteorite observations[2].

For re-entry velocities up to the escape speed, re-entry performance has generally been analyzed by considering the dynamics of a constant mass body. However, meteorite experience and elementary considerations indicate that mass loss due to consumption of heat protection material plays a significant role in the survival of bodies at high re-entry velocities. Allen[3] and Grant[4] have performed similar analyses to calculate the mass loss due to heat absorption during the re-entry as a function of the entry velocity. Such an analysis is reproduced in its simplest term here.

Following Allen[3], the equation of motion for a trajectory where the acceleration due to gravity can be neglected when compared to the drag, can be written as

$$m \frac{dV}{dt} = -\frac{1}{2} C_D \rho V^2 A \tag{1}$$

where m is the vehicle mass, t is the time, V is the velocity, C_D is the drag coefficient, ρ is the density and A is a characteristic area of the vehicle. The heat input can be written approximately as

$$\frac{dH}{dt} = \dot{q} A = \frac{1}{2} C_H \rho V^3 A \tag{2}$$

where H is the heat input and C_H is a heat transfer coefficient.

If we let Q^* be the heat capacity per unit mass of a heat protection system in which the material is being consumed, say by vaporization,

$$\frac{dH}{dt} = -Q^* \frac{dm}{dt} \tag{3}$$

Combining the above gives

$$\frac{dm}{m} = \left[\frac{C_H}{C_D Q^*}\right] V dV \tag{4}$$

If we can assume the quantity in the bracket to be essentially constant throughout a trajectory, then the mass at any instant, m, is related to the original mass, m_o, and the velocity at initial re-entry, V_i, by

$$m = m_o \, e^{\frac{C_H}{2 C_D Q^*}\left(V^2 - V_i^2\right)} \tag{5}$$

For the case of a nonconsumable heat protection system, i.e., a heat sink, the mass of the payload, M_p, to the original total weight is

$$\frac{m_p}{m_o} = 1 - \left(\frac{C_H}{C_D}\right)\left(\frac{V_i^2}{2Q^*}\right) \tag{6}$$

The payload mass ratios for both consumable and heat sink systems are shown in Fig. 2 as a function of the parameter $\frac{C_H}{C_D} \frac{V_i^2}{2Q^*}$. For high speed re-entry where radiative heating dominates, the heat transfer coefficient C_H approaches approximately 0.20 while C_D for blunt bodies is about 1.0. Thus, even if the heat sink were made of graphite with its relatively high heat capacity per unit mass, i.e., its high heat of vaporization, the parameter $\frac{C_H}{C_D} \frac{V_i^2}{2Q^*}$ will be approximately one at about 50,000 ft/sec and the no-mass-loss analysis yields vanishing payloads.

For the variable vehicle mass case, the payload mass ratio can be plotted as a function of re-entry velocity if reasonable values are assumed for the parameter $\frac{C_H}{C_D} \frac{1}{2Q^*}$. For the case of spherical bodies, with Q^*, C_D, and C_H, as previously stated the mass ratios are shown in Fig. 3. It can be seen that at re-entry velocities of about 80,000 ft/sec, the payload mass ratio of a sphere becomes vanishingly small.

Recently it has been pointed out by Allen that this limiting velocity at which the payload mass ratio vanishes can be overcome, or at least postponed to higher velocities, by considering more optimum vehicle geometries. When radiation from the gas behind the normal shock is the dominant form of heating in the stagnation region, then it becomes desirable to minimize the shock strength and, consequently, minimize the gas radiation, by utilizing a conical body. In doing this, we concede some ground to convective heating but we can drastically reduce the radiation and, consequently, reduce the total heat transfer.

The effectiveness of the conical re-entry vehicle in extending the limiting velocity is shown in Fig. 3 by the dashed line drawn for a very small cone half-angle, i.e., 8.5° from the data of Grant. The small cone angle favors high re-entry velocities. The results shown were obtained by Grant from a somewhat more complex numerical study including the effect of blunting the cone tip. This effect results in nearly equivalent performance for an end-on cylinder and a cone in this analysis. From the point of view of survival of a significant payload fraction alone, re-entry velocities of up to 100 or 150 x 10^3 fps are of interest for the heat shield materials presently envisioned.

The ratio of heat transfer to drag coefficient determines the limiting re-entry velocity, as shown by Eq. (5). As a result of the differences in dependence of C_H for convective and aerodynamic heating on cone angle, velocity and Reynolds number, this ratio will have a minimum value at some re-entry velocity and then will increase rapidly with higher velocities as radiative heating begins to dominate. The locus of these minima produces an optimum cone angle for each re-entry velocity, with optimum cone angles decreasing with increasing re-entry velocity. This effect is shown in Fig. 4 from an analysis of Allen[5]. It was also shown by Allen that the lowest value of ballistic coefficient possible for the optimum geometry produces the least heating.

The extreme re-entry conditions which are encountered in the above situations also involve very large loadings. In fact, the maximum decelerations are so large that most of the situations discussed will not be usable for manned re-entry vehicles. The maximum decelerations along the limiting velocity trajectories are of the order of several hundred to several thousand times the acceleration of gravity. Thus, manned re-entries will not utilize these very large velocities because the deceleration must be limited to values of the order of 10 g's.

The problem of the deceleration-limited manned re-entry vehicle is to produce a sufficient amount of drag during a single pass through the atmosphere to reduce the velocity below the local circular value without exceeding the maximum deceleration limit of 10 g's. The conditions required by such a maneuver have been investigated extensively by Chapman[6]. The entry corridor height concept, i.e., the distance between the vacuum perigees of the trajectory which just skips out of the atmosphere because of insufficient drag, and the vacuum perigee of the trajectory which achieves the maximum tolerable deceleration, was developed to illustrate this effect. Figure 5 shows this corridor height as a function of velocity for a ballistic as well as for lifting vehicles. Even an infinite lift/drag ratio cannot increase velocity at which the corridor vanishes for the 10 g deceleration limit above 85,000 ft/sec. For reasonable lift/drag ratios, such as one to two, the corridor vanishes at 80,000 ft/sec. If we can increase the physiological tolerance limit of the human passenger, or we develop a restraint system which would allow a maximum deceleration of 20 g's, the infinite L/D corridor would vanish at about 120,000 ft/sec. The above value applies for constant lift or drag trajectories. Use of moduiated forces could extend these capabilities somewhat [7].

Since the corridor for manned re-entry at very high velocity is very small, and our analysis of meteorite entries and high velocity missile re-entries indicates that very large fractions of the payload must be invested in heat protection, it appears reasonable to re-examine the competitive position of other deceleration schemes under these conditions. Preliminary results from a study of Yoshikawa and Allen indicate that the break-even point between aerodynamic and rocket-braking may well be reached in the 65,000 ft/sec re-entry velocity range for advanced propulsion systems, such as hydrogen nuclear rockets. Their preliminary results obtained from a very simple mass change equivalence are shown in Fig. 6. In addition to the use of rockets for braking, other schemes should be investigated. Among these are the use of a drag brake[8] or other light structure to produce deceleration at very high altitudes, or possibly an application of magnetohydrodynamic interactions[9] between the flow and the field of a magnet on board the vehicle to reduce heating and to increase drag. The latter possibility will be discussed more extensively in a later section.

The above considerations lead to the conclusion that although re-entry velocities higher than escape speed are certainly of interest for future entry systems, there are some very solid limitations which must be considered. Even the best presently known heat protection materials will be insufficient to allow survival of a significant payload fraction at entry velocities above 100,000 ft/sec. On the other hand, manned vehicles will probably be limited to re-entry velocities closer to 70,000 ft/sec. Above this velocity, rocket braking may become more efficient. In the latter case, the entry velocity of concern to the aerophysics problem, i.e., the velocity left for dissipation by aerodynamic braking, will be less than the above value. It must be concluded then that although very high velocities, i.e., greater than 100,000 fps, are logistically attractive, their utilization is presently doubtful because there are no obvious solutions to the problem of dissipating the resultant large energies. In order to remove this present restriction, new techniques of heat protection and dissipation must be invented and developed.

The problems discussed so far for entry into the Earth's atmosphere must also be considered when we talk about entering the atmospheres of other planets. The dynamical performance of a re-entry vehicle entering any planetary atmosphere is completely described by constants representing the scale height of the atmosphere, β, the radius of the planet, r_o, and the gravitational acceleration, g. A detailed look at the specific heating problem in a planetary atmosphere, of course, requires a detailed knowledge of the constituents of the planet's atmosphere for a better determination of the heat transfer coefficients, C_H.

Re-entry physics problems facing us for the future vehicle designs, consequently, must lie in the aerophysics of the environment encountered by a vehicle flying through atmospheres at velocities up to about two to three times the escape speed. Quite simply, the re-entry physics problem lies in the determination of the proper value of the heat transfer coefficient, C_H, in the previous discussions and achieving a thorough understanding of the basic phenomena underlying this determination. Other aspects of atmospheric entry, such as forces and stability, are determinable from the simplest Newtonian flow model tc reasonable accuracy, except perhaps for very slender bodies. It is a well known fact that the surface pressures do not depend strongly on the thermodynamic state of the gas but rather on the momentum transfer processes. Thus, these questions are essentially unaffected by the high temperature, high enthalpy re-entry environment in a direct way. However, through shape changes due to extreme heating, re-entry physics can interact with the body forces.

Much research has been done in the mechanisms by which thermal energy is transported from a high temperature gas to a body surface in hypersonic flow[10,11,12]. In this paper we shall concern ourselves with what are the difficulties and peculiarities of the high speed planetary entry problem and what are the areas where our knowledge is limited. The primary difference of the high speed entry problem is the fact that the air or other planetary atmosphere environment is probably highly ionized due to the large kinetic energy possessed by the re-entering body. Thus one large area of study must be the radiative and diffusive transport of ionization energy. Besides causing significant degrees of ionization, the large specific energy of the environment also necessitates consideration of the coupling between the various flow regions, i.e., viscous and inviscid, and the energy transport mechanism. The customary independent treatment of radiation and convection, inviscid flow and the boundary layer, as well as the shock front, can lead to erroneous conclusions. In certain situations we need to consider energy decay due to radiative transport, nonisothermal shock layer, radiation absorption in the boundary layer, in the inviscid flow and, even in the free stream for the case of strong ultraviolet radiation, the effect of all of these phenomena on the conditions at the boundary layer edge and, consequently, the convective transport, as well as the details of the shock structure.

In addition to this complex coupling of the flow and gas physics, we must also consider the surface interaction carefully. Due to the highly energetic environment and the predominance of radiative transfer under many situations, the interaction with the surface material may be quite complex. Although the material response is not, strictly speaking, a re-entry physics problem and is better treated elsewhere, there is very strong coupling particularly in the boundary layer where ablated surface material may dominate the air properties and the radiative, as well as convective processes, are appreciably altered. Thus surface interaction must be considered as an area of aerophysical research.

Research in re-entry physics has always been paced by our ability to produce and study the simulated environment in the laboratory. During the past few years significant advances have been made in several of our experimental techniques which now allow us to conduct experiments with gases duplicating the flight environment at velocities as high as 60,000 ft/sec. Arc-driven electric shock tubes have been operating with shock strengths up to 13 mm/μsec[13], giving stagnation point simulation of flight at up to 60,000 ft/sec.* Ballistic ranges are presently capable of accelerating small pellets up to velocities somewhat in excess of 30,000 ft/sec. When such a ballistic range is fired into the nozzle of a hypersonic shock tunnel in which the gases are flowing at 10,000 - 15,000 ft/sec, simulation of flight at the combined velocities is produced. Such a range has been operated successfully at NASA Ames[14] producing simulation of about 43,000 ft/sec and has the potential of fully utilizing the best that can be achieved in both these devices. This limit is probably of the order of 35,000 ft/sec for the launcher and 20,000 ft/sec for the shock tunnel. To accelerate pellets beyond these velocities, radical modifications of the hypervelocity launcher, as well as greatly improved shock tunnels, are probably required. The third experimental facility capable of contributing to research in this velocity range is an arc gas heater modified by the addition of magnetic fields to accelerate gases to high enthalpies without the need for creating the prohibitively high gas stagnation temperatures otherwise required. Development of such facilities is presently being pursued[15]. Early experimental versions of such devices are operating[16,17]. High enthalpy arc facilities can be expected to be useful, particularly for material research in the future.

To discuss re-entry in more detail, let us look at the thermodynamic state of a high temperature gas, such as air, under the conditions produced in high speed entry. The equilibrium state of gases of known composition can be calculated from thermodynamical and statistical mechanical considerations as a function of temperature and density. The conditions of a sample of gas brought to rest at the stagnation point, or anywhere else, of a vehicle flying through the atmosphere can then be calculated by the application of the Rankine-Hugoniot equations. Such calculations have been made for flight through the Earth's atmosphere with the results shown in Fig. 7. At approximately 30,000 ft/sec, air begins to ionize and is more than half ionized by 50,000 ft/sec. Temperatures of interest will range up to 20,000°K. It is this regime of partially ionized air in which our re-entry problems lie. In other planetary atmospheres, being largely composed of nitrogen, the other major constituent being fractions of CO_2 of the order of 10%, this picture will be very similar. The properties of both air and these atmospheres will be dominated by the properties of ionized atomic nitrogen at very high velocities. However, the details of entry into these atmospheres is necessarily clouded by our lack of precise knowledge about the constituents and their concentrations.

When dealing with transport of energy in gases, the properties of the gas of course play an important role. The most important of these is probably the thermal conductivity because of its strong influence on convective heating. Unfortunately the transport properties of high temperature gases are extremely difficult to calculate and even more difficult to measure directly. Our knowledge of the critical transport properties is thus restricted to extensive calculations of the collision cross sections of the various species present[18] and only a few measurements over limited temperature ranges of a very few of the species involved. This lack of knowledge of the transport properties of high temperature gases has kept us from achieving a really good understanding of the heat transfer mechanism. To date, we must be satisfied with an essentially integrated check involving both the boundary layer theory and transport properties. Some thermal

* The driver conditions which have been achieved in these shock tubes are capable of producing simulation of flights up to 85,000 ft/sec. At present the only limitation is the lack of a driver of sufficient size to eliminate the costly expansion of driver gases. The driver conditions would produce shock speeds of 16 mm/μsec at initial gas pressures of up to 1 mm of Hg (simulating stagnation point density at about 110,200 ft) and 18 mm/μsec (stagnation point simulation approximately 85,000 fps) at initial pressures of 0.1 mm of Hg, if sufficient energy were available to produce the same driver conditions in a larger diameter driver chamber.

conductivity measurements exist from measurements of the temperature distribution in an arc column[19], but probably the best and most extensive data lie in the comparison between convective heat transfer experiments and theory[20, 21].

Re-entry heating is basically from two sources, i.e., either due to the conduction of heat from the hot gas to the body or due to radiative transfer. The heat transfer coefficients for either phenomena can vary over a large range. In a free molecule flow, the heat transferred is essentially the complete kinetic energy of the air molecules relative to the body, $1/2\ V^2$, and the rate of arrival of molecules is ρV, assuming complete accommodation of the molecule to the conditions of the surface. Consequently,

$$\dot{q} = \frac{1}{A}\frac{dH}{dt} = C_H \frac{1}{2}\rho V^3 = \frac{1}{2}\rho V^3 \tag{7}$$

or

$$C_H \sim 1.0$$

For continuum flow, correlations of shock tube heat transfer measurements in dissociated air[10] and theoretical calculations[11] have been written in the form of

$$q\sqrt{R} = C_1 \rho^{1/2} V^{3.15} \tag{8}$$

or

$$C_H = C_2 \frac{V^{0.15}}{\rho^{0.5}} = C_2 \frac{V^{0.65}}{\mu^{0.5}(Re)^{0.5}} \tag{9}$$

Thus, the fraction of the flow energy transferred by convection decreases with increasing Reynolds number, Re, and increases only weakly with velocity. In our range of interest, the Reynolds number varies over a large range while the velocity varies only by a factor of 2 or 3. Although the above correlation equation was written to apply only over a limited range of velocity from 5 to 25,000 ft/sec, recent experimental data obtained under highly ionized conditions have indicated that the departures from this correlation equation are not very significant.

At the present time, our over-all knowledge of the convective heating process is quite good, even up to extremely high velocities. The available data and theories are shown in Fig.8. Over-all agreement is to plus or minus 20%, even up to the highest velocities investigated. As almost 60% of the air is ionized at this condition, it is unlikely that the ionization phenomena should introduce additional uncertainties to this agreement even to much higher velocities.

Upon closer inspection of the over-all agreement, one does not find quite such a satisfactory situation. Fay and Kemp[22] have shown a difference of a factor of 2 at stagnation pressures of about one atmosphere between an equilibrium boundary layer and one in which the chemistry and ionization processes are frozen at the values existing at the boundary layer edge. The exact thermodynamic state of the gas during the experiments[20] was not known accurately due to our ignorance of the ion-electron recombination mechanism at these high temperatures. Thus, it has not been possible to differentiate experimentally between the two states of the boundary layer gases for which the calculations have been made. The experiments were performed at stagnation densities at least one order of magnitude higher than the interesting flight regime. As most flight applications are in the higher heat transfer, frozen boundary layer regime, it is important to improve our understanding of the relation between the data and theory. In addition, the theory was calculated for nitrogen and the experiments were performed in air and some of the uncertainty can lie in the differences between the transport properties of the gases.

Radiative heat transport is more sensitive to the precise state of the gas. Whereas convective heating was roughly proportional to the flux of energy arriving at the vehicle, radiative heating is dominated by the temperature and density of the gas. Thus, radiation is not only sensitive to the degree of dissociation and other chemical processes, but also to changes in temperature and density caused by the degree of completion of these processes. Because of this dependence we must differentiate between radiation from gases in thermodynamic equilibrium and those in non-equilibrium states. Considering the equilibrium gas first, one must further differentiate between opaque and optically thin radiating layers. If the optical mean free path, i.e., the distance traveled by a photon before it is absorbed, is small compared to the thickness of the radiating layer, then the radiation will be the well known Stefan-Boltzman relationship,

$$q = \sigma T^4 \tag{10}$$

or

$$C_H = \frac{\sigma T^4}{\frac{1}{2}\rho V^3}$$

The more usual situation in re-entry problems is the optically thin gas. Extensive shock tube measurements[12] have been made of the radiative properties of air and several extensive tables have been collected from these data[23, 24, 25]. Over limited ranges of the variables, correlation equations can be written to fit these data in the form of

$$\dot{q} = C_3 R \rho^{1.7} V^8 \tag{11}$$

which is applicable in the 25,000 to 35,000 ft/sec regime.

Thus heat transfer coefficients for equilibrium radiation are roughly,

$$C_H = C_4 R \rho^{0.7} V^5 = C_4\, Re\, \frac{V^4 \mu}{\rho^{0.3}} \tag{12}$$

The radiative heat transfer coefficient increases rapidly with both velocity and Reynolds number. A limiting value of C_H due to radiative transfer can be estimated from simple considerations. If the flux of energy is $1/2\ \rho V^3$ per unit frontal area, then if all the energy is radiated away, at most one-half will be radiated toward a body. The other half will be radiated to space.

Yoshikawa and Chapman[26] have considered the radiative energy transfer from a two-dimensional layer, including the effects of radiation cooling, i.e., the loss of specific energy of the gas due to radiative transfer (sometimes called decay) and the reabsorption of radiative energy as the gas becomes more opaque. Heat transfer coefficients were found to reach a maximum value of about 0.25 from these data.

Research in radiation from high temperature gases revolves about determining the absorption coefficient of the gas. Shock tube investigations have allowed us to synthesize a model of molecular radiation for air which is known to approximately 30% up to gas temperatures of 8000°K. The molecular radiators shown in Fig.9 have been identified and quantitatively normalized as shown. Identification and normalization by the individual contributor is vital because each constituent has its own temperature and density dependence. Our knowledge of air radiation shown is quite complete up to temperatues of 8000°K.

For the high re-entry speeds of interest in the future, this knowledge will have to be extended to temperatures of 20,000°K. At these conditions, air is almost completely ionized and the dominant source of radiation is the capture of electrons by atomic ions to produce nitrogen atoms, i.e., Kramers' radiation, marked N in Fig. 9. The effective charge of the nucleus needed for quantitative calculation of this radiation is uncertain to a factor of 2 to 4, leaving the radiation intensity which is proportional to this parameter squared undetermined to a factor of 4 to 16. In addition, there may be significant contributions from the atomic line radiation. This source of radiation has only recently been identified[27] and quantitative results appear to bear out the theoretical predictions. Figure 10 shows data which have been collected at almost 10,000°K, which show both the lack of certainty of Kramers' radiation and the presence of atomic line radiation. At high temperatures the line radiation will be opaque when the background continuum is still optically thin and the opacity of these sources must be treated independently.

Despite the fact that our knowledge of the radiative properties of air is quite extensive, there do exist several regions of the spectrum which are not easily accessible to laboratory measurements. This fact casts a certain shadow of uncertainty over our knowledge of total radiation. To date, no good total radiation measurements exist. One of the best approaches to such a total measurement was made in a ballistic range at NASA Ames[28] by the use of 10 narrow band filters and phototubes covering the range of 0.2 to 1.0 micron in wavelength. These results are shown in Fig. 11. The agreement with the predictions is surprisingly good. The uncertainty due to the lack of coverage of the whole spectrum is accentuated at very high temperatures due to the shift of the blackbody peak into the ultraviolet and the overall higher level of blackbody radiation. Thus, until a good total measurement is made, a degree of doubt must exist.

The radiation from gases of other planetary atmospheres is an area of research which is only just being started. From our present knowledge of the atmospheric compositions of Venus and Mars, it must be concluded that the large amount of CO_2 present relative to Earth, and the strong tendency of the CN molecules to radiate, that these atmospheres will present a more severe radiation problem at equivalent velocities.* However, the uncertainties here are as yet considerably greater than air.

The last source of heating which must be considered is the radiation from the gas which has not yet achieved thermodynamic equilibrium. Initial observations of the existence of such an effect led to the speculation that the energies involved could easily be thermally significant[30]. However, detailed studies have shown that relatively little energy is transferred[27,28]. It was shown that the radiation in the non-equilibrium zone behind a shock front was due to species produced mainly as a result of binary reactions and consequently obeyed binary scaling laws. Consequently, the integrated non-equilibrium radiation intensity from a shock front is independent of density (peak intensity varies directly as density and zone width varies inversely) and depends on velocity only. In the 25,000 to 35,000 ft/sec range this dependence can be expressed roughly as

$$q_{N.E.} = C_5 V^{(5)} \qquad (13)$$

or

$$C_{H\,N.E.} = \frac{C_5 V^2}{\rho}$$

Recent measurements have shown that the velocity exponent in Eq. (12) increases sharply with temperature when the continuum radiation due to atomic ions starts to dominate, whereas the velocity dependence of the non-equilibrium effect does not show this effect. Consequently, the significance of non-equilibrium radiation should decrease still further at higher speeds until the overshoot is essentially not noticeable. This trend has been observed experimentally.

These three sources of heat transfer have in the past been considered as independent phenomena and their heating effects have been linearly additive. Several analyses have been

* Preliminary measurements in ballistic ranges by Carlton James[29] have confirmed this estimate. He found a maximum intensity at a composition of about 7.5% CO_2, which corresponds roughly to the maximum CN concentration. In a very gross way the data can be predicted by the known properties of CN radiation.

performed in the past year to show that this assumption can lead to serious errors when considering the high speed re-entry case.

Yoshikawa and Chapman[26] have analyzed the effect of energy loss due to radiation, or energy decay, on the overall radiative heating. In addition to considering this non-isoenergetic shock layer, they also accounted for the opacity of the gas by considering re-absorption of radiation. The geometry considered was a plane one-dimensional shock layer. Their results are summarized in the map of characteristic distances for absorption and decay shown in Fig. 12. In this figure L_{dec} is the length required to lose all the energy by radiation of constant intensity behind a normal shock, while L_{abs} is the length required to reach the blackbody radiation limit. At relatively high densities, i.e., low altitudes, $L_{abs} << L_{dec}$ so that absorption dominates decay while at high altitudes, $L_{dec} << L_{abs}$ and decay dominates. L_c represents the value of the characteristic length which dominates the radiation history and is thus a combination of L_{abs} and L_{dec}, whichever is smaller.

Another form of coupling has been considered by Howe and Viegas[31]. This paper treats the effect of both absorption and decay in the shock layer on the convective stagnation point heat transfer. The effects on radiative heating, as well as injection of foreign species into the boundary layer,are also considered in this reference. The non-dimensional convective heat transfer rates, i.e., heat transfer rate multiplied by the square root of the nose radius divided by the stagnation pressure, resulting from this analysis are shown in Fig. 13. The convective heat transfer is diminished significantly by the inclusion of radiation in the flow field, the effect being larger for larger bodies and higher pressure levels. It would appear as if this effect should be predictable by considering the change in the driving potential across the boundary layer caused by the energy loss in the non-isoenergetic shock layer. This is not found to be the case in Reference 31, but the cause is not well understood. These aspects of coupled convection and radiation will certainly require further consideration.

A by-product of this analysis touches upon the coupling of the radiation and the boundary layer gases which frequently are materials other than the ambient gas, i.e., gas which has been injected into the stream by ablation. The subject of the interaction of highly energetic flows with surface materials is another whole area of research much too involved to be covered in a thorough fashion in this paper. However, two problems which are dominantly aerophysical in nature should be singled out at this stage. The first is the possibility of blocking a significant amount of radiative transfer by the use of a relatively opaque boundary layer gas. The possibility of producing a surface material which will give off vapors with the required high absorption coefficient is not considered here. However, if such a vapor can be made in the boundary layer then, in the case where radiant heat transfer dominates, it may be possible to produce a shielding effect and reduce the total surface heating by radiation. However, this phenomena may also have the inverse effect, in that the radiant heat transfer may increase due to the increased absorptivity of the shock layer, as was the case in the examples discussed in Reference 31. The absorbing boundary layer will always tend to increase the convective heating over the corresponding value in a boundary layer where air is being injected in the same quantity as the absorbing vapor.

A second problem which arises out of material interactions is the simulation of the proper heating environment for sufficiently long times so that the material properties can be studied. For situations where radiative transport is the dominant mechanism, such, as for instance, for a Venusian entry at about 40,000 fps, the facility must supply radiant heating at rates of the order of 10 kw/cm^2. It is desirable to have the heat source be the gas itself so that the spectral energy distribution of the heating is also reproduced. The simulation facility should thus be capable of containing a gas at temperatures of the order of 15,000°K in such a manner that surface material interactions will be limited to the surface under study and the other walls of the container do not produce any effects. This would be the case if a model were placed in a hypersonic flow and the stagnation region gas were at the proper condition. Such a facility could well require extremely large energies.

One final coupling mechanism to be considered here is the absorption of ultraviolet radiation by the ambient gas ahead of the bow shock. At high speeds,and particularly when a large region of the shock front is not in thermodynamic equilibrium, a significant fraction of the energy is radiated at very short wavelengths. At these wavelengths, the mean free path of a photon is quite short even in cold gases and, consequently, a large amount of absorption may take place immediately forward of the body. Preheating will thus occur in the ambient flow and the gas cap energy will increase since it must accept this preheated gas. By this mechanism, sometimes called radiation trapping, the body must now be prepared to accept more than one-half of the incident energy flux. The work of V.A. Prokofiev[32] showed that a dramatic thickening of the shock wave can result from this situation. It can also be argued that since the gas ahead of the shock will now be heated and will expand even before the body arrives, the streamlines will diverge forward of the shock wave and a flow resembling a subsonic flow will result. A related phenomena, the precursor ionization due to the absorption of ultraviolet radiation emitted in the shock front, was predicted by Hammerling[33] and has been observed in the Glenn-Mercury re-entry by Lin[34].

It has been shown in earlier sections that high speed re-entry will require that large fractions of the total vehicle weight will be required for heat protection during entry. The analyses were based on use of the best conceivable ablating materials whose performance was in the 10, - 20,000 Btu/lb range. Pyrolytic graphite would be such a material. In situations where much of the vehicle weight will be required by the heat protection system based on this scheme, alternate approaches should be carefully scrutinized. The invention of a new and novel re-entry system could well change many of our conclusions.

A technique which could well radically alter the situation is the application of magnetohydrodynamic (MHD) interactions during re-entry. Flight applications of MHD have been a tempting possibility for some time. It is basically very attractive to be able to operate upon a hot gas stream without making physical contact. In the past the large power dissipation in the machinery for producing the strong magnetic fields was found to be prohibitive[9]. Over the past few years the understanding of the phenomenon of superconductivity has developed to a point where today relatively large superconducting magnets have been made and larger and even flyable ones are being designed[35]. With the power and dissipation problems removed, flight applications of MHD become more favorable.

In the laboratory MHD interactions have been produced to demonstrate drag and lift forces[36], as well as strong deflection of ionized hypersonic flows[9]. To assess the utility of MHD interactions, the pertinent parameters governing these flows have been calculated for the density-velocity range of interest[37]. The degree to which a flow may be altered by a magnetic field is measured by the interaction parameter, S, defined as the ratio of the work done by the magnetic body force in the interaction zone to the kinetic energy of the gas. When this parameter is approximately one, significant influences of the magnetic field on the flow should be expected. The magnetic Reynolds number is a measure of the disturbance of the primary field caused by the currents flowing in the gas. For large magnetic Reynolds numbers, R_m, the field is significantly influenced by the flow while at low values of R_m the flow has no apparent effect on the field. The Hall coefficient, $\omega\tau$, is a measure of the angle between the electric field and the current. When $\omega\tau$ is large, the currents (then called Hall currents) tend to flow in the direction of motion of the fluid and the resulting force vector is then rotated and the effectiveness of the interaction is reduced.

The above three parameters have been evaluated as shown on an altitude flight velocity grid in Fig. 14. As a typical distance over which MHD interactions take place is the shock stand-off distance, i.e., the thickness of the ionized shock layer, values of the characteristic dimension, ℓ, should be of the order of ϵR_s or the shock radius divided by the density ratio. Thus one might expect that for a one meter sphere with a 10 kilogauss magnetic field, the interaction parameter would be approximately one at 130,000 ft altitude at 30,000 fps. The Hall coefficient $\omega\tau$ would be approximately one and the magnetic Reynolds number would be 0.2. It appears from this map that MHD applications ought to be possible in the velocity range above 20,000 fps at altitudes of the order of 200,000, where most re-entries will encounter the peak forces.

A recent analysis by Levy and Petschek[38] predicts the interaction of a hypersonic flow with a magnetic field, such as the induced field of a current-carrying conductor. The analysis predicts that a shock layer will occur which is concentric with the conductor, and that the density in this shock layer will drop exponentially from the shock front toward the wire with local value of interaction parameter. Thus, for strong interactions this theory predicts very low densities in the vicinity of the wire. The physical picture of this flow is a shock layer which is of thickness ϵR_s, i.e., the density ratio times its radius, supported by the magnetic field lines and essentially lifted completely off the body surface. Convective heating behind such a wire would be almost completely eliminated. If this concept were to be used on a re-entry vehicle, the larger shock radius would result in added drag which, in turn, would cause deceleration at a higher altitude. Thus, radiative heating could also be reduced through the density dependence of this form of heating.

The theory sketched above has been verified experimentally in a shock tube[39]. A current was passed through a wire stretched across a shock tube (and out through the walls) timed to reach a peak value of about 2×10^5 amps during the steady flow time. A flow geometry as predicted by the theory, i.e., a concentric, thin shock layer located at a significant distance from the wire was observed. A picture of such a flow is shown in Fig. 15. The figure shows a top view of the flow over a magnetic field produced by the wire transverse to the flow. The wire is 4 mm in diameter and the shock front stands approximately 2 cm in front of the wire. This distance was about 30% greater than the distance predicted by the theory for these currents and conditions. The aerodynamic shock for the no-field case is marked on the figure to be very close to the wire. The luminous layer is seen to be considerably thinner than its radius despite the unfavorable view afforded by looking down at a two-dimensional geometry. It must be concluded that the theory predicts the flow geometry and, consequently, the heat transfer reduction predicted by the theory is also likely to be realized.

Applications of these new concepts have not as yet reached fruition. However, the understanding of these flows being gained from current research may point the way to future applications which can materially ease the difficulties of planetary entry at extreme velocities.

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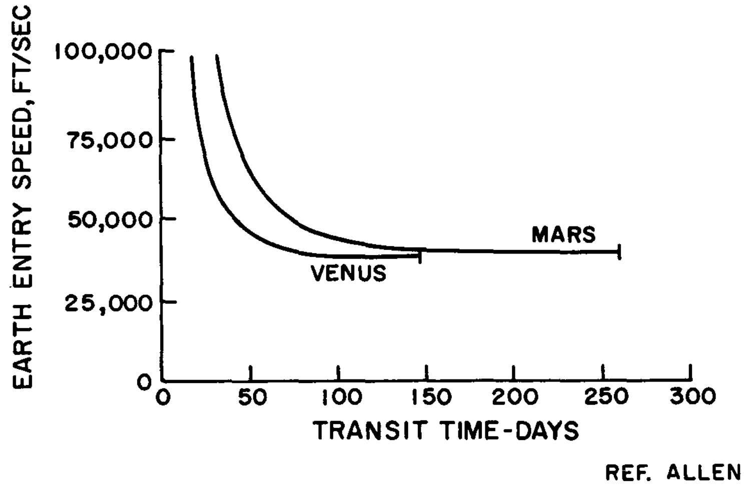

Fig. 1 Effect of velocity on the one way minimum transit time from Earth to Mars and Venus. (From Reference 3)

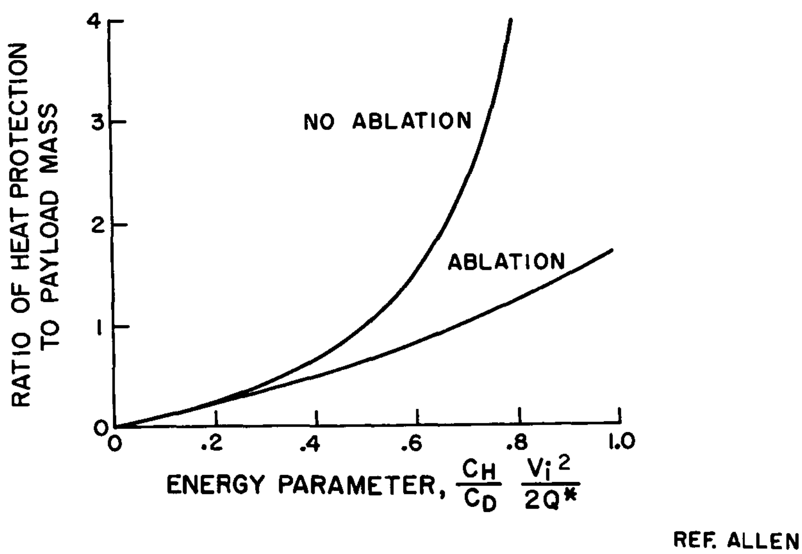

Fig. 2 Heat protection system mass requirements for ablative (mass is lost as it is consumed) and non-ablative heat shields. (From Reference 3)

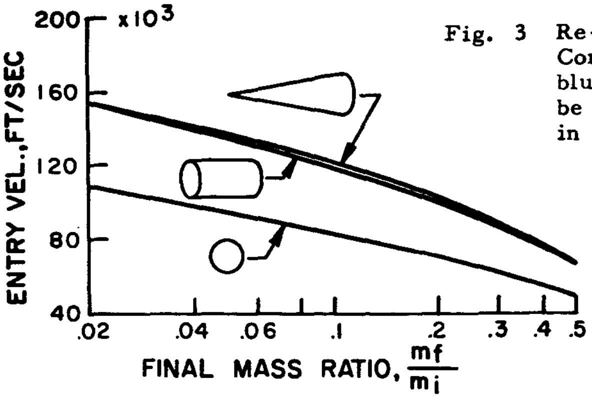

Fig. 3 Re-entry vehicle mass ratio variation with entry velocity. Cone tip is assumed to be initially small but finite and blunting due to heating is considered. Q^* was taken to be 15,000 Btu/lb and C_H constant at 0.25. Cone half angle in this calculation was 8.5°. (From Ref. 4)

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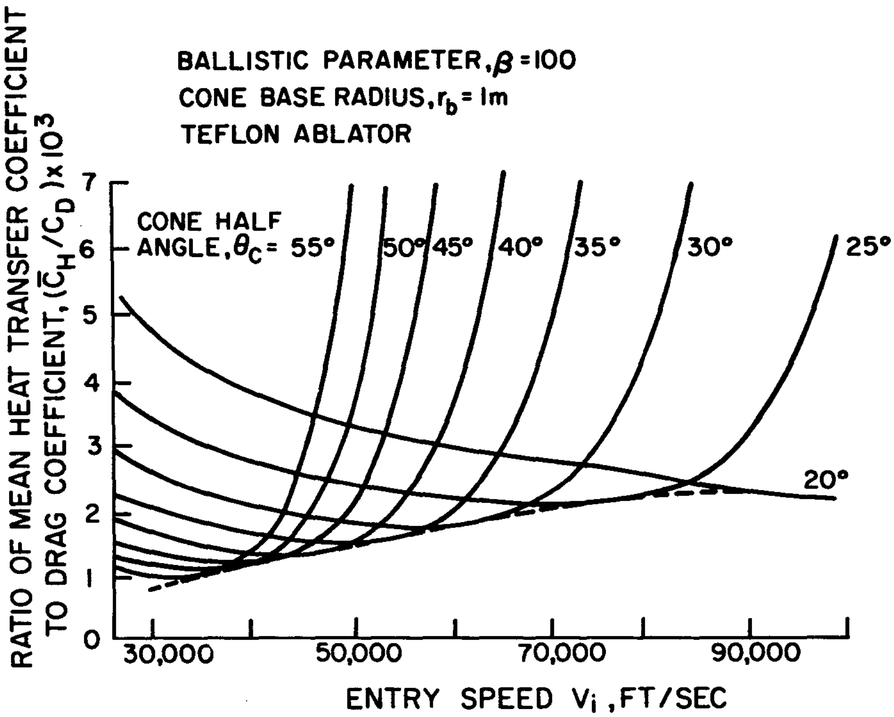

Fig. 4 Ratio of mean heat transfer coefficient to drag coefficient for various cone angles and entry velocities, showing the dominance of radiative heating (high C_H) at high velocities and the optimum geometry variation with entry velocity. (From Ref. 5)

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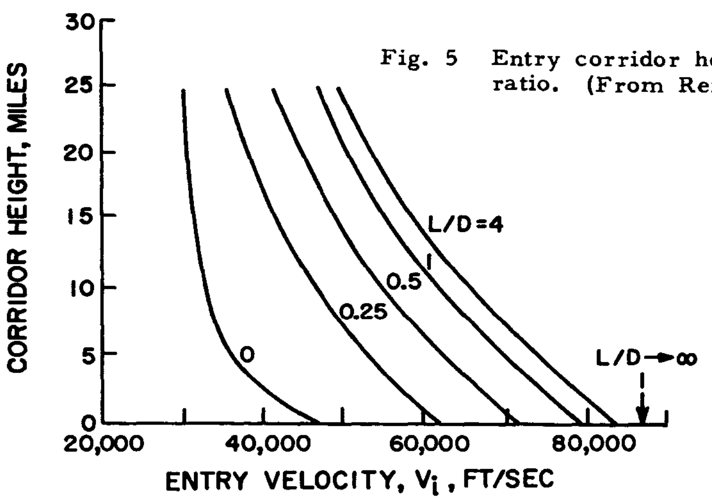

Fig. 5 Entry corridor height dependence on entry speed and lift/drag ratio. (From Ref. 6)

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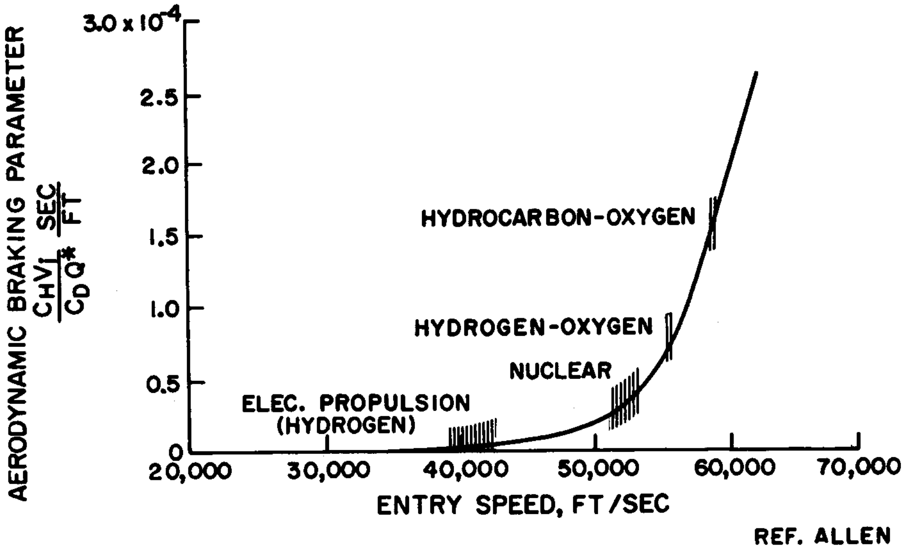

Fig. 6 Comparative performance of impulsive, rocket braking in terms of the velocity at which the heat protection system and the rocket will achieve the same mass ratios. (From Ref. 3)

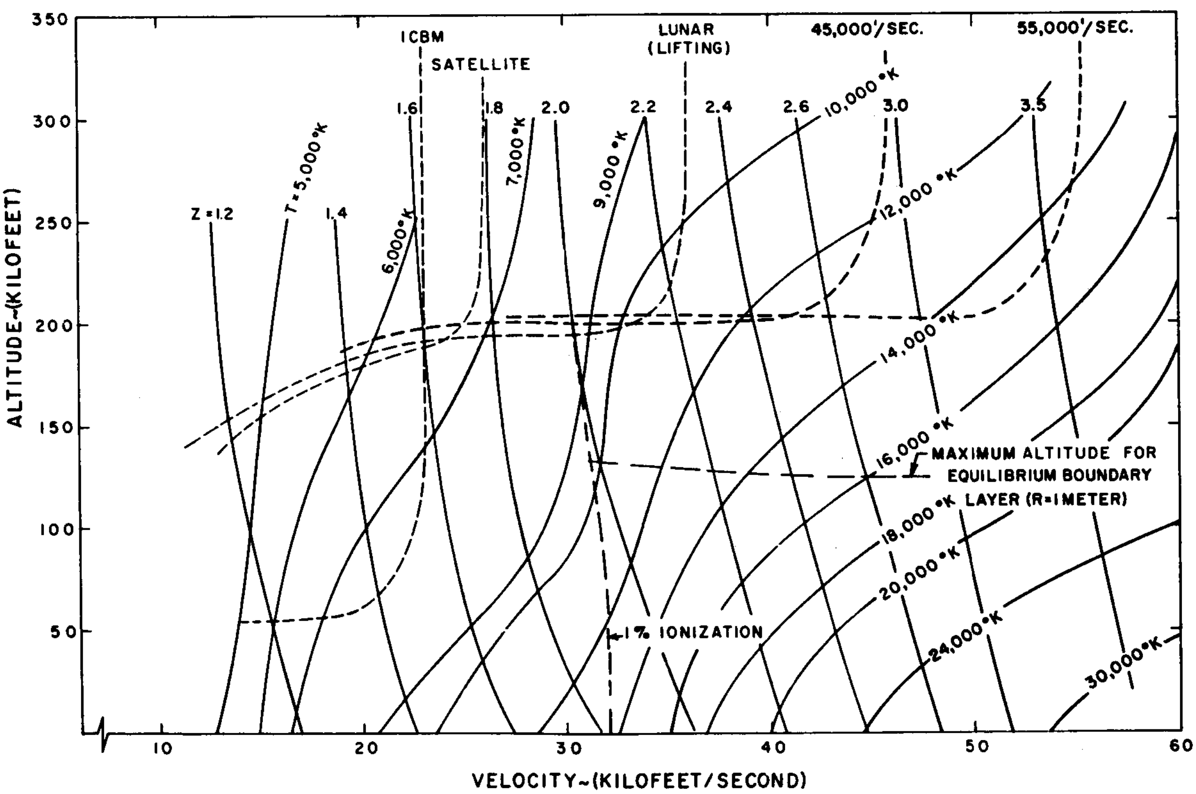

Fig. 7 Map of the stagnation temperature and compressibility function, Z, variation with altitude and flight velocity. Z is one for non-dissociated non-ionized air, approximately 2.0 for dissociated but non-ionized air and 4.0 for fully dissociated fully ionized air. Entry trajectories are 10 g undershoot boundaries for W/CDA ~ 50 L/D ~ 1.0 vehicles. Note all deceleration takes place in regions where the boundary layer chemistry is estimated to be frozen.

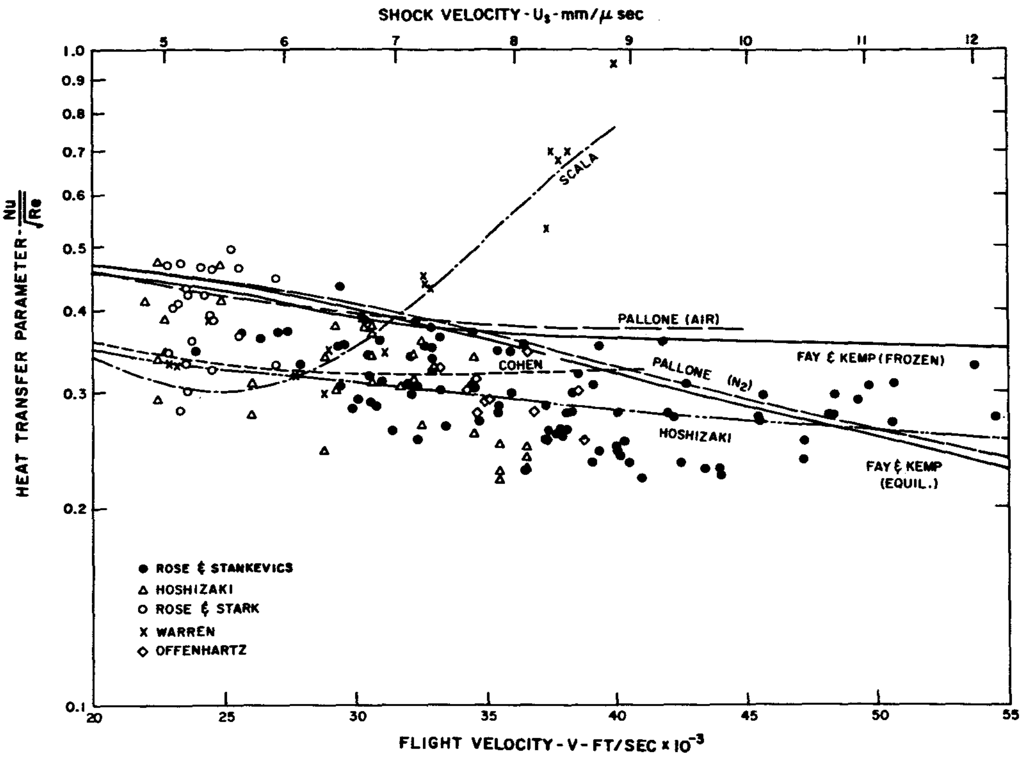

Fig. 8 Summary of available stagnation point heat transfer measurements and theories. Although a considerable amount of scatter exists in the data, the trend predicted by the theories of Fay and Kemp[22], Hoshizaki, Cohen and Pallone are clearly verified. The greatest uncertainties probably lie in the state of the boundary layer, i.e., Fay and Kemp equilibrium vs frozen result, and the differences in the transport properties of nitrogen and air.

RADIANCE OF AIR

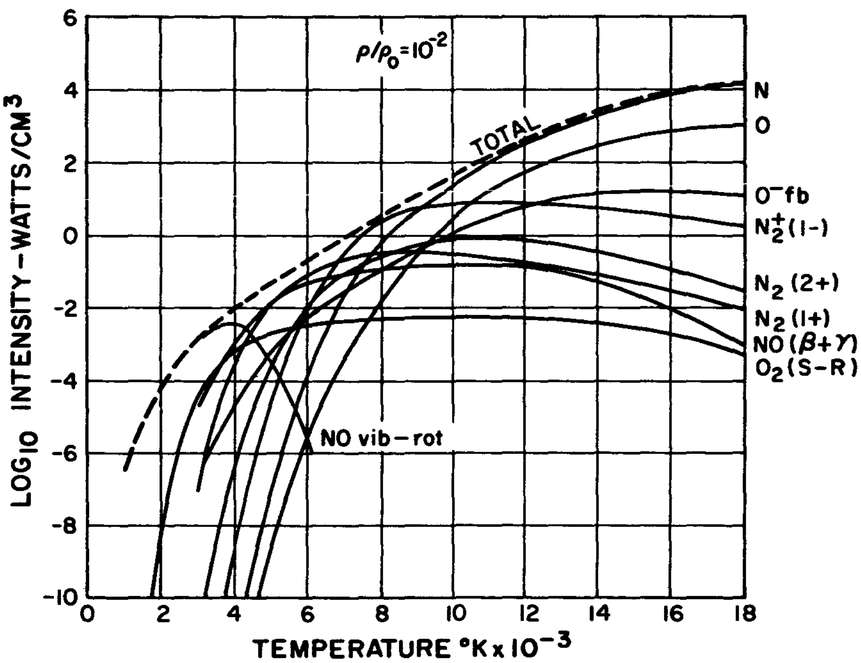

Fig. 9 Radiance of air as a function of temperature at ρ/ρ_0 of 10^{-2}. The contribution from various molecular bands making up this total is also shown. Because each band has its own temperature and density dependence, knowledge of total radiation is not enough to allow extrapolation to different conditions. An individual identification and normalization of each band system is required.

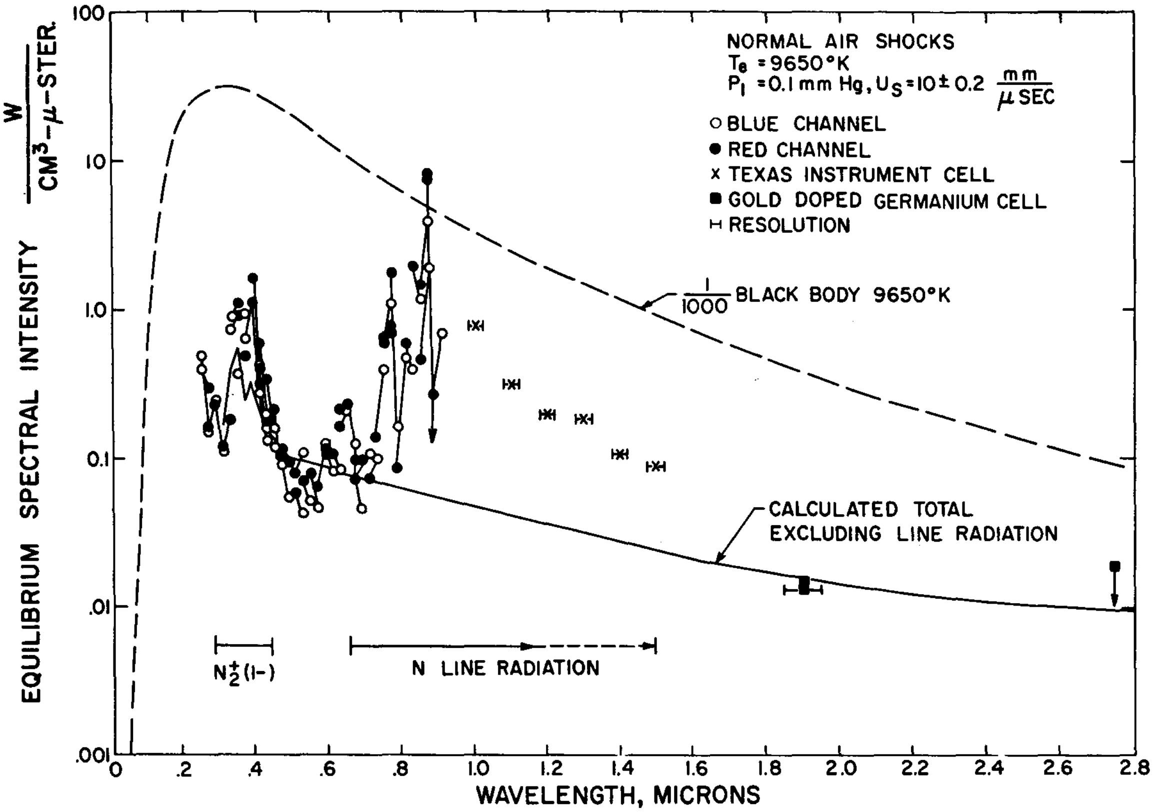

Fig. 10 Shock tube data showing the spectral distribution of radiation behind normal shocks from air at approximately 10,000°K. As can be seen from Fig. 9, only the N_2^+ (1-) band system contributes significantly. Other contributors are nitrogen atomic lines and continuum. Greatest uncertainty is the normalization of the level of the continuum radiation. Contribution in the near infrared is also not well defined.

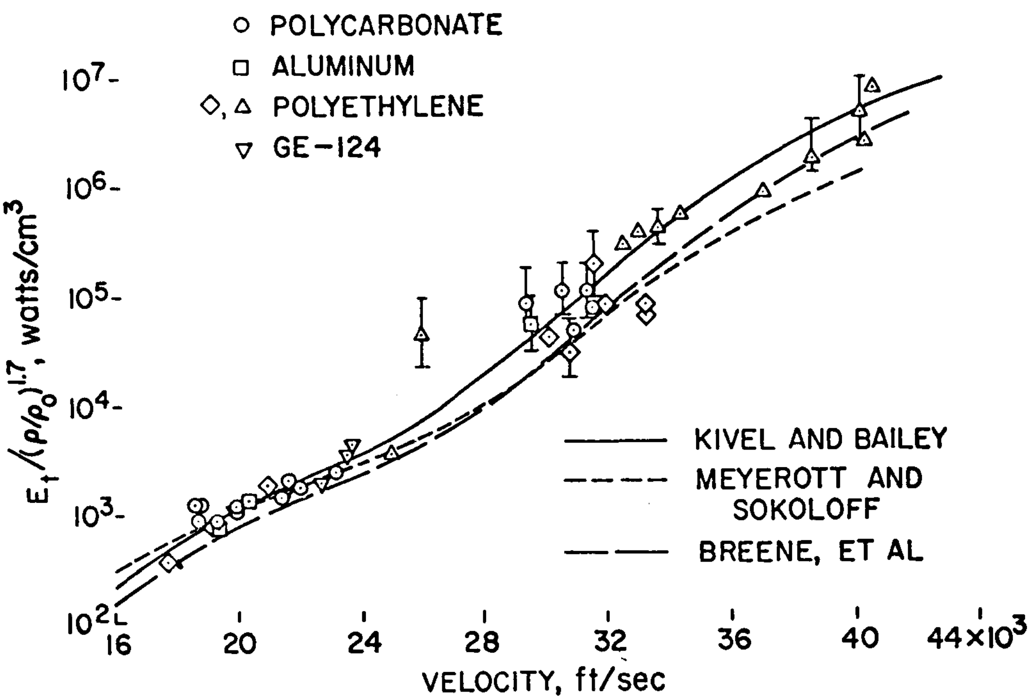

Fig. 11 Ballistic range measurements of the integrated radiation from the stagnation region in front of hypersonic pellets, compared to the predictions of Refs. 23, 24 and 25. The variations due to pellet material are attributable to contributions in the near infrared due to surface and boundary layer radiation. Wavelength region from 0.2 to 1.0 microns is covered by 10 narrow spectral and broad band detectors. (From Ref. 28)

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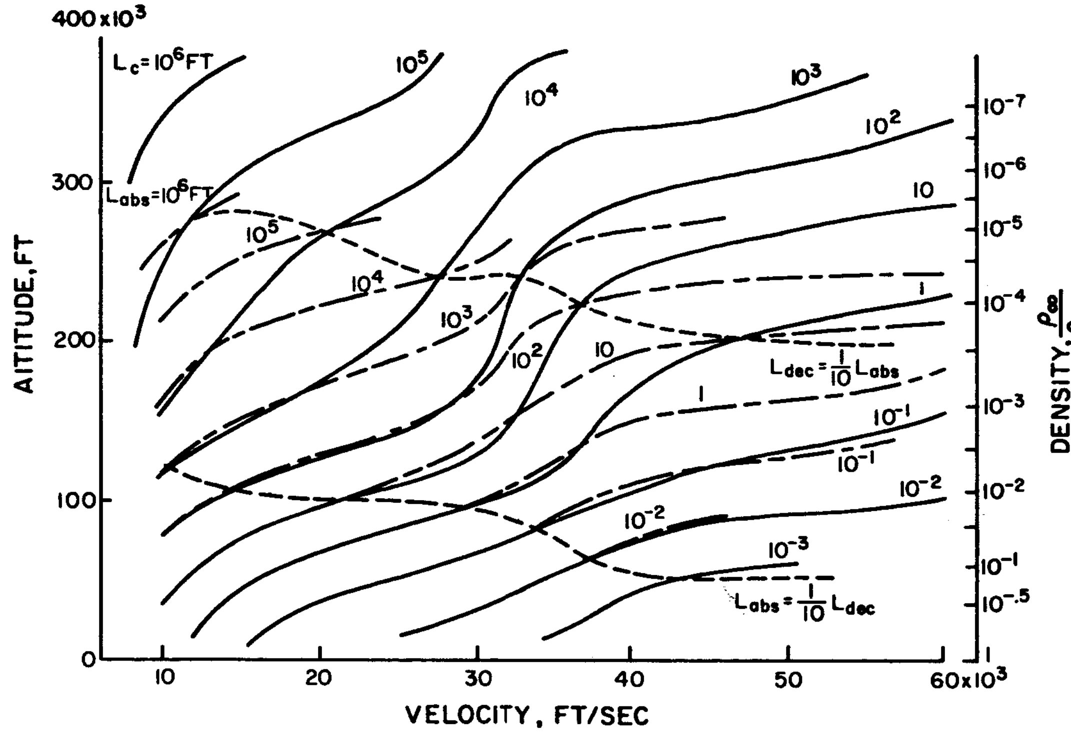

Fig. 12 Characteristic lengths for energy decay, L_{dec}, by radiation and for absorption, L_{abs}, of radiation mapped as a function of stagnation density and flight velocity. The combined characteristic radiation length, L_e, i.e., the dominant or shorter of the two, is also shown. (From Ref. 26)

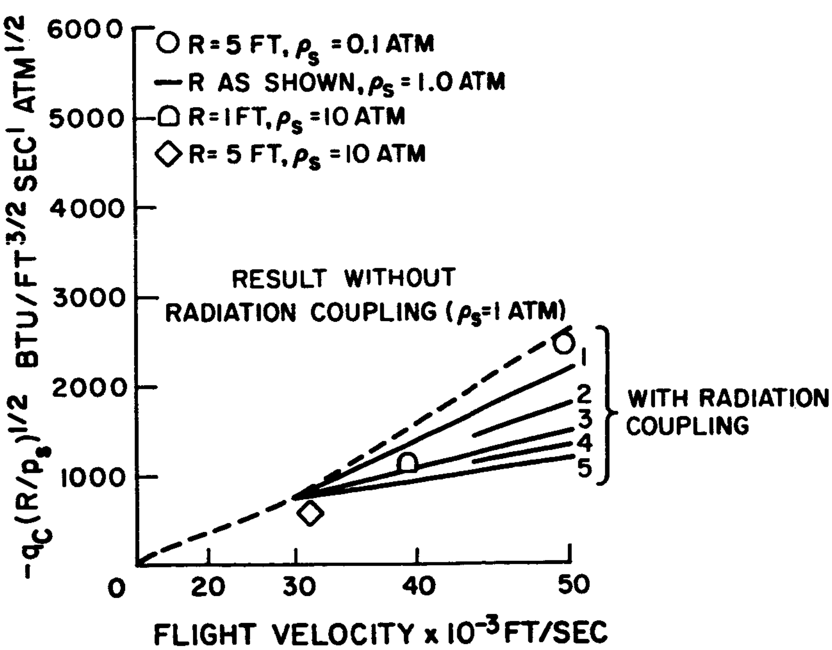

Fig. 13 Effect of radiative transport in the shock layer on the convective heat transfer rates at the stagnation point. The no coupling result shown is essentially in agreement with the data shown in Fig. 8. Radiation cooling of the shock layer decreases the convective heating significantly, especially at very high stagnation pressures and for large bodies. (From Ref. 31)

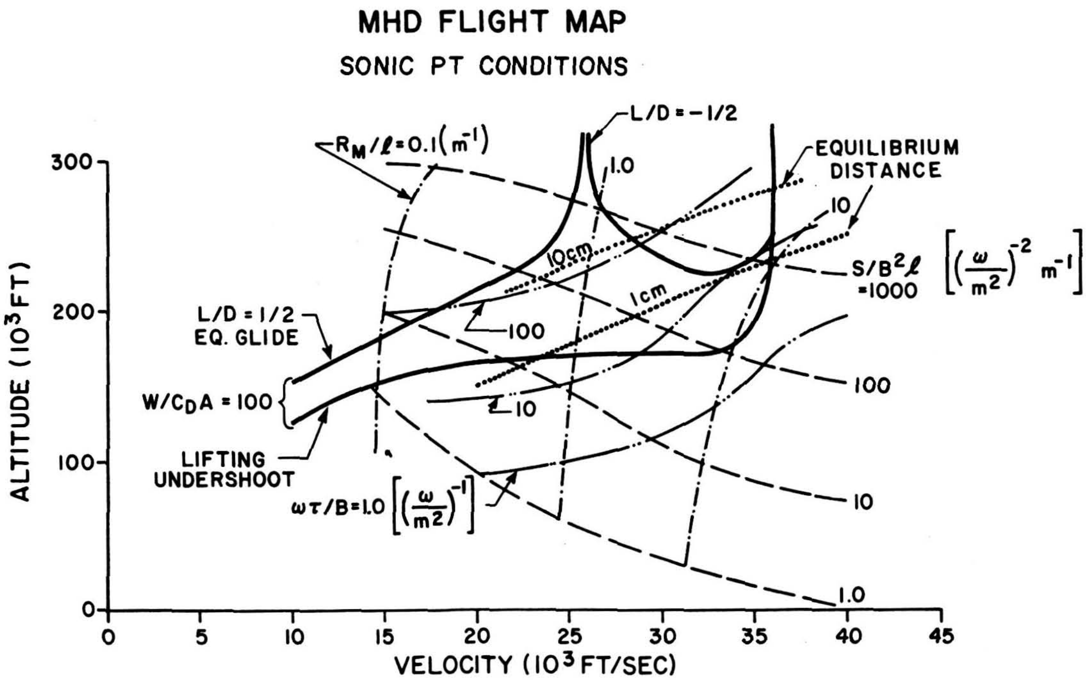

Fig. 14 Map of pertinent conditions, at the sonic point a hypersonic vehicle, useful for determining the character of the interaction of the flow with a magnetic field. All parameters; the interaction parameter, S, the Magnetic Reynolds number R_M, and the Hall coefficient, $\omega\tau$, are normalized for unit magnetic field strength, i.e., one weber per square meter, and unit length, i.e., one meter.

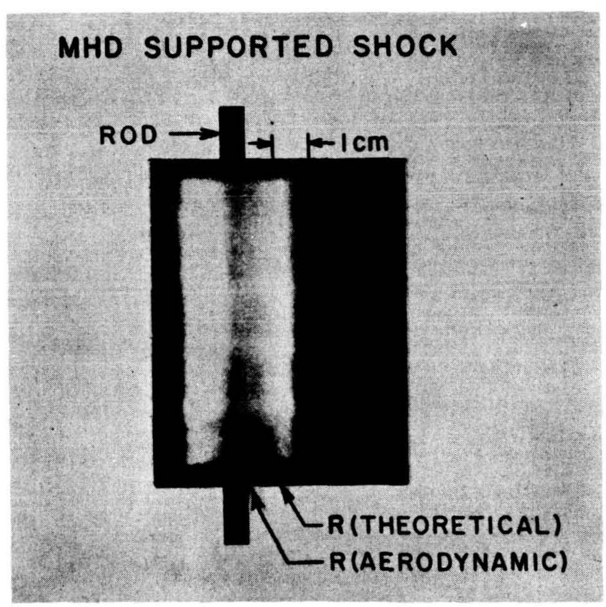

Fig. 15 Image converter photograph at an exposure of .05 μsec at the interaction of the hypersonic flow in a shock tube with a current-carrying wire aligned transverse to the flow. The shock layer which is formed is concentric with the wire, is of thickness ϵRs where ϵ is the density ratio and Rs the shock radius, and the density is very low in the immediate vicinity of the wire (evidenced by the lack of luminosity in front of the wire). The agreement between prediction and actual location of the shock front is evidence for anticipating the large heat transfer relief predicted by the theory.

THE X-15 PROGRAM

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Introduction

The X-15, which is the latest of a long series of research airplanes, is the nation's first piloted entry vehicle. The pilot is in complete control of the vehicle from launch to landing, thus making it possible to accomplish extensive research on the proper role of man in future space ventures.

Numerous articles and several books describe the development of the X-15 airplane and early flight-test results; therefore, only brief mention is made herein of some of the more important historical aspects of the X-15. Although the flight-test phase of the X-15 program is not yet complete, many tangible results have been achieved (see bibliography). One objective of this paper is to review some of the most important X-15 operational problems. The major emphasis, however, is placed on piloting aspects. Much has been learned in both of these areas that should be of great value in the development of future space vehicles.

History of Project

Design Development

The design requirements specified for the X-15 were a maximum velocity of 6,600 fps and an altitude of at least 250,000 feet. Representative areas of the primary structure were to experience temperatures of 1,200° F, and some portions of these representative structures were to achieve heating rates of 30 Btu/sq ft/sec. The airplane was designed to have satisfactory aerodynamics and structural characteristics relative to specific flight profiles, which resulted in attainment of the required performance and heating.

In December 1954, invitations were issued to contractors with experience in the design of high-speed, high-altitude aircraft to participate in the design competition. Proposals resulting from the invitation were received and evaluated during the summer and fall of 1955. A contract was awarded North American Aviation for construction of three "X-15" aircraft in June 1956, and a contract was given to Reaction Motors for development of a suitable rocket engine.

An extensive wind-tunnel and structural-component testing program was initiated in 1956. By September 1957, enough data had been collected so that construction of the first X-15 could be started. Much had been learned in these tests about hypersonic design considerations. This knowledge was reflected in numerous design changes instituted to accomplish the design requirements. The final X-15 configuration is illustrated in figure 1.

The X-15 propulsion system includes a 1,000-gallon liquid-oxygen tank, a 1,400-gallon anhydrous-ammonia fuel tank, and a YLR99 rocket engine. (The initial X-15 tests utilized an LR11 engine because of delays in the development of the design engine.) The YLR99 engine is throttleable from 40-percent to 100-percent thrust and can be shut down and restarted in flight. An important feature of the engine is the idle capability which allows about 85 percent of the engine starting cycle to be completed prior to launch.

To withstand the temperature environment expected, all external surfaces on the X-15 airplane are Inconel X, and the internal structure is a composite of Inconel X, titanium, and some aluminum, depending on the expected local temperature environment. Cooling for areas such as the flow-direction sensor on the nose, the cockpit, and electronic bay is supplied by liquid nitrogen.

Aerodynamic control is provided by the horizontal stabilizer deflected symmetrically for pitch and differentially for roll. Upper and lower movable vertical-tail sections are utilized for directional control. The vertical tail is a 10°-wedge section, and the lower movable section is jettisoned for landing clearance. Attitude control at low dynamic pressures is provided by rockets in the nose and wings. The stability augmentation is essentially a conventional damper system on the first (X-15-1) and second (X-15-2) X-15 airplanes. The third X-15 (X-15-3) utilizes the Minneapolis-Honeywell (MH-96) adaptive control system. The X-15 landing-gear system incorporates a nose gear and main-gear skids located well to the rear on the fuselage.

Flight Tests

Aerodynamic configurations. Of the 77 X-15 flights made prior to April 1, 1963, 68 were made with the design tail (lower ventral on). The last eight flights, however, were flown with the lower ventral removed. The reason for this configuration change is discussed subsequently.

Mode of operation. The mode of operation for the X-15 flight program is illustrated in figure 2. Two B-52 airplanes have been converted to carrier airplanes. The X-15 is launched from a location between the B-52 fuselage and inboard nacelle of the right wing.

The research flights were planned to be conducted along the instrumented range extending approximately 400 nautical miles northeast of Edwards, Calif., to Wendover, Utah. The X-15 is launched over various dry lakes, with all flights terminating at Edwards. A maximum range of 225 nautical miles has been required so far in the program.

Flight progress. The initial phase of the

X-15 flight program can best be described as simply a series of progressive steps to higher speeds and higher altitudes. Some deviations from this approach were made to investigate higher structural heating rates and stability and control at high angles of attack in order to insure a reasonable level of flight safety.

A summary of the flight progress for each of the three X-15 airplanes is presented in figure 3. Included are some of the events that significantly affected the program progress. The first glide flight was made in June 1959 by the contractor with the X-15-1; a powered flight with the X-15-2 was made in September 1959. In all, 11 contractor flights were made with the interim LR11 engine during 1959 and 1960 to evaluate the airplane and the various systems. During this period, the X-15-2 airplane was damaged in an emergency landing after a fire developed in the engine compartment during flight.

The first government flight with the X-15-1 airplane with the interim engine was performed in March 1960. This airplane was tested until February 1961, and the maximum speed and altitude for the interim engine were achieved. Six pilots, from the U. S. Air Force, the U. S. Navy, and the NASA, participated in this phase of the program.

The X-15-2 airplane was the first of the aircraft to be converted to the YLR99 engine and was flown three times by the contractor during November and December of 1960 to demonstrate engine throttling and engine restart capability.

The government first flew the X-15-2 with the YLR99 engine in March 1961 and continued the research program that had been started with the X-15-1 airplane. After engine conversion, the X-15-1 airplane was returned to the government and was flown again in August 1961. The X-15-1 and X-15-2 airplanes were used interchangeably in support of the flight program until August 1962 when the X-15-1 was removed from flight status and modified by the contractor to accommodate special cameras. The X-15-1 is currently in flight status. The X-15-2 was severely damaged in November 1962 in an emergency landing and is being rebuilt.

The X-15-3 airplane, which suffered major damage in June 1960 during a ground run of the YLR99 rocket engine, has been rebuilt and modified to accommodate the MH-96 adaptive control system. The first flight of this airplane was made in December 1961. Fourteen flights have been performed, and the MH-96 control system acceptance demonstration has been completed.

Performance. Figure 4, a plot of the X-15 performance envelope, indicates the extent of accomplished performance during the 77 flights made before April 1, 1963. The maximum velocity attainable with the X-15 is somewhat less than the design goal, inasmuch as the airplane is now heavier, the engine specific impulse is slightly lower, and available burning time is lower than originally planned. The solid curve shows the presently attainable altitude and velocity for the YLR99 engine. The lower dashed line, included for reference purposes, is for a constant dynamic pressure of 1,500 psf, although the design limit is 2,500 psf. The diagonally lined area shows that a maximum altitude of 314,750 feet and a maximum velocity of 6,020 fps have been attained. Although additional performance potential remains to be explored, attainment of altitudes of the order of 400,000 feet or greater (shaded area) may be limited by high dynamic pressure and high acceleration during recovery.

Operational Experience

A fairly complete summary of X-15 operational experience has been presented in several papers, the most recent at the March 1963 AIAA Space Flight Testing Conference in Cocoa Beach, Fla.[1] A less detailed summary of X-15 operational problems is presented in this paper.

Figure 5 shows most of the areas of concern in X-15 operations. These areas have been categorized as: structural and thermo-structural problems, the YLR99 rocket engine and propellant system, auxiliary power unit, and control and guidance systems. The bar graphs show the magnitude of particular problem areas and the time period in which such problems existed.

As might be anticipated, most of the operating problems were encountered when a specific system or X-15 structural component was first exposed to a meaningful operating environment. This generally occurred during early flights. However, in the thermo-structural area most of the problems were experienced when the X-15 attained a point in its performance envelope at which a significant thermal environment was reached. In some instances, such as with the stability augmentation system and the auxiliary power units, initial problems were resolved only to encounter different problems later in the flight program. With the YLR99 engine, a continuous series of problems of varying importance have been experienced.

As shown in the figure, the X-15 airplanes were grounded following the first landing accident after the X-15-3 engine explosion and briefly after the second landing accident. The most recent grounding was caused by problems with the auxiliary power unit.

Structural Problems

Most of the structural and thermal problems resulted from insufficient engineering data with which to establish realistic design criteria. Thus, they are representative of the state of the art at the time the X-15 was designed.

The deficiency of the landing gear was apparent after the first landing. After the fourth landing, the main-landing-gear shock struts were replaced by struts with greater energy-absorbing characteristics, and the gear backup structure was strengthened. High loading on the nose gear, caused by foaming of the oil within the oleo strut, was relieved by a modification which provided a floating piston to separate the oil from the gas. More recently, techniques have been devised to reduce landing-gear loads caused by the input of the stability augmentation system to the horizontal tail, inasmuch as such inputs were contributing factors in the latest X-15 accident.

Panel flutter was considered in the design of the X-15 using criteria then available and was not believed to pose a problem. Nevertheless, panel flutter has occurred in flight and has required modification of extensive areas of the fuselage side fairing and vertical tails, as indicated by the diagonal lines in figure 6.

Thermo-Structural Problems

The magnitude of temperatures resulting from turbulent flow in the vicinity of expansion-joint slots in the wing leading edge is not easily predictable. This condition contributed to the local permanent buckling, as shown in figure 7. In an effort to minimize the buckling problem, several design changes have been made. Two of the changes are shown in the lower sketch. An 0.008-inch-thick Inconel tab, welded along one edge, was installed over each slot to eliminate or at least minimize the turbulent flow and resulting local hot spots. A fastener was added at the slot to decrease the fastener spacing and to increase the skin resistance to buckling. To reduce the load that the skin splice must carry at each slot, a further change added expansion slots with cover tabs in three of the outboard segments of the leading edge. Internal shear ties were added at the new slots to prevent relative displacement of the leading-edge segments. These modifications have, essentially, solved the problem.

The X-15 windshield glass and retainer have undergone several changes in order to develop a combination which will withstand the temperatures encountered during certain high-speed or entry conditions. The first change was the replacement of soda-lime glass with alumino-silicate glass which has higher strength, better thermal properties, and can withstand thermal stress to temperatures 1.5 times greater than the expected flight values.

Failures of both types of glass were encountered in 1961. The fracture pattern of the alumino-silicate glass is shown in figure 8. It was observed that the retainer frame buckled near the center of the upper edge of the glass and that failure of both glass panels was initiated adjacent to this buckled area. The buckled area apparently created a local hot spot which induced higher thermal stresses than even the alumino-silicate glass could withstand. To eliminate this condition, new retainers were fabricated and installed which had twice the thickness of the original frame and were made of titanium instead of Inconel X. The reduced coefficient of expansion of titanium compensates better for the differential expansion associated with the cooler Inconel X substructure frame. Buckling of the frame was eliminated, but a new problem was encountered. After a flight last June, small cracks in the glass were observed near the forward edge of the rear section of the retainer, shown by the crosshatched area in the figure. Local heating resulting from stagnation ahead of the thicker retainer was suspected, and a glass-inspection procedure with polarized light verified the occurrence of glass annealing in this area. The rear section of the retainer (shaded region) was completely removed to eliminate the discontinuity and allow an undisturbed flow across the glass. Subsequent flights have been made with no evidence of annealing.

Rocket-Engine Experience

The YLR99 rocket engine has been the source of more flight-schedule delays than any other X-15 system. Preliminary Flight Rating Tests did not accomplish their objectives, as was subsequently shown by flight experience. Some of the more prominent problems have been premature chamber failures, pump-seal leaks, corrosion, metering-valve binding, and random failures of pressure switches, relays, and other small components. Perhaps the most insidious of all the X-15 engine problems involved detonations within the YLR99 engine second-stage igniter (fig. 9). The first-stage nozzle wall is also a wall of the second-stage liquid-oxygen manifold and curves around to form the injector face for the second-stage igniter.

On two flights, one in August 1962 and one in October 1962, the injector face was blown out of the igniter after shutdown. After the first occurrence, powerplant representatives concluded that a thread lubricant reacted with liquid oxygen by impact of a loose orifice plug. The second occurrence was with the orifice plug and fitting fabricated as an integral unit and with the system assembled with specially qualified thread lubricant. A NASA investigation team concluded that the only combustible material available was the fuel, liquid ammonia. For an explosion to occur would require a backflow of ammonia from the second-stage igniter chamber through the injector face into the liquid-oxygen manifold. Also, it was noted that both igniter detonations followed engine burnout due to liquid-oxygen depletion. This theory was proved in a series of special ground tests.

Design studies are being made toward sustaining second-stage operation for a longer period in the event of liquid-oxygen supply-pressure decay. Meanwhile, flights are being continued by servicing the X-15 with less than normal fuel to assure that oxygen-depletion burnouts do not occur in flight. This procedure reduces burning time and, therefore, imposes a limitation on maximum-performance capability. In addition, a shroud has been devised that will be placed around the igniter to retain any pieces that might separate, should more severe detonations occur; however, this configuration has not been tested.

Auxiliary-Power-Unit Experience

Early in the program, many minor problems with the auxiliary power units (APU) were encountered. These were primarily component failures which delayed flights until modifications and improvements increased their reliability. For approximately 2 years, fairly satisfactory operation was obtained. Then, in July 1962 a sudden rash of pinion-gear failures appeared in the units. Nine units were removed from service by mid-January of this year because of severe gear wear. Two X-15 pilots, on separate occasions, experienced complete gear failure and loss of one APU early in flight. Failure of both units is catastrophic; without hydraulic power, the aerodynamic flight control system cannot operate. After the second of the two flights, the pinion gear in the second APU was found to be badly worn. The X-15 aircraft were grounded until the problem was resolved.

Considerable effort was expended in seeking the cause of these failures. The sudden appearance of this problem after several years of satisfactory service under seemingly identical conditions led investigators to suspect a change in some operational factor. Fabrication, turbine-wheel balance, assembly, and overall inspection procedures were reviewed. Gear cases were checked for distortion or bearing misalinement. Data were scrutinized to determine if different aircraft flight parameters had imposed a new environment upon the units. The lubricating oil was investigated.

The study indicated that the pinion-gear failures were primarily attributable to insufficient lubrication at high altitude. Laboratory tests proved that pressurizing the gear box greatly alleviated the situation. Such a modification is now incorporated into the X-15 aircraft. In addition, procurement of gears fabricated from a more durable material and the design of a force-feed lubrication system were recently initiated as a long-range effort.

Experience With Control and Guidance Systems

The importance of careful system-development programs prior to flight testing has been dramatically illustrated by the X-15. In some instances, however, serious system problems can arise in spite of a thoroughly planned development program.

Before it was installed on the X-15 airplane, the stability augmentation system (SAS) underwent extensive developmental and environmental testing, including tests using the analog flight simulator and control-system mockup. In the simulator tests, a limit-cycle oscillation of 2.5 cps to 4 cps, primarily in roll, was detected but was not thought to be of sufficient amplitude to be objectionable. However, a tolerance level for limit-cycle amplitude had not been established. Figure 10 shows the limit-cycle amplitudes, as the control power is increased, for several signal-shaping filters in the stability augmentation system. When these oscillations occurred in flight, with the production filter, the pilots found them to be objectionable at conditions of high control power, so a modified filter was incorporated. Subsequently, during a relatively low-altitude entry, it was found that the lightly damped horizontal surfaces were excited at their natural frequency of 13 cps by control inputs. The X-15 inertial reaction to this oscillation was sensed by the augmentation-system gyros which closed the loop through the control system and produced a rather violent vibration. To alleviate this problem, a redesigned, or notched, filter was checked out on the fixed-base simulator to avert both the objectionable limit-cycle oscillations and the destructive higher-frequency resonance associated with the vehicle structure. This modification has been successful in flight.

Because of the great dependence placed on the stability augmentation system during certain critical flight phases, a backup, or auxiliary, system was developed on the simulator and incorporated in the aircraft control system. This system is automatically actuated in the event of a primary augmentation system failure. In flight, spurious signals generated by other electrical equipment actuated the system when the primary stability augmentation system had not malfunctioned. This problem was subsequently eliminated by altering the switching-network response. The system is now functioning satisfactorily.

An example of a more successful system development and vehicle integration program is the experience gained with the Minneapolis-Honeywell (MH-96) adaptive control system currently installed in the X-15-3. This system automatically adapts itself to the varying control effectiveness and changing basic-vehicle characteristics by providing an essentially invariant response to control input throughout the flight envelope, using automatically blended aerodynamic and reaction controls. A sketch of the pitch mode of the MH-96 adaptive system is shown in figure 11. A prototype of this system was flown in an F-101A airplane and was then extensively checked out in the X-15 simulator before the flight components were installed in the X-15 airplane. In general, few adjustments to the adaptive system have been required as a result of operational flight experience. The most significant changes were a reduction in the maximum level of usable system gain and the addition of a new notched filter. These modifications were required because of objectionable limit-cycle amplitudes and structural-resonance problems similar to those of the stability augmentation system. Extensive ground and prelaunch checkout procedures for each flight have facilitated the successful accomplishment of 14 flights.

For some systems, development requirements have been grossly underestimated. A good example is the experience with the X-15 inertial flight data system, which provides vehicle velocity, height, rate of climb, and attitude about the three axes to the pilot and to data recorders. This system was designed to be alined from the B-52 carrier airplane prior to launch of the X-15. Considerable flight experience was required to develop adequate procedures and control techniques for the proper alinement and erection of the system. Moreover, a number of engineering modifications and component improvements were necessary to obtain even acceptable operation of the system. As shown in figure 12, the velocity data are reasonably accurate for flight-control purposes over most of the flight regime. The altitude data are unacceptable during the final stages of flight. Therefore, it has become necessary to develop and utilize alternate techniques and procedures for completing various flight missions, using cues provided by other systems.

Summary

Many problems, both physical and procedural, experienced in the X-15 program could have been prevented had they been anticipated or more thoroughly accounted for in design and in qualification testing. A greatly decreased program cost and increased flight frequency would then have been possible, with attendant earlier attainment of research objectives. This statement is understandably broad, since it is impossible to provide for all contingencies. A research vehicle built in limited quantity with limited funds and incorporating untried systems is destined to experience many difficulties. Thorough, realistic system and component testing should be completed as early in the program as possible. Neither component nor system testing can stand by itself.

Piloting Aspects

The important phases of a typical X-15 altitude mission are shown in figure 13. Following rotation, the boost phase consists essentially of a constant pitch-attitude climb in which a peak dynamic pressure of from 500 psf to 1,000 psf is attained. The longitudinal acceleration on the pilot at burnout is about 4g. After burnout there is a relatively long coast period of up to 3 to 4 minutes during which reaction controls are used. Perhaps the most demanding phase of flight is the high-angle-of-attack entry in which peak normal accelerations of the order of 4g to 6g and longitudinal accelerations of -2g to -3g are imposed on the pilot in a flight regime where there is a rapid increase in dynamic pressure and wide variations in aerodynamic characteristics.

The final phase of an X-15 mission involves navigating to the Edwards area and performing the actual landing on the dry lake.

Boost

Figure 14 is a time history of the powered portion of flight for a 270,000-foot altitude mission made in January 1963. A maximum normal acceleration of about 2g was used in the roundout to attain the specified pitch angle of 38°. At this point (40 sec) the ground controller indicated that the flight path was somewhat high, so the pitch attitude was reduced to correct the trajectory. (The pilot is usually able to control the climb angle to within ±0.5° without difficulty, since control effectiveness is good and the airplane is relatively stable.) The controllability was rated as excellent. The acceleration along the longitudinal axis during the thrust period reached a maximum of 3.7g at burnout. In this instance, there was a 3-second delay in terminating thrust because the pilot experienced difficulty in reaching for the throttle. This extra burning time caused the vehicle to overshoot the designated altitude by some 20,000 feet. No significant burnout transient was evident in this flight, nor in any other flights. The transient in angle of attack and pitch attitude in this instance was probably induced by the pilot when he reached for the throttle.

How well the pilot has been able to achieve the peak velocity and altitude called for in a flight plan should be considered. Figure 15 compares the actual (flight) and planned burnout velocity and peak altitude. The mean discrepancy in velocity is about 200 fps with an extreme value of 500 fps. The mean dispersion in altitude is about 8,000 feet with an extreme difference of almost 35,000 feet. This piloting performance may seem crude compared to the 50 feet in 500,000 feet in altitude and 5 fps in 25,000 fps in velocity accomplished by the automatic systems used in Project Mercury. However, the X-15 mission, although designed to be controlled by a human pilot, has no requirement for this type of accuracy. Consequently, suitable instrumentation and display sensitivity have not been provided.

The X-15 experience certainly will not provide all of the information needed to determine the proper role of the pilot in future space boost operations. However, this experience will contribute to the fund of knowledge that will enable the final decision to be made.

In some X-15 situations, having the pilot in the loop was disadvantageous; in others, his presence was beneficial. On one flight, for example, the pilot had not flown the X-15 for 7 months. The trim selector in the airplane was different from that in the simulator. At launch, the pilot energized the wrong button. Without trim, a poor rotation was achieved which resulted in a burnout velocity that was 500 fps low and an altitude that was 30,000 feet lower than desired. In another instance, the pilot experienced pitch disorientation during boost. He disbelieved his displays and pushed over to check the horizon visually. By burning slightly longer than planned, the pilot was able to burn out within 8 fps of the desired velocity, but the final altitude was 22,000 feet below the flight plan.

Now, consider the benefits of having the pilot in the loop. In several instances, automatic damping systems have failed during flights and the pilot has been able to continue the flight plan to the successful completion of the flight. In one instance, during an altitude flight with the adaptive flight control system the plan called for a constant-pitch-attitude boost. When the system undershot the desired 38° pitch angle, the pilot overrode the automatic system and brought the airplane to the desired pitch angle. Final altitude was within 3,000 feet of the desired 250,000-foot peak.

On the basis of X-15 experience and also as a result of separate studies initiated and/or participated in by NASA Flight Research Center personnel, it is firmly believed that the pilot should be considered an integral part of the control loop. Of course, there is the possibility of introducing human error. However, by utilizing previous experience, such as that being accumulated on the X-15, and by careful training, it is felt that the advantages of active pilot participation in the boost control task outweigh the disadvantages. It has been repeatedly demonstrated that the control task of boost to orbital insertion is well within human capability.[2] It is imperative, however, that the pilot be included in the design from the beginning by providing an adequate display system and control modes to take full advantage of his presence.

Entry

The entry maneuver is perhaps the most challenging from the pilot's standpoint, since it is flown at a relatively high angle of attack and under rapidly changing conditions of dynamic pressure, temperature, and velocity, with the associated changes in aircraft stability and response.

Consider basic stability during the entry. Figure 16 shows the lateral-directional controllability boundary with the roll damper inoperative for the design configuration, that is, lower ventral on. The shaded area represents an uncontrollable pilot-airplane combination, defined largely from analytic methods and simulator runs but verified in limited flight tests. It is evident that a large portion of the entry is performed in an angle-of-attack range that is unflyable with the roll damper off. This condition is attributed to a negative-dihedral effect primarily caused by the lower ventral. The full significance of this problem was not appreciated until the X-15 was in flight status. For the same roll-damper-out emergency condition with the lower ventral off, the vehicle is controllable, although the handling qualities are not considered good in the entry angle-of-attack range.

Figure 17 is a comparison of a 314,750-foot entry made with the design configuration and a 271,700-foot entry made with the lower ventral removed. In both instances, the X-15-3 with the adaptive flight control system was used. For the basic (ventral on) configuration, the pilot was somewhat disturbed by the buildup in sideslip near peak dynamic pressure which was primarily manifested by a ±0.5g lateral acceleration. Largely because of this occurrence, the pilot rated the lateral-directional controllability as only marginally satisfactory.

In a recent entry made with the lower ventral off (right portion of fig. 17) no pronounced lateral-acceleration oscillation was evident, and the lateral-directional characteristics in the terminal phase of entry were rated as excellent by the pilot.

Present plans call for all future altitude missions to be performed with the lower ventral removed, although a study is being made to arrive at a different vertical-tail configuration. A more desirable design would provide a greater margin of directional stability at low angles of attack than provided by the present ventral-off configuration, but would retain the beneficial positive-dihedral effects at the higher angles of attack.

Figure 18 is a comparison of entries for two very similar flights to 250,000 feet. In each instance, the original design tail was used; however, in the flight shown on the left the standard stability augmentation system and the manually controlled reaction control system were employed. The flight on the right was made using the MH-96 adaptive flight control system, with the reaction controls and aerodynamic controls blended into a single controller. For both entries the maximum dynamic pressure was about 1,000 psf and the average angle of attack at entry was about 18°. The most significant difference between the two entries is the magnitude of the angle-of-sideslip oscillation as normal acceleration and dynamic pressure build up. The excursions are much smaller with the MH-96 adaptive system. The maximum normal accelerations were about the same (5g), with the adaptive system providing a somewhat smoother response. In general, the pilots favor the aerodynamic and reaction controls combined into a single controller. On these two flights, both pilots considered the handling qualities to be satisfactory, with a slight deterioration in the lateral-directional mode. However, at higher angles of attack for entry, the adaptive system is clearly superior on the X-15 simulator and will probably be used for all extreme-altitude missions.

Pilot opinion on the use of hold modes during entry appears to be mixed. Such modes can greatly reduce the pilot's concentration and workload, but this can boomerang. For, if a control-system emergency arose, some pilots would prefer to be actively "in the loop" prior to the emergency. Thus, the pilots might prefer to provide their own outer-loop stabilization.

One of the most critical problems that could occur is an augmentation failure during entry. A flight in which such a failure would occur, of course, would not be planned, and normally a single failure cannot produce a complete washout of augmentation. During the early stages of one entry from around 200,000 feet, however, an inadvertent disconnect of the critical roll damper occurred at an angle of attack of 17°. Inasmuch as the flight was being made with the design tail configuration, this angle was well inside the uncontrollable region, as illustrated in figure 19. Although some violent motions occurred, the pilot was able to recognize the problem and push over and reduce the angle of attack as shown, successfully completing the entry. Moreover, the pilot used this opportunity to evaluate the angle-of-attack level that might be controllable in a similar emergency from higher altitudes.

The simulation of entries and the pilot training provided by the fixed-base simulator have been completely adequate. This was predicted in 1958, on the basis of a comprehensive centrifuge program at the Naval Air Development Center, Johnsville, Pa., long before critical flight tests were conducted. All pilots do feel, however, that it is valuable to be exposed to the flight acceleration environment on a centrifuge at some time during a program, if for no other reason than that it contributes to the overall pilot experience.

In summary it can be said that, early in the program, entries from high altitude were anticipated with some misgivings, for here was a situation that combined all the problems of a potentially challenging control task with a difficult environment. X-15 experience has indicated that the entry piloting task is not nearly as formidable as once envisioned—not because the basic characteristics of the airplane were better than predicted, but because of intensive efforts to obtain a configuration and control modes where a single failure would not prove disastrous. The information being obtained from X-15 entries should be particularly useful for application to future lifting entry vehicles, such as the X-20 and lifting-body configurations.

Landing Techniques

Navigation to the landing site is started during the final stages of the X-15 entry. Pilot comments indicate that the Edwards lakebed, which is approximately 11 miles long and 5 miles wide, is easily discernible from as far out as 160 nautical miles at an altitude of 100,000 feet. The pilot is, therefore, in visual contact with his landing area during the entire approach phase. The maneuverability available to the pilot at 100,000 feet and a Mach number of 5 is defined in figure 20. In general, the landing site falls in region A, and the pilots have had little difficulty in managing their energy by using only their cockpit instruments and visual references. The energy-management situation is more critical in region B or C where either maximum maneuvering or a maximum range glide is required to reach the landing site. In such instances, radioed assistance is provided to the pilot based on ground-based energy-management calculations and radar tracking. Although the present methods for energy management are completely adequate for the X-15 under VFR conditions, flight tests will be made of several onboard energy-management systems that will provide research data pertinent to IFR approaches.

The final phase of each flight is, of course, the approach and landing. This area has progressed from one receiving a great deal of concern and attention in the first X-15 flights to routine operation based on the experience, procedures, and techniques developed.

Figure 21 illustrates the wide range in initial conditions that may exist at high key and the flexibility afforded the X-15 pilot in selecting a pattern that enables him to attain the designated touchdown point. The pattern is normally flown at an indicated airspeed of approximately 300 knots, and the handling qualities, including the control-system use and the airplane responses, are considered excellent. If less sink rate is desired, the aircraft can be flown at an indicated airspeed of 240 knots for the optimum lift-drag ratio, and, if necessary, excess altitude can be lost at constant airspeed by use of the speed brakes. Although rates of sink average 250 fps and have been as high as 475 fps prior to landing flare, none of the pilots has considered these values to be a limiting factor in the pattern.

The flare point is chosen so that the energy remaining after the flare will carry the aircraft to the intended touchdown spot. The flare altitude is not selected from the altimeter, but from the pilot's estimate of the height necessary to reduce the sink rate and arrive level in proximity to the ground. It is significant that a rather high flare speed is used to gain more time after the flare in which to make configuration changes, correct trim changes, and then execute the landing at acceptable values of angle of attack, sink rate, and in proximity to the intended landing point. Vertical velocities at touchdown are generally less than 6 fps.

During the last several years, the X-15 pilots have been requested to attempt spot landings to provide statistical data on landing accuracy for future low-lift-drag-ratio vehicles. Figure 22 summarizes the dispersions that have resulted from spot-landing attempts. Recently, the touchdown point has been well within ±1,000 feet of the intended point. Contributing factors to this achievement have been the excellent visibility from the X-15 cockpit, similarly excellent low-speed handling qualities, and the availability of effective speed brakes. Finally, before and during the X-15 flight program, landing simulations were made using the F-104 airplane configured to match the lift-drag ratio of the X-15. This experience has been invaluable and has allowed the pilots to establish geographic checkpoints and key altitudes around the landing pattern; thus, the pilots become familiar with the position and timing required in the pattern imposed by the low lift-drag ratios.

These glowing reports of X-15 approach and landing characteristics may prompt a question about the recent landing accident of X-15-2. This accident was traced to premature failure of the left gear, which, in turn, was the result of a hard landing triggered by a system failure which prevented flap deployment. Thus, the emergency situation was brought about by equipment failure, rather than pilot error.

Now, a few generalizations are in order on what has been learned from the approach and landing experience with the X-15 and other low-lift-drag-ratio gliders. Consider X-15 optimum pattern geometry. For VFR conditions, a general circular pattern has always provided ample pilot control over the touchdown point. Although a straight-in approach would be used in an emergency landing with minimum energy available, there is no reason to use this technique in normal operations. Under IFR conditions, however, with electronic aids the straight-in approach may be the more desirable technique.

In most instances when flying a pattern in a low-lift-drag-ratio vehicle, pilots choose a speed somewhat higher than that for maximum lift-drag ratio to allow a margin for final adjustments following the flare. Figure 23 compares the speed for maximum L/D and that selected as near optimum approach speed by pilots for a number of "gliders" having an $(L/D)_{max}$ of 3 to 4. It is apparent that the pilot generally chooses a speed at least 30 percent higher than that for $(L/D)_{max}$.

Finally, to what level can $(L/D)_{max}$ be reduced and still enable the pilot to complete a successful flared landing. Figure 24 summarizes some pertinent published[3] and unpublished NASA data. Pilot comments indicate that a minimum $(L/D)_{max}$ during the flare of from 2.5 to 3.0 is marginal, regardless of wing loading. Additional flight data are being obtained by the NASA Flight Research Center using lifting bodies for correlation with these results. The initial flight data will be obtained at a wing loading of about 6.5 psf.

Future Plans for the X-15

Basic Program

The original X-15 research program is about 90-percent complete. The remaining flights involve extension of the altitude envelope to perhaps 360,000 feet to 400,000 feet and additional heat-transfer flights. Also, a brief aerodynamic-noise investigation is planned using specially instrumented panels. Most of this program will utilize the X-15-3 airplane.

Follow-On Programs

Numerous experiments are planned which make use of the X-15 as a test bed to obtain heights greater than those reached by balloons, but lower than satellite altitudes. These experiments capitalize on the ability of the X-15 to provide on-the-spot pilot input in the conduct of the experiment and the return of the experiment to the ground for detailed evaluation and adjustment or correction of deficiencies. Some of the experiments will ride "free" in piggyback fashion; others will be grouped to share the cost of operation. Currently approved X-15 follow-on experiments are listed in the following tabulation:

Experiment	Mission
Ultraviolet stellar photography	High altitude
Ultraviolet exhaust plume characteristics[1]	Above 25 miles
Horizon definition	Above 40 miles
Optical-degradation measurements	Varied
Detachable high-temperature leading edges	High heating
Infrared exhaust signature[1]	100,000 feet to 130,000 feet
High-temperature windows	Varied
Atmospheric-density measurements	Above 125,000 feet
Micrometeorite collection	Above 150,000 feet
Advanced integrated data systems, and energy management	Varied
Vapor-cycle cooling	Long-duration zero g

[1]Piggyback package

The future X-15 program will be flexible and will be modified, extended, or terminated on the basis of timely reviews by NASA and Air Force personnel. At present, it would appear that the program will extend at least through 1966.

Concluding Remarks

The X-15 program has kept in proper perspective the role of the pilot in future programs of this nature. It has pointed the way to simplified operational concepts which should provide a high degree of redundancy and increased probability of success in future space missions. But, perhaps most important is the fact that a sizable segment of industry and government engineers and scientists has had to meet the problems of designing and building hardware and making it work. This has provided invaluable experience for the future aeronautical and space endeavors of this country.

Symbols

h	altitude, ft
h_{actual}	actual peak altitude, ft
$h_{planned}$	planned peak altitude, ft
L/D	lift-drag ratio
$(L/D)_{flare}$	maximum lift-drag ratio available during the flare
M	Mach number
q	dynamic pressure, psf
V_i	indicated airspeed, knots
$V_{(L/D)_{max}}$	velocity corresponding to maximum lift-drag ratio, KIAS
V_{actual}	achieved burnout velocity, fps
$V_{planned}$	planned burnout velocity, fps
$V_{approach}$	approach velocity, KIAS
W/S	wing loading, psf
α	angle of attack, deg
θ	pitch attitude, deg

Subscript:

max	maximum

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X-15 AIRPLANE

Figure 1

TYPICAL X-15 RESEARCH MISSIONS

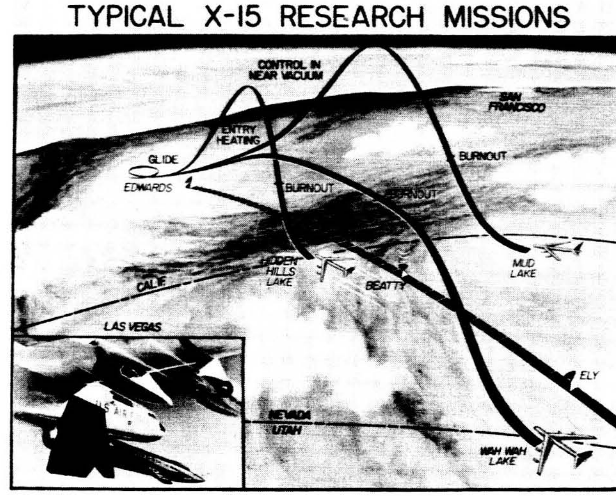


Figure 2

X-15 FLIGHT PROGRESS

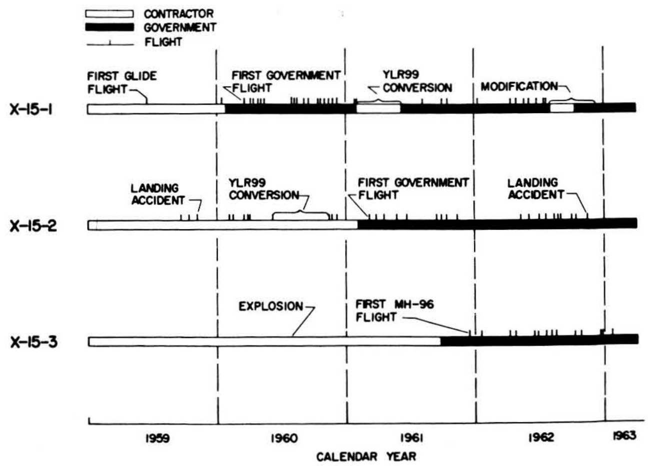


Figure 3

X-15 FLIGHT ENVELOPE

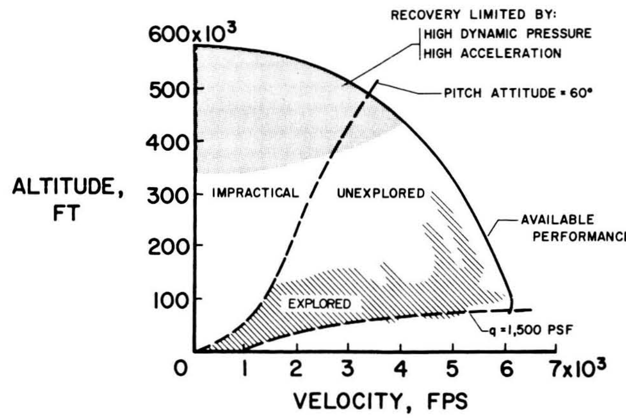


Figure 4

SUMMARY OF X-15 OPERATIONAL FLIGHT EXPERIENCE
JUNE 1959 TO APRIL 1963

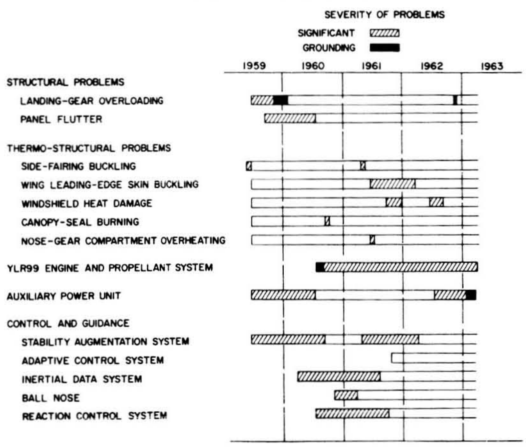


Figure 5

AREAS AFFECTED BY PANEL FLUTTER

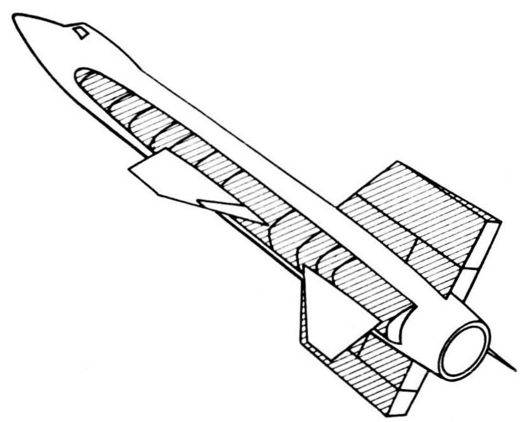


Figure 6

WING SKIN BUCKLE FOLLOWING FLIGHT TO MACH NUMBER OF 5.28

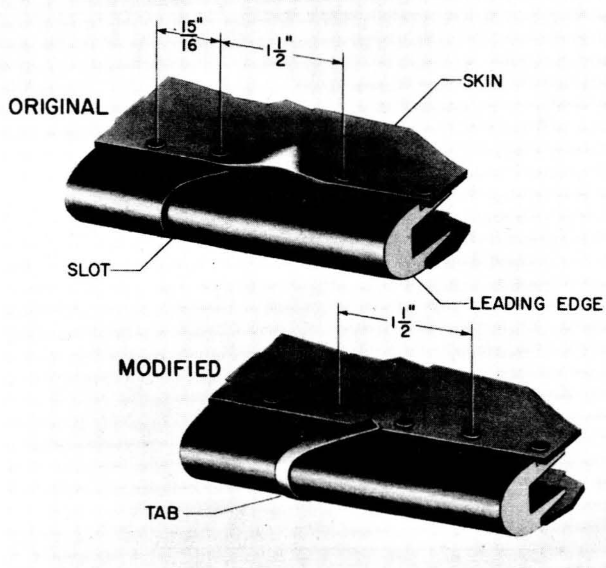


Figure 7

THERMAL PROBLEMS OF WINDSHIELD AND RETAINER

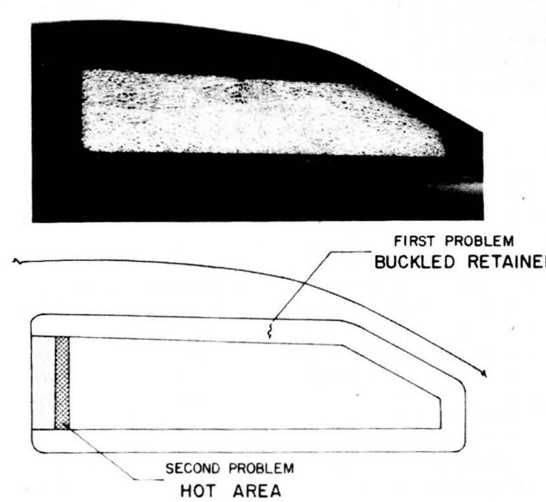


Figure 8

SCHEMATIC OF YLR99 ENGINE SECOND-STAGE IGNITER

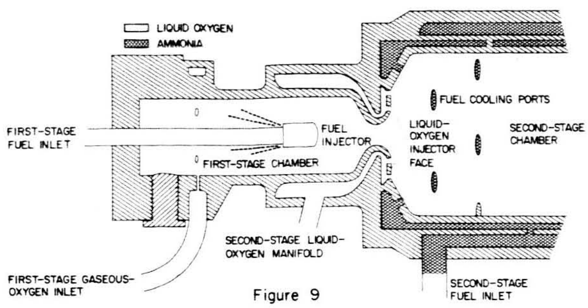


Figure 9

EFFECT OF SAS FILTERS ON LIMIT-CYCLE CHARACTERISTICS

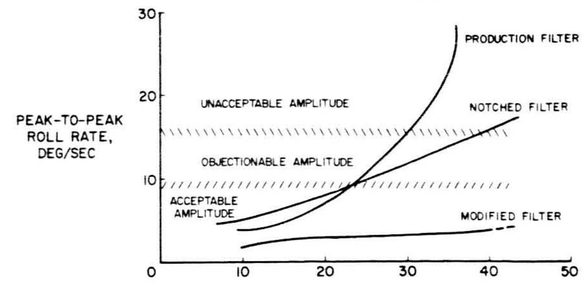


Figure 10

MH-96 ADAPTIVE CONTROL SYSTEM PITCH MODE

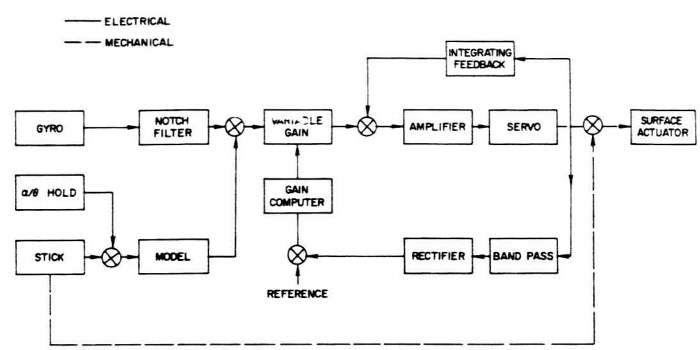


Figure 11

CHARACTERISTICS OF INERTIAL FLIGHT DATA SYSTEM

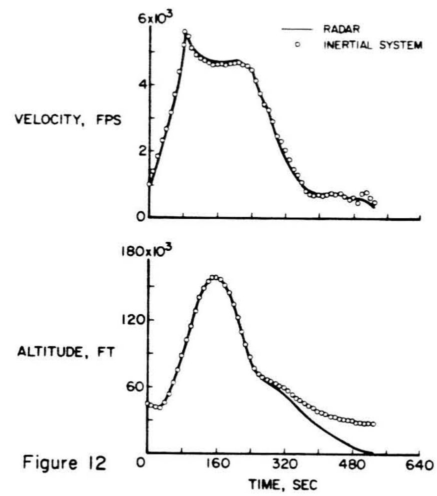
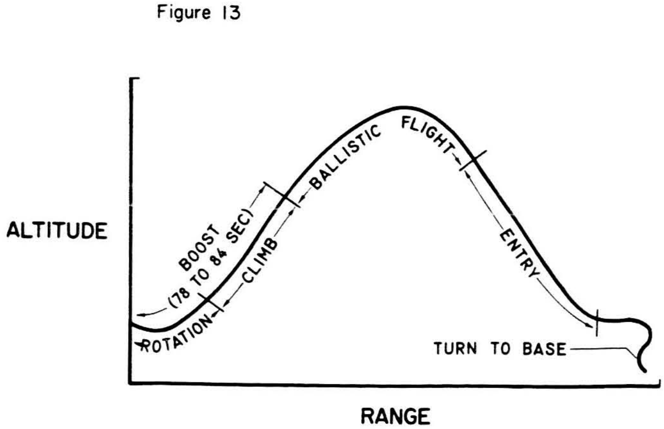


Figure 12

TYPICAL ALTITUDE PROFILE

Figure 13



REPRESENTATIVE X-15 BOOST

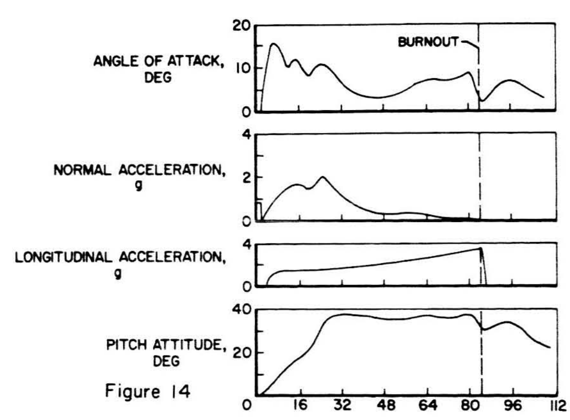


Figure 14

COMPARISON OF ACTUAL AND PLANNED PEAK PERFORMANCE

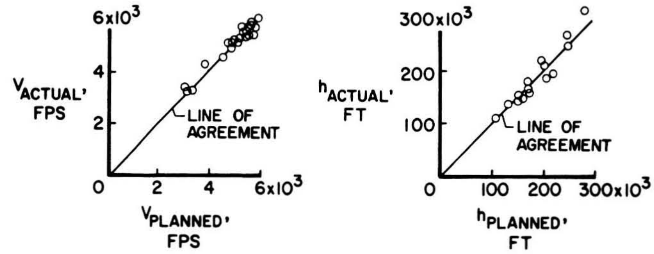


Figure 15

RELATION OF ENTRY TO CONTROLLABILITY BOUNDARY
LOWER RUDDER ON – ROLL DAMPER OFF

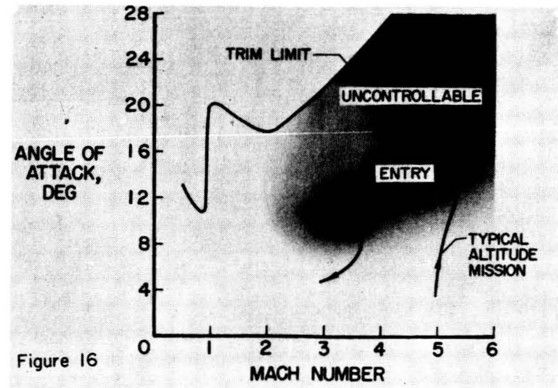


Figure 16

COMPARISON OF X-15 ENTRIES

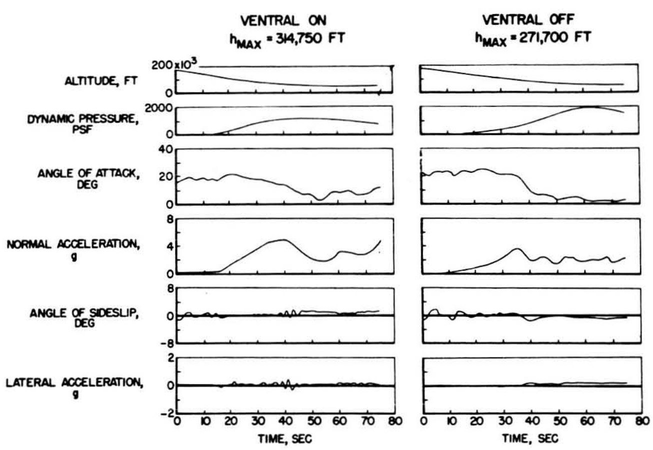


Figure 17

COMPARISON OF X-15 ENTRIES USING SAS AND MH-96

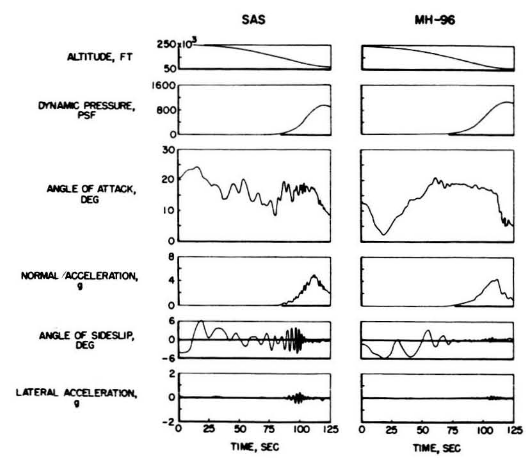


Figure 18

LATERAL-DIRECTIONAL PROBLEM AREA
LOWER RUDDER ON – ROLL DAMPER OFF

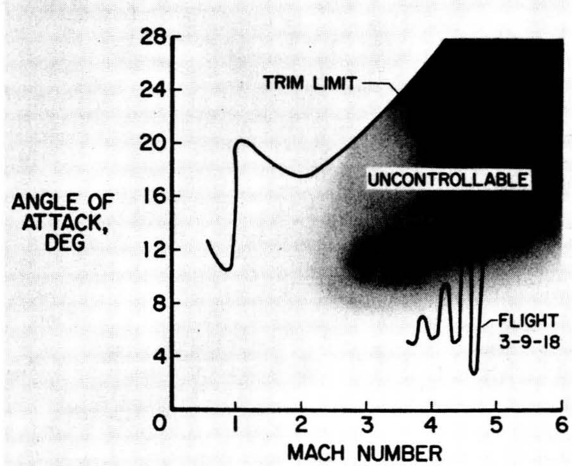


Figure 19

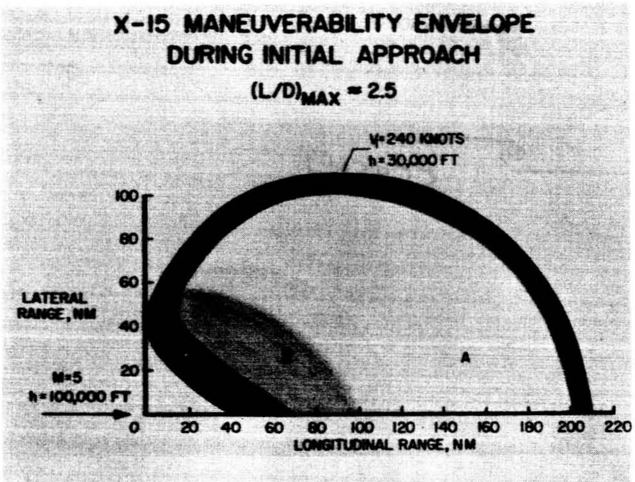


Figure 20

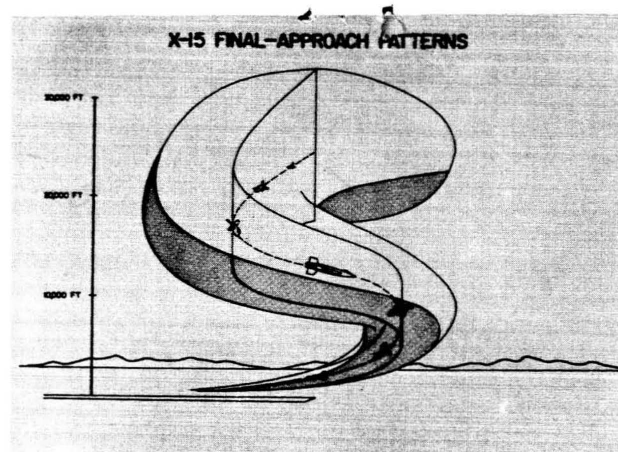


Figure 21

X-15 TOUCHDOWN ACCURACY

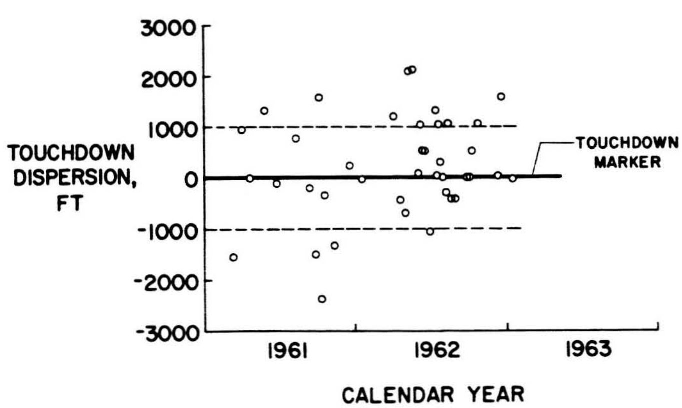


Figure 22

COMPARISON OF APPROACH SPEED AND $(L/D)_{MAX}$ SPEED

$(L/D)_{MAX} = 2.8$ TO 4.5

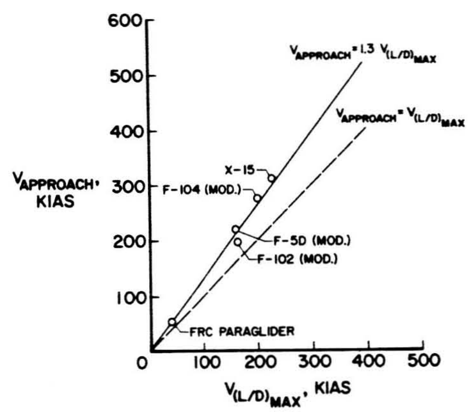


Figure 23

SUGGESTED CRITERIA FOR PILOTED FLARED LANDING

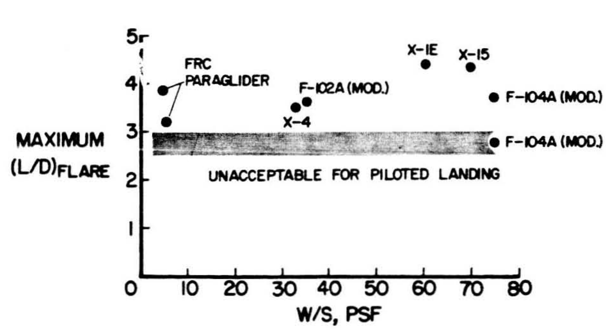


Figure 24

RESEARCH NEEDS FOR ENERGY CONVERSION SYSTEMS

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1.0 Introduction

The ability of man to travel in and explore outer space is determined by his having power sources that can meet minimum requirements for communication and mobility in any given space venture. Manned space missions that are currently in operational or developmental phases are Mercury, Gemini, and Apollo. The power requirements for these missions have been determined, and energy conversion systems to satisfy the mission requirements are under development. The establishment of research needs for energy conversion systems to carry out manned space exploration, should not come from space programs in the operational phase, but should come from projections of planned space programs for the future. These goals, coupled with programs designed to gain further knowledge of physical processes, regardless of their ultimate utilization, will yield the most effective energy conversion research program.

2.0 Review of Current Manned Space Missions Requirements

The basic classification of manned missions in which NASA is currently interested are [1,2] Earth orbit, Lunar and Planetary. In the earth orbit category, the Mercury and Gemini spacecraft are in the operational and developmental stages respectively.

The Mercury spacecraft was designed initially for a single astronaut to make three orbits. This has been extended to six orbits giving 9 hours exposure to near earth space environment, and it will be used in a flight covering 16 to 18 orbits. The Mercury spacecraft carries 144 lbs. of silver-zinc batteries to provide an average of 70 watts while in orbit, with an emergency reserve lasting 24 hours[3]. The peak demand is about 1 kw.

Project Gemini uses a two-man spacecraft and will be used for longer earth orbits than Mercury, including developing the art of rendezvousing and docking in space. The electric power consumption of the Gemini spacecraft is between 500 and 2000 watts. The time duration of a Gemini mission is up to two weeks. One additional requirement placed on the power supply was that it produce a supply of potable water. On the basis of these requirements, an ion exchange membrane hydrogen-oxygen fuel cell was chosen as the electric power source. Comparative studies of total system weight including primary energy source, energy converter, heat rejection system and auxiliary controls, showed that the Gemini fuel cell system was superior to a Ni-Cd battery solar cell system beyond 20 hours operation. The Gemini system is well suited to mission times of 20 to 1000 hours[4].

Project Apollo uses a three-man spacecraft and embraces a number of missions including earth orbit, circumlunar, orbital rendezvous, and lunar landing and return. Power requirements are 1.2 kw with peak power estimated at 4 kw. Mission times extend up to a period of 2 weeks. The power source chosen is the medium temperature and pressure modified Bacon cell using an aqueous electrolyte of potassium hydroxide. The Bacon cell is a hydrogen-oxygen fuel cell and its by-product is water, which is recoverable and useable as potable water for the astronauts.

All the current manned space missions for which hardware is under development have modest power requirements and relatively short mission durations; that is, power requirements of up to a few kilowatts and durations of several weeks. The electric power sources for these missions are either rechargeable batteries or hydrogen-oxygen fuel cells.

2.1 Review of Future Manned Mission Requirements

The present and future missions for space exploration have been outlined by Hyatt[2] and are summarized by him in Table 1. The manned space missions, Mercury, Gemini and Apollo, are operational or in the developmental stage. Energy conversion systems for these missions have been chosen and are in the design, test and evaluation stages. Before 1970 a lunar landing is anticipated and orbiting manned laboratories will be in the planning or operational phases. In the decade 1970 to 1980 a manned lunar station, a Mars landing and various deep space probes are possibilities. Each new undertaking places additional demands upon electric energy conversion systems.

In a lunar station, a permanently installed power station of a megawatt capacity is a possibility and portable power sources for exploration of the lunar surface will be needed. In deep space probes the electrical power source will be needed not only for communication, telemetering and control, but also for powering electric propulsion units. An estimate of electric power requirements versus time has been made by Sanders, et al[5], and is shown in Fig. 1. The estimate is for gross power requirements of over 100 kw by 1974 and over a megawatt when electric propulsion power is used. This high power level coupled with very long mission times of a year or more, points to a nuclear power source as the only solution.

In the cases of manned space flight the electric power system must have major emergency reserve capacity to give the human cargo a high probability of survival in case of a power failure. Typical mission times and survival times are shown in Table 2[6]. In all cases except earth orbits, survival time and mission time are close to the same duration. This points to excessive emergency power or more realistically, extreme reliability of the primary power source. Such reliability can only be obtained by long development and test programs on operational equipment. This precludes the fast utilization of new systems in operational vehicles where reliability is a prime requisite.

2.3 Energy Conversion Systems for Manned Space Flight

Wide varieties of electrical energy conversion systems are needed to meet all requirements of manned space travel. The choice of an energy conversion system is partially determined by the power level and duration of the mission. It is also dependent upon environmental conditions, relative reliability, available source of primary power, use of by-products, heat rejection, etc. A listing of various factors which play a dominant role in determining feasibility of a given system is given in Table 3. To fully evaluate these factors, detailed information of each mission is needed.

Examples of typical power applications are given in Table 4. These are not meant to be all inclusive, but rather to serve as an indication of power ranges and mission requirements which must be served by energy conversion systems. The diversity of power levels, mission times and refueling possibilities rules out any single energy conversion system as being optimum for all possible space power applications.

Studies of the requirements of specific missions were made by Rykar for several manned space flights. Schutte et al[7] have studied the relative advantages of fuel cell systems, solar cell systems, and cryogenic chemical dynamic systems for missions similar to those considered by Rykar. He concludes that solar cell systems give the highest kw/lb, but above 1-2 kw the size of solar collectors and radiators make their use doubtful. The weight of a fuel cell which increases directly with kw demand, presents upper limits on fuel cell systems and gives advantages to chemical dynamic at higher than 5 kw rating when below 1000 kw hr. In the range of 100 kw hr to 1000 kw hr and peak demands of 1 to 5 kw either fuel cells or chemical dynamic may be preferred, depending on the specific application. For higher power levels and longer durations nuclear powered energy converters offer the best, and possibly the only solution.

An auxiliary condition inherent in using a nuclear power system in space, is extreme reliability of the total energy conversion system. In many instances the nuclear power plant will be used where survival times of human cargo are of the same duration as mission times. Absolute reliability and long life of the conversion system is, in such instances, imperative.

While the nuclear power source takes a dominant position in manned space travel, it is not an optimum energy source for all missions. Solar cells plus rechargeable battery systems are feasible where portable power supplies are required. Solar powered conversion systems in the kilowatt range are also feasible in space probes where solar energy is non-cyclic. The upper power limit on solar systems are as yet undetermined, being critically dependent upon fabrication and transportation in space of large area collectors.

The needs of human cargo in manned space flight for water and food, require the transporting of some forms of consumable products. Thus fuel cell systems with by-products of potable water are desirable for medium duration missions and possibly for long duration missions. Similarly biochemical or bioelectric energy sources offer special advantages when integrated with the needs of human cargoes.

A first consideration in limiting the number of applicable energy conversion systems, is the type of primary energy source that can be used for a given mission. Possible primary energy sources and their range of applicability are listed in Table 5. Comparing Table 5 with the power applications listed in Table 4, indicates that the whole range of primary energy sources known, may find application in manned space travel. Considering all classes of needs of manned space flight listed in Table 4, it does not appear possible to materially limit the types of energy conversion systems applicable in special situations to manned space flight.

The only significant conversion systems which the author would eliminate on the basis of his survey of manned space flight requirements, are thermochemical systems consuming hydrocarbons. There is no obvious advantage in having thermochemical or electrochemical systems which substitute hydrocarbon systems for hydrogen-oxygen systems, with the possible exception of bio-cell systems. Hydrogen-oxygen fuel cells seem preferable to the yet undeveloped hydrocarbon fuel cell for manned space application. The same general conclusions appear to apply to thermochemical dynamic systems, and open cycle hydrocarbon consuming MHD systems. For the rest of the direct energy conversion processes and a large class of dynamic systems, there is enough diversity in manned space applications to indicate potential usefulness of all these systems. This does not mean that the development of operational systems on a very broad base is indicated without a specific application. It does, however, indicate that there is no simple formula to restrict fundamental research on processes, merely on the basis of the needs for power sources in manned space missions.

3.0 Status Review and Research Needs Energy Conversion Systems and Devices

The energy converters of long term potential interest in space application include solar cells, thermionic converters, thermoelectric generators, fuel cells, batteries, bio-cells, magnetohydrodynamic generators, and dynamic converters of all types. Extending energy converters to include electrical propulsion engines, adds to this list, electrostatic thrustors (ion engines), electromagnetic thrustors (plasma engines), and electrothermal thrustors (arc engines).

An extensive range of scientific and engineering problems are encountered in covering a field this broad. The state of basic knowledge ranges from the very merger as in the case of biochemical or bio-electrical processes to extreme scientific and engineering sophistication in the well known and highly developed dynamic systems. Any status review and statement of research needs covering such a wide range of scientific and engineering levels of competence is at best sketchy. It is difficult to maintain a balance between established processes and new concepts which are many years from basic understanding, and farther from operational feasibility. To help offset bias by the author, it is recommended that the following status reports and symposia be reviewed by the critical reader.

1. IRE Transactions on Military Electronics, Vol. Mil. 6, No. 1, January 1962, Issue on Direct Energy Conversion Covering Solar Cells, Thermoelectricity, Thermionic Conversion,

Fuel Cells, Batteries, MHD Generators, and Radio Active Isotope Powered Converters.

2. Astronautics, Vol. 7, No. 11, November 1962, Issue on "State of the Art, 1962".

3. Third status report on Fuel Cells, 1 June 1962, No. AD 286686 USASRDL.

4. Status Report on Thermoelectricity NRL, Memorandum Report No. 1361, January 1963.

5. Selected Papers on New Techniques for Energy Conversion, Edited by Sumner N. Levine, Dover Publication, Inc., New York, 1961.

6. Proceedings 16th Annual Power Sources Conference, 22-24 May 1962.

7. Advanced Energy Conversion - Pergamon Press, Symposium on Thermoelectric Energy Conversion
   vol. 1, parts 1-4, 1961
   vol. 2, January-June 1962
Symposium on Thermionic Power Conversion
   vol. 2, July-September 1962
   vol. 2, October-December 1962

8. Programs in Astronautics and Rocketry by ARS, published Academic Press, Vol. 3, Energy Conversion for Space Power 1961, Vol. 4, Space Power Systems, 1961.

9. Space Power Systems Conference, American Rocket Society, Santa Monica, California, September 25-28, 1962.

10. Proceedings of Specialists meeting on Photovoltaic Energy Conversion, IEEE, AIAA, Washington, D.C., April 10-11, 1963.

11. 4th Annual Symposium on Engineering Aspects on MHD, Berkeley, California, April 10-12, 1963.

3.1 Solar Cells

The silicon solar cell is technologically the most advanced of the direct energy converters. They have been used as the power source on space satellites and space probes with satisfactory performance. Silicon solar cells are subject to deterioration in performance when subjected to high energy radiation. On unprotected cells a 25% reduction in output after 10 days exposure to the Van Allen radiation belt has been reported. Studies to minimize radiation damage have included using n-on-p instead of p-on-n structures, base material of oxygen free silicon and various protective coatings. The use of protective coatings has significantly reduced cell deterioration by radiation damage. The efficiency of silicon cells range from 10% to 15%. The efficiency of operational solar cell systems are lower than these figures with 8% being typical for an oriented system and non-oriented system dropping off to under 1%. Typical solar cell systems will run approximately 9 watts per pound.

In addition to silicon, other semiconductor materials with energy gaps between 0.7 ev and 1.5 ev have been studied for solar cell applications. Of these, GaAs, CdS, and CdTe have shown some promise. GaAs was originally proposed as a cell with high radiation resistance but current tests do not bear out this prediction. Thin film cells of CdS are being investigated with the aim of developing flexible arrays which can serve as very large area collectors in space. The best efficiencies of solar cells made from materials other than silicon, range between 2% and 11% and most are not yet operational nor competitive in performance with silicon cells.

The solar cell is today useful and offers continued long term promise in space applications. Continued development of solar cell structures to fit the special needs of space travel and last in the space environment is of major importance. Solar cell research divides into three major classifications: 1) investigation of fundamental properties of materials; 2) device structure to utilize efficiently solar radiation; 3) studies of radiation effects. Research in all areas is necessary.

The dominant material properties in solar cell performance are the energy gap, carrier mobility, carrier lifetime, and absorption coefficient. Fundamental studies of material systems with particular reference to the dependence of electronic properties on composition, crystal structure, lattice defects, temperature, etc., form an important foundation for solar cell research.

In the area of device studies the detailed understanding of the mechanisms of hetrojunctions and graded gap structures is in its infancy. Improved utilization of broad band radiant energy in variable energy gap structure may result from understanding the transport mechanism in such structures. Studies of contact resistance, anti reflective coatings, broad band windows, thin film structures, etc., are also an essential part of solar cell research. The minimization of radiation damage through choosing materials with transport properties less sensitive to radiation, induced defects or the design of structures to shield radiation damage from the junction area, are of vital importance to ensure operational solar cell systems in the space environment.

The solar cell holds such an important place in space travel that a broad spectrum of fundamental and applied research is justified to ensure the development of solar cells with both improved efficiency and longer life in a space environment.

3.2 Thermionic Converters

The major classes of thermionic diodes are the close spaced (0.001 cm) vacuum diode and the cesium diode operated in the space charge neutralizing, plasma, and arc modes. Current research and development work is concentrated on the cesium diode and significant advances in theoretical understanding and device construction have been obtained.

The longest measured life of practical thermionic converters tested so far, is about 2000 hours. The life of most thermionic converters is significantly shorter than this, and in most cases failures are due to the corrosive effects of cesium on structural parts. Eventually it is believed that structural problems will be solved and the evaporation of the emitter will be the factor which determines converter lifetime.

The present state of the art in vacuum thermionic converters indicate that efficiencies of approximately 5%, and power densities 1/2 watt/cm^2 are possible at emitter temperatures of 1500°K with useful emitter lifetime of up to 500 hours. The emitter-collector spacings are of the order of 10 microns and present tremendous problems in the fabrication of devices. Both because of low efficiency and construction difficulties, the vacuum converter is not considered promising, except for very special conditions requiring a heat source below 1500°K and a sink temperature around 1000°K.

The cesium thermionic converter with emitter temperatures of 1600°K to 2000°K has obtained efficiencies of 6% to 16% with power densities of 2 to 14 watts/cm^2.

Laboratory thermionic converters have been built for solar, chemical and nuclear sources. The operating temperature while compatible with solar heating is still somewhat high for nuclear sources and chemical sources. The low efficiencies and high energy density place extreme requirements upon the design of the heat source and method of coupling it to the converter. For space applications the thermionic system promises superior power to weight ratios. This is due to a relatively low converter weight per kw plus the high sink temperature leading to minimum radiator areas for disposal of waste heat in space. Because of its high operating temperatures 1500°C to 2000°C it also offers considerable promise as a topping device to be used in conjunction with conventional turbine generator systems and nuclear sources. The thermionic converter is emerging from the laboratory for use in some space application as evidenced by the SNAP 13 program and the SET program. In the SET system an output of 6.2 watts per pound was obtained.

The areas of research which are indicated by the present status of thermionic converters are: 1) emitter materials and surface properties; 2) transport properties of plasmas; 3) dynamics of arc mode operation; 4) structural and heat transfer design of converters. In most of these areas extensive research activities are currently underway. These include theoretical analyses giving the work function of cesium coated surfaces; observing the plasma conditions in the interelectrode spacing by studying the optical properties of the plasma; and measuring voltage-current curves in a systematic way over wide ranges of emitter temperature, collector temperature interelectrode spacing, and cesium pressure, for a number of different emitter materials. Some of the materials which have been tested are niobium, tantalum, molybdenum, tungsten, rhenium, and iridium.

While impressive gains in the performance of thermionic converters have been made, there is still a great deal to be done in understanding fundamental mechanisms as well as construction of practical converters. There is evidence of a 50% decrease in output due to voltage drops in the interelectrode plasma sheaths. Polycrystalline emitters result in highly non-uniform emission patterns with indications of significant reduction in effective emitter area. Materials of construction and seals are needed which can withstand cesium vapors and the high heat fluxes and temperature gradiants of practical converters. Research in the thermionic area must still embrace fundamental studies of emission and transport mechanisms as well as engineering studies of integrated device and system design, if operational thermionic systems for space application are going to be obtained.

3.3 Thermoelectric Generators

The research and development effort in thermoelectrics has decreased significantly. The massive shotgun type search for new thermoelectric materials with high figures of merit has not yielded results as rapidly as first predicted. Device development programs have shown that in the actual construction of physical generators electrical contacts, material contamination, chemical and structural stability, and mechanical strength, all play a dominant role in determining the ultimate utility of a given material. Segmented generators in which several materials are placed physically in series in a given n or p leg of a thermocouple junction have been built, and couple efficiencies of 11% obtained. While segmentation has advantages from the viewpoint of best utilization of thermoelectric properties over a given temperature range, the problems of electrical contacts, thermal expansion of dissimilar materials, etc. limit the practicality of segmentation. Consequently the only thermoelectric generators that have advanced past the laboratory stage have been composed of homogenous materials in each leg.

The principle generator material used has been the long established PbTe system doped both p and n type. Other materials used include ZnSb, GeTe which are p type and are more stable than p type PbTe. The alloy GeSi developed by R.C.A. has moved into a position of prominence as a generator material, due principally to its ability to operate over wide temperature ranges and remain chemically and mechanically stable. The 500 watt SNAP 10-A system uses GeSi thermoelements and should be tested in a spacecraft by 1966. Other thermoelectric generators in the SNAP series include SNAP 3, 7A, 7B, 7C, 7D, 7F, 9A, and 11. Thermoelectric generators have been used with solar concentrators and large area solar thermoelectric conversion panels have been built.

Thermoelectric generators have relatively low efficiencies. Thermocouple efficiency of approximately 10% and overall efficiencies of 5% to 6% when used with a radioactive isotope heat source or 2% to 3% with a combustible fuel heat source. The advantage of thermoelectric generators with a nuclear heat source is their long life. More than two years of continuous operation has been obtained with a SNAP thermoelectric generator. No other energy conversion system is today in a state of development that can exceed nuclear powered thermoelectric generators for long term, reliable, maintenance free, unattended operation. This factor alone makes thermoelectric generators an important part of the manned space power program.

The research and development program in thermoelectricity should consist of two main parts: 1) studies of materials and transport mechanisms; 2) design and fabrication of thermoelectric structures to meet operational conditions using existing materials.

Further advances in thermoelectric materials require detailed studies of electronic transport mechanisms and galvano-magnetic effects in semiconductors and semimetals. The most useful thermoelectric materials are all compound semiconductors and semimetals used individually or as solid solutions of several systems. Binary, ternary and quartanary systems have been found to be good thermoelectric materials. The band structure of these materials is known in only the meagerest form. The Hall mobility measured in Bi_2Te_3, the most studied thermoelectric material, does not correlate with the proposed energy band model. Improvements in figure of merit by a factor of two have been observed in the BiSb system at liquid nitrogen temperatures, but theoretical explanations are not yet available. Long term detailed studies of the electronic properties of thermoelectric materials are necessary to lay the foundation for further advances.

An all inclusive list of material systems to study is hard to establish, and it is not evident that all promising materials should be investigated in depth. Any fundamental study should certainly include some of the best known thermoelectric materials as PbTe, Bi_2Te_3, ZnSb, BiSb, GeTe, and GeSi, if increased insight into thermoelectric processes are to be obtained.

In the device area, continued work on junction fabrication, material degradation with time and temperature, fabricating arrays of couples, physical and thermal shock resistance of materials, design of heat transfer structures, etc., are all necessary to develop operational systems. These studies are best conducted in the context of specific applications as is currently being done in the SNAP 7 and 10 programs. Devices studied must be based on today's known thermoelectric materials. The effect on devices of new materials which may come from fundamental studies cannot be realistically predicted, but the potential value of such materials in the space program justifies significant fundamental research on the electronic and physical properties of thermoelectric materials.

3.4 MHD Power Generation

Progress is being made toward the goal of making MHD power generation practical, but much remains to be done.

Nuclear reactors cannot, at present, operate at temperatures at which equilibrium electrical conductivity is sufficient for efficient MHD power generation. In order to use a nuclear reactor as the primary energy source, either for central station power generation or for power generation in space, non-equilibrium ionization has been proposed. This involves the use of a gas mixture (usually a noble gas with alkali metal seed) in which the temperature of electrons can be maintained well above the gas temperature with little expenditure of power. When this is achieved, the electron density and electrical conductivity are increased enough to make application to MHD power generation attractive.

For central station power generation with fossil fuels, the major problems are still associated with the high temperatures necessary for acceptable electrical conductivity and the attendant materials problems that result. One approach has been to use oxygen enrichment to lower the preheat temperature to that obtainable with readily-available, economic, heat-exchanger materials. It has been shown that the addition of an oxygen plant increases the capital cost slightly and reduces the plant efficiency but not by a large enough factor to make this approach unreasonable.

The use of MHD power generation in space applications is limited to relatively large power installations, a megawatt and upwards. The MHD system that seems most feasible for space power is a closed cycle whose working fluid is either a seeded noble gas or an alki metal, using a nuclear power heat source. The relatively high exhaust temperature (2500-3000°F) of MHD systems minimize the radiator size necessary to dispose of the waste heat in space. Studies have shown that because of the T^4 factor in the radiation disposal of heat there is a gain in overall output power per pound of system weight by using lower temperature differentials between heat source and sink. Considering the radiator weight plus generator weight an ideal carnot engine should operate at 3/4 of maximum carnot efficiency to minimize total system weight. This results from a trade between increased generator size due to reduced carnot efficiency and reduced size of radiator. The radiant disposal of heat desiring a high exhaust temperature matches the inherent characteristics of the MHD power generator very well since the minimum temperature is determined by the useful conductivity of the gas. The MHD generator is, therefore, well matched to one of the major limitations of a space environment.

Superconducting magnet coils are essential for MHD power generation because of the high fields obtainable and the low power input required. Work is progressing both in the area of materials development and in the area of fabrication technology. At present it appears that fields of 10-30 kilogausses are obtainable with negligible power requirements for large generators.

Present experimental work is directed toward a better understanding of the fluid mechanics of MHD generators and toward reliable and long-lived channel construction. Characteristic results to date are the production of 1500 kilowatts of electric power in a combustion-fired generator that operates for roughly ten seconds, and the successful testing of an MHD channel design for 140 hours at gas temperatures of 5000°F.

The research in MHD of specific interest to space power is primarily in the following areas: 1) high temperature nuclear heat sources; 2) transport processes under non-equilibrium conditions; 3) superconducting magnet design including insulation; 4) MHD boundary layer phenomena. The development of the MHD generator for space application still contains many areas of basic research in addition to many engineering development programs. Its long term usefulness in space justifies continued research on this system but the state of the art today precludes operational systems in the near future.

3.5 Dynamic Systems

The dynamic systems, which include a thermal mechanical engine and an electromechanical generator, have a wealth of operational data on which to draw. The heat engines may be either open or closed cycle systems. The open systems use chemical fuels and are most attractive for short missions where the working fluid may be used as a heat sink for cooling. As a rule of thumb 500 kw hr is the transition between open and closed cycle dynamic systems.

The open cycles investigated have used working fluids of hydrazine, hydrogen-oxygen and hydrogen. The hydrogen-oxygen system can be used in a number of cycles which include reheat and regeneration and which are integrated with the cooling, requirements of the flight vehicles. Upper temperature limits of operation, due primarily to materials, are around 2000°F, however, when the design of a conversion system is integrated with the vehicles cooling requirements, a much lower temperature may be desired as in the cryocycle which use cryogenic super critical hydrogen heated by the waste heat from the spacecraft and then expanded through several cycles in a gas turbine system.

In closed cycle systems for space application, the primary source of energy is either solar or nuclear energy. The working cycles considered have been Rankine or Stirling cycles, using working fluids of Mercury, Rubidium, Cesium, Potasium, Indium, Helium, Lithium, Bismuth, etc. The mercury system is applicable up to 1600°F and the alkali metals cover the temperature range of 1500°F to 2600°F. The higher temperature cycles are fundamentally limited by materials. The Rankine (condensible vapor) cycles with metal vapors present difficulties with condensing and boiling in a zero gravity environment plus severe erosion problems. The Brayton single phase gas cycle has been investigated as an alternate to Rankine cycles for the space environment. Studies indicate that overall system weight including radiator is larger in a gas cycle system because of the variable exhaust temperature. For the same maximum temperature a gas cycle has four times the size radiator as a Rankine cycle. Considering the higher operating temperatures possible with gas cycles radiator sizes are approximately twice those of Rankine cycles. The compressors, turbines, and bearings in gas cycles are more reliable. The gas cycle may prove advantageous for nuclear MHD closed cycle systems particularly with a helium working fluid which simplifies the materials problem at high temperatures.

In terms of non-space applications reliable open cycle systems are available in a large range of ratings, and closed cycle regenerative Brayton systems of megawatt capacity have been operated reliably for over a decade. In the high power range 300 kw to 1 mw the joint SNAP 50/spur program is laying the ground work for operation in a space environment. For high power space application the dynamic systems are prime contenders as the feasible power source.

The requirements for research in the dynamic systems are quite different than most of the direct conversion systems. The scientific principles of the dynamic converters are well known. The problems areas include: 1) materials for higher temperature operation; 2) properties of new working fluids; 3) development of cycles coupled with environment control of the entire spacecraft; 4) development of bearing, seals; etc. for use in a space environment; 5) test and reliability data proving feasibility of long term maintenance free operation in a space environment.

The research needs for dynamic systems other than in materials, are hardware development programs in which specific systems are constructed and tested until all the weak links are eliminated. A dynamic system is fundamentally an engine and must follow a development and test pattern analogous to that of aircraft piston and jet engines. In aircraft the engine development has always been separate from the airframe development. The similarities between engines and dynamic converters indicates that the preferred approach is to develop standard dynamic systems which will be used in a wide range of space missions. To tie development programs on dynamic systems to specific missions will make it both difficult and wasteful to establish the long term test programs needed to develop reliable dynamic power systems.

3.6 Fuel Cells

The hydrogen-oxygen system is still the most successful reagent combination for low temperature fuel cell use. The Gemini spacecraft operates on a hydrogen-oxygen cell equipped with an organic cationic ion exchange membrane as electrolyte. Interest in different types of solid ion exchange systems as electrolytes is high. The Apollo spacecraft operates with a Bacon type hydrogen-oxygen cell, using an aqueous electrolyte at relatively high temperature and pressure and dual porosity electrodes.

Other low temperature cell systems are under study. Some interesting results have been reported for hydrazine and methanol cells. Hydrocarbon cells continue to be the subject of intensive effort, but despite rumors of success, no low temperature cell system working directly on hydrocarbons and showing reasonable performance has been announced. Work is also being done on low temperature reforming of hydrocarbons to a mixture of hydrogen and carbon oxides for subsequent use in a low temperature fuel cell system. A key problem here is the development of a suitable reforming catalyst. Also under study are two types of high temperature cells for burning hydrocarbons, namely those using solid electrolytes and those using molten salts as electrolytes. Calcium zirconate in massive or film form shows promise as a solid electrolyte, but temperatures in excess of 1000°C are required in order to secure reasonable mobility of oxygen ions in the zirconate lattice. Molten salt systems operate at lower temperatures, generally around 500-700°C. The salts used are usually carbonates, through which oxygen travels as a carbonate ion. To simplify the corrosion and containment problems encountered with such molten salts, there is currently much interest in electrolyte pastes made by incorporating a finely divided solid like magnesia into the melt.

To minimize the weight and volume in any fuel cell the electrode current densities at the desired operating voltage must be maximized and the plate spacing must be minimized. The electrode current densities are determined by transport and ionization processes at the electrodes. The following steps occurring in series are generally involved: 1) Transport of reagent to the electrode. If a gas, then transport through the gas phase, followed by solution in the electrode and diffusion to the electrode surface is involved. If a porous gas diffusion type electrode is used, then this transport is through the electrode pores. 2) Chemisorption on the electrode surface. 3) Occurrence of the electrochemical reaction on the electrode surface, involving electron transfer. 4) Travel of the ion product to or away from the electrode. 5) Transport of other reaction products away from the electrodes, countercurrent to the flow of reagents. Anyone of these steps can limit the operation of an electrode. For hydrocarbon fuels the electrochemical reaction at low temperatures is the rate-limiting step. In the hydrogen-oxygen cell, both the transport steps and the electrochemical reaction rate determine the performance of present cells.

Continued improvement in fuel cells requires fundamental studies of the various transport processes in both solid and liquid electrolytes, and investigation of electrochemical catalysis. Extensive studies of this nature are currently underway.

In the case of hydrogen-oxygen fuel cells operational systems are under development. These programs will establish valuable data on total fuel cell system operation in a space environment. While hydrogen-oxygen fuel cells are becoming operational there are many steps in the process which are not well understood. System development programs must not be allowed to replace or eliminate research programs on specific processes in fuel cells. Continued basic research on all limiting transport and reaction processes are essential to further improvement in fuel cell systems.

3.7 Batteries

The battery is the oldest of the electric power sources. It has the advantage of being an integrated energy storage and conversion device and as such has very wide usefulness in space power application.

Batteries are generally classified in two groups as 1) primary batteries, 2) secondary batteries. Primary batteries involve irreversible electrochemical reactions in which the reagents are consumed without replacement. Secondary batteries are designed to operate on reversible electrochemical reactions, so that the system can be restored to the charged state by reversing the direction of current flow. Regenerative fuel cells, in which the products of the cell are converted to the original fuel by thermal energy, also deserved mention here.

For space power the application of primary batteries will be on relatively short term missions or as reserve emergency power. The use of secondary batteries and regenerative fuel cells can cover a wide range of applications from supplying the total energy demand in solar powered system during periods of no solar radiation to a wide range of special purpose and emergency power application. Batteries have such diversity of use that on almost all manned as well as unmanned space missions batteries will find some range of application.

In terms of research the battery is in the technological state of development where the limiting problems are those of design and fabrication to obtain very long life, under a variety of conditions such as depth of discharge, number of discharge cycles, temperatures, sealed construction, shock and vibration, etc. The program needed to improve batteries is primarily that of hardware development and test. The needs for fundamental research is in detailed investigation of mechanisms much as outlined for fuel cells. Such programs should be carried on with the results from the research being used for future systems rather than radical modification of existing systems. Battery research and development programs should, therefore, follow the general pattern used for any highly developed operational device or system.

3.8 Bio-cells

The bio-cell under study today are is of two general types: 1) bio-chemical process yielding chemical by-products which serve as fuels and 2) direct bio-electric conversion. In both cases the unique feature is that the electrode processes are promoted or catalyzed by bio-chemical agents rather than by conventional chemical catalysts. These bio-chemical catalysts are fundamentally the enzymes, which are introduced into a cell either by living micro-organisms or are added directly as crude extracts made from micro-organisms. The exact form of the enzymes used depends upon the type of reactions to be operated. The primary fuels for bio-cells can be a wide variety of relatively inert materials such a vegetable products, human waste and petroleum, while ultimate oxidants will be oxygen. Photo synthesis may also be included in the bio-cell process in some instances.

The power densities from bio-cells are low and the temperature range is primarily that of life on the earth's surface. The use of bio-cells as primary power sources seems doubtful, but they offer potential on very long term manned space missions for combined use as special power sources integrated with environmental control.

The state of the art in bio-cells is many years from operational systems. The research needed is fundamental studies of bio-chemical and bio-electric processes. The design and test of actual systems today is premature and should be made secondary to all fundamental research which will give increased understanding of the mechanisms in bio-cell energy converters.

4.0 Research Planning

Research and development programs on energy conversion processes can be subdivided into four major steps.

a. The understanding of fundamental processes

b. The development of materials to implement item (a)

c. The conception and fabrication of systems to use items (a) and (b) efficiently

d. The development and test of hardware to obtain performance characteristics, life, and reliability data under both laboratory and actual operating conditions.

The progression from fundamental research items (a) and (b) through development and fabrication of operational systems, items (c) and (d), is seldom a linear time sequence. In some cases an understanding of the science of fundamental processes lags significantly the fabrication of a working device, while in others the knowledge of fundamental processes leads to extensive programs to develop a working energy conversion system. Thermoelectric and biological conversion are examples of devices preceeding a knowledge of fundamental processes while the MHD generator and thermionic converter is an example of a knowledge of processes preceeding the development of devices. Regardless of the sequence of events leading up to the development of an operational system there is always an interplay between hardware development work and fundamental research. Just as in scientific research where experimental and theoretical studies supplement and complement each other, and both must go on simultaneously, so must there be some overlap between development work and fundamental research on energy conversion systems. The exact amount of overlap is determined by many factors and a quantative answer to the optimum sequence is difficult to establish.

Some guide lines of the time sequence between research and development and operational systems can be deduced from the historical evolution of various energy conversion systems. Table 6 presents a historical development of four direct energy converters. Several factors are apparent from the Tables. 1) The fundamental concepts necessary to establish that an energy conversion process exists is usually very old dating back to the 1800's. 2) The establishment of practical feasibility results from additional concepts about details of internal mechanisms and often these concepts come from research on other systems. 3) The time delay between laboratory working devices and a knowledge of those factors establishing feasibility has shortened materially in the last decades. 4) Fully operational systems lag a nûmber of years behind a laboratory model with five to ten years being a representative time depending upon the complexity of the equipment and the reliability desired.

Fully operational systems, have not been developed for all the direct conversion processes shown in Table 6. The most significant conclusions from those data for planning research on energy conversion systems is: 1) the acceleration that has occurred in developing laboratory models of energy converters once basic processes are conceived, and 2) the significant time lag that still exists between laboratory models and operational system. The lapse of five to ten years between conception plus laboratory feasibility studies and a full scale, tested operational system is apparently a real time lag even when extensive programs to develop specific devices are supported. This natural time delay must not be forgotten in anticipating research results or starting research programs that will be useable in future operational systems.

5.0 Conclusions

The operational requirements of manned space flight place extreme demands upon energy conversion systems. These include long flight times, high operational power requirements, large power and long operating times for emergency systems, a wide diversity of environmental conditions and a broad spectrum of operational specifications ranging from a small communication system to a spacecraft main propulsion system or a large lunar based main power plant.

The diversity and complexity of the operational and environmental conditions of manned space travel minimize the probability of signifcantly shortening the time between initial research results and operational systems. This points strongly to the conclusion that energy conversion systems for manned space travel during the next decade must come from conversion processes whose basic fundamentals are known today, and in which working laboratory models exist.

A second conclusion based on the review of manned space flight requirements and the present status of energy conversion systems is that the power requirements needed for manned space exploration in the next two to three decades cannot be met with existing systems. It has been shown that a lead time of one to two decades exists between the initial conception of an energy conversion process, the fabrication of a laboratory prototype, and its ultimate development into a tested fully reliable operational system. Such being the case it is imperative to actively support fundamental research in energy conversion processes to supply concepts for the decades ahead.

If the fundamental research is neglected, future decades will see research and development of specific processes conducted simultaneously, with a resultant loss in efficiency, increase in cost, and high probability of not having at any cost the energy conversion systems needed for manned space flight from 1980-onward. The necessary fundamental research may be conducted by studies to increase the knowledge of processes involved in specific energy conversion systems. It can also be conducted as fundamental studies of limitating processes occurring in existing systems such as boundary layer phenomena, emission characteristics of materials, temperature and physical limitation of materials, thermodynamics of cycles, basic transport phenomena in liquids solids and gases, etc. The needed concepts may also come from totally independent research not oriented toward energy conversion systems such as biochemistry, intersteller plasma physics, etc. The necessary fundamental research need not be tied to hardware development or mission requirements. In fact to do so will only increase its expense and delay its results.

The choice of power supply is very much determined by specific mission requirements and environmental conditions. To carry out meaningful studies of a given mission or type of mission specific devices and systems must be used. The results of such studies can indicate which of several systems, whose design parameters are known, would be most promising for a specific mission. Mission oriented research, however, by its very preoccupation with specific needs for operating systems is the worst possible platform from which to plan fundamental research.

It is equally dangerous to review the present status of energy conversion systems and "blue sky" which systems will be the most significant based on operating characteristics which have yet to be obtained in the laboratory. The research analysist who anticipates break throughs by surveying the state of the art and tries to orient research accordingly seldom makes a wise choice of research programs. The problem areas in known energy conversion processes can be pinpointed and sound research programs planned. The danger in research planning is to predict needed "break throughs" and try to force their occurrance by large expenditures on crash programs. This particular disease, which today inflicts many research programs, seems to have been caught from hardware development programs where such an approach has some justification. The result is short term research which changes every three to five years with each change in mission requirement. The crash type, mission oriented, combined fundamental research, and hardware development program is an enormous sink of talent and dollars with significant results seldom forthcoming. Energy conversion research by its very nature is an applied science, as is all device research, and is particularly vulnerable to this mixing of research and hardware goals.

Fundamental research programs cannot replace the hardware development programs needed to supply operational systems in the next decade. Without extensive and expensive development programs, reliable tested operational systems will not be available for tomorrows manned spacecraft. Such hardware programs, however, must be based on known operating processes and not on extrapolated results of unproven processes.

The funding of energy conversion programs has historically evolved from hardware needs. Two major factors indicate that a change in philosophy is necessary. One is that energy conversion has extended its scope into processes whose basic mechanisms are not fully understood. The other is that long term projected applications for manned space flight require energy conversion systems not obtainable by extrapolation of existing systems. The solution is a recognization of energy conversion as a science whose research results are needed today as a contribution to knowledge, without reference to any specific mission. The accumulated results of this research when assimilated and understood will be the foundation on which future energy conversion systems can be based. Then and only then can sound hardware development programs with specific mission goals be planned and the very real problems of operational characteristics, life and reliability be faced and solved.

Long term fundamental research programs on basic mechanisms and processes as yet not understood and hardware development programs converting known processes into hardware is the program needed to meet both the ten year and thirty year goals of energy conversion systems in manned space flight.

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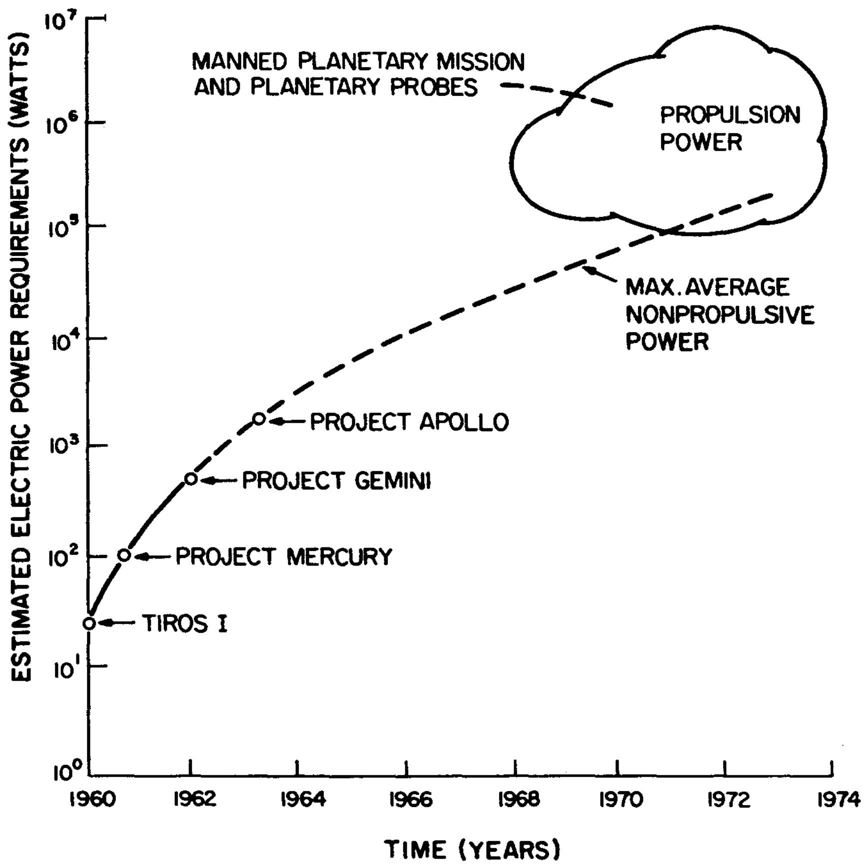

Fig.1 Anticipated Nonpropulsive and Propulsive Space - Power Requirements

TABLE 1

SOME SPACE EXPLORATION POSSIBILITIES

MISSIONS	EARTH ORBIT	LUNAR	PLANETARY
UNMANNED	1958- 1 UNMANNED SATELLITES Scientific Satellites •Small Special Purposes •Orbiting Observatories Application Satellites •Communication •Meteorology •Navigation Engineering Research	1962-68 2 LUNAR PROBES •Ranger •Surveyor Intermediate Space Probes	1962- 3 DEEP SPACE PROBES •Mariner Voyager Search for Extra-Terrestial Life Out of Ecliptic Gravitational Experiment Outer Planets and Their Satellites Leave Solar System
MANNED DEVELOP-MENTAL (• AUTHORIZED PROGRAMS) ↑	1962-68 4 MANNED SATELLITES Ballistic Reentry •Mercury •Gemini Maneuvering Reentry Interim Orbital Labs.	Before 1970 5 MANNED LANDING •Apollo Lunar Logistic System UNMANNED	After 1975 6 MANNED EXPEDITIONS Mars Landing Venus Reconnaissance Search for Life on Planets
↓ (NO AUTHORIZED PROGRAMS YET) MANNED OPERATIONAL	After 1968 7 ORBITAL OPERATION Manned Orbiting Labs. Operational Ferry Vehicle Recoverable Boosters Engineering Experiment And Development	After 1970 8 LUNAR STATION Scientific Observations Lunar Explorations	After 1980 9 PLANETARY OPERATIONS Mars Station Advanced Manned Expeditions Jupiter Satellites Mercury and Others

TABLE 2

MISSION DURATION AND TIME TO SURVIVAL

MISSION	DURATION	TIME TO SURVIVAL*
Earth orbit	1 day	30 minutes
Space laboratory	6 to 12 months	30 minutes
Circumlunar	5 days	3½ days
Lunar landing	6½ to 14 days	3½ days
Venus-Mars fly-by	1 year	1 year or less

(*) Time to survival is defined as time required to return to surface of earth without regard to geographic location of earth landing point.

TABLE 3

FACTORS AFFECTING ENERGY CONVERSION SYSTEMS FOR MANNED SPACE FLIGHT

THE PERFORMANCE REQUIREMENTS

power demand
primary power for propulsion, auxiliary power in main spacecraft, power for special missions either portable or installed at fixed locations

COMPATABILITY WITH THE SPECIAL ENVIRONMENT OF SPACE AND LUNAR OR OTHER PLANETARY SURFACES

high vacuums
irradiation from high energy particles
meteoroids, micrometeoroids, and cosmic waste
radiant disposal of waste heat
temperature ranges
lunar or planetary surface conditions
solar energy density and its periods of availability

TIME AND RELIABILITY OF OPERATIONAL PHASES

minimum time and power for emergency conditions and survival
short term exploration from main spacecraft
long term reliability of fixed installation or main spacecraft power
availability or absence of equipment maintenance or repair
absolute reliability of power source for human survival

UTILIZATION OF BY-PRODUCTS OR SPECIAL FEATURES OF POWER SOURCES

by-products of value to sustain life
cyrogenic system as heat sink or environmental control

TABLE 4

TYPICAL POWER APPLICATIONS FOR MANNED SPACE TRAVEL

SPACECRAFT AUXILIARY POWER

10^2 to 10^5 watts for communications, control and environment
mission times from 1 to 10^2 days -- no refueling

ORBITING SPACE PLATFORMS

10^3 to 10^6 watts for communication, control, experimental laboratories and environment
mission times from 10^2 to 10^4 + days -- refueling possible

SINGLE ASTRONAUT PORTABLE POWER SUPPLIES

1 to 10 watts for communication
10 to 10^3 watts for special experimental apparatus
mission times 0.1 to 10 days -- refueling or recharging normal

SPACECRAFT AUXILIARY AND ELECTRIC PROPULSION POWER

10^5 to 10^8 watts for communication, control, propulsion and environment
mission times from 10 to 10^3 days -- no refueling

LUNAR BASED POWER PLANTS

10^6 to 10^{10} + watts for central power station at fixed installation
mission times from 10^3 to 10^4 + days -- refueling possible

ASTRONAUT EXPLORATORY TYPE VEHICLES OPERATING FROM MAIN SPACECRAFT

10^2 to 10^5 watts for communication, propulsion, and experimental apparatus
mission times 0.1 to 10 + days -- refueling or recharging possible

TABLE 5

PRIMARY ENERGY SOURCES FOR MANNED SPACE FLIGHT

NUCLEAR FISSION ENERGY SOURCE

high total energy demand
- low power, long operating life
- high power, short to long operating life

relatively independent of space environment

CHEMICAL ENERGY SOURCES

low to medium total energy demand
- low power, long duration, high power, short duration or refuelable

relatively indepependent of space environment
dependent upon rechargeable system or availability of transportable fuels
by-products such as water useful to human cargo

SOLAR ENERGY SOURCE

high total energy demand
- low to medium power, long operating life
- maximum power set by collector area

strongly dependent upon space environment
- radiation damage
- cyclic availability of solar radiation

BIOLOGICAL ENERGY SOURCES

low to medium energy demand
- very low power

possibly dependent upon space environment
- solar energy needed for photo synthesis processes

human waste one possible fuel supply
by-products useful to human cargo

TABLE 6a - HISTORICAL DEVELOPMENT
SOLAR CELLS

Fundamental Concepts	Additional New Concepts Needed For Application	Laboratory Models	Operational Systems
1834 Faraday negative coefficient of resistance in silver sulphide	1910-1930 General Studies of semiconductors	1904 J.C. Bose Cats Whisker r.f. rectifier using galena and silicon	1958 Vangard Earth Satellite
1839 Becqueral photovoltaic effect in electrolytes	1927 L.O. Grondahl P.H. Geiger Copper Oxide Rectifiers	1930 W. Shottky Cuprous Oxide photoelectric cells	
1873 W. Smith photo conduction in selenium	1940-1950 Extensive work on crystal preparation of silicon and germanium, Bell Telephone, Labs, and Purdue University	1948 W. Shockley J. Bardean W.H. Brattin Transistor	
1874 F. Brown Rectification in lead sulphide and iron pyrite	1955-1963 Extensive work on III-V compounds	1954 D.M. Chapin C.S. Fuller G.L. Pearson p-n junction solar cell	
1879 E.H. Hall Hall Effect			
1928 Sommerfield Quantum Mechanical Treatment of Electrical Conduction in Metals			
1931 A.H. Wilson Quantum Theory of Electron Motion in Semiconductors			

TABLE 6b - HISTORICAL DEVELOPMENT FUEL CELLS

Fundamental Concepts	Additional New Concepts Needed for Application	Laboratory Models	Operational Systems
1805 Davy's low temp. cell, using solid C as fuel O_2, and Pt electrodes in nitric acid	1880 Gas diffusion type electrode	1930 Railway signal battery (air breathing cathode, replaceable zinc anode	1930 Railway Signal batteries
1839 Croves first H_2-O_2 cell using Pt electrodes in sulfuric acid	1900 Rasch, solid electrolyte with gaseous reagents	1952 Bacon H_2-O_2 fuel cell	1964 Gemini Fuel Cell
	1930 Union carbide porous carbon electrodes	1954 Grubb, ion exchange fuel cell	1966 Apollo Fuel Cell
	1937 Bauer & Preis solid zirconium yttrinum membranes	1957 Calcium zirconate membrane cell	
	1947 Davtyan, impregnated porous ceramic discs		
	1954 Grubb, ion exchange membranes		

TABLE 6c - HISTORICAL DEVELOPMENT
THERMIONIC CONVERTERS

Fundamental Concepts	Additional New Concepts Needed For Application	Laboratory Models	Operational Systems
1884 T.A. Edison Thermionic Emission	1950 "L" Cathodes or dispensor cathodes	1958 Hernquist-et al space change neutralization with cesium	No operational systems yet
1901 O.W. Richardson Richardson- Dushman Equation	1951 Champeix qualitive dis- cussion of con- verter - ruled impractical	1958 V.C. Wilson Cs plasma diode	
1915 S.W. Schlicter Statement of basic concept of thermionic converter	1956 Hatsopoulos vacuum diodes and crossed E and H field triode	1958 G.N. Hatsopoulos J. Kaye vacuum diode	
1923 I. Langmuir Diode space charge	1957 Moss-analysis of planar diode	1958 Grover, Pidd etal fissionable emitters	
1933 J.B. Taylor I. Langmuir Emission from W in presence of Cs		1959-1963 Extensive research on emitters, cesium plasma and materials of construction - life test on prototypes	

TABLE 6d - HISTORICAL DEVELOPMENT
THERMOELECTRIC CONVERTERS

Fundamental Concepts	Additional New Concepts Needed For Application	Laboratory Models	Operational Systems
1822 Seebeck Effect	1949 A.F. Ioffe Theory of semiconductor thermoelements	1947 M. Telkes SnSb-Pbs generator	1958 SNAP III Thermoelectric generator
1834 Peltier Effect	1954 H.J. Goldsmith Bi_2Te_3	1953 A.F. Ioffe Russian prototype thermoelectric refrigerator	1966 SNAP 10A Thermoelectric generator
1857 W. Thompson Theory of Thermoelectricity	1955 R.W. Fritts PbTe	1956 A.F. Ioffe Russian Kerosene Thermoelectric lamp	
1948 H.B. Callen Onsager's reciprocal relations for Thermoelectricity	1956 A.F. Ioffe, et al Thermal scattering by neutral defects studies of ternary solid solutions	1957-1963 Extensive development on materials and devices ranging from 1 to 5000 watts	
1953 Herring-Thermoelectric power of semiconductors		1962 GeSi alloy	
1956 J.R. Drabble R. Wolfe Band structure of Bi_2Te_3			

PROBLEMS AND PROGRESS WITH LONG-DURATION LIFE-SUPPORT SYSTEMS

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Introduction

Life-support systems designed for long durations will be required for such future manned missions as the space station, the lunar base, and the expeditions to Mars and Venus. In order to realize the full scientific potentialities for these missions, the life-support systems should be designed for essentially indefinite operation, with a minimum consumption of stored or resupplied expendables, to reserve vehicle payload capability for scientifically meaningful equipment. Of course, there are state-of-art limitations imposed on system design for a particular mission.

Spacecraft life support encompasses the functions of thermal and atmospheric control, water management, waste management, food supply, and personal hygiene. Because these functions are fundamentally interrelated by energy and material balances that may override other considerations for long-duration missions, functional comparisons on a system basis are necessary. The relationships between the life-support system and other vehicle systems such as the power system are extremely important in arriving at an integrated optimum vehicle system.

For missions beyond the two-week class now under development there will be a strong tendency toward further conservation of expendables. This development is, necessarily, at the expense of power consumption substantially higher than the power requirements for current life-support systems.

In advanced life-support systems for long-duration missions, emphasis will be placed on the processing of waste products to conserve essential materials such as oxygen and water. Closure of the oxygen and water loops will be essential to avoid excessive penalties. Complete closure of the material loop by food production appears to be in the distant future because of the complexity of man's dietary requirements.

This paper discusses recent progress and present status, and presents many of the development problems that must be solved to achieve a completely integrated and reliable life-support system for long-duration missions.

Material Balances

A life-support system for long-duration missions must provide the various materials consumed by the crew, including a gaseous environment that is controlled with respect to temperature, pressure, and composition. In addition, the system must provide the means for removing the wastes produced by man.

Assuming an average metabolic rate of the order of 500 Btu/hr, the following figures are obtained for material inputs and waste production:

Material Inputs		Waste Production	
O_2	= 1.9 lb/man-day	CO_2	= 2.25 lb/man-day
H_2O	= 4.5 to 10.0 lb/man-day	H_2O	= 2.2 to 7.7 lb/man-day
Food	= 1.8 lb/man-day	Urine	= 3.35 lb/man-day
		Feces	= 0.4 lb/man-day

The production of water vapor (perspiration and respiration) will vary, depending ~~on~~ the method of cooling the man. The lower value of the H_2O listed under waste production, above, is applicable to a shirtsleeve environment in which the latent heat load is maintained at the minimum level by a suitable mean radiative sink temperature environment[1]. The higher value applies to a man in a ventilated, thermally insulated pressure suit. Of these two, a shirtsleeve environment will be desired from the standpoint of comfort.

Table I shows the recommended atmospheric design conditions for a shirtsleeve environment.

TABLE I

RECOMMENDED NOMINAL ATMOSPHERIC DESIGN VARIABLES FOR A SHIRTSLEEVE ENVIRONMENT

Parameter	Missions Up to 3-4 Weeks	Missions Beyond 3-4 Weeks
Total atmospheric pressure, psia	5	10
Temperature, °F	75	75
Partial pressure, mm Hg		
Nitrogen	0	340
Oxygen	240	170
Carbon dioxide	≤7	≤5
Water vapor	10	10

Food consumption will vary from 1.3 to 5.0 lb/man-day, depending upon diet, for caloric intake of 3000 kcal/day. The lower value applies to a dry food diet; the higher value is applicable to a normal food diet. The required water consumption will depend upon the amount of water contained in the food. A diet corresponding to a food intake of 1.8 lb/man-day will be assumed compatible with the nutritional and palatability requirements for long-duration missions. The amount of water available from the food will determine the efficiency required of some of the recovery processes, in closed-cycle systems[2].

The above material balance figures are a rough estimate of the incentive for loop closure to conserve expendables. Associated weight and power penalties, and basic considerations of feasibility and reliability, will strongly influence specific decisions regarding loop closure.

Breathing Gas

With an open-cycle breathing gas system, carbon dioxide is removed by a washout process using a sufficient throughflow of breathing gas to maintain the carbon dioxide partial pressure within the desired limits. Assuming mixing of the products of respiration with the breathing gas, the required open-cycle breathing gas flow is given by

$$\dot{w}_g = \dot{w}_c \frac{(\pi - p_c)\, m_g}{p_c\, m_c}$$

where $\dot{w}_c$ = CO_2 production rate, lb/hr

p_c = CO_2 partial pressure, psi

π = total pressure, psia

m_c = molecular weight of CO_2

m_g = molecular weight of breathing gas

Therefore, depending upon total pressure and breathing gas composition, the required throughflow will vary from 6.1 lb/man-hr with air at 14.7 psia to 1.6 lb/man-hr with pure oxygen at 3.5 psia, for a carbon dioxide partial pressure of 7.6 mm Hg. By recirculation and carbon dioxide removal from the breathing gas, the material requirements are reduced from the 37 to 146 lb/man-day range to 4 lb/man-day (allowing for a leakage rate of 2 lb/man-day). Because of their low fixed weight and power requirements, open-cycle breathing gas systems will be optimum for short durations (up to 12 hours).

Water

In vehicles using fuel cells for power, the potable water produced as a byproduct of power generation may be in excess of requirements. In such vehicles, close water management will not be required, and an open-cycle water system will be optimum. For long-duration missions, where solar, nuclear, or isotope power systems will be used, water reclamation will be required. The condensate obtained from the heat exchanger in the atmosphere control system will be relatively pure and will require little treatment, probably only filtration over activated charcoal and ultraviolet irradiation, to be suitable for drinking purposes. Urine is 95 percent water and requires more extensive processing, such as distillation, to be usable. The water balance will be an important factor in the design of any closed-cycle system.

Comparison of human production and consumption of the molecular species water (see Table II) shows that for any cooling-ventilating method (shirt-sleeve or pressure suit) the excess of water production over water consumption is about 1.2 lb/man-day. This quantity is referred to as metabolic water. Roughly two-thirds of the metabolic water (0.9 lb/man-day) comes from partially oxidized foods such as carbohydrates. The remaining one-third (0.3 lb/man-day) comes from oxidation of 0.035 lb/man-day of hydrogen in food by part of man's oxygen supply. This represents $0.3 \times \frac{16}{18}$ = 0.27 lb/man-day of the man's oxygen consumption. Limiting consideration to stored food supply, the 0.9 lb/man-day of food-derived water can be regarded as a tolerable margin of error in the closure of the water and oxygen loops. This represents essentially an allowance for physical leakages from the spacecraft and functional leakages in the various recovery systems due to process inefficiencies. Considering the nature of the contemplated recovery processes, this is not a large margin, but it should be sufficient, if the cabin leakage rates can be kept low. Incidentally, water will almost surely be an intermediate product in processes for recovery of oxygen from carbon dioxide; therefore, a surplus of water can be converted into a surplus of oxygen by electrolysis.

It should be emphasized that the amount of metabolic water depends upon the diet and this will determine the requirements for the processes used in water reclamation and oxygen recovery. For example, using a completely dehydrated food diet (representing a food intake of 1.32 lb/man-day), the water derived from the food drops to 0.74 lb/man-day. Obviously, with a reduction in metabolic water, the material conservation requirements for the life-support system become more stringent.

TABLE II

FOOD-WATER MATERIAL BALANCE

Oxygen consumption = 1.90 lb/man-day

Carbon dioxide generation = 2.25 lb/man-day

Urine production = 3.35 lb/man-day

Fecal output = 0.40 lb/man-day

Food consumption = 1.80 lb/man-day

Water generated by oxidation of food =
$(1.90 - 2.25 \frac{32.0}{44.0}) \frac{18.0}{16.0} = 0.31$ lb/man-day

Fecal water (assume 75%) = 0.30 lb/man-day

Urine water (assume 95%) = 3.20 lb/man-day

Total solids in wastes = (0.40 - 0.30) + (3.35 - 3.20) = 0.25 lb/man-day

Carbon in food converted into $CO_2 = 2.25 \frac{12.0}{44.0}$ = 0.615 lb/man-day

Hydrogen converted into $H_2O = 0.31 \frac{2.02}{18.0}$ = 0.035 lb/man-day

H_2O derived from food (metabolic H_2O) = 1.80 - 0.615 -0.035 - 0.25 + 0.31 = 1.22 lb/man-day

If the feces is not processed for water recovery and 95 percent of the water in the urine is recovered, 0.46 lb/man-day of water will be lost, reducing the water available to the oxygen recovery system, as shown in Table III.

TABLE III

WATER AVAILABLE TO OXYGEN RECOVERY SYSTEM
95 PERCENT RECLAMATION OF URINE,
NO FECAL WATER RECLAMATION

Food Consumption, lb/man-day	Metabolic Water, lb/man-day	Available Water, lb/man-day	Available O_2 in H_2O, lb/man-day
1.32	0.74	0.28	0.25
1.80	1.22	0.76	0.67
5.00	4.42	3.96	3.52

Since the oxygen available in the carbon dioxide produced by the man amounts to 1.64 lb/man-day, it is impossible to close the oxygen cycle using the dry food diet, unless the fecal water is recovered. On the other hand, with a normal food diet (which is probably impractical for a long-term mission), there is no need to process the carbon dioxide for oxygen recovery; ample oxygen is available in the surplus water. With the assumed diet of 1.80 lb/man-day, the oxygen recovery system is required to recover 75 percent of the oxygen contained in the carbon dioxide for makeup of metabolic consumption. If the oxygen recovery system is additionally required to provide for makeup of cabin leakage, higher oxygen recovery efficiencies will be necessary[2].

Carbon Dioxide

Absorption of carbon dioxide on lithium hydroxide requires 2.6 lb/man-day in expendable absorbent. Lithium hydroxide will be optimum for mission durations up to about 14 days[3]. Regenerable carbon dioxide removal systems of various types can be used for long durations.

Atmospheric Fluid Storage

High-pressure gaseous storage will be used for oxygen supply in vehicles designed for short-duration missions. If the utilization rate is above the minimum set by boiloff, cryogenic storage methods will offer significant weight advantages when the amount of fluid stored is in excess of a few pounds. On vehicles using oxygen recovery, atmospheric fluid storage will be required for makeup of leakage and for repressurization.

A space laboratory could logically require fairly large quantities of stored atmosphere, to meet emergency situations, but have a low normal demand rate. This combination of requirements could be met with either high pressure gas storage (which will be relatively heavy) or cryogenic systems with refrigeration (which will be relatively high in power consumption). The choice will depend on weight-power tradeoffs and continued progress in the development of storage techniques involved.

Oxygen Recovery

For mission durations where solar or nuclear power systems will be optimum, oxygen recovery can be used to advantage. Most oxygen recovery systems utilize the electrolysis of water for production of oxygen. The hydrogen that is produced simultaneously with the oxygen is used to reduce the carbon dioxide which is separated from the recirculated breathing gas. A variety of reactions can be used in the intermediate steps. It is convenient to classify these systems according to the carbon-containing end product. For example, using the Sabatier reaction, with an end product of methane, a hydrogen deficit of 0.172 lb/man-day results if all the carbon dioxide is reduced stoichiometrically. This hydrogen deficit can be related to the water balance, if the hydrogen is produced by electrolysis of water. To produce 1.90 lb/man-day of oxygen, it will be necessary to electrolyze 2.14 lb/man-day of water. The 0.24 lb/man-day of hydrogen produced as a byproduct of oxygen generation will provide the water balances (see Table IV) for the specified end products.

TABLE IV

WATER BALANCE FOR OXYGEN RECOVERY SYSTEM

End Product	H_2O by CO_2 Reduction lb/man-day	Available Metabolic Water lb/man-day	Water Balance lb/man-day
C	1.84	0.76	+0.46
CH_4	1.07	0.76	-0.31
$C_nH_{0.8n}$	1.38	0.76	0

Material Loop Closure

Based upon the material balance considerations, Table V shows the priorities assigned to various processes for long-duration missions.

TABLE V

RELATIVE PRIORITY FOR MATERIAL LOOP CLOSURE
FOR LONG-DURATION MISSIONS

Function	Process	Maximum Material Saving Incentive lb/man-day
Recycle breathing gas	Remove CO_2	33-142
Reclaim water	Reuse condensate	2.2-7.7
Reclaim water	Process urine	3.2
CO_2 removal	Regenerable	2.6
O_2 recovery	CH_4 end-product	1.6
H_2 recovery	C end-product	0.3
Reclaim water	Process feces	0.3

It should be emphasized that the material deficits which may appear to be small on a percentage or a lb/man-day basis can represent substantial weights for a multi-man, long-duration mission. That is, because of the cost of transportation for space cargo, it will be desired to use an efficient (in terms of materials conservation) life-support system; the payload capacity available can then be reserved for equipment or supplies which increase mission capability. This will probably be desired whether or not resupply is possible.

Power Penalties

The weight penalties for power consumption exert a highly significant influence on life-support system design. This is particularly true for short-duration missions where relatively open-cycle systems will be optimum because of their low fixed weight and low power consumption. As mission durations increase, the weight of expendables mount, necessitating closing the cycles. For long-duration missions, power consumption will still be important, even where power is available at reasonably low penalties, because of the weight penalties associated with rejection of waste heat. High thermal efficiency will, therefore, be desired when compatible with the requirements for high material recovery efficiency.

Priority must also be given in life-support system design and process selection for use of low-cost energy where it will suffice. Low-cost energy, in this case, is the energy available at small vehicle-weight penalty. Waste heat (from the electronic equipment, for example) is usually the cheapest form of energy, although its temperature level may not be adequately high for many processes. Thermal energy provided by a solar absorber and transmitted by a liquid heat transfer fluid loop may be obtainable at lower penalties than electrical power. However, for such purposes as the electrolysis of water and operation of fluid circulating devices, electrical power will be required.

Electrical power may be available at more than one penalty, depending upon conversion (voltage from d-c to a-c), regulation, and energy storage. Much of the electrical power required by an advanced life-support system can be in the form of unregulated d-c, which is obtained with some types of power systems at significantly lower penalties than closely regulated a-c. In vehicles with large free cabin volumes, much of the power-consuming process equipment can be operated on a cyclic basis to take advantage of power load schedules or periodic intervals of power abundance. An example of this would be an earth orbital space station (using a solar power system) in which the oxygen recovery system, which consumes most of the power, is operated only during the sunlight phase to avoid the weight penalties associated with energy storage for dark-side operation.

Carbon Dioxide Removal

One of the major problem areas in design of oxygen recovery systems has involved integration of the regenerable carbon dioxide removal system with the carbon dioxide processing system. Regenerable carbon dioxide removal systems based upon use of the synthetic zeolite molecular sieve adsorbents have been extensively studied[3-9]. In all of these systems, the adsorbent is regenerated by vacuum desorption (with vacuums of the order of 100 microns and less) under adiabatic conditions. This approach appears to be satisfactory if the carbon dioxide is dumped overboard and the space vacuum can be used for the desorption process. Where the carbon dioxide is to be processed for oxygen recovery, the removal system must supply the carbon dioxide at a relatively high pressure level, preferably without use of vacuum pumps that would have prohibitive weight and power requirements[10]. Adsorbent regeneration to provide the carbon dioxide at the pressure levels required for processing can be accomplished by heating the bed, by purging the bed with gas, or by a combination of the two methods.

Purge-gas desorption by itself, using hydrogen, has been shown to be impractical as a means of removing carbon dioxide from molecular sieves because of the high gas flow required[11]. The purpose of using hydrogen as the purge gas is for the ultimate integration into an oxygen recovery system.

Satisfactory desorption by combined heating and purging has been demonstrated with hydrogen flows corresponding to stoichiometric reaction of the hydrogen and carbon dioxide to form methane. However, the energy requirements and bad temperature levels are comparable to those required for thermal desorption by itself. Since control is more difficult and the hydrogen purge gas must be extremely dry to avoid poisoning the molecular sieve, it appears that purge gas desorption for carbon dioxide removal from molecular sieve beds offers no advantage over thermal desorption.

Figure 1 shows typical thermal desorption performance for a molecular sieve canister at 14.7 psia. Thermal desorption was accomplished by an integral electrical heating element located internally in the bed. During the 26-minute preheat period with a 600-watt power input, relatively little carbon dioxide is evolved until the bed reaches a temperature of approximately 400°F. At the point where carbon dioxide is evolved at the required rate, the automatic control reduces power input to approximately 240 watts to maintain an essentially constant flow of carbon dioxide. Near the end of the desorption cycle, the power input to the bed increases to 300 watts to complete desorption. Residual carbon dioxide content is less than 1.0 percent at the end of desorption. The saw-tooth shape of the carbon dioxide production curve results from the off-on type of power control. More uniform delivery could be achieved by a proportional control with a lead or anticipatory circuit. However, as will be discussed, high conversion efficiencies have been obtained with the simple type of control.

Figure 2 is a schematic diagram of a regenerable carbon dioxide removal system in which the molecular sieve is thermally desorbed. In this system, the process gas is first dehumidified before passing to the molecular sieve bed, where the carbon dioxide is removed. After leaving the carbon dioxide adsorber, the process gas flows through the desiccant bed that is being regenerated. The carbon dioxide adsorbent bed is regenerated by heating. The heat remaining in this bed at the end of the desorption cycle is used for desorption of water from the desiccant bed. Part of the power expended in the carbon dioxide desoprtion process is used twice.

The thermal energy required for thermal desorption of a molecular sieve at 14.7 psia is ideally around 1000 watt-hours/lb of carbon dioxide at a temperature level in the 400 to 500°F range. The performance shown in Figure 1 indicates a thermal energy requirement of approximately 2000 watt-hours/lb of carbon dioxide. By use of more efficient thermal insulation to reduce heat leak, it has been possible to reduce the energy input for desorption to the neighborhood of 1200 to 1400 watt-hours/lb of carbon dioxide. If desorption is to a lower pressure than 14.7 psia or if it is aided by a vacuum pump, the thermal energy requirement can be reduced to around 700 watt-hours/lb of carbon dioxide.

Since thermal energy is required for the desorption process, the electrical energe requirements can be reduced by use of waste heat. Waste heat will be obtainable at the proper temperature level from either the power system or the oxygen recovery system (in particular, the hydrogenation of carbon dioxide reaction). Optimization of the carbon dioxide removal system will involve system integration and detailed consideration of many factors, including the availability of waste heat from power, thermal control, and oxygen recovery systems; power consumption penalties; and the heat dissipation penalties.

Oxygen Recovery by Methanation of Carbon Dioxide

Methanation of carbon dioxide (the Sabatier reaction) involves its reaction with hydrogen to yield methane and water:

$$CO_2 + 4H_2 = CH_4 + 2H_2O$$

The reaction is exothermic (1600 Btu/lb carbon dioxide reacted) and is self-sustaining with high conversion efficiencies (about 95 percent) at moderate temperature levels (less than 500°F). Because of these advantages, use of the Sabatier reaction has been intensively studied for systems where it is the end reaction and where it is an intermediate reaction. Extensive experimental studies in this area have been recently conducted[12,13,14].

A typical system using the methanation process is shown schematically in Figure 3. Carbon dioxide is mixed with electrolytically-generated hydrogen and reacted in a reactor using a nickel catalyst. The water produced by the reaction is condensed in a heat exchanger and the methane is either dumped overboard or is processed for hydrogen recovery. Figure 4 shows typical catalyst performance.

Considerable variation in catalyst activity has been observed between different batches of presumably identical formulations. However, with suitable preconditioning and design allowances in system operating parameters (bed temperatures), this should not be a difficult problem.

Table VI shows typical performance that has been obtained with an integrated carbon dioxide removal and methanation system.

TABLE VI

INTEGRATED CARBON DIOXIDE REMOVAL AND METHANATION SYSTEM TYPICAL PERFORMANCE

Adsorption-desorption cycle time	85 min
Molecular sieve effective CO_2 loading	4.4 wt percent
Average thermal energy input	4900 Btu/lb CO_2
Methanation catalyst weight	2.5 lb-hr/lb CO_2
Overall system O_2 recovery efficiency	97 percent

The system efficiency refers to the percentage of the carbon dioxide adsorbed that is converted to methane. Since the catalytic reactor under the operating conditions obtained in system provides essentially complete conversion, the slight loss in system efficiency is attributed to the inefficiencies of the control system in maintaining a stoichiometric ratio at the reactor inlet. All oxygen recovery systems, if they are to be efficient, will demand precise proportioning of reactants at the extremely low flow rates of the order of a few tenths of a lb per hr. System controls, including sensors in particular, may represent a significant problem area for long-duration operation, although recent tests have been highly encouraging in this area. Figure 5 shows typical performance of an integrated system for one desorption cycle.

As indicated in Table IV, a water deficit of 0.31 lb/man-day results where methane is the end product of the oxygen recovery process (assumed diet = 1.80 lb/man-day, 95 percent urine water recovery, no fecal water recovery). If all of the waste water is recovered, a surplus of 0.15 lb/man-day is obtained. It may be possible to obtain essentially 100 percent water recovery. If this water reclamation is accomplished and if the assumed diet is representative, an oxygen recovery system with methane as the end product may be optimum. However, if, as now seems likely, a somewhat drier diet is used, it will be necessary to process the methane to recover some of the hydrogen for reuse. Two processes for hydrogen recovery will be considered: methane pyrolysis and methane oxidation with carbon dioxide.

Methane Pyrolysis

The most direct way of recovering the hydrogen from the methane produced by the Sabatier reaction involves pyrolysis of methane to carbon and hydrogen:

$$CH_4 = C + 2H_2$$

This reaction is endothermic, requiring 2340 Btu/lb of methane (at 1000°F), with equilibrium conversion favored by high temperatures and low pressures, as shown in Figure 6. By use of metallic catalysts such as palladium, cobalt, or nickel, reasonable decomposition rates can be obtained at temperatures below 700°F.

Low temperature operation of the pyrolysis reactor is desired to minimize side reactions, to promote nucleation and precipitation of carbon, to improve reliability, and to reduce power consumption. However, low equilibrium conversions will be obtained, necessitating recycling for low-temperature operation.

Figure 7 is a schematic diagram of an oxygen recovery system using carbon dioxide methanation followed by methane pyrolysis. The methane pyrolysis is accomplished in a recycling loop that uses a palladium diffusion cell for separation of the methane and hydrogen.

The alternative arrangement for a methane pyrolysis system involves operation at elevated temperatures (or low pressures) where high conversion efficiencies are obtained.

Buildup of carbon on the catalyst will eventually clog the catalyst surfaces, reduce its effectiveness, and increase the pressure drop through the bed. Therefore, the catalyst must be periodically replaced or cleaned.

In order to eliminate the water deficit incurred by the methanation reaction, it will be necessary to achieve the conversion efficiencies given in Table VII.

TABLE VII

REQUIRED CONVERSION EFFICIENCIES FOR MATERIAL BALANCE CARBON DIOXIDE METHANATION FOLLOWED BY METHANE PYROLYSIS

Food Consumption lb/man-day	H_2O Deficit CH_4 End Product lb/man-day	CO_2 Reduction to CH_4 percent	CH_4 Pyrolysis to C + H_2 percent
1.32	-0.79	100	80.0
1.80	-0.31	74.8	41.7

For the dry food diet, high material recovery efficiencies will be required. For the assumed diet of 1.80 lb/man-day, it may be possible to eliminate the recycle and separator in the methane pyrolysis loop. The thermal energy input for methane pyrolysis will range from 173 to 458 watts/man for the conditions given in Table VII.

Methane Reaction with Carbon Dioxide

Another method of recovering the hydrogen from the methane produced by the Sabatier reaction involves reacting it with carbon dioxide according to the following reaction:

$$CH_4 + CO_2 = 2H_2O + 2C$$

This reaction is mildly exothermic, yielding 142 Btu/lb of carbon dioxide (at 100°F).

As shown in Figure 8, equilibrium conversion is favored at low temperature, although conversion reaction rates will be enhanced by high temperatures. This reaction offers the possibility of accomplishing the decomposition of methane at lower temperatures than required by highly endothermic methane pyrolysis reaction previously discussed. However, the carbon fouling problems may be similar.

Figure 9 is a schematic diagram of an oxygen recovery system using the Sabatier reaction followed by methane oxidation by carbon dioxide. A recycle loop is provided for the methane-carbon dioxide reactor to obtain high system efficiencies.

In order to obtain a material balance with the methanation reaction followed by reaction of methane and carbon dioxide, the conversion efficiencies shown in Table VIII are required.

TABLE VIII

REQUIRED CONVERSION EFFICIENCIES FOR MATERIAL BALANCE CARBON DIOXIDE METHANATION FOLLOWED BY REACTION OF METHANE AND CARBON DIOXIDE

Food Consumption lb/man-day	CO_2 Reduction to CH_4 percent	CH_4 Oxidation to C + H_2O percent
1.32	58.2	72.8
1.80	58.2	27.8

Because of the relatively low conversion efficiency required for the methane reaction with carbon dioxide, the recycle may not be required for the assumed diet of 1.80 lb/man-day.

Oxygen Recovery by Carbon Dioxide Reduction

The reverse water-gas reaction (Bosch reaction) involves the reduction of carbon dioxide by hydrogen to carbon and water.

$$CO_2 + 2H_2 = C + 2H_2O$$

This reaction is exothermic, evolving 921 Btu/lb of carbon dioxide. Iron pellets or steel wool are effective catalysts for the reaction which optimally takes place at temperatures in the range from 930 to 1130°F. Carbon monoxide and methane are both produced in the reaction, which provides conversion of approximately 30 percent of the carbon dioxide per pass, requiring recirculation of the reaction products, as shown in the system schematic of Figure 10. The catalyst requires activation by heating with hydrogen prior to use. Periodic reactivation and cleaning of the catalyst is required. The formation of carbon monoxide and the relatively high catalyst consumption rates are the major disadvantages of this system for oxygen recovery.

Oxygen Recovery by Direct Pyrolysis of Carbon Dioxide

Direct thermal decomposition of carbon dioxide into carbon and oxygen requires extremely high temperatures for appreciable reaction rates. Carbon dioxide can be more readily dissociated into carbon monoxide and oxygen, as follows.

$$2CO_2 = 2CO + O_2$$

A conversion of about 5 percent has been obtained at a temperature of 3500°R and a pressure of 1 psia. Conversion is favored at low pressures. The carbon monoxide produced by the pyrolysis reaction is then separated from the oxygen and passed through another reaction to give

$$2CO = CO_2 + C$$

This reaction proceeds readily at 900°F to provide good yields, using iron carbide or the product carbon as catalysts.

Figure 11 shows a possible system arrangement which uses a molecular sieve for separation of the carbon monoxide and oxygen. Among the numerous disadvantages of this system are low operating pressures, high operating temperatures, and the presence of carbon monoxide as a product of reaction to be separated from oxygen.

Oxygen Recovery by Electrolysis of Carbon Dioxide

The electrolytic dissociation of carbon dioxide requires the carbon dioxide to be in an ionic state. Liquid and gaseous carbon dioxide are nonelectrolytic and do not conduct electric current. Carbon dioxide does not dissolve in many electrolytes to form ionic solutions that can be electrolyzed for carbon dioxide reduction. Following are descriptions of two electrolysis systems that have been investigated.

Electrolysis of Molten Lithium Carbonate

Lithium carbonate is formed by addition of carbon dioxide to molten lithium oxide:

$$CO_2 + Li_2O = Li_2CO_3$$

The lithium carbonate is electrolyzed to produce carbon and oxygen and regenerate the lithium oxide:

$$\text{Anode: } 2CO_3^{=} = 2CO_2 + O_2 + 4e^-$$

$$\text{Cathode: } 2Li^+ + 2e^- = 2Li$$

$$\text{Overall: } 4Li + 2CO_2 = 2Li_2O + 2C + O_2$$

The electrolytic cell is required to operate at temperatures in excess of 1000°F to maintain the electrolyte in the molten condition. The theoretical energy requirement for the electrolysis amounts to 1.1 kw-hr/lb of carbon dioxide electrolyzed. Actual cell efficiencies that have been obtained in experimental units have been low, with an energy requirement of 8.5 kw-hr/lb of carbon dioxide indicated. Problems anticipated with this type of system include development of filters for carbon separation, and development of components and materials resistant to the hot, highly corrosive electrolyte.

Electrolysis of Carbon Dioxide in Solid Electrolytes

This involves the use of semiconductors as solid electrolytes for the conveyance of oxygen ions from the anode to the cathode:

$$\text{Cathode: } CO_2 + 2e^- = CO + O^{=}$$

$$\text{Anode: } 2O^{=} = O_2 + 4e^-$$

$$\text{Overall: } 2CO_2 = 2CO + O_2$$

The carbon monoxide appears at the cathode and oxygen appears at the anode. The carbon monoxide is decomposed catalytically in a separate reactor according to the reaction:

$$2CO = CO_2 + C$$

The carbon dioxide produced by this reaction is returned to the inlet of the electrolytic cell.

The feasibility of this process has been demonstrated[15], although much research and development work remains to be done before a practical unit can be fabricated. An electrical power requirement of 6 kw-hr/lb of carbon dioxide is estimated for the process.

Oxygen Recovery by Fischer-Tropsch Process

The Fischer-Tropsch reaction is normally associated with the commerical production of liquid hydrocarbons by the reaction of carbon monoxide and hydrogen. Depending upon pressure, temperature, and catalyst, a variety of organic materials can be produced ranging from simple hydrocarbons to alcohols and other oxygenated compounds. The Fischer-Tropsch process can be applied to carbon dioxide hydrogenation to yield water and a hydrocarbon product. If the carbon-containing product is of higher molecular weight than methane, less hydrogen will be lost by dumping this product overboard than would be the case using the Sabatier reaction. The Fischer-Tropsch process, therefore, offers the possibility of a single-reaction oxygen-recovery system that meets the material conservation requirements for long-duration missions. For example, the waste products from the oxygen recovery system will ideally have an H/C mole ratio of 0.80. If a Fischer-Tropsch reaction can be made to provide a hydrocarbon with that composition, no hydrogen recovery reaction will be required.

Considerable research work will be required with catalysts and with reactor design to apply the Fischer-Tropsch process to an oxygen recovery system. Anticipated problem areas include obtaining high reaction yields at moderate pressure levels, and preventing polymerization or other reactions that may give physically undesirable products.

Oxygen Recovery by Photosynthetic Gas Exchangers

Photosynthetic gas exchangers use the photochemical reactions of green plants to reduce carbon dioxide and produce oxygen. Most of the work with photosynthetic gas exchangers has involved algae, although recently there has developed increasing interest in higher plants such as endive and chinese cabbage[15] and duckweed[16]. A potential major advantage associated with photosynthetic systems is the capability for production of food, using bacterially digested wastes to supply most of the plant nutrients. The problems here involve development of a nutritionally adequate and palatable diet, using these rather simple food sources. The requirement for high growth rate excludes the use of many common crop plants. Methods have been developed to bleach and make a "flour" of algae that can be used in baking or as a filler in other foods. With development, it should ultimately be possible for algae to provide at least one-half of the total daily food requirements. One of the major problems has involved elimination of bacterial contamination that has apparently caused the digestive disturbances reported by test subjects on an algael diet.

Other problems associated with photosynthetic systems include control of growth rate, elimination of toxins, and control of pathogenic microorganisms.

Trace Contaminant Control

In addition to the water vapor and carbon dioxide produced by the man, other contaminants are generated by man and by the equipment. These trace contaminants, if allowed to build up, can lead to noxious, toxic, or explosive atmospheres.

Contaminants with high molecular weight, such as indole, skatole, and methylmercapton, are efficiently removed by activated carbon. The capacity of activated carbon for high-molecular-weight contaminants is large relative to the contaminant production rate. It has been estimated that the activated carbon required for trace contaminant control will amount to only 2 lb/man year[10]. Consequently, regeneration will not be required.

Contaminants with low molecular weight, such as hydrogen, carbon monoxide, and methane, are relatively poorly adsorbed by activated carbon. Catalytic combustion with oxygen appears to be the most effective means of preventing buildup of low-molecular-weight contaminants. Of these substances, methane is the most difficult to oxidize. Hopcalite catalyst has been used for trace contaminant control on submarines. As shown in Figure 12, a catalyst bed temperature of approximately 830°F is required for complete oxidation of methane with hopcalite. Since operation at lower bed temperatures should be preferable from the standpoints of reliability and power consumption, an investigation of catalysts for oxidation of trace contaminants has been undertaken[10]. Several platinum catalysts were tested and found to be inferior to hopcalite. A palladium catalyst was obtained that gave complete conversion of methane at 620°F.

Present knowledge of the trace contaminant problem for long-duration space capsules is incomplete. In particular, the contaminants produced by the electronic and other equipment are not known. The "maximum allowable concentrations" are not known for many contaminants, or are based upon industrial safety standards that may not be applicable to long-term, continuous exposure for man-in-space.

Microbiological Contamination Control

Although the space vehicle can be aseptic at the start of a mission, the production of undesirable microorganisms by the crew during an extended flight is a matter of concern. These microorganisms can impose a health hazard, produce a noxious atmosphere, or degrade equipment and supplies. Since all microorganisms can be assumed to be introduced into the vehicle by the crew, it might appear that the menace to health would be insignificant, assuming a proper preflight quarantine period. However, in some species of bacteria, high mutation rates from benign to pathogenic forms are experienced. Therefore, provisions should be made in spacecraft life-support systems for elimination or control of growth of microorganisms.

Ultraviolet radiation of the 2537 Å wavelength is very effective in killing microorganisms. Power requirement for a suitable ultraviolet lamp is low, around 5 watts. The lamp will generate some ozone that can be decomposed catalytically. The process gas flowing through the trace contaminant catalytic oxidizer will be sterilized, but the flow required for trace contaminant control purposes will not be adequate for microbiological control. Germicides can be incorporated into filters and surface coatings to prevent growth of microorganisms in areas where favorable conditions exist for their propagation.

Waste Management

The waste management system consists of equipment for the collection, treatment, and processing of storage of the various waste products. Much of the previous work in waste management has been concerned with waste collection; here, the emphasis will be on the waste processing that will be required in long-duration missions.

Feces

The composition and the amount of feces depend upon diet and physical condition of the individual. An average amount is 0.4 lb/man-day, 70 to 85 percent of which is water, with the remainder consisting of food residues, digestive fluids, cell fragments, bacteria, and products generated by bacteria. Gases that are produced include odorous products such as indole, skatole, methylmercapton, and hydrogen sulfide, in addition to hydrogen, carbon dioxide, and methane.

A primary function of the waste handling systems involves treatment to inert the living bacteria that produce gas and toxic products. This can be accomplished by freezing, heating, combustion, or chemical or biological treatment. For short-duration missions, it will suffice to freeze and store the feces. For the long-duration missions, the storage space required may be excessive and other methods may be necessary. Dehydration followed by combustion of feces leaves very small residue and permits recovery of the water and oxygen contained therein. The penalties in terms of oxygen consumption and increased heat load for feces combustion are very nominal. Bacterial digestion of feces also provides water recovery and supplies a nutrient for a photosynthetic system.

Urine

Urine production depends upon diet, water consumption, metabolic rate, and perspiration rate. An average rate of 3.3 lb/man-day can be used for design purposes. The solids in urine represent about 5 percent by weight and include amino acids, salts, hormones, organic acids, and numerous, different compounds. Normally, no pathogenic bacteria are present.

With respect to urine treatment, the waste management system has the function of processing to prevent evolution of toxic and odorous substances. For short-duration missions, the urine will be stored or vented overboard through an evaporator. This will be optimum where fuel cells are used for power and potable water is produced as a byproduct of power generation. For long-duration missions, in which solar or nuclear power systems are used, it will be necessary to recover the water in urine for reuse. Many different processes have been studied for water reclamation from urine. Perhaps the simplest, and one that involves essentially no power or heat rejection penalty, consists of atmospheric or vacuum distillation integrated into the intermediate heat transport fluid loop to take advantage of the different temperature levels that exist at various points in the loop. That is, the heat of condensation is added to the coolant at one point and is removed as the heat of evaporation at another point with a different temperature level.

Wash Water

The amount of wash water required will depend upon the extent of personal hygiene measures. For short missions, the personal hygiene provisions can be very rudimentary (such as sponge baths) without any serious discomfort. For longer durations, it may be necessary to provide arrangements approaching those obtained by the individual in his normal habitat. An effective bactericide and detergent should be used in the wash water to control bacteria on the skin and increase the efficiency of soil removal. The used wash water will contain skin cells, bacteria, and small amounts of solutes. The amount of solid material removed from the skin by washing will amount to approximately 0.007 lb/man-day, 0.003 lb/man-day of which will be removed by filtration and treatment with activated carbon.

The solutes remaining in the wash water after this treatment will be principally sodium and potassium chloride. Assuming a wash water requirement of 3.0 lb/man-day and a tolerable limit on solutes of 1.0 percent, the wash water could be recycled 7.5 days before requiring more extensive processing than filtering and treatment with activated charcoal. Whether or not the saving effected by this approach to processing of the waste water is worthwhile depends upon the overall vehicle water management problem and the resultant penalties for water processing.

Dehumidification Water

The amount of condensate collected in the heat exchangers and used for moisture removal will depend upon the latent heat load from the crew and the loss of water by evaporation in food preparation and personal hygiene.

The condensate will be quite pure chemically, but may be contaminated with small amounts of organic substances that can support bacteria growth. Filtration of the condensate through an activated charcoal bed and irradiation by ultraviolet should provide adequate treatment to make the condensate suitable for drinking and food preparation purposes.

Conclusions and Recommendations

The following comments apply to the present status of long-duration life-support systems for spacecraft.

Carbon Dioxide Removal

Thermally-desorbed molecular sieve adsorbents appear to offer a satisfactory means of regenerable carbon dioxide removal that is suitable for integration with oxygen recovery systems.

Because of its static nature, electrodialysis carbon dioxide removal is promising for long durations, but requires more research concerning ion exchange resins and membranes, and internal water management within the cell.

Atmospheric Fluid Storage

Subcritical cryogenic storage of oxygen and nitrogen would offer greater weight savings than supercritical storage for long-duration missions involving storage of large amounts of fluid. Atmospheric fluid storage will be required for leakage makeup and for repressurization in vehicles using oxygen recovery. It will probably be necessary to use cryogenic refrigerators to prevent excessive boiloff losses with cryogenic tankage for spacecraft with large pressurized volumes and low leakage rates. Chemical repressurization systems may be used to advantage in some applications.

Oxygen Recovery

A number of different processes have been proposed for oxygen recovery from carbon dioxide. At the present time, carbon dioxide methanation (Sabatier reaction) followed by reaction of the product methane with carbon dioxide appears to be the most attractive of the various schemes. The water produced by both reactions is electrolyzed to produce oxygen and the hydrogen used in the methanation reaction. Sabatier reaction catalysts and integration of the reactor with a regenerable carbon dioxide removal system have been extensively investigated, with highly encouraging results. The second step in the system, which involves recovery of hydrogen from methane, has the additional problems with catalyst life and rejuvenation. Continued research work with catalysts and reactor design will be desirable to attain the full system capabilities with respect to weight, reliability, life, and power consumption.

Food Supply

Freeze-dried food will provide the major part of the diet for the foreseeable future. The amount of water derived from food can exert an important effect on the choice of the processes to be used in oxygen recovery and water reclamation. Photosynthetic systems offer promise of providing at least part of the food requirements for very long missions, but will require considerable research work with regard to plant physiology, system engineering, and hardware design.

Trace Contaminant Control

Low-molecular-weight contaminants can be catalytically oxidized, and high-molecular-weight contaminants can be efficiently adsorbed on activated carbon. More data are needed concerning the nature of the contaminants that will be produced and the maximum allowable concentrations for many contaminants.

Microbiological Contamination Control

This represents a potential problem area where little information is available. The seriousness of the problem has not yet been properly assessed.

Waste Management

Most of the work to date in the area of waste management has involved collection, handling, and storage of human wastes. For long-duration missions, it will be necessary to process the urine for water recovery. It may also be desirable to process feces in some way to reduce the storage space required (if for no other reason). If a very dry food diet is used, reclamation of the fecal water may be essential to avoid a water deficit. Dehydration and combustion appear to offer an effective means of solid waste disposal that provides conservation of the essential materials contained therein. Biological treatment of wastes will provide both conservation of materials and a nutrient for a photosynthetic system.

Water Reclamation

Water can be recovered from four waste sources: heat exchanger condensate, wash water, feces, and urine. Different processes can be used for reclamation of water, depending upon the source of the waste water and the use to which the product water will be put. Condensate can be used for drinking after filtration and treatment to kill microorganisms. Wash water can be reused as wash water after filtration. Reclamation of fecal water is the most difficult and is probably best accomplished by combustion-vaporization. The water contained in the urine can be recovered by a number of different methods. Perhaps the simplest of these involve a vacuum or atmospheric distillation process that is integrated into the coolant loop to take advantage of the different temperature levels in the loop.

Energy Requirements

An important consideration in life-support system optimization involves the preferential use of low-weight-penalty energy. This requires determining the penalties associated with use of various types of energy such as low-temperature waste heat, high-temperature thermal energy, unregulated d-c electrical power, and regulated a-c electrical power. The type and quantity of energy required may exert a significant influence on the process selected for a given function. Life-support system optimization involves integration of the various subsystem energy inputs and outputs to make maximum use of the internal energy available. Overall optimization involves integration of the life-support system energy requirements with the power system, considering the availability and penalties for various types of power. Because the life-support system is a large consumer of power, its requirements may be reflected in the design of the power conversion system.

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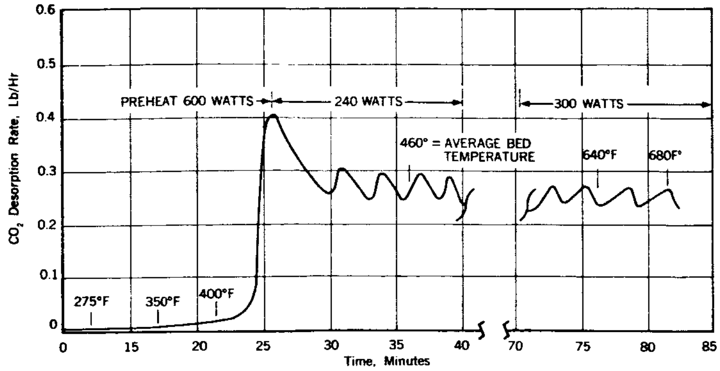

FIG. 1 - THERMAL DESORPTION OF MOLECULAR SIEVE

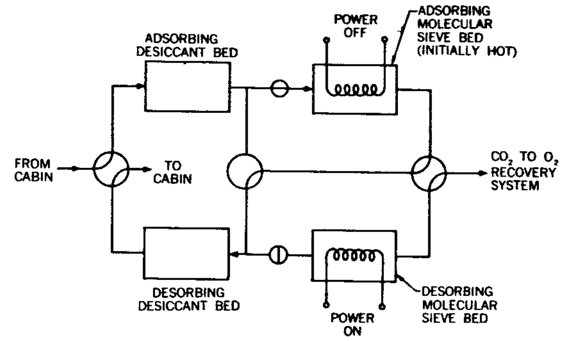

FIG. 2 - Regenable CO_2 Removal System Suitable for Integration with O_2 Recovery System

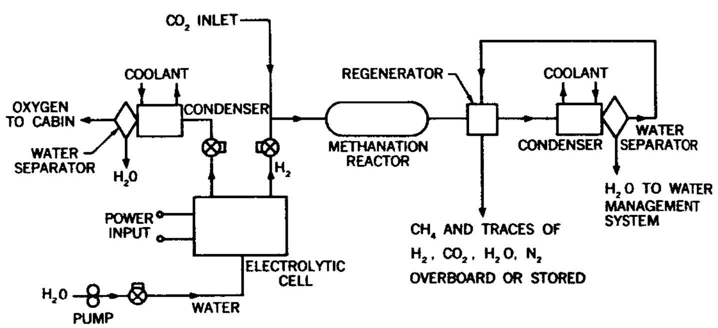

FIG. 3 Methanation System

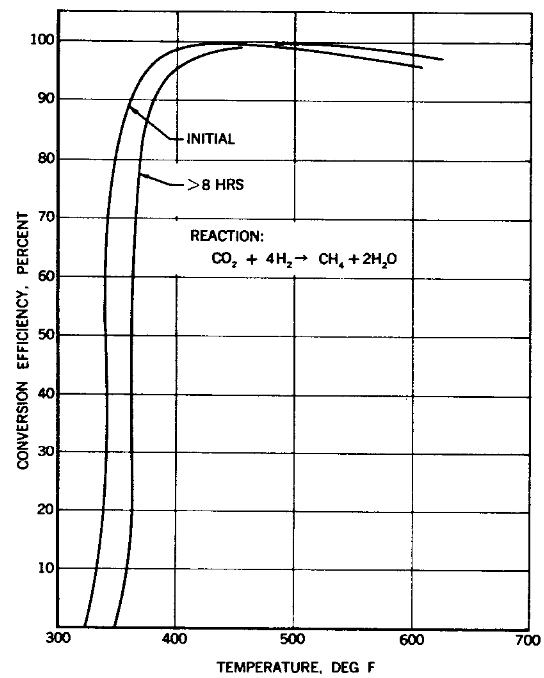

FIG. 4 CO_2 Methanation Catalytic Reaction

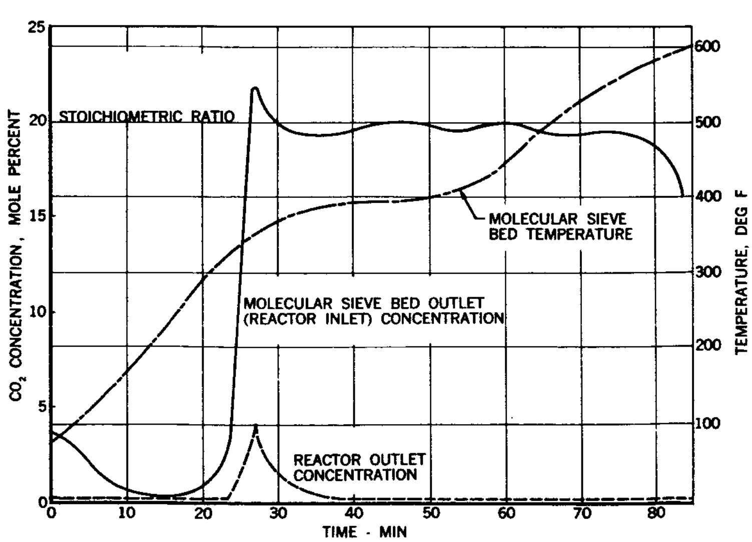

FIG. 5 Integrated System Characteristics

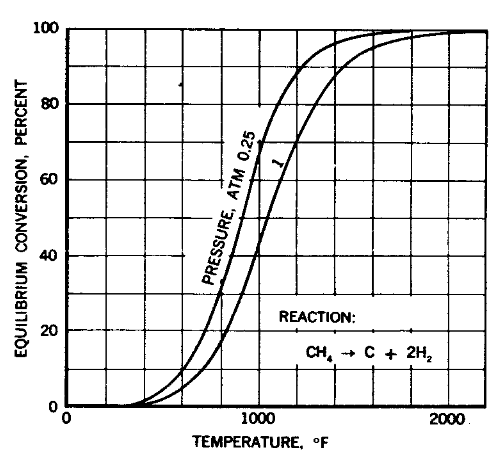

FIG. 6
Reaction Equilibrium – Hydrogen Recovery from Methane

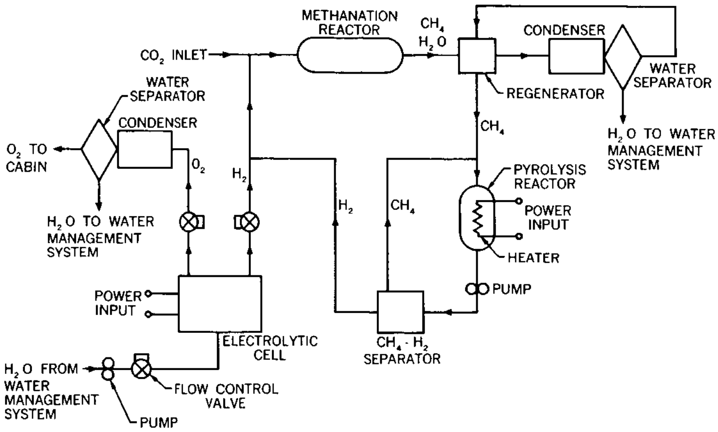

FIG. 7 - Oxygen Recovery by CO_2 Methanation CH_4 Pyrolysis

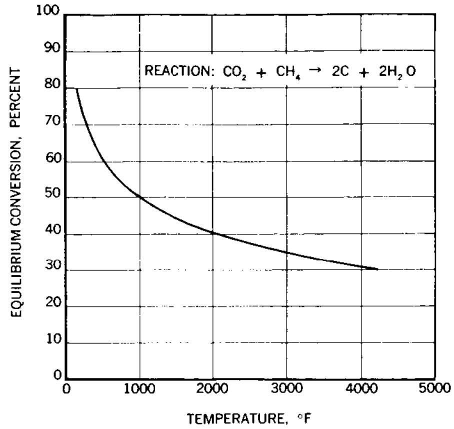

FIG. 8 - $CO_2 \cdot CH_4$ Conversion as a Function of Temperature

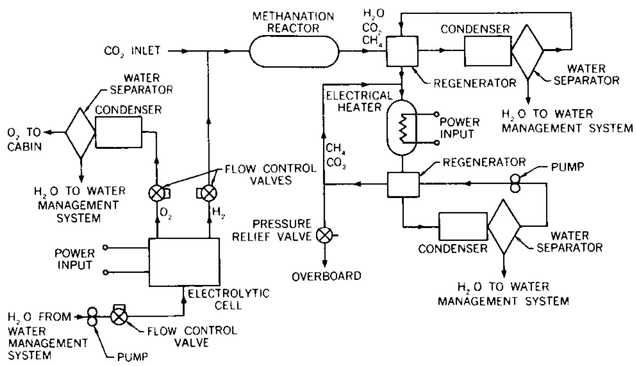

FIG. 9 - Oxygen Recovery by CO_2 Methanation and Reaction of CO_2 and CH_4

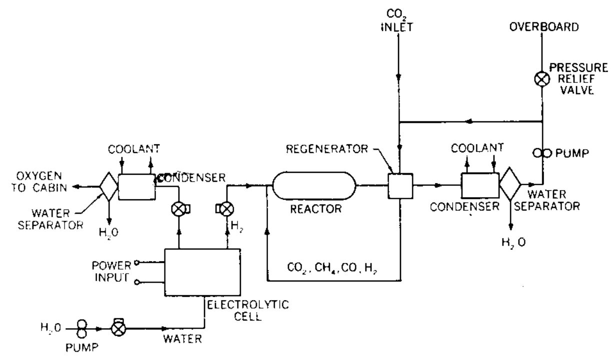

FIG. 10 Oxygen Recovery By Reverse Water Gas Reaction

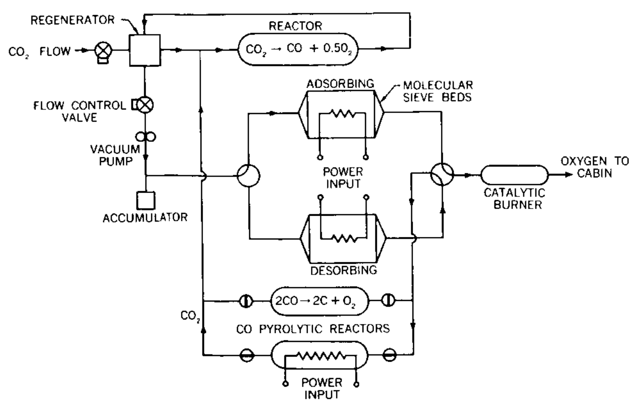

FIG 11 - Catalytic Decomposition of Carbon Dioxide to Carbon and Oxygen

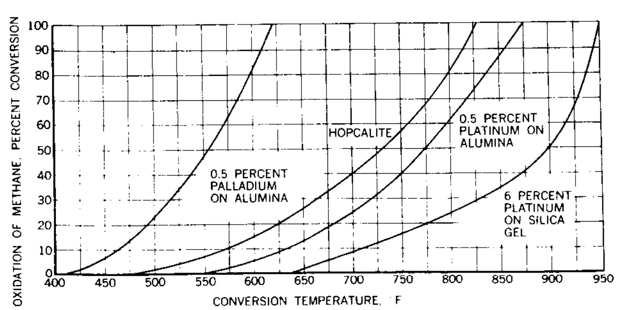

FIG. 12 Catalytic Methane Conversion

STATUS AND FUTURE ENGINEERING PROBLEMS
OF ELECTRIC PROPULSION SYSTEMS

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Introduction

Many types and sizes of chemical rockets are now in use after many years of development. Nuclear rockets though not yet a reality, are expected to become available in the next five to ten years. Why, then, the interest in electric propulsion?

The answer lies in the high values of specific impulse that can be achieved with electric propulsion. Chemical rockets are very simple and light weight, but they are limited in specific impulse to 400 or 500 seconds. As a consequence, an extremely large portion of the total vehicle weight must be devoted to propellant, limiting the payload. The nuclear rocket is somewhat better than a chemical rocket, with a potential of achieving a 1,000 second specific impulse, but this may still not be high enough for the very sophisticated missions proposed for the future. The electric system represents a new approach to the propulsion problem. Rather than attempting to add thermal energy directly to the exhaust gas, the thermal energy is first transformed into electrical power. This power is then employed in any one of a number of possible thrust devices to produce vehicle acceleration. Broadly, these can be described as (1) electrothermal thrustors in which the electrical power is simply used to heat an exhaust jet providing specific impulses of 1000 to 2000 seconds, (2) electrostatic devices in which the electric power accelerates the particles by electric fields to specific impulses of several thousand seconds, and (3) magneto-hydrodynamic devices in which magnetic fields are used to accelerate a neutral plasma to specific impulses of a few thousand seconds. Because the MHD devices have a great deal of research and development work ahead of them, they will not be discussed further in this paper.

The benefits of reduced propellant consumption are not, however, obtained without cost. For a given thrust level, the electrical power that must be expended increases directly with specific impulse, and the powerplant weight increases correspondingly. Therefore, it is necessary to optimize thrust level and specific impulse in order to arrive at a balance between propellant weight and powerplant weight to yield a maximum payload. When this optimization is performed, vehicle acceleration is typically found to be on the order of 10^{-3} to 10^{-4} G's at a specific impulse from a few thousand up to near 20,000 seconds. As a result, the electric spacecraft cannot take off from a planet's surface; it must be boosted into orbit by a chemical or nuclear rocket. An additional problem resulting from the low thrust is that the electric vehicle requires long propulsion times to achieve high velocities. Thus, in contrast to high thrust trajectories, which consist of short power bursts followed by a long coast period, the electric system operates almost continuously with little or no coasting. This, naturally, imposes a stringent reliability and life requirement on the electric powerplant.

Engine Characteristics

A typical electric engine is shown schematically in Figure 1. It consists of the thrustor, propellant feed system, electrical power supply, electrical power conditioning, a nuclear-reactor heat source, and a radiator for rejecting waste heat. From mission considerations, general specifications for the engines can also be defined. Figure 2, a comparison of advanced nuclear rockets and electric propulsion for a manned Mars mission, serves as an example to point out general requirements[1, 2, 3]. In order to perform the manned Mars mission, an engine life of 10,000 to 15,000 hours is required. Specific weights for the entire engine on the order of 10 to 25 lb/KW of thrust must be achieved in order to compete successfully with the nuclear rocket for most missions. In the case of many higher energy scientific probes, a specific weight as high as 40 lb/KW would make electric propulsion superior to the nuclear rocket[4]. Reliability goals of .9 for the unmanned missions and .99 for the manned missions become extremely difficult when the long life requirements are considered.

The power levels of interest for electric-propelled spacecraft are a function of the mission type and the booster used to place the electric spacecraft in orbit. A Saturn C-1B booster can place approximately 28,000 pounds of payload in a 300-mile orbit. With this size spacecraft, 300 to 500 kilowatts of electric power can be utilized effectively by the electric engine for performing unmanned scientific probes to the planets. A Saturn C-5 booster can place 200,000 pounds of payload in a 300-mile orbit, or 85,000 pounds to escape. With this size spacecraft, 1 to 5 megawatts of electric power can be utilized by an electric engine to deliver much larger unmanned payloads to the planets. It is estimated that for a manned interplanetary mission, 1,000,000 to 2,000,000 pounds of payload must be placed in a 300-mile orbit. Ten to 30 megawatts of electric power can be utilized by the electric engine with this range of spacecraft sizes.

Thrustors

There are several electric thrustors which are sufficiently advanced to be moving into engineering phases. These several types are illustrated in Figure 3. The electrothermal thrustor has two approaches, both having nearly the same set of problems. The arc jet heats the propellant (hydrogen or ammonia) by use of an arc drawn from a center body to the chamber or nozzle walls. The resistojet heats the propellant by passing it around hot refractory metal heating coils. No arc occurs in the resistojet.

The contact-ionization electrostatic engine produces ions by passing cesium vapor through a 2000° F porous tungsten plug where the ions are formed on the surface of the tungsten. These ions are then electrostatically accelerated to a specific impulse of several thousand seconds to form the exhaust beam.

The electron-bombardment electrostatic source forms ions by a different process. Here the propellant vapor, which may be mercury, cesium, argon, xenon (or many other materials), is fed into the ionization chamber. Electrons are emitted from the cathode at high velocity. These electrons strike the propellant atoms and ionize them. The ions then drift out the back of the engine where an electrostatic field then accelerates them to a specific impulse of several thousand seconds.

The Present Status of Thrustors

The present status of thrustors for electric propulsion is quite encouraging. Useful power efficiencies are being attained and the weight of the thrustor unit is relatively good. Table 1 summarizes the conditions in March, 1963.

The thrustor specific areas are only a small fraction of the cross section of the boosters which could be used to launch a system of interest to electric propulsion. A comparison within the same family, such as between the two Lewis electron bombardment, the two Electro-Optical Systems electron bombardment, or the two Hughes contact ionization thrustors, shows that small increases in power level bring large reductions in the specific area. This area reduction comes about primarily because a change in power level is not accompanied by a change in voltage. Hence, the insulators for low power thrustors are the same size as for power thrustors, and a significant amount of space is occupied by insulators. It is unlikely that this trend will continue to much higher power levels because a near optimum size seems to exist in the neighborhood of a few kilowatts so that large powers would be attained by clustering modules rather than by scaling. This would then mean that future specific areas would not be much improved over the best values in Table 1.

The specific weight of these thrustors is too high but in the case of the electrothermal thrustors, is encouraging. The EOS unit, at 1.3 lb/KW, is the lightest electrostatic thrustor, but it is the one which has received the most light-weight design effort[5]. (It should be noted that this particular EOS thrustor has yet to be built and tested.) Even the most optimistic mission analyses do not assume power generating specific weights much lighter than 10 lb/KW. In order for electric propulsion to be competitive, the system weight must be of the order of 10 to 25 lb/KW. With the bulk of the electrostatic thrustors weighing 5 lb/KW, considerable effort on thrustors will be needed to keep the overall engine weight competitive.

Thrustor Problems

The various electric propulsion devices all have unique technology problems. These problems are either in lifetime, power efficiency, or propellant utilization, or combinations of these three.

Lifetime Until very recently, the lifetime of arc jets was only minutes but a recent AVCO test ran several hundred hours at 30 kilowatts. The development of low power (3 kilowatt) arc jets is not proceeding as well, however, primarily because of nozzle erosion. Resistojets are not quite as far along and lifetime data is not available. Contact-ionization thrustors are plagued with heater failure, ionizer embrittlement, and ionizer property changes with time. The electron-bombardment thrustors, too, have lifetime problems. The most critical area is the electron emitter. It is possible to build an emitter with the necessary lifetime, but not at a useful power efficiency. The lifetime of the accelerator structure is another problem area for both the contact and the electron bombardment ionization thrustors. The lifetimes that have been observed on some of the thrustors and components are shown in Table II. There is much more development work to be done in improving lifetime for all electric thrustors and component parts.

Power Efficiency The power efficiency of the various electric thrustors is shown in Figure 4 [6, 7, 8, 9, 10]. At the low specific impulse near 1000 seconds, the electrothermal thrustor seems likely to provide power efficiencies of better than 75%. In the specific impulse range from 1000 seconds to 5000 seconds the power efficiency is low, but above 5000 seconds the power efficiency of electric thrustors again becomes high.

The power losses in the ion rockets can almost all be assigned to the cost of producing the ions, because the acceleration of these ions is practically 100% efficient. A comparison of the theoretical ion production energy to the actually observed ion production energy for several propellants is given in Table III. When the ionization energy in a practical ion source is a hundred or more times the theoretical values, it is well worth an extensive effort to correct this loss.

Propellant Utilization Probably the most troublesome area in the ion-thrustor is that of propellant utilization. Currently, the typical experience ranges from 80% to about 95%[6, 10,11,12]. This loss of propellant causes two difficulties, the most obvious being that the wasted propellant must be placed in orbit in place of payload. Fortunately, the high specific impulse of ion thrustors makes this less of a problem than with other devices operating at low specific impulses.

For example, even at the low propellant consumptions of ion thrustors, a utilization of only 80% would lead to a decrease of payload in the parking orbit of about 6%.

Beyond any doubt, the more serious drawback of less-than-perfect propellant utilization results from the presence of neutrals in the exhaust. These neutrals are struck by the high velocity ions and ionized. In effect, this causes an ion having zero velocity to suddenly appear at a place in the acceleration system where it should have an axial velocity of thousands of feet per second directed along the line of thrust. But in this case the new ion can only accelerate toward the accelerator structure where it will impact at high velocity causing erosion and subsequent loss of the critical electrode shape. This action can easily limit the life of the accelerators to a few hundred hours[13].

Supporting Research No specific solutions to the contact-ionization source lifetime problems are just around the corner. Much research is being conducted in these areas, primarily at Hughes Research Laboratories and Electro-Optical Systems, and progress in all areas is being made.

The electron-bombardment source lifetime problems may not be as far from solution. An NASA contract (NAS 3-2516) with EOS is for the development of a cesium electron-bombardment ionization source which employs a cesium-coated cathode. Since the propellant is also cesium, the cathode can even be bombarded by energetic cesium to maintain its temperature[6]. This configuration is very promising for long-life performance, but has yet to be tested for more than a few hours.

The lifetime of the accelerator structure in the electron-bombardment thrustor can be improved by reducing the current density in the exhaust beam. This increases the specific area and weight of the thrustors. However, accelerator lifetime and performance can also be improved through increased knowledge of the accelerator ion optics. This is being studied at Hughes Research Laboratories under an NASA contract (NAS 3-2511).

The ion production losses as well as the poor propellant utilization are being attacked by extensive analytical and the supporting experimental studies of the ionization chamber processes. Work on this part of the source is being conducted at Jet Propulsion Laboratory, Hughes Research Laboratories, Space Technology Laboratory, Electro-Optical Systems, Ion Physics, and Lewis Research Center.

The power efficiency of the arc jet is low because a stabilizing impedance must be used in series with the arc. In a D.C. arc this is a resistor which dissipates nearly as much power as the arc. Development work in this area has been pursuing stabilization of the arc before rectification of the primary A.C. power, through the use of low dissipation inductors or thyratrons, but this has yet to be demonstrated. Operations of the arc directly from A.C. has been used also but has not been very successful.

Flight Tests

The NASA flight test plans were extensively reviewed in a recent paper[14] and will not be discussed in detail here. Table IV shows the current plans for tests in both ballistic and orbiting shots.

Most, but not all questions regarding electric propulsion can be satisfactorily answered by tests in vacuum systems on the ground. For example, an ion beam in space is a stream of positive particles. In a short time (milliseconds) the vehicle would accumulate an excess of negative charges which would cause the beam to return to the vehicle. This phenomenon is not generally observed in vacuum tank tests because of free electrons in the "vacuum," but through indirect tests[15] using pulsed beams, it has been shown that neutralization of the beam can be affected in a tank. However, there are still a sufficient number of known vacuum tank-ion beam interactions to cause some doubt as to the completeness of the knowledge of the processes involved. Flight testing seems to be the only means of resolving these difficulties.

A second area needing flight test confirmation is the ability of systems, and especially propellant systems, to operate in a zero-gravity environment. This is a problem, well known to chemical propulsion systems, and is nearly as serious for the electrothermal thrustors which use a relatively large amount of propellant and, to a lesser degree, is serious for the electrostatic thrustors.

There is also a group of long-term effects, which only extended exposure as in orbiting flights, can adequately test. These include the effect of meteorites and corpuscular radiation. Aside from the direct damage that would be caused, there is a possibility that these foreign materials could trigger electrical breakdown between the high-voltage electrodes.

It is known that the impact of a high velocity particle will initiate an electrical discharge pulse. If the acceleration system of an ion thrustor is operating near its limiting electrical field strength, such a pulse could lead to a continuous discharge. This might require momentary thrustor shut down in order to clear the fault.

If foreign material either of meteorite origin or that sputtered from the structure by meteorites, accumulates on the ion emitter of a contact ionization thrustor, "poisoning" of the ionizer will result. In time, this would reduce the ion current and lead to a performance loss.

Status of Electric Power Supplies

It was shown earlier, in the discussion of missions, that the electric engine must be light weight, have high reliability, and operate for long periods of time. The electrical generating system (comprising the major portion of the engine weight) must achieve specific weights of 10 to 20 lb/KW for manned missions and 40 lb/KW for unmanned missions in order to be competitive with other propulsion schemes. The electric generating systems must also achieve the required reliability for operating times in excess of 10,000 hours. Power levels of interest are dictated by both missions and boosters, but powers in the range $\frac{1}{2}$ to 30 megawatts are presently of interest. Based on these requirements, the following can be concluded:

1. A closed cycle is required. In an open cycle, the weight the cycle fluid lost during the 10 to 30 thousand hours of operation would be prohibitive.

2. The system must utilize a nuclear heat source. A chemical heat source is too heavy for long missions. A solar heat source requires a large collector (about 70 ft^2/KW) which is heavier, because of the size, than nuclear sources at high powers. Isotopes are not available in sufficient quantities for a manned mission. A fusion reactor is not yet available, even in the laboratory. Thus, a nuclear source, specifically a fission reactor, is the only heat source of interest.

3. The powerplant must operate at very high temperatures. In space, a closed-cycle heat engine must reject waste heat by radiation. The radiator area required is directly proportional to the fourth power of the absolute temperature. At a temperature of 1500° F, approximately 1000 square feet of area are required per megawatt of electric power. In contrast, at 700° F approximately 20,000 square feet, or one-half an acre of radiating surface are required per megawatt. Thus, to achieve a tolerable radiator size, the rejection temperature must be high, which in turn also implies heat-source temperatures near 2000° F to achieve reasonable cycle efficiencies.

4. The combination of high reliability and long life is extremely severe. The testing, alone, required to develop and prove a reliability of 99% for thousands of hours is prohibitively expensive. It may well be that once a given system has been established as the best that technology can provide, the only way of achieving any better reliability is through the use of redundancy in critical components and on-board repair and maintenance.

Power Conversion Techniques

Power-conversion schemes of potential interest for electric propulsion systems are the Rankine cycle, the Brayton cycle, thermionic conversion, thermal electric, and magneto-hydrodynamics (MHD). Of these five schemes, only the Rankine cycle and the thermionic presently show promise. The Brayton cycle requires relatively low radiator temperatures, and consequently is too heavy. The thermal-electric conversion scheme operates at low temperatures, is relatively low in efficiency, and consequently is too heavy. MHD shows considerable promise, but the technology is presently not far enough advanced to realistically discuss at this time.

Rankine-Cycle Powerplant The Rankine-cycle power-conversion system shown schematically in Figure V is typified by the steam-turbine cycle used in ground power-generating systems. Thermal energy is removed from the nuclear reactor by a liquid-metal heat transfer fluid. This fluid passes through a boiler heat exchanger, where it supplies heat to vaporize the cycle working fluid. This energy is then converted by a turbine to mechanical-shaft energy driving an electric generator. Wet vapor leaving the turbine is condensed and pumped back through the boiler. The waste heat is rejected to space through a radiator. The high cycle temperatures necessitate the use of unconventional turbine fluids such as potassium, sodium, rubidium, or cesium to prevent excessive pressures. The severe thermal, environmental, and life requirements imposed on the nuclear-power supply in turn result in many severe problem areas.

Multi-megawatt space systems require the use of very high temperature reactors cooled by liquid metal to achieve light-weight reactors and shields. This imposes reactor control and fuel-element problems, both of which are under extensive investigation by the Atomic Energy Commission and its contractors.

The high-temperature alkali metals required for reactor coolants and for turbine fluids are corrosive. Ordinary materials are not satisfactory on a strength-to-weight basis. High-strength, corrosion-resistant refractory alloys and sophisticated handling techniques must thus be developed before long life can be assured. Some promising test results have been obtained with Cb-1% Zr, but there are no data for the more advanced alloys likely to be required by systems for manned missions.

Radiator areas on the order of 1,000 to 2,000 square feet per megawatt of electric power must be provided for the electric engines. Coolant must be circulated through this surface to transfer heat from the powerplant to the radiating surface. This radiator must be protected from meteoroids because punctures resulting in the loss of the engine coolant cause a powerplant failure. Integration and deployment of these very large structures from the spacecraft present additional problems.

The Rankine-cycle powerplant requires a number of high-speed rotating units, including a turbo-alternator and two or more pumps. Long-lived, high-temperature bearings must be developed for these units and complex packaging problems involving the integration of high-temperature (2000° F) turbines and low-temperature electrical components in close proximity must be solved. Additional problems are the prevention of turbine-blade erosion by the high-temperature wet cycle fluid and the development of turbine geometries and materials for the high-temperature, high-stress environment.

The electric engine requires a large number of electrical components including the alternator, controls, and power conditioning for converting the alternator output to match the electric thrustor. The present-day state-of-the-art limits most of these components to temperatures in the 100° to 200° F range. This can be a major problem when it is noted that the rejection of 35 KW of electrical losses at 140° F require approximately 1,000 sq. ft. of radiating surface. Thus, the development of high-temperature electrical components becomes a necessity.

Thermionic Powerplant A thermionic converter consists of two electrodes, a hot cathode and a relatively cold anode, as shown in Figure 6. Electrons are emitted from the hot surface and collected on the cold surface, producing a potential of about one volt. In the thermionic reactor, the converter is built into a cylindrical fuel element. Heat is supplied by fission of the fuel contained within the cathode, as shown in Figure 7. The cold electrode, or anode, is cooled by a liquid metal circulated around the thermionic fuel elements, and through the radiator where heat is rejected to space. For power systems producing megawatts of electric power, hundreds of these fuel elements are contained within the reactor and each fuel element can contain from 10 to 20 converters. The converters are connected in series and series-parallel combinations to provide a reasonable voltage output and some measure of redundancy through the parallel connections. To achieve reasonable efficiencies and system weights, heat is rejected at 1300° F to 1800° F and the cathode operated at temperatures from 3000° to 3500° F.

At present, nuclear-thermionic systems are in the very early technology phase. The program is more concerned with understanding the basic physics of the thermionic converter and in developing nuclear fuels and fuel-clad combinations which can withstand the high-temperature nuclear environment than in investigating the power system. Very little work has been undertaken on the problems of integrating a large number of converters in the reactor or toward developing a high-temperature, fast thermionic reactor. So it is too early in the development program to predict the magnitude or criticality of the problems that will be encountered in the thermionic system. An attempt is made in the following paragraphs to summarize what is presently thought to be the more critical problems.

The fuel-element converter combination seems to present the most critical development problems, primarily because it must operate at higher temperatures than any long-life nuclear reactor. Long life for the fuel-clad emitter combination may prove to be very difficult to obtain. Short-time tests have shown promise, but long-term operation in a nuclear environment where diffusion of the fuel into the clad, chemical combination of the two, vaporization of the fuel, swelling of the fuel due to fission product, etc. may cause serious difficulties.

Assuming that the problems in the fuel element can be solved, there still remain many unknowns on how converters connected in series and series-parallel combinations will act. Electrical stability of these combinations may prove to be a difficult problem.

Another problem which may be the major one encountered by thermionic systems is the ability of converter circuits to withstand failure of a single converter. Preliminary analysis has indicated that an open circuit failure of a converter in a nuclear fuel element can cause overheating of the fuel element with a high probability of catastrophic failure. To achieve reliability necessary for electric propulsion missions, it is mandatory that a system containing many thousands of individual converters be capable of tolerating the failure of one or more converters during the mission's life.

Radiator and containment material problems in the thermionic system are fairly similar to those encountered in a Rankine-cycle system. The electrical components also present similar problems. However, it should be noted that a thermionic system produces low-voltage D.C. power which must be converted to a higher voltage before it can be effectively utilized by the thrustor. This requires an inverter for converting the D.C. to A.C. power. Using the present state-of-the-art, these inverters are extremely heavy and sensitive to high-temperature nuclear environments. Considerable development is still required in this area before they can be used in a high-powered space system.

Power Supply Status The development of "moderate temperature, low power" nuclear-reactor electric power supplies is proceeding with considerable promise of ultimate success. The specific weights and power levels of power systems now under development are summarized in Figure 8, where they are compared with the estimated weights of larger, more advanced powerplants.

Atomics International under AEC contract is developing SNAP-10A and SNAP-2. SNAP-10A is a 500-watt, 900° F, nuclear reactor system incorporating thermoelectric conversion, that weighs about 800 lb/KW. SNAP-2 is a 3-KW, 1200° F nuclear reactor system, with a mercury Rankine-cycle power conversion system, that weighs approximately 250 lb/KW[16,17].

Aerojet-General, under NASA contract, is developing SNAP-8, a 35-KW, 1300° F, mercury Rankine system similar to SNAP-2. It weighs about 150 lb/KW with nominal shielding.

The SNAP-2 + 10A development reactor has been successfully operated for long periods of time of which 2000 hours were at full power. The SNAP-8 reactor should operate early this year. The SNAP-10A power-conversion system has shown some success and is currently scheduled to be integrated with the nuclear system this year, with flight tests within two years. SNAP-2 integration tests and flight tests are scheduled to follow SNAP-10A by one to two years.

The development of high-temperature high-power Rankine-cycle powerplants is presently in the research and technology phase. The program is concentrated on obtaining the basic data required before initiation of design. This includes the measurement of fluid properties and heat-transfer coefficients, the investigation of material compatibility with the alkali metals, development of fabrication techniques for refractory metal, fuel-element development, turbine-erosion investigations, the development of high-temperature electrical components, etc. Technology programs have been underway in this area since the late '50's under the Air Force (SPUR), NASA, and more recently the AEC-AF SNAP-50 programs. These programs are being funded in the tens of millions of dollars per year.

Referring to Figure 8 again, it is apparent that considerable improvement in powerplant performance is required before the goals for electric engines are achieved. Present estimates of the performance levels that may be achieved with advanced systems when redundancy, all potential losses and spacecraft integration considerations are factored into the performance estimates are also shown. These weights are considerably higher than some published values, primarily because they include 25 percent component redundancy, structure weights, power-conditioning weights and losses, and a radiator meteor probability of non-puncture of 99 percent.

Despite these conservative assumptions, the electric propulsion system still offers potential for manned missions if long life can be obtained.

Conclusions

In spite of all the problem areas being studied, there are some which are recognized but about which little has been accomplished and doubtless still many more which have yet to be defined.

It was pointed out that in order to design an electric engine system of long flight capabilities, a multipronged research, development, and testing effort of extensive proportions must be completed. The success of all parts of the electric propulsion system depends, as do all aspects of space propulsion, on the development of high temperature, high strength, low weight structural materials. Important electric thrustor component problems are the heater, the ionizer, and the accelerator structures. The power supplies must utilize corrosive liquid metals and be designed and qualified to operate in a high temperature, nuclear environment. To establish that the total system has the required reliability for many thousands of hours, it will be necessary to test all of the components, subsystems and final systems many more thousands of hours.

Systems will have to incorporate redundancy concepts and a tolerance of component failures and may have to be capable of onboard repair and maintenance in flight. The systems will also have to remain operable and useful despite possible deterioration in performance which aging will produce in components and subsystems.

The state-of-the-art for small electric thrustors is promising and with the possible exception of the lifetime problem, it should not be difficult to extrapolate the present performance to higher power levels. Unfortunately, the nuclear electric power supply will present considerably more difficulty. A number of low power sources are being developed but these are generally characterized by low temperature and high specific weight. The technology will require considerable extension before the light-weight systems needed for electric propulsion are available. Progress in securing and expanding the technology is being made steadily and it is expected that the technology necessary to allow the initiation of the design of advanced systems for electric propulsion will be available in the late 1960's.

It has been shown that if the research and development programs now being pursued prove successful and result in a long-lived electric engine system, a unique and important element will have been added to our future space propulsion capability.

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FIG. 1 ELECTRIC ENGINE SCHEMATIC

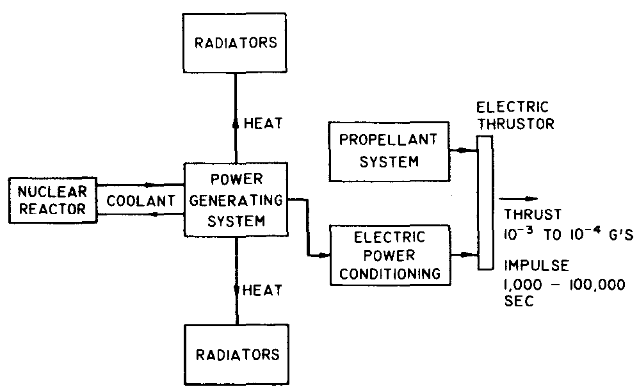


FIG. 2 MANNED MARS MISSION

CREW SHIELDING FOR 100 REM DOSE, METEOROID SHIELDING FOR P_o = 0.999

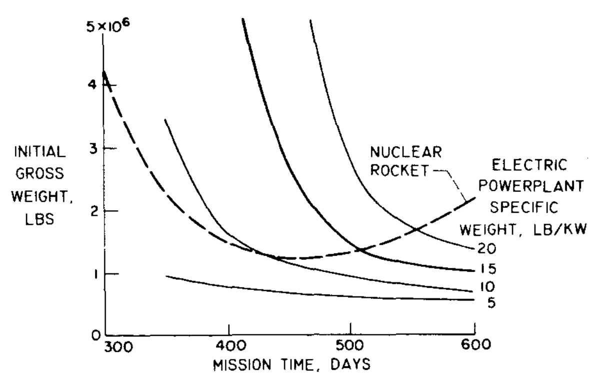


TABLE I

PRESENT STATUS OF THRUSTORS

Thrustor	Approximate Power (KW)	Specific Impulse (Sec)	Power Efficiency %	Specific Area (Ft^2/KW)	Specific Weight (Lb/KW)	Reference
Lewis Resistojet	15.0	850	75	0.003	1.0	7
AVCO Arc Jet	30.0	1050	45	0.0004	0.22	8
TAPCO Electron Bombardment (1)	3.0	5000	75	0.05	3.3	18
EOS Electron Bombardment	1.0	5500	82	0.34	4.0	5
EOS Electron Bombardment (1)	2.3	7000	75	0.10	1.3	5
Lewis Electron Bombardment	0.5	5000	70	0.88	6.0	19
Lewis Electron Bombardment	2.0	5900	75	0.17	5.0	20
Hughes Contact Ionization (1)	0.5	4500	40	0.54	4.1	22
Hughes Contact Ionization (1)	2.5	7600	62	0.08	4.8	21

Note: (1) These performance figures have not been attained (4-63), but are believed to be realistic estimates of performance within 12 months.

TABLE II

LIFE TESTS

Item	Observed Lifetime
AVCO Arc Jet (30 KW)	720 hours
Lewis Electron Bombardment (4 MLB, 5000 sec)	150 hours
Hughes Contact Ionization (3 MLB, 8000 sec)	125 hours
IPC Electron Bombardment Engine Cathode Heater	2000 hours
Hughes Contact Ionization Ionizer Button	8000 hours

TABLE III

IONIZATION ENERGY

(ELECTRON VOLTS)

Propellant	Theory	Practice
Argon	15.7	600
Cesium (Electron Bombardment)	3.9	600
Cesium (Contact Ionization)	3.9	1500
Mercury	10.4	500 to 800

TABLE IV

NASA FLIGHT TEST PLANS

FOR ELECTRIC PROPULSION

Name	Date	Power	Objective
SERT I	Late '63	1/2 KW	Thrust and Neutralization in Space
SERT II	Mid '66	3 KW	Long Term Space Effects
SERT III	Mid '65	1 KW	Attitude Control by Ion Rockets

FIG. 3 ELECTRIC THRUSTORS

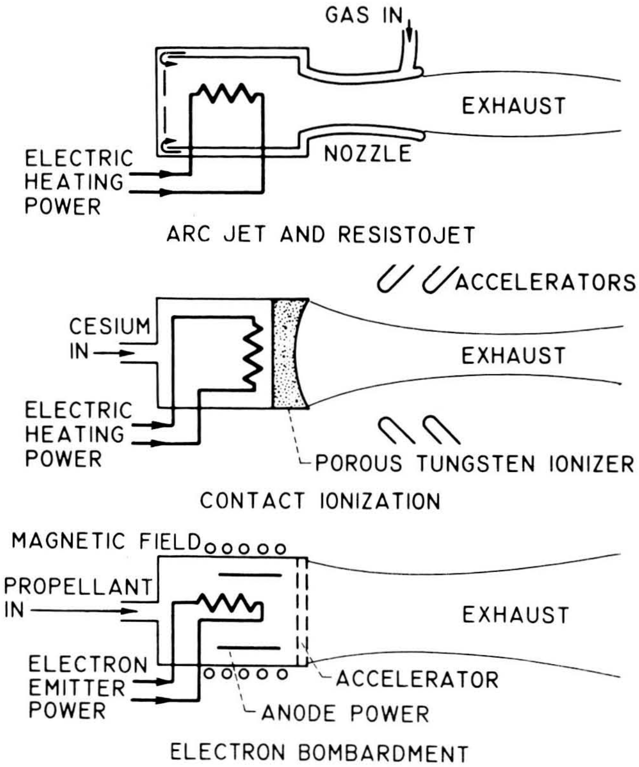


FIG. 4 THRUSTOR POWER EFFICIENCY

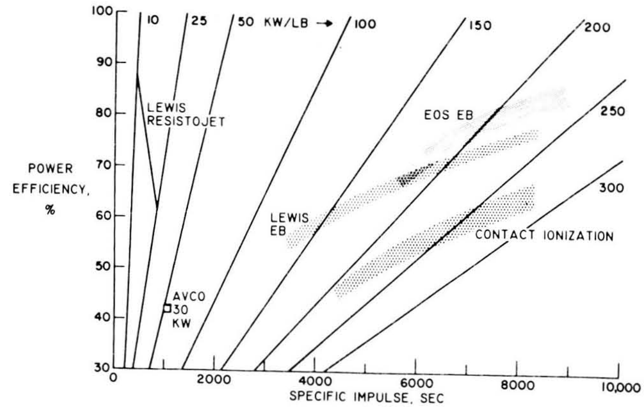


FIG. 5 SCHEMATIC OF RANKINE CYCLE SPACE POWER SYSTEM

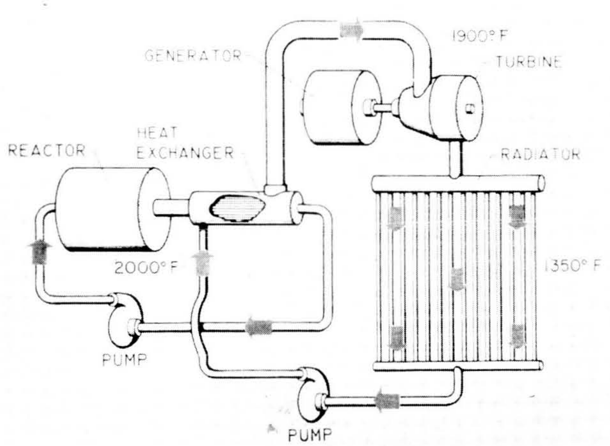


FIG. 6 THERMIONIC CONVERTER

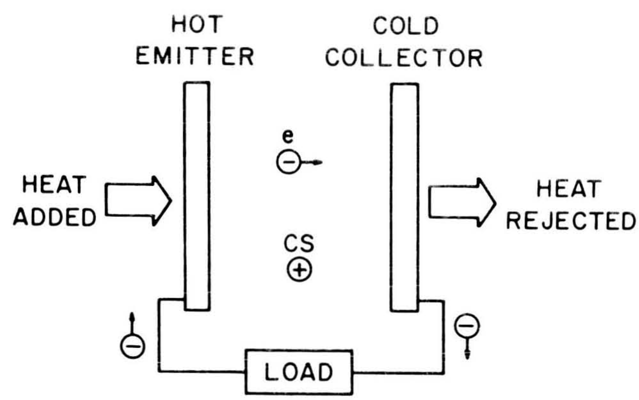


FIG. 7 NUCLEAR THERMIONIC POWER SYSTEM

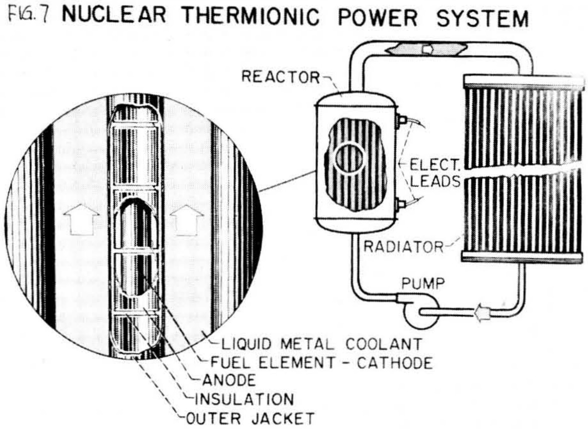
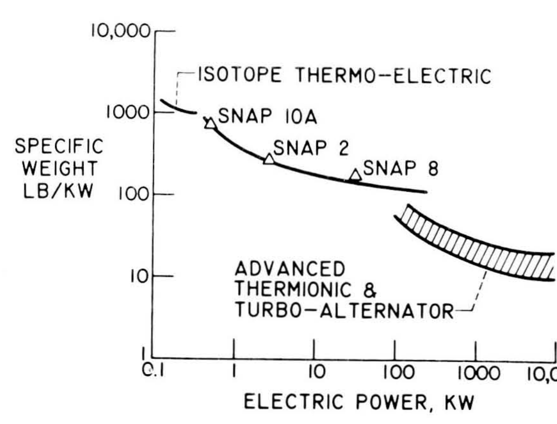


FIG. 8
ESTIMATED SPECIFIC WEIGHTS OF NUCLEAR POWER SYSTEMS



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INTRODUCTION

Design requirements of a vehicle or system are linked integrally to the overall mission objectives for which the vehicle or system is being developed. Therefore, before characteristics of future manned spacecraft are discussed, a brief review of these objectives is in order.

The existence of an aggressive space program for both exploration and possible military missions is beyond question. Vast sums of money have been appropriated by the government for research and development. Large and small administrative organizations have been established to direct and coordinate the efforts of industry and the military in the advancement of space technology. The largest peacetime buildup of technological capabilities in history is under way. Within the next decade or two, at the present pace, manned deep-space exploration in the interest of scientific advancement is inevitable. This trend will continue and accelerate as new knowledge of space is developed.

Manned military missions in space present a problem; they are difficult to justify now. It is historically true, however, that all major technology advancements have important effects on a country's military strength and direction. The obvious analogy to justify manned military space systems is the military history of the airplane. Though the parallel of the airplane and the manned re-entry vehicle is bound to be faulty here and there, to ignore the possible lessons to be derived from such a comparison could be unwise.

Therefore, operational considerations for military as well as scientific missions are considered in this paper. Since the use of manned spacecraft for military missions is still uncertain, it is expected that during the foreseeable future the character of manned space programs will be mostly research and development, involving a limited number of vehicle designs, numerous and frequent changes, and a constantly increasing emphasis on cost and performance improvement.

SPACE PROGRAM PLAN

In Figure 1 is shown, in condensed form, the estimated schedule for future manned space program development to be used for this discussion. The timing shown may be optimistic for the near-term events and lacking sufficient perspective for events a decade from now, but the interrelationships shown are valid. The plan assumes an orderly buildup of technological capability, environmental data, and operational experience in each area (i.e., high orbits, space stations, lunar programs, etc.). No effect of political pressures or other unforeseen stimuli is considered on the timing or phasing of the various programs.

Examination of these programs reveals a prime interest in future high-orbit missions for both military capability and research. It is believed that this interest will result from the vulnerability aspects of low-orbiting spacecraft to hostile action and the desire to explore regions more remote than near-Earth orbits.

SPACE PROGRAM PLAN

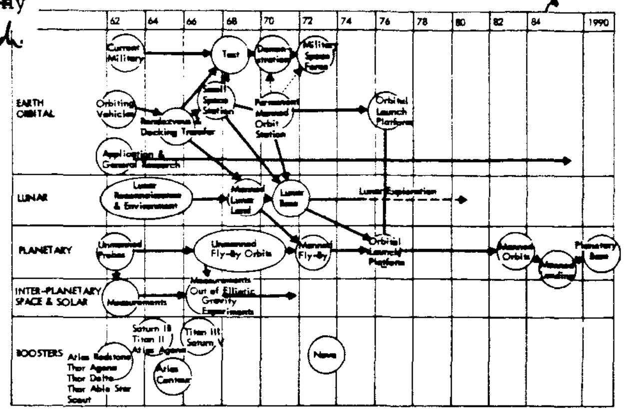


Figure 1

The programs shown will include Earth-to-orbit shuttle-type spacecraft, permanent orbiting stations, orbit-to-orbit shuttles, long-range interspace transports, and fixed lunar bases. It is important that these programs be developed in an orderly fashion, with sufficient emphasis on securing environmental data early upon which realistic engineering design can be formulated. Also, it would be wise to develop our knowledge of man's capabilities in logical steps. Here it will be advisable to develop a small space station early in which variable-g effects for long periods could be studied. Then, after exploration of the moon has been accomplished, and its value as a potential space base is more fully understood, the decision regarding use of a large space station or orbital assembly, or a lunar staging base for space explorations can be made with confidence.

RELATIONSHIP OF DESIGN TO MISSION REQUIREMENT

In the past, because of severe booster limitations, spacecraft design was constrained within narrow bounds. With the advent of C-1, C-5, and Titan III launch systems, we now have more flexibility for several alternate approaches — the choice will depend on accurate assessment of mission requirements. The impact of varying these requirements is readily seen in the case of possible transportation systems to support lunar exploration. Combinations of lunar excursion modules, alone or combined with direct shot lunar logistic vehicles, are feasible. A study was made to compare these modes, assuming that personnel are carried by lunar excursion modules only and cargo by either means.

The concept of lunar transportation systems that combine crew- and cargo-carrying capability complicates the determination of the relative efficiency of various system concepts. This results principally from the uncertainty in annual crew and cargo transportation requirements for lunar-base support and is graphically shown in Figure 2.

Three different-size lunar excursion modules were assumed for the study to carry the crew, and the cargo was assumed carried by either the LEM's or by a lunar logistic vehicle. A comparison was then made on the basis of C-5 firings required to support lunar missions of varying size.

The performance of all-crew system concepts is represented by horizontal lines, with intercepts equal to the reciprocal of crew number carried per launching. Also, all-cargo system concepts are represented by straight lines through the origin, with slope equal to the reciprocal of cargo transported per launching. In either of these cases, the relative efficiency of various systems is unequivocal, being independent of mission requirements.

LUNAR BASE SUPPLY PARAMETERS

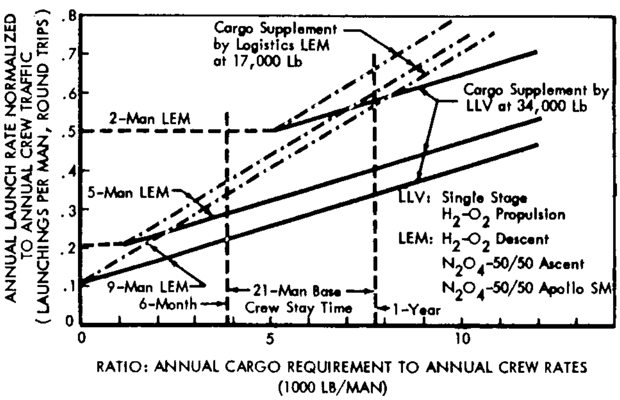


Figure 2

However, when system elements that combine crew and cargo capability are introduced into the total transportation concept, the relative efficiencies become mission-sensitive. This is illustrated above by showing the nondimensionalized launch rate requirements for families of LEM/logistics, LEM, and LEM/LLV transportation systems as a function of the required ratio of cargo to crew.

The following significant observations may be derived from these data: The relative advantage of multiple-man transportation systems is a strong function of the crew/cargo ratio. Thus, for crew transportation only, a two-man LEM suffers a 2.5-to-1 launch rate penalty over a five-man LEM. For support of a nominal 21-man lunar base with one-year crew rotation interval, this penalty is decreased to 1.09-to-1 (with logistics LEM supplement) or 1.42-to-1 (with LLV supplement). Therefore, the choice for selection will depend upon the initial R&D costs for system development and the numbers of launches required.

ENVIRONMENTAL DATA REQUIREMENTS

Meteoroids. The National Aeronautics and Space Administration has a well-conceived program for obtaining environmental data of all types. From an engineering design standpoint, however, major data deficiencies still exist. For instance, with regard to meteoroids the existing data is plotted on Figures 3 and 3A in terms of meteoroid flux versus particle mass. The actual data points from U.S. satellites gathering this data are shown in comparison to the current estimated variation. It will be noted that the band of uncertainty encompasses five orders of magnitude and that considerable extrapolation is required to reach the minimum size particle that will cause structural damage

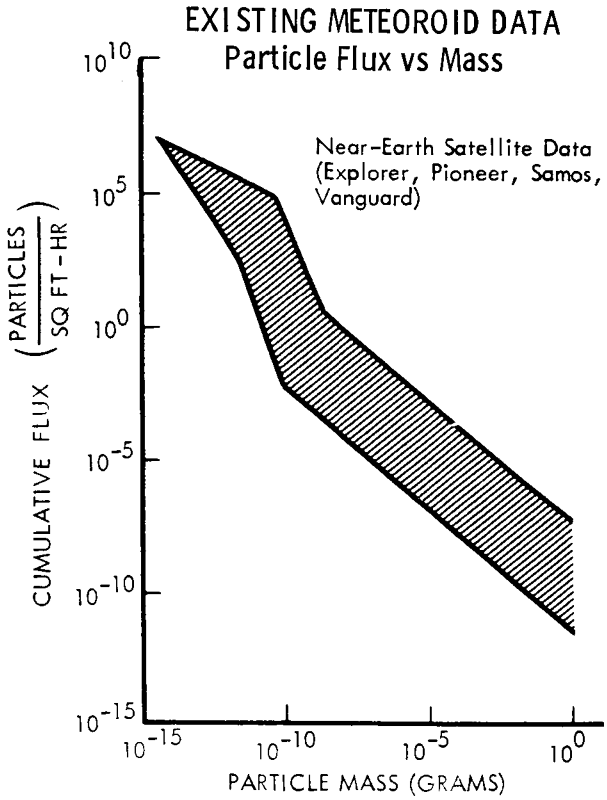


Figure 3

— 10^{-5} grams. Further, most of this data is secured in near-Earth orbit; very little information is known in the region of the Moon or cislunar space (Figure 3). There is little data available that can be used for predicting meteoroid storms or planning space flights. The meteoroid protection required for a space vehicle cannot be predicted with confidence based on available data today. A satellite program capable of measuring particles weighing 10^{-5} grams and larger is required.

The effects on human occupants of high-velocity particle penetrations of pressure chambers need further study. This should include evaluation of the spallation or fragmentation hazards. It is expected that the criteria for the pressurized compartments of spacecraft will be similar to those for a commercial jet aircraft with regard to damage protection (i.e., the pressure vessel designed to be penetrated by particles weighing as much as a gram without catastrophic rupture), and, in addition, provisions for repairing this damage on board will be required.

Radiation. Radiation hazards caused by solar flares, Van Allen belt radiation, or nuclear bursts are well recognized, but it is difficult to establish a required level of protection. Vehicles operating in the region below the Van Allen belt probably will not carry the additional weight of shielding required for any man-made radiation. It is difficult to see any need for operation of manned spacecraft for extended exposure to the Van Allen belt radiation. Therefore, the radiation hazard of most concern is solar flare activity for cislunar and space flight. It is recognized that better radiation data is required, and NASA has a program to develop it. In addition, the criteria for which protection must be provided are not well understood.

EXISTING METEOROID DATA
Particle Flux vs Altitude

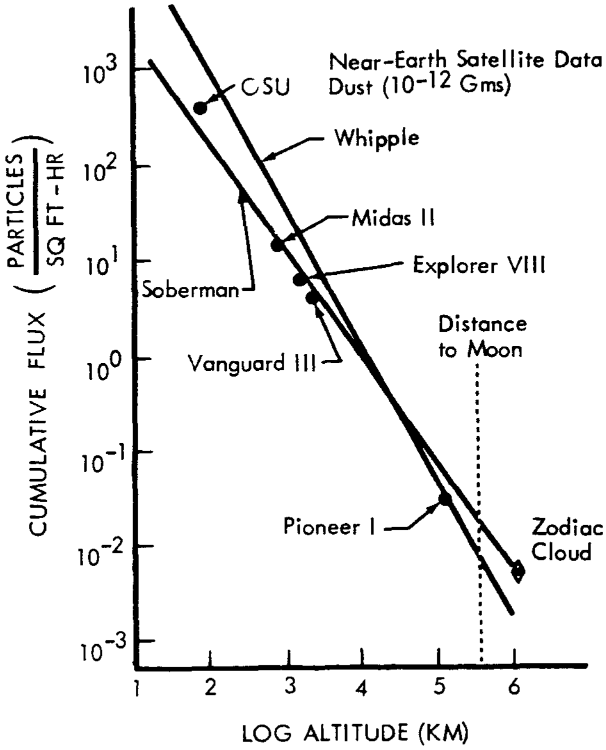


Figure 3A

To illustrate the problem, a design analysis was made for a level of solar flare activity approximating an average level between the low- and high-level periods of flare activity. It was assumed that flights of long duration would be made during those periods in cislunar space, and that the degradation of the probability of mission success would be determined for different amounts of radiation shielding. A peak survival value of .995 was selected as being representative of the maximum degradation level that might be tolerated for any one of the many other kinds of potential failures affecting mission success. Figure 4 shows the amount of protection required versus survival probability. Survival is defined as receiving less than 400 rads of dosage at a 4-cm body tissue depth. The mission length is assumed to be one month. Short-time flare detection will reduce shielding requirements. Provision for localized protected areas will doubtless be required for future spacecraft. The astronauts can enter them and ride out solar radiation storms. The data upon which this analysis is based varies greatly with calender time and, in order to have the confidence one should have in the validity of the results, more information is required.

OPERATIONAL ALTITUDES

The Apollo program represents a major milestone in man's astronautical advancement even greater than the Mercury flights. It is man's first true space exploration and is the forerunner of manned space operations through regions remote from the Earth.

SOLAR-FLARE PROTON SURVIVAL PROBABILITY PER MONTH AS A FUNCTION OF ALUMINUM SHIELD THICKNESS

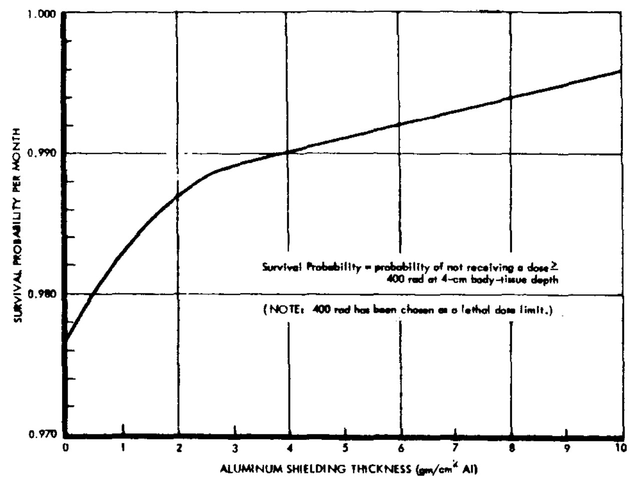


Figure 4

Also important is the potential these remote regions provide for possible manned military systems. The vulnerability of manned space systems will require their dispersal to the remote regions of cislunar space to escape enemy detection. In Figure 5 is plotted the expected detection range for advanced radars versus relative radar cross sections as related to small manned spacecraft. It can be seen that very high altitudes will be required to escape radar detection. Future development of optical scanners and trackers will increase the detection altitudes still further.

ANTICIPATED DETECTION RANGE — FUTURE RADARS FOR 10^{-5} PROBABILITY OF DETECTION

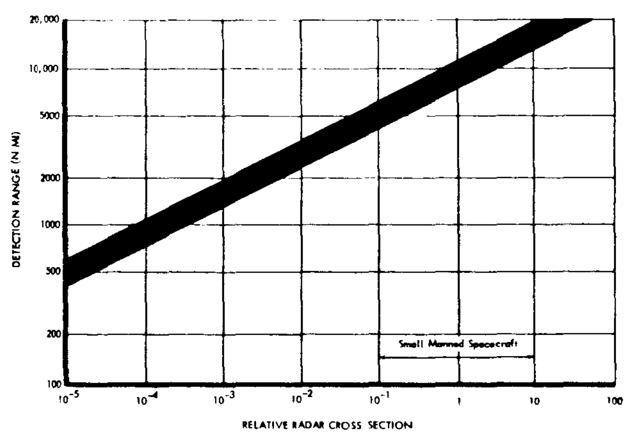


Figure 5

Shaping and coatings can drastically modify the radar return from the spacecraft. Even with the best techniques, however, avoidance of detection is most difficult at low altitudes if continuous-search methods are used. Decoys can be used to make identification of the real target difficult, but they become complex if they are required to be effective for extended periods. They will be effective, however, for flyover-type missions. Therefore, for manned military systems having extended mission durations, high-altitude operation will be required.

SUPERORBITAL OPERATIONS

In development of flight operations for high-altitude orbit injection and return, velocities ranging from 30,000 to 45,000 feet per second must be considered. These are some of the basic problem areas, from an operational sense, which then result in:

1) Greater launch and injection propulsion requirements;
2) Larger space maneuvering requirements;
3) Wider re-entry corridor width;
4) Improved landing site acquisition;
5) All weather landing.

They will be discussed to show their impact on spacecraft trends and design requirements. It is recognized that mission performance and equipment requirements are equally important in selection of the specific spacecraft, but since missions beyond Apollo are as yet barely defined, no discussion of these problems will be attempted.

Launch and Injection Operation Trend

Many studies have been made comparing the various ways of putting large amounts of payloads into orbit, including reuseable launch systems, large expendable systems, and modular launch vehicles. Studies have also been made, from the cost standpoint, to evaluate the use of Earth-orbit rendezvous techniques for spacecraft assembly. It has been found that rendezvous assembly is by far the most economical method for putting large amounts of payload in orbit using expendable boosters. Most studies today show that for such large amounts of payloads in orbit, recoverable boosters offer even greater cost savings. In either case, rendez vous is an integral part of assembly of large payloads, the only limitation being the maximum number of flights that are operationally feasible for that mission.

Assumptions and ground rules used for a recent study are listed below:

1) LO_2-LH_2 two-stage boosters, steel construction, single thrust chamber;
2) 300-nautical-mile, circular-orbit mission;
3) Booster reliability of 1.0 for the boost phase;
4) Cost and booster payload data based on present-day estimates;
5) Payload boosted into orbit in more than one piece and assembled by rendezvous technique. The rendezvous maneuver penalizes the payload weight and reduces reliability. The payload weight penalty was assumed to be 0.1 and 0.2 as defined by the ratio of unsuccessful tries divided by total tries;
6) Cost figures, including vehicle, facilities, GSE, R&D, and propellant costs. Each vehicle was costed independently of other sizes.

The results of this study, shown in Figure 6, show a cost advantage for multiple launches and rendezvous over the total payload range. One important factor causing this is the high initial R&D cost for developing a new booster and indicates the value of getting as much use from a design as possible. It also illustrates the advantages of uprating boosters until the cost of uprating approaches the cost of developing a new larger booster.

PAYLOAD COST AT RENDEZVOUS RELIABILITY OF 0.8

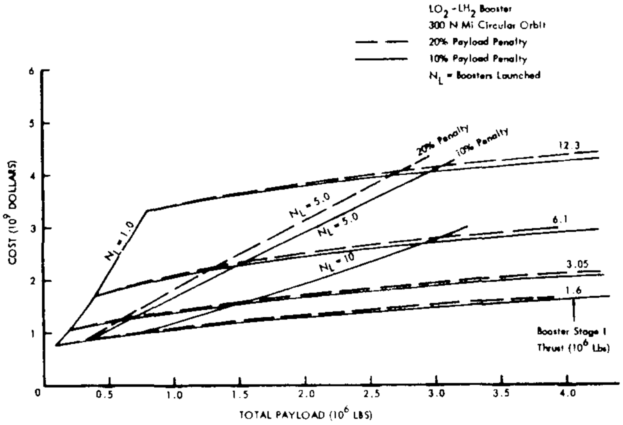


Figure 6

Booster reliability was assumed to be 1.0 for this study. If this were degraded a certain increment, the curve set would shift upward uniformly and not change the basic conclusions. In actuality the booster reliability is poorer for the initial flights, becoming asymptotic to some high level as the number of launches increases. Thus, if a reliability factor that follows this trend were used, the advantage would be in favor of large number of launches and favor the smaller booster cost comparison even more.

Space Propulsion Trend

Analyses of the many potential manned missions in space reveal very few cases in which space propulsion will not become necessary. This holds true for both military and scientific activities. Propulsion systems considered, of course, run the gamut from electromagnetic to nuclear, and from integral installations such as the Aerospaceplane to separable propulsion units.

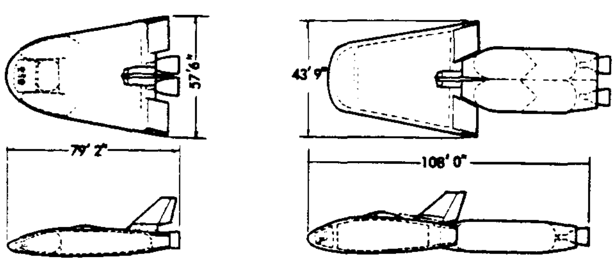
Other than for interplanetary missions it appears that the extremely low thrust-to-weight ratio propulsion systems have limited application. For cislunar operations there will be required thrust levels ranging from 0.10 to 1.0, preferably with a throttling range of approximately 10 to 1. The ratio of the national propulsion effort aimed at chemical or nuclear propulsion to meet such a requirement is proportionally too low when compared to the efforts on the extremely low-level thrust-level types.

There is considerable debate in the industry concerning onboard space propulsion versus separable units. Aerospaceplane advocates envision a completely recoverable space system, which they believe will be the ultimate development of space activity when it reaches large-scale operation. It becomes a matter of economics much the same as the recoverable first-stage boost problem, although operational problems will no doubt be an important factor. Without a completely recoverable space system, it is possible to visualize hundreds of spacecraft in operation, with man-made debris scattered throughout space. The problems of traffic control or surveillance would be impossible in such a case.

A study was conducted to assess the comparison between an all-recoverable upper stage containing on-board propulsion and an equivalent spacecraft with a separable propulsion unit. Each was assumed to carry 10,000 pounds of payload in a 10,000-cubic-foot payload compartment. The propulsion system was sized to provide a 21,000-fps velocity increment in both instances. Two conditions were then evaluated. The first can assume an initial stage propulsion system providing 5000 to 6000 fps velocity. The upper stage then provides the impulse for injection into a 300-nautical-mile circular orbit. The second case assumes that the upper stage is injected by other means into the same orbit so that the 21,000-fps increment is available for space travel and return.

In Figure 7 are shown the results of this study. In the first case the total weight in orbit is 126,000 pounds for the recoverable system and 95,000 pounds for the nonrecoverable units. For each flight, the weight differential is 31,000 pounds in favor of the disposable propulsion system. Inasmuch as the disposable propulsion stage weight was 28,000 pounds, 1.1 pounds of vehicle weight were required to recover each pound of booster. Or in other terms, if the cost of placing 1 pound into orbit can be reduced to $100 per pound, as advocates of recoverable boost systems promise, then it will be an economic advantage to use the all-recoverable system if the cost of the disposable propulsion unit is more than $110 per pound unit weight.

INTEGRAL VS EXPENDABLE PROPULSION
Aerospaceplane Second Stage



10,000 LBS	PAYLOAD WEIGHT	10,000 LBS
10,000 CU FT	PAYLOAD VOLUME	10,000 CU FT
21,000 FPS	ΔV	21,000 FPS
370,000 LBS LF_2/LH_2	PROPELLANT	251,000 LBS LF_2/LH_2
—	EXP. PROP. SYS	28,000 LBS
126,000 LBS	RE-ENTRY WEIGHT	67,000 LBS

NOTE: 59,000 lbs are required to recover 28,000 lbs of expendable propulsion system or 1-1 lbs of vehicle required to recover 1 lb of empty booster

Figure 7

In the second case, the total weight in orbit for the recoverable system is 496,000 pounds and for the nonrecoverable system, 320,000 pounds. The difference in favor of the nonrecoverable system is 176,000 pounds. Again, if the cost per pound in orbit is assumed to be $100, the economic advantage will be in favor of the recoverable system, if the cost of the disposable propulsion stage exceeds $630 per pound unit weight. For reference it is believed that the cost per pound of an empty stage presently approximates $500 to $1000, the cost to launch a pound into orbit is in the $1000 plus range. The economic advantage of the disposable system will then be considerably greater, based on today's launch costs, than that shown by the above analysis.

The analysis reveals another trend. The relative advantage of a disposable propulsion unit increases with the amount of space propulsion being provided. For instance, in the first case approximately 1500-fps orbital velocity was provided and in the second case 21,000 fps. The break-even unit cost of the disposable stage ran from $110 to $630 per pound. This indicates that low-orbit vehicles with a minimum propulsion requirement on all recoverable spacecraft will make sense when launch costs are reduced to attainable values; but for high-orbit space operations requiring velocities of 35,000 fps and higher, the disposable propulsion unit will be used. The use of nuclear stages will also favor separation of spacecraft from the propulsion unit. Figure 7 shows a comparative layout of the all-recoverable and separable spacecraft.

Re-entry Vehicles Trend

The question of optimum re-entry vehicle configuration approach is one that is not yet finally settled, and it may never be settled to the satisfaction of all concerned. The proponents of near-ballistic shapes emphasize the superior weight efficiency and basic simplicity of their vehicles. On the other hand, advocates of lifting re-entry vehicles, such as X-20, emphasize re-entry flexibility comfort and horizontal landing. The arguments consider only part of the mission. There is no perfect all-purpose re-entry shape any more than there is a perfect all-purpose airplane. The choice will develop around the mission requirements. We could design higher performance into an airplane if we were willing to compromise its takeoff, landing, and hold requirements. Likewise, an airplane optimized for landing and takeoff would be a poor high-performance fighter unless some form of variable geometry is used.

The weight advantage of a low L/D re-entry vehicle is illustrated in Figure 8. The data shown is for a series of vehicles configurated for a common set of ground rules and for two re-entry speeds. A vehicle with an L/D of 1.5 will weigh approximately 30 percent more than one having an L/D of 0.5. This ratio holds true for two- and three-man vehicles, but will become smaller as the size of the spacecraft increases. The additional weight results from the heat protection system and control system requirements based on today's knowledge. The projection of state-of-art development also reduces this ratio.

RE-ENTRY MANEUVERABILITY WEIGHT REQUIREMENTS

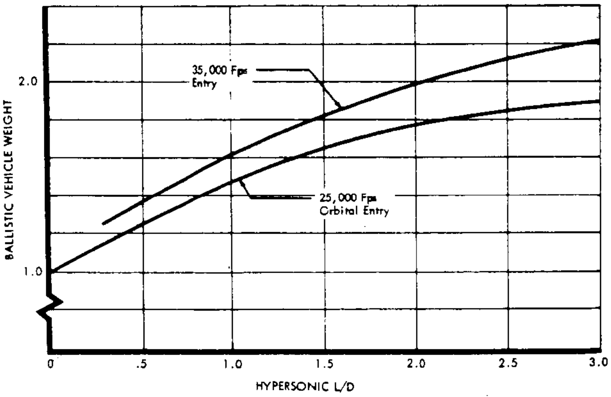
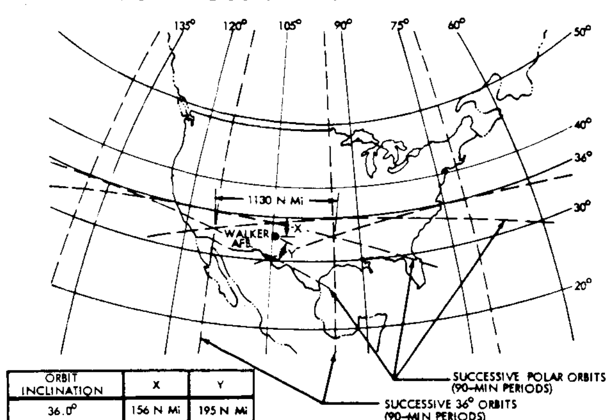


Figure 8

An acceptable level of operational flexibility can be achieved by low L/D space vehicles by the choice of orbit altitudes and inclinations. In Figure 9 is shown orbit paths for an inclination of 36 degrees. A side range of only 200 nautical miles is required to be able to land at Walker Air Force Base in Texas from any of four orbits, For a 34-degree orbit a recovery can be made with a vehicle having a side range of approximately 35 nautical miles.

36° ORBIT TRACK CROSSINGS



ORBIT INCLINATION	X	Y
36.0°	156 N Mi	195 N Mi
36.5°	186 N Mi	170 N Mi

Figure 9

On the other hand, to return to such a base from a 90-degree orbit will require a side range exceeding 600 nautical miles for two orbits each day. Likewise, to return to the same base from an equatorial orbit will require a side range of approximately 2000 nautical miles. This side range can be achieved either by orbit plane change prior to re-entry or by maneuver during re-entry. Our trade studies have shown that for low side ranges the minimum weight system has a low L/D. For side ranges over 700 to 1,000 nautical miles the advantage favors vehicles that can achieve the required side range using maneuverability during re-entry. So again, operational mission requirements must be defined before an approach is selected. It is hard to believe that at this stage of space development we should reduce our efforts to understand better the capabilities of both. Understanding connotes actual flight development, not merely analytical trade studies and cost-effectiveness comparisons.

When re-entry from high orbits is considered, the problem of achieving a re-entry window width compatible with guidance and control capabilities is difficult. The Mercury vehicle is capable of entering the Earth's atmosphere and has sufficient corridor width at 25,000 fps without the use of lift. The Apollo vehicle, on the other hand, to maintain an acceptable minimum corridor for re-entry from the Moon, will require an L/D equal to 0.5. If the re-entry velocity is to be increased, deceleration force reduced, or corridor width increased, L/D values must be increased. The use of propulsion for providing this tolerance is no longer competive from the weight standpoint, since the impulse to change the direction of a body traveling at 35,000 fps has become prohibitive. Figure 10 shows the variation of corridor width for various values of L/D at different re-entry speeds. A maximum force of 6 g's was used to limit allowable maneuvers so as to keep the comfort level within acceptable values. No lift modulation was assumed. If modulation is considered, significant increase in corridor width results for the higher L/D.

L/D REQUIRED FOR 6-G CORRIDOR

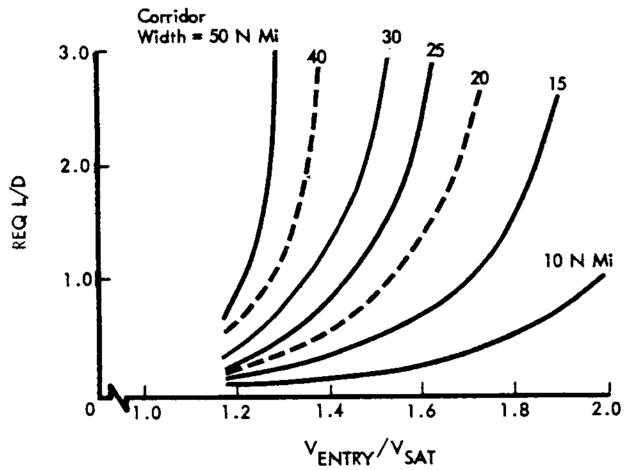


Figure 10

When the total problem of re-entry from superorbital flight is considered (i.e., (a) meeting the extremely small tolerance between skip-out and exceeding either the g limits or temperature limits, (b) being on the orbit which passes near the desired landing site on first pass, (c) decelerating at the right rate to stop exactly at the site rather than undershooting or overshooting due to angle of entry, atmospheric variations, or timing, and (d) having weather conditions sufficiently good to land), the only possible conclusion is that research should be aimed toward opening up the tolerance to much wider limits than those attainable with L/D of 0.5 for future operational flexibility.

Re-entry Vehicles — Landing Phase

Although specific operational requirements will dictate the terminal recovery of future spacecraft, the trend will be toward greater range control to correct re-entry errors. Experience with X-20 will show it to have this capability satisfactory for a research program. It may also be adequate for operational landing systems as well that depend on ground-controlled guidance used for the terminal phase. However, improvements in the terminal phase L/D can and will be developed by use of thrust, variable geometry, or base drag reduction.

A good design must seek resolution of the conflicting design requirements of re-entry and landing. For re-entry, the most efficient spacecraft will minimize the surface area exposed to high heating. For the landing phase, the L/D should be high enough to give the pilot time to evaluate his position with reference to the landing field at "high key" and also at "low key," to have visual reference of the end of the runway. With an L/D of 4.5 the high and low key positions are 30,000 and 15,000 feet, respectively. Raising the L/D to 10 changes high key to 4000 and low key to 2000 feet. If pilots are to control the spacecraft manually, this reduction in high and low key altitudes becomes most important, particularly in variable weather conditions. Future systems doubtless will utilize some form of automatic landing guidance that will minimize pilot function. However, even with automatic terminal guidance, the vehicle will require a high enough L/D to allow the pilot to take over guidance function if he wishes. Otherwise he is nothing more than a passenger, and it is difficult to believe that such degree of dependence upon automation is desirable in the foreseeable future.

Wind-tunnel tests have been conducted on many variable geometry configurations to examine the gain in subsonic L/D possible for reasonable weights. In Figure 11 is shown the remarkable improvement possible at less than 5 percent increase in vehicle weight for a configuration that has a hypersonic L/D equal to approximately 1.25. With such an L/D, the pilot will be able to make range corrections of 200 miles after reaching transonic speeds. This method of providing range and heading corrections of such magnitudes was found to be the lightest of the many studied and one that could also reliably be deployed, with the spacecraft controllable to the desired limits. It is believed the re-entry spacecraft of the future will incorporate such variable geometry devices.

If improved L/D is desired only to correct the flight path during approach phase prior to flare and during flare, rocket thrust augmentation can be used. From the weight standpoint, such installations can be competitive with variable geometry designs, but they lack the large range-correction potential. They are, however, simpler and require less R&D for availability. The effect on the basic L/D of a vehicle for various thrust rockets is shown in Figure 12.

VARIABLE GEOMETRY RE-ENTRY VEHICLES
L/D Characteristics

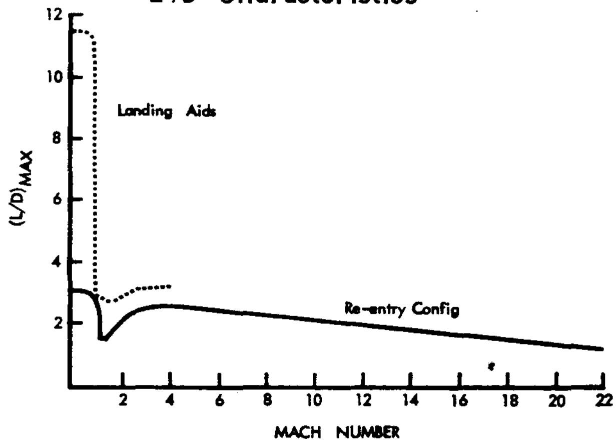


Figure 11

THRUST-AUGMENTED L/D

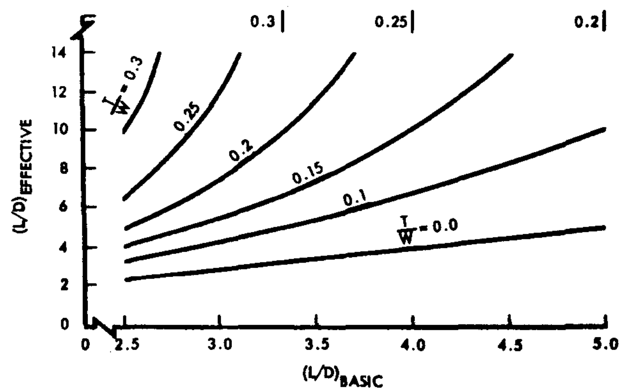


Figure 12

RANGE EXTENSION WITH TURBOJET

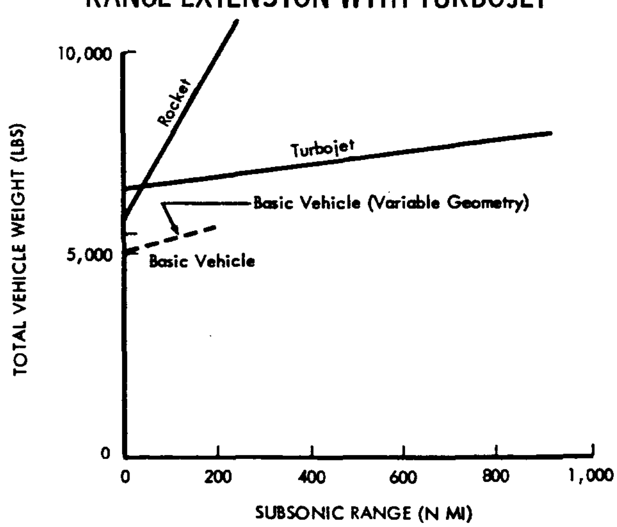


Figure 13

Figure 13 shows a comparison of weights for the various propulsion methods plotted versus range. It is believed that future spacecraft will run a combination of variable geometry plus a modest amount of rocket impulse during the terminal and landing phases of flight, which can also be used for other phases of the mission in a back-up mode if required.

DESIGN FOR RELIABILITY

If large-scale space systems operations are to develop, their reliability will have to reach levels that provide more confidence of mission success than now exists. Currently, major improvements are being developed in component design to increase their mean time to failure. System redundancy, parallel load paths, and minimum system activity are other approaches also being incorporated into spacecraft to improve mission success probability. Yet these steps alone will not be sufficient for complex spacecraft requiring mission durations lasting several weeks or months.

Our analyses show that typical vehicles, whose missions require good guidance, accuracy, attitude stabilization, space maneuvers, antenna sensor and solar panel orientation, environmental control, communication, power supplies, etc., are difficult to design with a high degree of confidence of operational lifetimes exceeding several weeks. Even assuming major improvements in the state-of-art of component design, this is still true. It is our conclusion that when long mission duration is essential for such spacecraft, manned vehicles with in-flight maintenance offer the best solution.

A comparison of such spacecraft is shown in Figure 14 where five different design concepts are plotted. System reliability versus a complexity factor which is the product of mission time and failure rate are shown. The longer the mission time or the higher the failure rate, the greater is the complexity factor.

MANNED VS UNMANNED DESIGN ENVELOPES

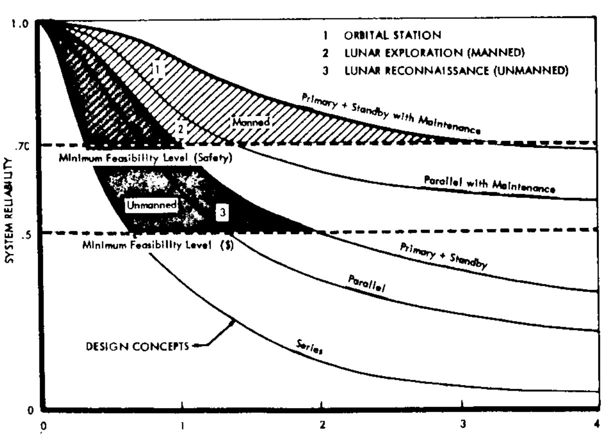


Figure 14

The systems shown are:

1) A series-type system in which failure of a major element causes failure of the spacecraft;
2) A parallel-system concept, in which dual subsystems are used. Both subsystems function at all times, but either alone is sufficient to handle the function;
3) A primary subsystem with a failure detector, a switching means, and a backup subsystem to handle the function if failure occurs on the primary subsystem;
4) A parallel-subsystem concept with a means of repair for the malfunctioning subsystem;
5) A primary-plus-standby concept with a capability to repair a malfunctioning subsystem.

For reference, three points are shown. Point 1 is typical of the reliability that can be obtained for a manned space station with a mission duration of 6 months. Point 2 represents a manned lunar reconnaissance module having from 200 to 400 hours' operating time. Point 3 represents a sophisticated, unmanned lunar reconnaissance vehicle with a mission duration of from 400 to 600 hours.

A line is drawn arbitrarily at 0.5 reliability to indicate that below such a value the system concept is not considered to be feasible. Another line drawn at 0.7 is considered, from the safety standpoint, to be the minimum feasibility level for a manned system. Below that value the escape system would be used too often for the vehicle to be considered a good risk.

From this data it can be seen that in-flight maintenance is the most effective factor for attaining the level of reliability necessary for complex spacecraft. Whereas there have been instances of reviving malfunctioning elements by clever remote diagnosis and manipulation, for the more complex vehicles a man on board will be required. The design of both spacecraft and subsystems will have to provide space and modular construction to permit this maintenance capability.

FUTURE STRUCTURAL REQUIREMENTS AND TRENDS

The most important factor influencing construction of re-entry vehicles is the hypersonic L/D desired. As stated earlier, it is believed that L/D varying from 1.0 to 1.5 will be the most typical for the future. Consequently, a composite construction will be required to provide the minimum weight solution for orbital or superorbital re-entry.

Re-entry vehicles that experience a peak cold wall heat flux of 40 BTU/ft^2-sec or less, except in stagnation regions, can use refractory metal shields backed by fibrous insulation with current state of art. For these configurations the pressure compartment would also serve as body structure and probably be of weldable aluminum alloy. Either a water-wick system or a circulating system with water boiler would be required to absorb heat coming through the insulation. The thermal protection — including refractory shield and support, insulation, water, and water system — would weigh 3 to

5 pounds per square foot of surface area, depending on the refractory-shield-to-structure attachment. This system is primarily influenced by the peak heat flux with relatively small effects on weight of vehicle L/D.

The peak heat flux exceeds 40 BTU/ft^2-sec in at least the forward regions of lifting re-entry shapes that have been examined with an L/D of 1.2 and less. A charring ablator can be used in the higher heat flux regions. For the vehicle with an L/D of approximately 1.2, 8 pounds per square foot is required where the peak heat flux is 40 BTU/ft^2-sec. Because of the significantly higher weights for the ablation system, a composite system with ablation only in forward areas and radiation shields for the remainder is attractive for vehicles such as shown in Figure 15. A relatively thin layer of low-density ablation material may be bonded to the refractory metal to protect the coating from the meteoroid environment. Since this material will be ablated early in re-entry, control surfaces can be operated during re-entry within heat limits.

F6-C — APPROXIMATE REGION OF THERMAL SYSTEMS

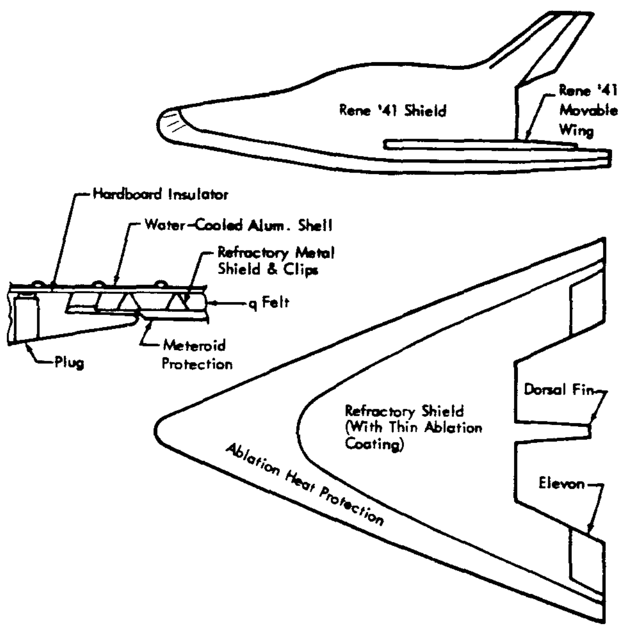


Figure 15

Transpiration systems are being evaluated and compared with other systems. The efficiency of a transpiration system is strongly a function of temperature allowed at the porous surface. Another important parameter is extent of optimization of effluent flow rates for each local area and versus time. Figure 16 compares thermal protection weights for a charring ablator with four different transpiration systems. The higher char surface temperatures, with heat rejection by radiation as max heat flux is increased, results in best efficiency for a char ablator at higher heat fluxes and best efficiency for transpiration at lower fluxes. The total heat flux is based on a heating-time curve for equilibrium glide from low orbit at L/D = 1.0. The relationship of transpiration and ablation is similar at other values of L/D, since weights for both systems increase almost directly with total heat flux. Transpiration systems are attractive because they are reusable and maintain a fixed aerodynamic shape. Depending on the type of system and design heat flux, transpiration may be competitive or superior on a weight basis to charring ablation.

COMPARISON OF TRANSPIRATION AND ABLATION SYSTEMS

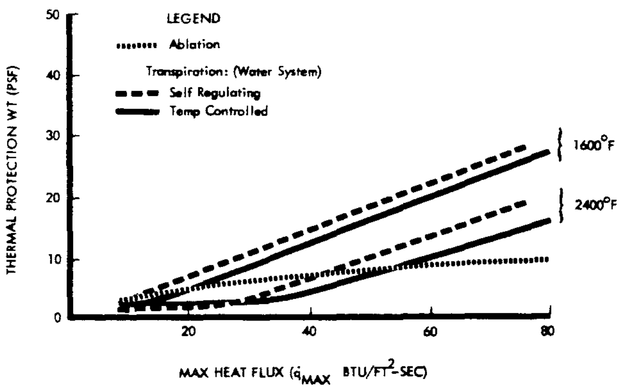


Figure 16

Vehicles designed for re-entry from low orbits using refractory radiation shields can be used for atmospheric entry at escape velocities by bonding ablation material to the refractory shield for absorbing the short duration peak heating at superorbital velocities. Tests have demonstrated the compatibility of selected ablators and bonding agents with disilicide refractory coating.

CONCLUSION

In summary, it is expected that manned spacecraft of the future will emphasize operation at high-orbit altitudes. They will require use of space propulsion for rendezvous, space maneuvers, and return. The propulsion units will be separable and disposable for high-altitude operation, even when recoverable first stages are used. Rendezvous is most promising for assembly of large space payloads. It is more economical than the use of single launches of new large boosters.

Re-entry spacecraft will require use of lift to provide sufficient corridor width for superorbital re-entry. For operational systems, the L/D range over 1.0 appears the most attractive, considering all the variables affecting re-entry accuracy. Variable geometry plus rocket augmentation will be used to provide landing-site acquisition and all-weather landings. Composite structure appears most attractive for such vehicles.

THE RELIABILITY ASPECTS OF SPACE PROGRAMS

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On many occasions during the past 5 years I have stated that the reliability of spaceborne equipment and devices may become a pacing factor in the achievement of our national objectives in military and civilian space. To introduce my subject this afternoon, I wish to emphatically repeat this observation.

Further, I am convinced that reliability is of such crucial importance to all our space-related activities that it should be considered a matter of national policy. It is essential, I believe, that the highest levels of government and industry management be made fully aware that the attainable level of space-system reliability not only may place physical restrictions on the extent of our space explorations, but will directly control the time required to accomplish any space program, as well as its cost. For example, we must recognize that the ultimate practicability of a satellite communications system, whether military or commercial, will depend not so much upon our ability to launch and guide satellites into optimum orbits or to demonstrate that usable communications circuits can be established via those satellites, but on how much it will cost to maintain in orbit the number of satellites required to furnish a reliable communications service. And, of course, this cost will be directly dependent upon the operating lifetime of the satellites in orbit.

The degree to which reliability will limit or delay the practical attainment of our space objectives, or affect their cost, will depend critically upon these factors: (1) how well the factors involved in obtaining high systems reliability are understood throughout the government and industrial agencies engaged in our space programs; (2) how thoroughly this knowledge is applied to design, engineering, construction and testing; and (3) the level of support and adequacy of direction of general reliability programs.

I shall review today some of the more significant aspects of management and technology involved in obtaining reliable systems. After noting a few of the efforts underway in the Department of Defense, including some in which the National Aeronautics and Space Administration (NASA) is jointly engaged, I shall suggest some program concepts that might help us to reach our space programs' goals more effectively.

The fundamental limitation on space systems reliability is the relatively high failure rate of parts now available to the system designer. Some space systems under development, or planned, have requirements for operating lifetime ranging from 1 to 5 years. This demands a level of reliability of individual parts several orders of magnitude greater than most parts now available can attain.

To put this in perspective, let us consider a space vehicle incorporating 20,000 parts, whose mission calls for a 90-percent probability that it will not fail for 1 year. Without redundancy, this would require parts having an average failure rate of better than 0.0001 percent per 1000 hours. The failure rate of standard electronic parts now generally used in military systems ranges from 0.1 to 0.001 percent per 1000 hours--one to three orders of magnitude poorer than needed for our assumed space craft.

As many of you know, some $30 million has been spent in the MINUTEMAN program to increase the life of electronic parts. This effort has provided an average improvement of roughly 10 to 1 so that failure rates of MINUTEMAN parts range from 0.0002 percent per 1000 hours for diodes to 0.01 percent per 1000 hours for power transistors. Specifications for these improved parts are being prepared by the Air Force so that they will be available to all system designers.

But the use of even these improved parts--which are generally the best available today--would give a 90-percent probability that our 20,000 part space vehicle would operate without failure for only about 1 month.

It is obvious that reducing the complexity of spaceborne devices will greatly increase their probable operating lifetime. For example, if parts of MINUTEMAN quality were used in a spacecraft having only 2000 parts, one could reasonably expect a 90-percent probability that it would operate without failure for 1 year. The use of redundancy would also provide improved reliability.

I realize that predicting a system's reliability on the basis of the number of parts and their failure rates may give results that are overly pessimistic when compared to the system's actual performance. The combination of statistics and recent experience with space and weapons systems, however, convinces me that, to meet the requirements of many of our space programs, we will need parts of considerably better quality than those produced for the MINUTEMAN program.

In addition to preparing specifications for MINUTEMAN-quality parts, the Department of Defense is supporting several other parts-improvement efforts. A Space Parts Working Group, with representation from the Military Departments, NASA and space-system contractors, has been established under Air Force leadership. The Group's aim is to determine common requirements for parts used in space systems. Availability of these requirements will permit us to concentrate development and testing work on these parts in an effort to raise their reliability level.

In carrying out the recommendations of the "Darnell Report,"[1] the Department of Defense is preparing parts specifications that include stringent requirements for reliability. We are also supporting a research effort to identify mechanisms of parts failure and to develop methods of short-term testing to determine failure rates without having to test large quantities of parts for long time periods.

All these efforts will certainly improve the general level of parts reliability, but I seriously doubt that we will ever fully meet reliability requirements for space systems via the route of conventional electronic-parts improvement. I am convinced that the only promise of really meeting these requirements lies in the widespread use of microelectronic integral circuits. There is good evidence that this new electronic technology offers significantly greater reliability than comparable circuits employing the best conventional parts available today. True, the present state of the art does not permit the use of microelectronics in many types of circuit applications. But the technology is advancing rapidly, and we may expect that microelectronics will be usable to a much greater extent during the next few years. In addition to its inherently high reliability, the reduced size, weight and power requirement of microelectronics will permit a more generous use of circuit redundancy in space vehicles.

I strongly urge the space industry to very carefully consider the application of this new technology in the development of new systems. Also, wherever it seems to offer technical promise, I recommend that more emphasis be placed on the development of solid-state devices to replace klystrons, magnetrons and traveling-wave tubes.

Giving too little attention to reliability during the design, engineering and test of a system has been a major source of operational failures. Even if all parts needed for a system have adequate reliability, improper application of these parts will inevitably result in an unreliable system.

In considering reliability engineering for spaceborne systems we should recognize that there is a fundamental factor--other than long life and maintenance-free operation--that is different from most other systems; that is the fact that space systems are produced in relatively small numbers.

In designing devices that are to be produced in quantity, the engineer has tended to rely--perhaps too heavily--upon the possibility that design deficiencies can be corrected or inadequate components replaced during production, or that his design mistakes can be rectified by field modifications during the equipment's service life. In many of our guided-missile programs (NIKE and TERRIER, for instance), missiles have been produced in quantities of several hundred for developmental firings. Designers have used the information gained in this production experience and the numerous early firings to improve equipment design and to test its reliability. To a certain extent, this practice has relieved the engineer of pressure to do a thorough design job in the first place; unfortunately this has had the effect of delaying general recognition of the need for intensive reliability engineering.

Because space systems are not "produced" in the usual sense of the word, and also in view of the high cost of launching the vehicles, this correct-as-you-go design procedure is not practical. The system and all its components must be properly designed in the beginning; then, before the first launch is attempted, the system's performance and reliability should be proved as far as possible in the laboratory under simulated operating conditions. This means, of course, that the designer must put much more emphasis on those engineering and test phases that are vital to incorporating and measuring reliability. Let the designer's philosophy be not "fly and fix," but "design for reliable operation from the start."

We have learned a great deal during the past few years about the elements in design and engineering that affect systems reliability. Today most competent design engineers know how important it is to select the best parts, to avoid tight tolerances in electrical and mechanical systems, and to conservatively load parts, electrically and mechanically. They know how to protect parts and circuits against environmental stresses. They know how tremendously the application of redundancy in areas of critical circuits or weak individual parts can contribute to reliability. They know how to predict with reasonable accuracy the life and reliability of systems and subsystems even before putting them together for the first time. And they know, by testing at subsystem and system levels, how to determine whether their design meets the desired standard of reliability.

In my opinion, a mature and experienced engineer knows enough today about the principles of design, engineering and test associated with reliability that he should be able to design a system whose inherent reliability is controlled almost entirely by its complexity and the reliability of the parts he has to work with. I suspect that system failures from design deficiencies are caused not so much by what we don't know as by what we don't do.

Now, if we have this much know-how in design and engineering, why do we still make mistakes in those areas that lead to failures in our space programs?

The answer, simple in fact but very complex in cause and remedy, involves three basic conditions:

First, on a particular job we may not have enough engineers with the requisite know-how.

Second, even if the engineers have the knowledge, we may not give them enough time or money for thorough design, engineering and test.

And, third, the company or government program-management people may not be sufficiently enlightened, either technically or policywise, to see that the available know-how is effectively applied.

The first of these is a problem of great national concern. As a result of the recent rapid expansion in our military and space programs, we have created such a vast amount of development and engineering work that our limited supply of competent, experienced technical people is seriously overextended. I won't go deeper into that question right now, because it would only lead to a discussion of our educational systems and the multitude of reasons for the present inefficient use of engineers. I can suggest, though, that on every major project there must be enough engineers with the proper know-how and with the authority to ensure that all the engineering done on that job is right in every detail.

The second situation, in which we know how to engineer the job but don't do it right for lack of time or funds, also presents a sticky problem. More often than not, a contractor is faced by program schedules and money limitations that make it impossible for him to design, engineer and test the system with the thoroughness needed to satisfy the mission reliability requirement. Too often, overriding schedules are based on policy criteria and do not allow sufficient time for a sound engineering and test program. I believe this is a characteristic national fault; it is certainly not unique to our space programs.

We must all feel the great national urgency that presses us toward early success in our major space and weapons-development programs. Over and over again, however, experience has taught us that too much haste in engineering and testing can bring on serious delays in a program, often far exceeding the time it would have taken initially to do a thorough job in the essential design, engineering and test. Worse than that, hurried and inadequate engineering and testing may lead to repeated failures in the initial operational tests, and this in turn may cause the entire project to be abandoned--a very undesirable and often unnecessary waste of human and economic resources.

The solution of this whole problem--adequate design, engineering and test leading to acceptable levels of reliability--depends largely upon the attitudes and enlightenment of management people in both government and industry. Management is called upon to recognize and respect the critical importance of establishing realistic and essential reliability goals and, through sound planning and administration of all phases of a program, to ensure that these goals are reached.

When first planning a space program, the government must very carefully determine the minimum level of reliability essential to meet the mission objectives. Once established, the reliability factor must be considered just as critically and conscientiously as any other performance characteristic in decisions regarding the cost of the program and the time required to complete it--even the feasibility of initiating the program at all!

In his proposals, a prospective contractor must fully take into account the required reliability levels and must include realistic estimates of the time needed to satisfy those requirements and the cost of the work. The government in its final selection of the contractor, must adequately weigh the contractor's understanding of the reliability requirements and associated technical and managerial factors.

Once a contract has been awarded, the contractor's management must make sure that the program is staffed, throughout all design, engineering and testing phases of the program, with the engineering know-how necessary to meet the reliability requirements; moreover through incentives and disciplines, the proper motivation must be instilled in the project people, one that will keep them constantly working toward the designated reliability goals. The program directors in the government must watch over the status and progress of the program's reliability aspects just as carefully as they review the contractor's adherence to any other performance or management requirement.

From the over-all standpoint of design and RDT&E management, we in the Defense Department are emphasizing, in all major development programs, the use of quantitative reliability requirements and suitable demonstration plans. We are stressing design reviews, reliability monitoring, system analysis and the inclusion of a program-definition phase. The more recent contracts are incorporating incentive clauses on reliability. (We are also advocating that PERT analyses be made during the program.)

I appreciate, of course, that in certain programs impelling factors may justify the early firing of space vehicles, even though the risk of failure due to the lack of reliability may be fairly high. We should, however, know reasonably well how great a risk is involved so that we can balance the cost of failure against the true value of early success. On the other hand, there is considerable doubt that it would be wise to carry the development of some systems, such as communications satellites, into the very expensive phases of final engineering and satellite launching until certain precautionary steps have been taken; that is, it should be reasonably well established, through careful prediction, analysis and preliminary design, that it will be possible with the current technology and components to eventually attain the level of reliability necessary to make the proposed system operationally and economically feasible.

Another concept that could help toward making space systems perform reliably involves the greater usage of common devices or subsystems whose reliability is well established. To encourage the application of this "building-block" philosophy, the Department of Defense and NASA are cooperating in an effort to develop launch-vehicle components that can be used in several space programs. Mr. John Rubel, Assistant Secretary of Defense for Research and Development, recently had this to say about the joint project:

> Both NASA and the Defense Department have coordinated efforts leading to the standardization of some of the most important of these "building-blocks." The SCOUT launch vehicle is an example. The standardized ATLAS D, being built by the Defense Department (under a program towards which NASA is contributing approximately fifty percent of the funds), is another example. The Defense Department has also developed a standardized AGENA upper stage, the AGENA D, which is programmed for use both by NASA and the Defense Department for a great variety of missions. No single factor is likely to contribute as much to improved launch vehicle reliability as the standardization of launch vehicles and their repetitive use.

We can likewise make significant gains in reliability and cost reduction by more common usage of guidance systems and other space craft devices. Administratively, the attainment of such goals in the various areas of guidance, telemetry, communications, stabilization and power supplies will probably be much more difficult, but we feel it is so vitally important to meeting our reliability goals and reducing the mounting proliferation of R&D projects, that it deserves very careful consideration. In support of this conviction, NASA and the Defense Department have undertaken a coordinated program aimed at bringing about the common usage of advanced guidance systems in space craft and launch vehicles at the earliest possible time.

We all recognize that the introduction of new components or techniques into a system development before their reliability and performance have been firmly established is a prime source of system operational unreliability. On the other hand, we know that the development and use of new components, techniques and devices is essential to meet the advancing performance requirements of our space programs. A vigorous exploratory and advanced development program to provide these new components, techniques and devices must be maintained. But we need to increase our efforts to perfect them through extensive engineering and reliability assurance testing before they are introduced into specific system developments. For a long time I have thought that all our space programs could profit immensely from a continuing "test bed" program wherein new devices and components could be tested for life and performance in a real space environment before being committed to a particular space mission. Probably much of this kind of testing could be done rather inexpensively by "piggyback" rides on vehicles launched for other missions. This "space testing," of course, should not replace thorough testing of new devices in a simulated environment in the laboratory.

As I close I would like to repeat the areas requiring our greatest attention in attaining the level of reliability demanded for our space programs.

First, we must place highest priority on the development and application of microelectronics as our best hope of meeting space program reliability objectives.

Second, we must continue and increase our joint DOD-NASA efforts to establish common requirements for electronic, mechanical and electromechanical parts applicable to space systems and to foster the development, test and application of these common parts.

Third, in all phases of space programs, we must place more emphasis on the use of proven design techniques, engineering practices and methods of reliability-assurance testing.

Fourth, we must make greater use of commonly applicable space devices whose reliability and performance have been demonstrated.

Fifth, we in the government must carefully determine the mission reliability requirements for a new system and specify these requirements in our contracts in quantitative terms, along with tests to prove compliance. When reliability is thus specified, the contract must contain reasonable incentives involving penalties or a premium on fees that will motivate the contractor to meet the reliability specifications.

Finally, reliability must be considered a matter of national policy and its importance to the successful accomplishment of our space objectives recognized by line management at all levels of government and industry involved in the conduct of our space programs.

[1] Office of the Director of Defense Research and Engineering and Office of the Assistant Secretary of Defense (Supply & Logistics), Parts Specification Management for Reliability, May 1960 (2 vol.). (Paul H. Darnell was chairman of the ad hoc group that made the study.)

GENERAL MANAGEMENT OF SPACE PROGRAMS INDUSTRY VIEWPOINT

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Introductory Remarks

American industry has risen to every occasion demanded by either national security or national prestige goals set by the governing bodies of these United States. Space leadership will be no exception. Historically, American industry has provided a standard of living in this country which overshadows anything ever before experienced in history. This has been possible through the exercise of freedom of thought unhampered by authoritarian type governmental control, but emanating from the competitive spirit of a free people each one of whom has the opportunity to pull himself up by the grace of Almighty God to whatever level of advancement he sets as his goal.

It is in this arena that the American aerospace industry now faces a task which will demand of it a dedication greater than ever before required. In assessing the challenges of "General Management of Space Programs," Industry will do well to view management in the light of Noah Webster's definition, "The judicious use of means to accomplish an end (by) skillful treatment." Although it is unlikely that Webster in the early 1800's foresaw the age of astronautics clearly, it is strikingly true that he very explicitly set down in black and white the task of management from the industry point of view.

We might well paraphrase Webster and say that the challenge to management is to

> "foresee, select, train, utilize in the most economical manner possible the proper proportions of manpower, facilities, working capital and lastly, but of great importance, team-spirited leadership to accomplish America's stated goal, 'Leadership in Space'."

This function of management may be symbolically represented by the wheel (see Figure 1) in which the hub is the task, the spokes are the balanced distribution of talents, and the rim is the well-rounded management team exerting a uniform pressure on all the spokes, holding them to the task, and thoroughly aware of the many varied up and down external conditions which are experienced by any forward moving, pioneering team. A fault in the rim is even more disastrous than a crack in a spoke. In the brief time available, only a few of the many facets of this task can be touched upon and it is hoped that they will be found to be of common concern.

AEROSPACE INDUSTRY GROWTH

Forecast

First and foremost is the necessity to grasp the very scope of the aerospace industry as it is today and its growth from its relatively humble beginnings in the early 50's. Figure 2, compiled from various governmental bureaus, clearly depicts this progression through the 1950's and forecasts its extension through the 1960's. However, judging from past experience with such projections, it will in all probability be exceeded by a significant amount. Graphically displaying this data on Figure 3, it can be recognized that the basic "jumps" in rate of expenditure have been brought about by specific events such as Sputnik I and the successful Project Mercury manned flights. Recognizing the fact that space exploration has barely started, it is more to be expected that there will be at least two more step inputs this decade. These could result from far more ambitious space exploration effort reinforced by successful programs on Gemini and later, Apollo.

As an example of the unlimited vista from today's porthole, consider the possibilities of a suggestion made by Dandridge Cole in an article entitled "Capturing the Asteroid," from which I take the liberty to quote "Even so remarkable an achievement as capturing a minor planet should be considered a possibility in the next decade, especially with the advent of nuclear spacecraft." This article appeared in the March 1963 issue of Astronautics and Aerospace Engineering and is indicative of the ever accelerating pace of space thinking. In all likelihood, some of this thinking will come to fruition and it is not at all unreasonable that the actual rate will reach 14 billion dollars per year by 1970 as appended on Figure 3. With a gross national product projection of 767 billion dollars for 1970, this still represents merely 1.82% allocated to aerospace. From the point of view of industry, management must look forward at least as daringly as its technological personnel, or be outstripped.

Expansion or Merger?

In the past an expansion of such proportions in any industry was in general created by a simple numerical increase in the number of personnel doing the already existing tasks for the most part. This is not the challenge to management now. For aerospace technology has brought with it a vast increase in the number of associated skills and technical disciplines. In support of this statement, the experience of McDonnell Aircraft Corp. is presented. Whereas in 1950, exempt salaried employees, representative of these technical disciplines, accounted for 11.32% of total employment, they now in 1963 represent 27.9% of total employment. This increase has brought about an even greater than proportionate increase in job classifications as changing technology and increasing specialization create new demands. McDonnell experience to date from 1950 is shown on Figures 4 and 5 with a forecast for the 1970 period extended at its present rate of expansion. Whereas a decade ago engineering fell into a few broad fields such as aeronautical, civil, mechanical, electrical and chemical, we find today such fields as human engineering, aerospace medicine, psychology, physiology, anthropology, astronomy and the pure sciences playing an increasingly important part in the integrated effort required for successful spacecraft design.

Looking constructively ahead, management has two basic choices: either expand from within using its present staff as a nucleus or expand by the acquisition of complete staffs from without. Conditions which favor expansion from within are:

1. Broad range of basic skills already available.
2. Inspiring "programs-in-being."
3. Favorable employment situation.
4. Steady rather than "explosively" programmed growth.
5. Financial strength sufficient to carry new operations through "growing pain" status.

Faced with these two alternatives, what has the aerospace industry done? A survey was made from the best sources available of a selected group of 21 companies commonly considered to be the "hard core" of the aerospace industry, exclusive of strictly electronics or propulsion manufacturers. These companies collectively absorbed a total of 176 supplementary organizations during the period of 1950-1961 as shown on Figure 6 for an average of 8.4 acquisitions per company. However, it is particularly significant that six of the eight companies doing by far the major share of the aerospace business acquired a total of only fourteen operations for an average of 2.33 acquisitions per company. Obviously, acquisition is not the complete answer, but each management group must face up to this problem, and be flexible enough to adjust to changing conditions.

Growth in Test Facilities

Along with this growing technology, management inherits the problem of providing progressively extensive physical facilities, with particular emphasis on those of a laboratory or test nature. Although statistics are not generally available for the whole industry with respect to capital investment for facilities and the portion of the total required for research and development, the specific case of McDonnell Aircraft Corp. is presented for consideration. For the acquisition of its present facilities, M.A.C. has reinvested from earnings 70.6 million dollars for land, buildings, machinery and equipment. During this same period, the government has provided 44 million dollars, or 38% of the total 114.6 million dollars of facilities. This 70.6 million dollars represents the reinvestment of 65% of total corporate earnings to date. Figure 7 shows the distribution of this investment between what may be classified as production type facilities and R & D type facilities. Research and development facilities include engineering campus office buildings and all laboratory buildings and laboratory equipment. In an effort to make the values more meaningful, it was decided to compare these costs versus the pounds of "production" and "R&D" craft produced to date. On this basis the reinvested earnings were $11.20/lb. for R&D and $2.40/lb. for production. This results in a ratio of R&D to production investment of 4.7 to 1. When the presently programmed installations are completed, this ratio will have increased to 5.1, and over the next five years is forecast to become 7.5. Under these circumstances, management is confronted with the challenge of allocating capital asset investments where they are best able to contribute to the corporate long term growth on a sound financial basis. In pursuit of this, M.A.C. has invested heavily in test facilities capable of simulating space environment for manned space vehicles and will shortly have 15 space chambers in operation ranging in size and capability from 32" dia. x 72" long @ 10^{-10} torr. to 30' dia. x 35' long @ 10^{-8} torr., including two chambers capable of testing a Gemini spacecraft and adapter in complete orbital configuration, including provisions for manned occupancy during the duration of the tests.

Despite the very substantial investment in test facilities on hand at the start of the Mercury program, it was found necessary to further increase this amount. See Figure 8.

AVAILABILITY OF TRAINED MANPOWER

For the past several years we have been continually made aware of the projected shortage of engineering and science college graduates. A recent publication makes the following contrast (Figure 9) for the 1960-1970 period. Practically all fields are going to double. This should be looked forward to with enthusiasm by management as a forecast of an expanding economy, but it must be planned for.

The questions management must ask itself are:

1. Have the educational agencies of the land recognized this requirement?
2. Are the educational facilities and curricula paced to satisfy this requirement? and
3. What part is industry taking in the solution of the problem?

In a recent presentation to the St. Louis Junior College District Technical Education Advisory Committee, Norman C. Harris, of the University of Michigan, presented some rather revealing facts on the matter of educational levels. As shown on Figure 10 and Figure 11, taken from this presentation, the occupational spectrum will have been so drastically changed by 1970 that there will have to be an entirely new emphasis on middle level or junior college education. Aerospace management level personnel have here a very inviting challenge to work with community groups to bring about this transition. Particularly because junior colleges tend to satisfy local needs, aerospace firms which will be a very significant employment factor should accept a prominent role in this endeavor through counseling, publicizing, and taking an active part in obtaining the facilities needed to train this major share of their workforce.

The aerospace industry today is a very active proponent of employee development through training programs completely operated by each company, and through college and graduate level assistance by means of co-op programs or reimbursement type off-hour degree programs. Typical of this interest in the development of its employees is the example of McDonnell Aircraft as summarized in Figure 12.

MANAGEMENT CONTROLS

As industrial enterprises become more complex in keeping with advancing technology, there is a concurrent need to develop better systems of management controls. The very breadth of ground needing coverage automatically has increased the amount of effort required in this area and is reflected in the growth of administrative personnel. As an example of the growth in emphasis in this area it is interesting to examine M. A. C. experience in overhead ratios during the past five years as shown on Figure 13. These figures are for almost identical total employment levels of approximately 27,000 personnel. In general, such a growth is looked at as undesirable when thought of as added burden. However, it should be remembered that controls, when properly installed, serve the purpose of enabling more efficient management, and there should be a net gain in operating efficiency through their proper use. Thus, any increase in administrative expense is unwarranted unless it is more than offset by a decrease in operating costs. Herein lies the problem. Originators of management systems are to be highly commended for their analytical approaches in many instances, and it is the top management's job to assess the proper degree of penetration of "systemitis" and assure that the system reports the progress, not delays it.

An increase of the order of 32% in any cost area is always a source of major concern and has resulted in a very detailed study to determine the causes. A major contributor to this increase is the necessity of producing a multiplicity of reports and analyses for the various governmental agencies, with a great deal of overlapping data. Figures 14 and 15 depict this problem and show clearly that a reduction in reporting formats will permit a definite reduction of cost in this area. This challenge is one to be shared by both industry and government.

The above mentioned few challenges must and can be met and properly dealt with by an agressive management. Their solutions will definitely produce more aerospace program for the same total dollars.

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The author greatfully acknowledges the assistance of Mr. R. C. Krone, Director of Personnel, and Mr. P. T. Rafter, Manager of Plantwide Planning, McDonnell Aircraft Corp., in compiling the data presented herein.

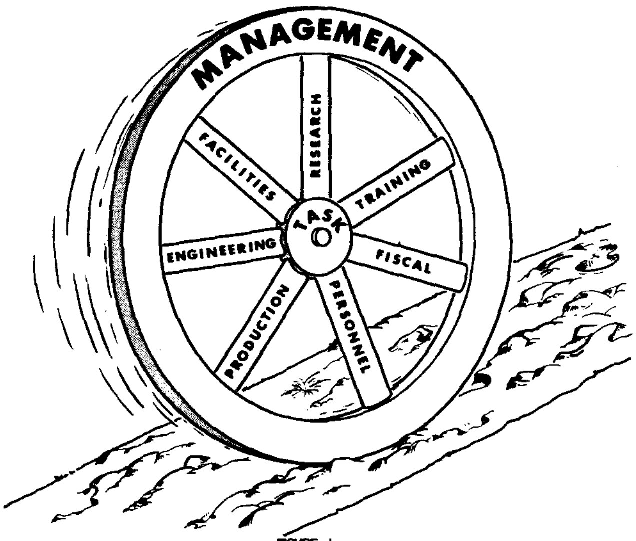

FIGURE 1

MISSILE & SPACE EXPENDITURES

IN DOLLARS FOR FISCAL YEAR

YEAR	TOTAL SPACE BILLIONS	GNP BILLIONS	%
1952	0.2	347.0	.07
1956	1.1	419.2	.26
1960	3.4	503.2	.68
1964	7.9	606.0	1.30
1968	9.8	709.0	1.38
1970	10.0	767.0	1.30

FIGURE 2

MISSILE & SPACE EXPENDITURES

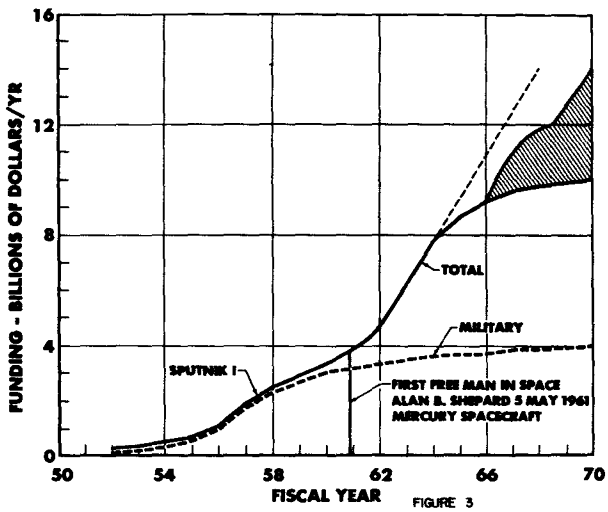

FIGURE 3

PERCENTAGE OF EXEMPT SALARIED EMPLOYEES IN RELATION TO TOTAL MCDONNELL EMPLOYMENT

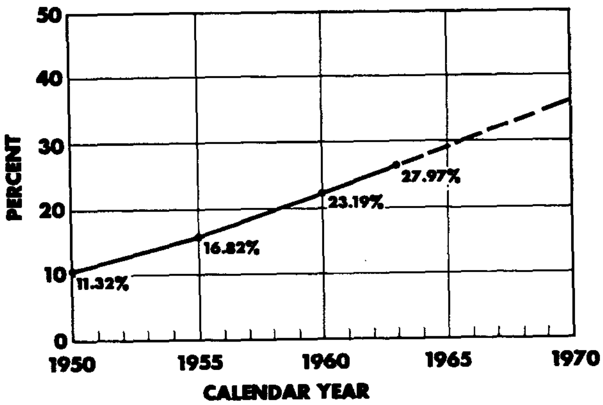

FIGURE 4

NUMBER OF SALARY CLASSIFICATIONS REQUIRED TO KEEP PACE WITH INCREASED SPECIALIZATION AND CHANGING TECHNOLOGY

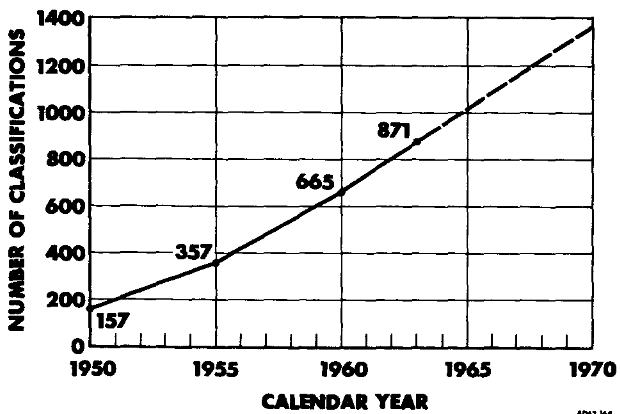

FIGURE 5

COMPANIES GAINED BY ACQUISITION

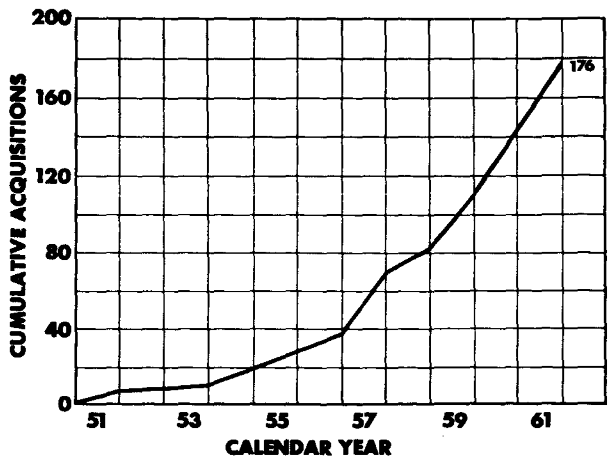

FIGURE 6

CAPITAL ASSET INVESTMENT

BREAKDOWN ANALYSIS FOR PRODUCTION VS RESEARCH & DEVELOPMENT

(MILLIONS OF DOLLARS)

	EXPENDED AS OF 28 FEBRUARY 1963 PROD.	R&D	TOTAL	UNEXPENDED PROD.	R&D
M.A.C. INVESTMENT					
REAL PROPERTY	$ 33.0	$ 20.3	$ 53.3	$ 4.3	$ 4.2
INDUSTRIAL FACILITIES	9.5	7.8	17.3	18.3	5.5
TOTAL	$ 42.5	$ 28.1	$ 70.6	$ 22.6	$ 9.7
GOVERNMENT INVESTMENT					
REAL PROPERTY	$ 10.3	0	$ 10.3	0	0
INDUSTRIAL FACILITIES	33.9	0	33.9	5.9	0
TOTAL	$ 44.2	0	$ 44.2	$ 5.9	0
TOTAL INVESTMENT					
REAL PROPERTY	$ 43.4	$ 20.3	$ 63.6	$ 4.3	$ 4.2
INDUSTRIAL FACILITIES	43.4	7.7	51.2	24.3	5.5
TOTAL	$ 86.8	$ 28.0	$ 114.8	$ 28.6	$ 9.7
POUNDS PRODUCED JAN. 1947 - DEC. 1962	36.2	2.5	38.7	-	-
INVESTMENT PER PRODUCED POUND	$ 2.40	$ 11.20			

FIGURE 7

FACILITY GROWTH

	MILLIONS OF DOLLARS	DOLLARS/LB.
FACILITIES AVAILABLE (PRIOR TO MERCURY)	86.0	—
ADDITIONS FOR MERCURY	0.545	182.0
ADDITIONS FOR GEMINI	8.290	1180.0

GROWTH RATIO = $\frac{1180}{182} = 6.5$

FIGURE 8

TYPE OF SCIENTISTS, ENGINEERS AND TECHNICIANS IN AIRCRAFT, MISSILE & SPACECRAFT INDUSTRY

	EMPLOYMENT 1960	REQUIREMENT 1970	PERCENT INCREASE
ENGINEERS	88,300	155,600	76.2
SCIENTISTS			
CHEMISTS	3,000	5,400	80.0
PHYSICISTS	5,800	12,600	117.2
METALLURGISTS	1,400	2,600	85.7
EARTH SCIENTISTS	100	200	100.1
MATHEMATICIANS	4,000	10,100	152.5
MEDICAL SCIENTISTS	100	1,200	1100.0
BIOLOGICAL SCIENTISTS	100	900	800.0
OTHER SCIENTISTS	3,600	5,800	61.1
SUB-TOTAL	18,100	38,800	114.2
TECHNICIANS	49,000	107,000	118.4
TOTAL	155,400	301,400	93.0

FIGURE 9

OCCUPATIONAL TRENDS AND EDUCATIONAL REQUIREMENTS

(U. S. LABOR FORCE – 1930)

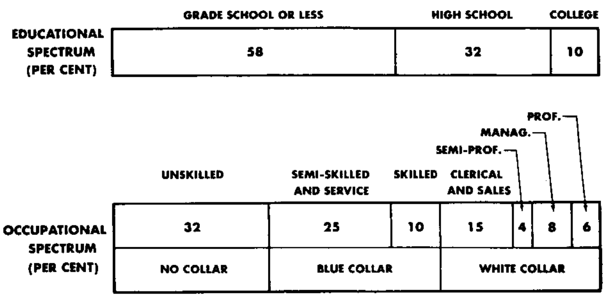

FIGURE 10

OCCUPATIONAL TRENDS AND EDUCATIONAL REQUIREMENTS

(U. S. LABOR FORCE – 1970)

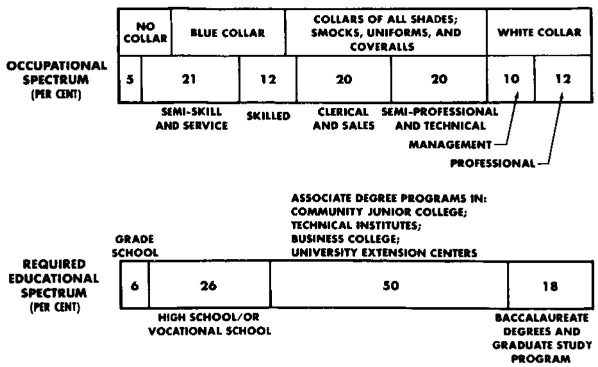

FIGURE 11

ANNUAL TRAINING SUMMARY

IN-HOUSE PROGRAM

NO. OF COURSES	NO. OF CLASSES	MANHOURS	ORIGINAL ENROLLMENT	COMPLETING	%
105	315	105,000	6462	5362	83

CO-OP PROGRAM

NO. OF SCHOOLS	NO. ON PROGRAM	NO. GRADUATES THIS YEAR
6	81	12

COLLEGE STUDY PROGRAM

NO. OF SCHOOLS	NO. REIMBURSED	DEGREES RECEIVED (FY 62)
21	980	43

FIGURE 12

CHANGE IN OVERHEAD RATIO

AREA	1958	1963
GENERAL ADMINISTRATION	1.0	1.32
MANUFACTURING	1.0	1.06
ENGINEERING	1.0	0.94
PROCUREMENT	1.0	1.00

FIGURE 13

ADMINISTRATIVE TYPE REPORTING SYSTEMS

	SCHEDULE & MILESTONE STATUS	FINANCIAL DATA DD-1097	COST DATA	MAN-HOUR MANPOWER DATA	FACILITY DATA	REMOTE SITE DATA
BSD/SSD EXHIBIT 62-1 CONTRACTOR REPORT EXHIBIT	X	X	X	X	X	X
AF/SC CONTRACTOR COST STUDY		X	X	X	X	X
AMCM 70-5 & ARDC HANDBOOK CONTRACTOR PROGRAM PROGRESS REPORTING	X		X	X		
ASO COST MANAGEMENT MANUAL	X		X			
DCPR-DEFENSE CONTRACTOR PLANNING REPORT				X	X	X
PERT TIME & COST	X		X	X		

FIGURE 14

PERT TIME AND COST REPORTING SYSTEMS

	PERT TIME	PERT COST
NASA PERT SYSTEMS MANUAL	ONE TIME ESTIMATES	NO
AIR FORCE ASD EXHIBIT AS2061-1 PERT MANAGER'S HANDBOOK	THREE TIME ESTIMATES	NO
NAVY MIL-P-23189A MIL. SPEC. PERT/TIME AND PERT/COST	ONE TIME OR THREE TIME ESTIMATES	YES-COORDINATED WITH PERT TIME
DOD & NASA GUIDE-PERT COST	ONE TIME OR THREE TIME ESTIMATES	YES-COORDINATED WITH PERT TIME

AD63-164

FIGURE 15

GOVERNMENT VIEWPOINT
OF THE
MANAGEMENT OF THE AEROSPACE PROGRAMS

James N. Davis
Deputy Assistant Secretary of Defense
(Installations & Logistics)

As weapons become more complex, a method for managing their acquisition has developed which is intended to insure that the government's investment is effective. Programs to acquire such weapons give rise to such problems as:

1. Schedule compression and overlap of hundreds of traditionally sequential tasks, synchronizing hundreds of distinct task-activities in one major program;
2. Forecasting program difficulties with sufficient time to develop alternate solutions;
3. Executing large numbers of contracts on one program;
4. Evaluating the risk of untried technology as it is designed into sub-systems and tested;
5. Projecting dollar needs in advance so as to respond to the annual budget cycle;
6. Committing funds and knowing what they will buy;
7. Avoiding overruns;
8. Obtaining the special industrial facilities required;
9. Scheduling the use of test facilities;
10. Supporting the test program and later the inventory with sufficient spare parts to keep the weapon in operation;
11. Making sure that test, ground handling and maintenance equipment is developed and produced on schedule;
12. Insuring attention to reliability and quality control during design;
13. Ordering, financing and shipping government-furnished equipment on time to prime contractors.
14. Analyzing and preparing forecasts of cost and schedule impact caused by alternate force levels;
15. Determining needed engineering changes and scheduling them into the program; and
16. Insuring that contractors respond to the system plan so that all components work together.

Regardless of organization, this imposing array of problems must be solved.

About 125 weapon systems are monitored systematically by the Office of the Secretary of Defense. The contribution of each is significant to our military posture. Strong emphasis on schedule synchronization within these projects is essential. Likewise, costs are sufficiently large on each, that constant attention to economy and avoidance of overruns is critical. Normal budget constraints and limits on our ability to finance escalating program costs require accurate long range dollar projections by managers who are familiar in depth with their programs. In turn, this can be done only if our program managers can accurately assess the solution of technical problems.

The above discussion is intended to illustrate the problem: executing a program which is characterized by complexity, concurrency and military urgency. For those weapons programs which have these characteristics (by no means all of them), a management organization is needed for each which centralizes the program authority and responsibility in one office. The program manager should be carefully selected on the basis of his general intelligence, boldness, proven willingness to make decisions. He should become the visible center of program authority and information with his career affected by his performance.

All priority weapons programs do not require the same composition in their central program offices. There is a gradation which can be applied to delegations of authority. Nevertheless, these are the delegations, modified by degree, which should be centralized under the program manager, if he is to be held responsible (to any degree) for execution of the program:

1. Budget control;
2. Contract administration;
3. Authority to buy in-house services;
4. Selection of his support personnel;
5. Control of the tenure of his staff;
6. Technical evaluation and system decisions;
7. Control of engineering changes, etc.;
8. Measurement of achievement on contract incentives; and
9. Use of Government facilities

In addition, he should be a major participant in selection of contractors, negotiation of contracts and construction of contract incentives on performance. Likewise, the Armed Services Procurement Regulations should be in harmony and in support of this centralized weapons management concept.

The personnel practices likewise should develop classifications and the career potentials which support it. We believe the experience gained by personnel in a weapons program office will be much broader than that obtained in other Defense assignments. Program office personnel are required to cope with the simultaneous management of time, dollars and technical progress. They are faced with the subtleties of technical interface problems, the importance of forecasting technical and schedule difficulties, and the conservation of program time as a precious commodity. These are the characteristics of the new personnel challange in Government, and likewise in industry.

The rank of the military program manager should be commensurate with the gravity and the magnitude of his task. In a study last year, nine programs were checked and, in only one case, was the rank of the manager and his delegations of authority considered commensurate with his task. It will be found that on most of our major programs, the manager works with men of much higher rank on his more important problems. These men are by no means all in his direct line of Service command. The Services are encouraged, therefore, to upgrade the position of the program manager, and to develop distinct career patterns in this field.

The above description of an "in-house" management concept is not unique or unrelated to industry. In fact, we expect it to be merely a continuation of the management structure within industry which is under contract to create weapons for us. Therefore, in industry, we expect a complimentary program management organization, with similar rank, responsibility and delegation of authority. The industrial manager must assume the responsibility of technical design, work loading, budgeting, scheduling and program decisions in order to create the sub-system, or as prime contractor, to integrate the work of all contractors in response to the Government statement of work.

Each manager, industrial and military, therefore, must have the other as a primary point of contact in Government and company in order to resolve problems and execute the program.

The Government shall continue to work with industry in developing managerial tools which assist in creating a complex weapon system. Neither the Government nor industry manager can expect to anticipate problems and make decisions without means of rapid analysis of technical activities, their cost and their timing. Some time-honored methods are not believed adequate, and such new approaches PERT-COST will be encouraged. Tools such as this can only become useful as industry assists Government in their refinement. An energetic campaign is being developed to insure the newer, mechanized methods of work and resources planning and analysis are not layered on top of older manual systems of reporting. The latter should be eliminated.

In addition, an open channel of technical and programming audit will be encouraged. Neither the Government nor the industry program manager is safe in his heavy responsibility if he has only one source of information. Trust is implicit, yet the shade of difference in motives between customer and vendor, the semantics of work agreements and descriptions of program status warrant each seeking a small, but distinct, surveillance activity under their jurisdiction which provides a separate accounting of progress and problems.

To this end, the Government has sought to collect advanced technical skills within the program manager's staff, or at least in-house. This has not always been successful, and outside organizations, non-participants in the contract as such, continue to augment our in-house skills and supply advice on program activities of advanced complexity and criticality to program success.

Policies governing the in-house management of major weapon systems will become more clearly described, and procedures for their conduct as well. As an example, a new procedure outlines the method for conducting a program definition activity at the beginning, so as to permit the industrial groups which might be associated in a weapons development, to explore in depth the technical, schedule and cost feasibility of the proposed program. This directive is in the final stage of coordination.

Finally, the search continues for a useful format of program data which can be viewed periodically by top Service and Defense management and which insures that, even on an exception basis, all major problems requiring high-level decision are forecast early and plans developed to accommodate them.

PROBLEMS FACING THE AEROSPACE PROJECT ENGINEER - INDUSTRY VIEWPOINT

Joseph G. Gavin, Jr., Vice-President, Director LEM Program,
Grumman Aircraft Engineering Corporation

Rather than pursue the problems of the aerospace project engineer at a distant philosophical level, I would like to examine them from a very personal point of view. To begin with, let's establish a definition. The Project Engineer referred to here is the senior technical person holding line authority in a major program. Sometimes this person is called the Engineering Manager of a program. This distinction is necessary because occasionally the term 'project engineer' is applied to levels of engineering supervision more traditionally known as group leaders. This Project Engineer, of which I speak, carries an immense responsibility, and must at various times display talents worthy of Albert Einstein and John Foster Dulles.

Let's first examine his technical problems. While he cannot be expected to be expert in all disciplines, he must be reasonably at ease in considerations ranging from heat transfer to digital data handling. His comprehension level must be sufficient to earn the respect of the various specialists within his organization. Modern, complex systems require difficult trade-off and integration compromises. With the support of his group leaders, the project engineer must define the proper compromises without inordinately lengthy studies. He must require from his crew adequate, useful, and convincing information; he has to resist the sometimes easier course of asking for further investigation - beyond the level of real significance. For example, in the LEM program, we are now examining a very interesting compromise - should weight be invested in a stronger landing gear to permit rougher landings or in more propellant to permit better landings? We could continue to embroider this study for months; but we won't; we must avoid this temptation.

Another technical hurdle for the project engineer is the undefined or "floating" requirement. Designing to provide margin for such requirements requires conservative boldness - or is it bold conservatism - and strong convictions. Pursuing the example of the LEM, we are currently wrestling with the problem of what constitutes reasonably safe assumptions with regard to the lunar surface. How high a coefficient of friction might an assumed dust layer provide? A course of action will have to be taken long before all the answers are available; our solution must provide a reasonable degree of flexibility to cover the range of possibilities.

A further technical demand on our project engineer is a clear understanding of those areas within the project which press the state-of-the-art. The problem usually occurs in two steps; first to recognize these areas, and second to limit them. Our Orbiting Astronomical Observatory is an example of a program made rather difficult by the necessity of pressing the state-of-the-art in a number of areas simultaneously in order to achieve the desired results. In this case, astronomical precision has placed unusual demands on such things as star tracker gimbal angle accuracy, control of heat flux to minimize structural distortion, and data handling and storage capacity - all at unprecedented reliability levels. Again, without the proper evaluation and approach, we could not have progressed from analysis to hardware.

In reviewing the project engineer's role, it is sometimes surprising to see how much of his efforts are devoted to administrative problems. He must maintain a delicate balance of emphasis between project and discipline - his specialists must be clearly project oriented, yet they must benefit from their ties with colleagues on other projects. The project engineer must resist the tendency for the myriad of insignificant, and therefore easier, administrative demands to dilute his attention to the significant and frequently thorny technical questions. At the same time, he must exercise judgement with respect to delegation of both technical and administrative responsibilities - he must resist the temptation to carry out each study himself; he cannot funnel every detail through his office. By these last comments, I do not mean to imply that his administrative role is less important than his technical role. He must take a leading part in cost and schedule estimates - otherwise neither he nor his subordinates will live up to these seriously. He must demonstrate administrative as well as technical control to limit over-elaboration, to resolve group interfaces, and to insure coordinated milestone accomplishment.

While engineering education seldom stresses this point, a surprising proportion of the project engineer's trials and tribulations are in reality people problems. He must be able to approach each subordinate in the manner which will result in optimum performance. He must be able to apply the appropriate "filter" to each subordinate's comments so that the information is "normalized". He must exhibit leadership, must be able to inspire others to lead, and must be able to evaluate performance objectively. He must be able to communicate effectively within his engineering project, within the program organization, with representatives of the procuring agency, and with sub-contractors. One of his toughest tasks is to recognize and acknowledge those occasions when he is wrong.

In the case of manned vehicles, he also is confronted with the necessity of working with, understanding, and communicating with pilots or astronauts, as the case may be. Success for the project demands the development of mutual respect.

Having progressed from technical problems to a discussion of human relations, I may as well go all the way and reduce the project engineer's considerations to a few very basic questions which he must answer in almost every instance:

> "If I permit the project to progress in this direction -- would I go as pilot?
> -- would I ask my best friend to go as pilot?
> -- would I invest my own money?
> -- does this action really count?"

The project engineer can make use of the most refined methods - systems studies, multi-variable mathematical analysis, elaborate simulations and tests - but, in the end, he has to satisfy these questions.

In principle, everything I have said was just as true 10 to 15 years ago as it is today. What then are the differences which make the job of today's project engineer more difficult? Here are a few:

(a) Today's major program is larger, represents a greater technical step ahead, and is one among a smaller number of national programs. This makes every decision more significant in terms of either money or effort. Each decision requires greater justification and more careful analysis of its implications.

(b) The quest for performance - of all kinds - inspired by mission requirements and industrial competitiveness has increased the level of effort as well as the calibre of talent required to do all but the simplest engineering tasks.

(c) Flight testing has always been expensive and potentially dangerous. With the advent of manned space flight the magnitude of these conditions has increased drastically. More patience and ingenuity must be exercised in testing on the ground. The Probability of Mission success and mission safety must be explored with far greater care and understanding.

(d) And finally, I am convinced that, under the pressure of these more demanding programs, a better professional engineering job is being accomplished today - not easier, but better.

More detailed technical study supporting the decision-making process, more detailed test programs with additional emphasis on extracting the maximum amount of information from every level of testing. Efforts such as these, and the multitude of others covering every technical - and human - aspect of the program - are the responsibility - and the salvation - of the project engineer of today's space programs.

SPACE POLICY AND SPACE MANAGEMENT

By
Dr. Edward C. Welsh, Executive Secretary
National Aeronautics and Space Council

It is a pleasure for me to speak briefly to this distinguished audience on a very broad subject: "Space Policy and Space Management." Indeed the subject is so big that I plan to take less than the allotted time and simply make a few capsuled observations.

By way of introduction, I would observe that combining space management and space policy in the same discussion is in itself a point of significance. They belong together, but they are often treated as being unrelated. It is true that one can evolve and expound policy without making provisions for management -- but one cannot have effective policy without competent management. The ingredients of policy are the objectives, the guidelines, the purposes -- plus the will and intent to carry them out. The ingredients of management are the competences--including the people, the procedures, and the funds -- necessary to make the policy work. Not so incidentally, management is more than supervision of current performance. It is also planning ahead for future performance.

At the risk of unwarranted misinterpretation on the part of scientists and engineers present, I suggest that there is as great a scarcity of top-notch managers as there is of top-notch technologists. It is perhaps unfortunate but nevertheless fact that in the space business nothing less than the best can be considered adequate. That applies to management as well as technology.

Positive vs. Negative Policies

This country, as great as it is, cannot afford avoidable handicaps in the space race. Consequently, we should avoid the bad practice of management based upon fallacious budgetary philosophies and rely instead upon constructive policy objectives. Few things are more wasteful than the starting and stopping, the delaying and hastening of programs to suit the whims of those who place a halo on inaction and place a wall of ignorance around the vital surge of technological competence.

What can be more harmful to this country's future than a proposal that we curtail our space efforts, stifle our productive initiative, and freeze the U. S. in a second place position? There are those who are so enamored with slogans of false economy that they would deny our future generations the power, prestige, and profit to be derived from an adequate space investment now.

Who is it that would suffer from a policy based upon the antediluvian ideas of those who recommend that we not invest our public and private capital in space exploration? Surely not the Soviets. They would gain, if we followed such policy suggestions. No, the ones who would lose from such short-sighted views would be the citizens of the U. S. who believe freedom and first place are the basic rights and privileges of Americans.

In case there is any doubt as to what I am referring, I want to make it clear that I am not criticizing those who believe they detect waste or needless duplication in our space program. I would be pleased to have every such item exposed and have the funds so identified re-invested in the program so as to accelerate performance. Those who place a balanced budget and a first-place Russia ahead of our space accomplishment are the persons toward whom I point my criticism. When those persons finally enter the twentieth century, I hope they will wake up and learn that the head-hiding ostrich is not our national bird.

Multi-Project Program

It is not feasible to try to give a detailed exposition of our space policy on this occasion, but there are some features of our policy which I want to mention. I do so while emphasizing that there are various aspects of policy which will undoubtedly be omitted in the interest of brevity and generalization.

First of all, there is an over-all policy objective, which the President has expressed as the determination of this country "to become the world's leading space-faring nation." It is worth mentioning that this is a broad umbrella, under which one can find many space projects -- some of which have been clearly defined, such as the lunar mission, and some of which are either being identified, such as space stations, or are yet to be specified. Suffice to say that the space program is not a one-project design nor a short-run episode. It is a growing, expanding, multi-project program, which I predict will become an increasingly significant part of our way of life. As important as it is, the moon project is not the whole space program; it is just one portion and one phase of an over-all program which

has breadth through variety and depth far into the future.

Features of Policy

A few features of this broad space policy are:

1. Its objectives are peaceful. The law of the land states that "activities in space should be devoted to peaceful purposes for the benefit of all mankind." That same legislation specifically includes, as part of the peaceful purposes, space activities devoted to "the defense of the United States." In other words, the distinction as to what is peaceful and what is not peaceful is a matter of intent and is not determined by what agency of the government engages in them. Space activities devoted to deterring war and maintaining peace are as peaceful, in the light of law and policy, as space activities devoted to augmenting our scientific knowledge about the solar system. U. S. space policy is distinctive in its determination to improve the opportunities for peace rather than to increase the threats of world domination and possible aggression.

2. Our policy asserts that space travel and space exploration are subject to international law and that such activity should be consistent with the provisions of the United Nations charter. Among other things this means that outer space and celestial bodies are not subject to national appropriation and that space is not to be used for aggressive purposes. Even though it is feasible, the U. S. has no intention of placing weapons of mass destruction in orbit and will only do so if compelled to such action by the aggressive activities of others.

3. It is our policy to seek increased international cooperation in, and mutually advantageous agreements for, the orderly and open conduct of space and space-related activities. To meet our world leadership responsibilities, it is necessary both to compete with as well as to cooperate with other nations in space.

4. It is also our policy that the concept and performance of our space responsibilities shall be maintained and strengthened as a national program rather than a series of separate and unrelated projects. To this purpose increasingly close and coordinated relationships are being maintained among the government agencies concerned. This has the merit of improving the flow of data, experience, and technical knowledge between agencies and also of minimizing the likelihood of duplication. When I refer to avoidance of duplication I include duplication of omission (where no one is doing something needed) as well as duplication of commission (where two or more agencies are doing the same thing). The former inhibits our space progress even more than the latter.

5. There are inherent risks in space explorations, particularly in manned flight, and they must be recognized. It is basic to our space policy that we accept these risks, take reasonable safeguards, absorb the unavoidable losses which will occur, and proceed with courage toward our objectives. If we had been unwilling to take risks and proceed on course, this country would never have become the great nation it is. If we become satisfied with our power and wealth and timid in our efforts, we will lose our world leadership position and along with it our freedom. Space is a challenge to both our courage and our initiative. If the ways of the timid are followed, the race is lost.

6. While we must continue to improve on state-of-the-art competences, it is essential that we strive just as hard to make so-called quantum jumps in technology and in performance. In fact, as a policy matter, I would stress the drive for the "leap-frogging" type of improvements rather than giving inordinate attention to making a piece of equipment slightly better, to adding a few more pounds of thrust to a rocket engine, or to wedding ourselves to existing technology. The space program is a program of vitality, in which the rate of obsolescence is being pushed by a vigorous competitor on the one hand and by the unlimited opportunities for greater accomplishment on the other.

7. It is a significant element of policy that progress be the joint product of government and private enterprise. Our chief competitor in the space race operates under a different system and, in a sense, our relative success in the space race is and will be a test of how well our system stands up in the eyes of the rest of the world. I am convinced that we can survive the test and show anew the superiority of our system, but it will not be possible to do so if we fail to use effectively the assets we have. That we started late in the race was not a fault of private enterprise. The blame there must rest on the government. But, now that the government has awakened to its task and has allocated resources to the effort, the free enterprise system per se is under the gun. The space challenge is indeed an opportunity, but we can flunk its test if we are dilatory or if we place false fiscal objectives ahead of space success.

Future Characteristics

It is so easy to confuse the space program with some particular aspect of it that I will make a few more observations which should be obvious, but sometimes are not. Emphasis upon liquid chemical rockets, for example, is a phase of current space development and should not be interpreted to mean that the space program as such has made a choice to the exclusion of solids or nuclear propulsion. All three and possibly other techniques will become essential and standard in the program. Moreover, the concentration of attention and funds on the manned lunar project is temporary, so far as the space program is concerned. It is, however, a current objective and a basic step in building our longer run space capabilities. Space stations, planetary visitations, and even terrestrial travel of both personnel and freight via space will have future priorities at least as great as the lunar project. Moreover, operational space applications in communications, meteorology, navigation, and observation will become standard and reliable instead of experimental. The distinction between scientific and manned expeditions will dissolve and be replaced with a recognition of their complementary characteristics.

Space Management

In these brief comments, most references to management have been indirect rather than specific. Perhaps the major point I want to make is that we can assemble all the essential ingredients of trained scientists and engineers, of funds and facilities, of materials and equipment ... and we will still lose the space race if the application of managerial skills is deficient. If the government delays decisions, fails to exploit all technology, engages in intra-mural conflict and confusion, and underpays its key personnel, performance suffers. If private contractors underrate precision and reliability, deliberately under-estimate bids, waste unduly their best talents in selling rather than producing, refuse to assume some of the risks of innovations, performance also suffers. If both government and private management fail to plan ahead with confidence that the space race is a long race, a more far-sighted competitor will be found in the winner's circle.

Conclusion

In conclusion, the national space program is an essential and dynamic feature of our economy. It benefits the many rather than the few. It strengthens the nation, both at home and abroad. It looks forward rather than backward. It highlights the contributions of the most able technologists and the most able managers and gives them an unprecedented peacetime opportunity to repay more fully the special bonus they have received by living in this great country.

We can afford the space program. We must afford it. We can-be satisfied with nothing less than first place -- to the moon, to the planets, and throughout the solar system.

CONGRESSIONAL VIEWPOINT ON SPACE PROGRAMS

By

Olin E. Teague
U. S. House of Representatives
Chairman Sub-Committee for Manned Space Flight
House Science and Astronautics Committee

I want to begin these remarks by asking for your understanding. There are many of you, I am sure, who think that service on the House Committee on Science and Astronautics is a glamorous assignment. It is true, there is an element of glamor involved but -- you can take my word for it -- very little of it rubs off on our constituents.

It is true that in my own District, there are many people who believe whole-heartedly in our space program, -- particularly in our plans to put a man on the moon. But, it should be made very clear to you that there are also many people in my District who think the space program is a lot of nonsense -- and who believe quite sincerely that we are wasting a prodigious amount of money in trying to put men on other planets.

One of the newspapers in my District has already taken me firmly to task for my preoccupation with the problems of space. The Editor feels I should concentrate on more earthly problems.

And, let me add also, that today as we sit here, the House of Representatives is voting on a Feed-Grain Bill. This bill is of the utmost importance to my constituents -- particularly in a time of drought such as we are experiencing at the present time. And, as sure as you are born, I shall have an opponent next year who will attack me on the grounds that -- instead of staying in Washington and voting for my constituents, I was located in the plush Marriott Motel, in Dallas, dreaming about the moon.

I mention these facts, not out of self-pity, but as a reminder that we, in Congress have our problems, just as you have yours. Service on the Science and Astronautics Committee is not altogether a bed of roses. It's a pretty controversial assignment.

There is another point to remember. In the entire Membership of the Congress we do not have a single Nobel Prize winner. We do have a smattering of PH.D.'s -- but most of these are honorary. We have a few Rhodes Scholars and a handful of Phi Beta Kappas, but most of the Members who have achieved any degree of academic eminence have done so in the Humanities: Law, Literature or History.

There are very few scientists in our ranks. And, to be perfectly frank, I don't think many scientists could get elected to Congress.

Congress, in short, is comprised of 435 ordinary American citizens. And, among these there are at least 435 different points of view regarding the space program.

I would like you to put yourselves in our shoes for a few moments. Think, for just a little, how much we in Congress have to take on faith. We discuss budgets in the billions of dollars. Trying to serve the best interests of the Nation and of the American taxpayer, -- without ever being perfectly sure that we are right in our decisions.

And, I need make no apology for that -- because the scientists who come before us often have conflicting ideas about what is right and what is wrong.

As Chairman of the Manned Space Flight Subcommittee, I can speak for my colleagues when I say that we are greatly impressed with the team of scientists with whom we have the privilege of working. I refer to such men as Dr. Werner Von Braun, D. Brainard Holmes, George M. Low, William E. Lilly, Dr. Kurt H. Debus, Robert R. Gilruth, and others.

This is a great and dedicated group of men and I want to pay tribute to them at this time.

Now, I want to make myself clear about the space budget.

As you so well know, a day doesn't go by during which some Republican Members of the Congress don't get up and declare that the President's budget should be attacked, not with a paring knife, but with a hatchet. The attack is invariably answered by Democratic Members who claim that the budget cannot be cut by so much as a hundred dollars without seriously impairing the safety and the economic health of the nation.

The Members of the Committee on Science and Astronautics have attacked the budget problem with great seriousness and dedication. We have held exhaustive interviews with private industry and with every interested agency of government in order to assemble our information. I can tell you this: if we find any soft spots in the space budget, we are going to eliminate them. If we find any money that does not need to be spent this year -- but can be postponed until next year -- we are going to recommend that the appropriation be delayed.

The Members of the Committee on Science and Astronautics are enthusiastically determined to spend as much money as is necessary to insure that the United States become permanently the leading sky-faring nation on earth.

However, the Members are also determined to prevent the space race from becoming an extravagent and wasteful boondoggle. There is always a danger of this, since Congressmen are not equipped to know and understand the needs of the space program to the same extent, let us say, that they are equipped to assess the amount of expenditures necessary for a familiar program such as the construction of dams. On the other hand, there is the danger that some of the scientists who come before us might see in this program a veritable bonanza of appropriations -- by far surpassing the dreams of every science faculty in every major university in the history of the world.

We have the responsibility of providing for a program that will insure our permanent supremacy in space, and, at the same time, preventing the possibility of the U. S. Treasury from being looted by inspired visionaries.

The learned gentlemen come before our Committee and demand -- and I think the verb is not too strong -- a budget of 5.7 billion dollars for the coming year. We are told that this is the absolute minimum and that if we cut the request by as much as $100, we shall be crippling and endangering the entire space exploration program for the future.

Now, at first glance, this is pretty frightening talk. However, we -- and every Congressman -- know that the budget preparation began eighteen months ago. The whole concept of space exploration has been changed -- in many important respects -- in the meantime. The direct ascent approach to the moon has -- for better or for worse -- been unceremoniously junked. The earth-rendezvous approach has been considered and discussed and the lunar-rendezvous has been embraced -- with joy by some, with misgivings by others. All this has happened between the creation of the budget and its presentation -- so, can one blame a Congressman for questioning the assurances of these witnesses when they say this requested amount is the minimum that should be granted -- with no questions asked?

For all we know, the request should be for a billion dollars more -- or a billion dollars less. Or, indeed, it could be that the request is absolutely right. We really have no way of being certain.

As you know, the Bureau of the Budget saw fit to cut the request from 6.2 billion to 4.2 billion dollars. In doing this, I am sure the Bureau of the Budget was working from just as much a scarcity of positive information as was the Congress -- or the scientists.

Can Congressmen be blamed for being skeptical?

To be brutally frank -- the scientists -- who talk so boldly about communicating with other planets -- have done a very poor job of communicating with the people on <u>this</u> planet who are expected to foot the bill.

They seem almost intent upon bedazzling the Congress and the public with a glamorous and bizarre jargon. They talk of diodes and vidicons, command modules and magnetohydrodynamics, bleed-off and burnout -- almost as if they were deadly afraid of lapsing into comprehensibility.

There have been many times when I have wished with all my heart that more Congressmen had studied science -- and that more scientists had studied English.

It would, I think, be quite helpful for the space scientists to admit, quite frankly, what they do not know. Such admissions would not only be good for their souls, but they would increase confidence in their probity.

A touch of humility is a much felt want -- and with that humility there should come a greatly enlarged desire to share all knowledge and techniques with all organizations -- military and civilian -- who are seeking an American break-through in space.

As of now, the space program -- and particularly the program for lunar exploration -- is in trouble in the Congress. The trouble is not necessarily fatal, but it could grow into something very serious indeed.

First of all, there is a large and growing group of Congressmen who are against the program, period. They feel that the moon-race is -- in the words of former President Eisenhower -- "A mad effort to win a stunt race." I don't agree with this point of view, but, on the other hand, it would be foolish to brush it aside as simple-minded provincialism. Some Very distinguished people have expressed serious doubts about the moon race -- including Doctor Killian, Dr. Van Allan, Lewis Strauss, Dr. Condon, Dr. Vannevar Rush -- and many others. The opposition of such people as these cannot be simply ignored -- expecially when they have been more articulate in expressing their opposition than the proponents of lunar exploration have been in expressing their support.

Then, there are other Congressmen who feel that lunar exploration is a too-expensive luxury. They feel that the money to be spent on placing a man on the moon could be better spent here on earth -- on feeding the hungry, sheltering the homeless, healing the sick and educating the ignorant.

There are even some people in Congress who are so old-fashioned as to think the money should be <u>saved</u> -- and applied against the national debt.

There is still another school of thought -- and to this one I subscribe -- that believes that the only possible justification for the expenditures of these vast sums of money is that it will lead to a permanent and impregnable reinforcement of the defense of our nation.

We cannot justify the program purely in terms of international prestige. There are ways of building our prestige better, for less money and in terms that people can more easily understand.

If the program is to succeed, the scientists -- both military and civilian -- must do a far better job of convincing the American people along these lines -- and doing it in terms that they can understand.

Continuing Congressional support depends upon this -- and so do continuing appropriations.

This is the job that must be done -- and the scientists must do their share in getting the message over to the public. You are all educators of specialists, -- you must also become educators of the average man.

APOLLO PROGRAM EVOLUTION AND BACKGROUND

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INTRODUCTION

The Apollo Spacecraft Project has been in progress for approximately three years and considerable progress has been made. A significant milestone was reached with the awarding of the contract for the Lunar Excursion Module (LEM) to the Grumman Aircraft Engineering Corporation. The award of this contract completed the selection of the major industrial team members that will carry out the Apollo Project. Concurrent with this completion of the selection of the industrial team, flight tests of full-scale Command and Service Module hardware have begun. As the emphasis starts to shift to flight testing, it is interesting to recapitulate chronologically the activities of the Project to date.

PROGRAM EVOLUTION

The accomplishments of the Project to date may be summarized as follows: First, the mission objective has been established to land men on the moon and return them to earth in this decade; the lunar orbit rendezvous technique has been selected as the mission mode and design approaches to the spacecraft have been determined. Secondly, the spacecraft work statements have been completed and contractors selected. Most of the contracts have been definitized, with accompanying program definition. The spacecraft design is about 50 percent complete, with the Command and Service Modules being further along than the Lunar Excursion Module. Spacecraft fabrication is underway, with boilerplate and prototype spacecraft hardware coming off the manufacturing line. The first major flight test will be conducted within the next several months. It will be a flight test of the Command Module and the Launch Escape System (LES) to measure the dynamic behavior of the configuration and to qualify the LES in the case of a simulated pad abort. Last, organizations have been developed at MSC, at NASA, and at various contractors to implement the Program.

Significant Project activities and related events may be described in some cases as periods of work and in others as discrete milestones. Figure 1 presents a summary of both since the beginning of the Project.

Project Mercury was officially announced as a program the latter part of 1958, and assigned to NASA for execution. The Apollo Project was announced in mid-1960. At that time, Apollo was planned as a circumlunar flight, and not as a lunar-landing mission. Shortly thereafter, the first formal Apollo organization was formed. Alan Shepard's Redstone flight in May of 1961 was followed very closely by a Presidential decision to carry out the manned lunar-landing mission. Several months later, the Office of Manned Space Flight was created within NASA to direct the Apollo Program, including spacecraft, launch vehicle, and other program elements. In the latter part of 1961, the decision was made that the Saturn C-5, or Saturn V, would be the launch vehicle for the lunar-landing mission. The Apollo spacecraft project was moved to Houston, Texas, in early 1962; and shortly thereafter John Glenn's orbital flight took place. The last and one of the most significant decisions was the decision that the mission mode would be the lunar-orbital-rendezvous technique. This decision was made in mid-1962. Simultaneously with these discrete milestones were various developmental activities. Some of the more significant activities during this time period are presented in the lower half of figure 1.

As stated previously, the Mercury Project started in 1958. Shortly after the Mercury Project started, NASA formed the Goett Committee under the chairmanship of Dr. Harry Goett of Goddard Space Flight Center. The purpose of this committee was to study and recommend what should be the next manned space-flight program. This information would provide the necessary guidance to the various NASA Centers in setting up their research programs. In May of 1959, after a study of some nine missions, the committee decided that the lunar mission should be the next manned mission carried out by the NASA. This recommendation, after some months of close study, was presented to the Administrator of NASA in early 1960. The Manned Spacecraft Center (Space Task Group at that time) was represented on this committee and quite a few of the Apollo spacecraft concepts were developed during this period. The next big effort was a coordinated study effort. This study effort lasted for about 18 months, in the period of 1960 and early 1961. It involved the Space Task Group, various Research Centers, and industrial concerns. This effort started with a series of technical guidelines. These guidelines dealt with such things as the flight time, the number of crewmen, the launch vehicle, landing capabilities, and similar basic requirements. These technical guidelines formed a frame of reference, within which various technical studies were carried out. These technical guidelines were presented to the various Research Centers in April 1960. Later in 1960, several industrial studies were started. These studies were built around the same guidelines. This study period was summarized in mid-1961 with an Apollo Technical Conference in Washington, D.C. This was the formal part of the Apollo study effort, leading up to the first development contractual effort.

SPACECRAFT DEVELOPMENT

Prior to completion of the various studies in mid-1961, work statements and specifications were being prepared for the development of the Command and Service Modules. The gray part of the bar on figure 1 for this module indicates the period of time in preparation of the work statements, bidders' briefings, proposal preparation, evaluations, and selection of the Space and Information Systems Division of North American Aviation, Inc. to develop the Command and Service Modules. This contract, after a period of negotiations, was awarded at the end of 1961. Since that time, North American has continued to design and develop these modules, leading to this first test which should be conducted in several months.

The Navigation and Guidance System had been studied by the Massachusetts Institute of Technology (MIT) on NASA contract starting in early 1961, and the contract for system development was awarded in mid-1961. MIT has continued the design and development of this system to the present time. Participating with MIT in the development and manufacture of the Guidance System are three industrial contractors: the AC Spark Plug Division of General Motors, Raytheon, and the Kollsman Instrument Corporation. Shortly after MIT was awarded the contract for the Navigation and Guidance System, they prepared work statements; and after bidders' briefings and evaluations, the contractors were selected in 1962.

The last major element of the spacecraft is the Lunar Excursion Module. The block of time in early 1962 under the study effort bar represents a concentrated study of the lunar orbital rendezvous mode and associated spacecraft design. This effort, combined with the effort of OMSF, MSFC, and industrial groups, led to the selection of the lunar orbital rendezvous mode. As before, a work statement was prepared and, after a period of evaluation, the Grumman Aircraft Engineering Corporation was selected to carry out the development of the Lunar Excursion Module. This contract was awarded in late 1962, and the contractor is now initiating the design of the LEM.

There are two areas to which particular attention should be given in the development of the Project. These areas are the evolution of the mission mode, and the evolution of the spacecraft configuration. The mission evolution is depicted in figure 2. As stated previously, Apollo was first announced in mid-1960 as a circumlunar reconnaissance mission. It was recognized then that prior to a lunar-landing mission, a reconnaissance mission would be desirable. In addition, the planned launch vehicle payload capability at the time was 16,000 pounds, which limited the mission to a circumlunar mission. Studies in the period referred to, therefore, dealt primarily with a circumlunar mission and the associated spacecraft. In May of 1961, the decision was made that the mission would be a manned lunar-landing mission. At the time, however, the mode was not established, but only that the mission would be a lunar-landing mission. From May 1961, at the time of the Presidential decision, until November 1962, a period of intensive study was in progress which culminated with the decision to develop the Saturn C-5 (S-V) launch vehicle. This decision meant that the direct-landing, single launch vehicle mission would not be carried out. It was tacitly assumed at the time that the mode would be earth-orbital rendezvous. For the period of time of late 1961 until mid-1962, the studies dwelt on two things: how the spacecraft would be developed to carry out the mission where the complete spacecraft landed directly on the moon, or alternatively, the lunar orbital rendezvous technique where only a portion of the spacecraft was landed on the lunar surface. In the middle of 1962, after such studies were made by various groups, the decision was made to use the lunar-orbital mission.

The spacecraft has evolved parallel to the evolution of the mission and mission mode. Figure 3 presents the spacecraft configuration as it appeared at various times. On the far left is a sketch which was put in the original study work statement. At that time, it was thought that the spacecraft would probably be in three modules, including a reentry module; a module with space propulsion capabilities for maneuvering; and possibly, a module related to the specific mission that the vehicle was carrying out. During the first study period, considerable effort was directed toward the reentry module. At the bottom of figure 3 are shown four shapes that received considerable attention. The configuration on the right was the one subsequently selected as the Command Module. This module configuration has remained constant during the evolution of the spacecraft, along with the launch escape tower. In the early studies, the propulsion module was geared mainly to an abort capability, rather than lunar landing. After the lunar landing decision was made in May, efforts to evolve a system for lunar landing were intensified. The first such effort which appeared in the 7/61 work statement is shown on the figure. It consisted of a landing or descent stage of hydrogen-oxygen and an ascent stage of a cluster of solid rockets and liquid verniers. By the end of 1961 when the contract was awarded to North American, the configuration (identified as 12/61) had been evolved; namely, a single engine liquid-propulsion system for the Service Module. The landing stage had evolved in detail, but was still a large hydrogen-oxygen stage. In order to reduce some of the problems associated with landing this very large stage on the moon, the decision was made to break this stage in two parts, as shown in the figure. The lower portion (identified as 4/62) would provide the primary propulsion during the descent to the lunar surface, but would be jettisoned prior to touchdown. The second stage, with landing gear, would provide the propulsion for the hovering and landing, itself; and then the upper stage or Service Module would be the launch stage. That was in April 1962. At the same time, however, studies were continuing on the LEM and the decision was made that the LEM would be the route taken. The last configuration was evolved in July 1962, and this is the overall configuration that is being worked on today.

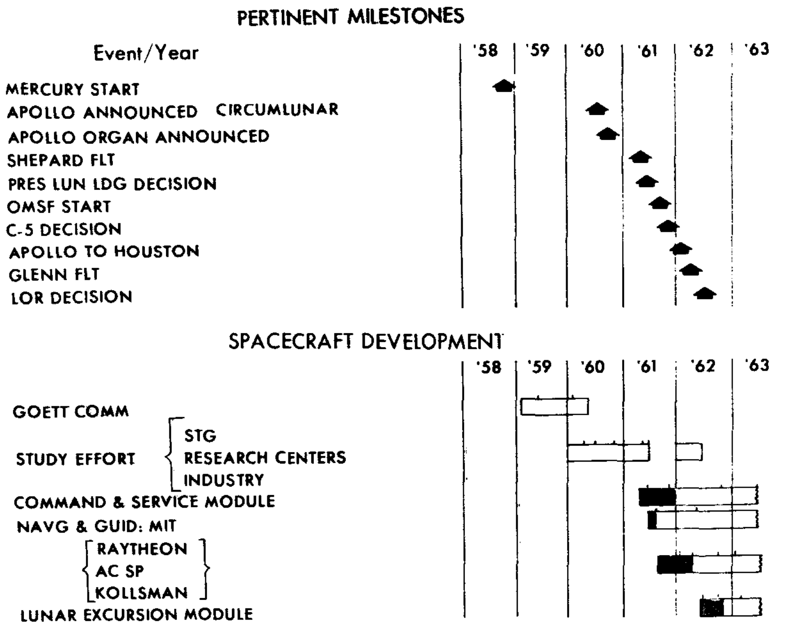

Figure 1.- Project Milestones and Development

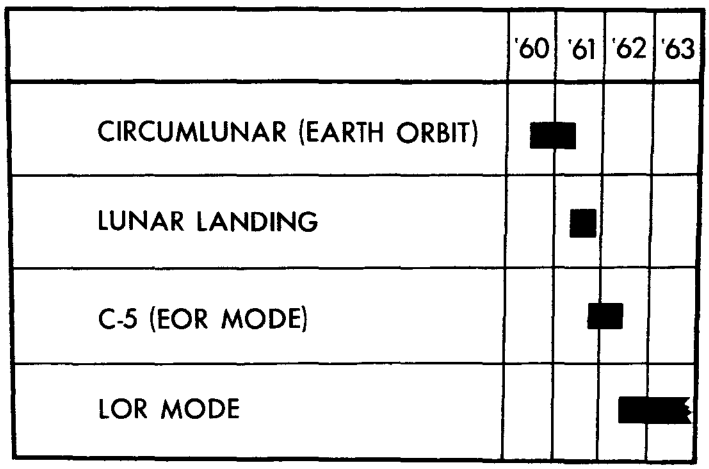

Figure 2.- Apollo Mission and Mission Mode Evolution

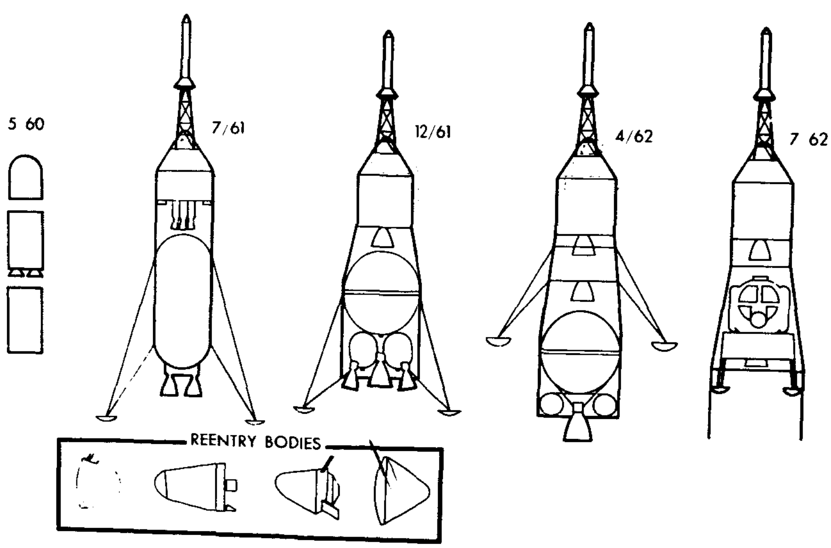

Figure 3.- Apollo Spacecraft Configuration Evolution

MATERIALS PROBLEMS IN MANNED NEAR-SPACE OPERATIONS

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The important and often decisive role of materials in space technology has been widely discussed and is generally accepted. There is an abundance of proposed research topics in the field of materials which call for pursuance at a high rate of effort, if serious setbacks in space programs are to be avoided. This abundance is a result of the variety and severity of often unexplored environments, of the close relationship between materials and design and of unconventional performance criteria, introduced by the associated disciplines.

It is difficult to judge the relative importance of these proposed research topics, primarily due to the absence of clearly established and numerically defined design requirements. This leaves us with an enormous number of materials problems, much too large to be pursued at a high rate of effort. Materials research and development is expensive and, once started, represents a considerable commitment with regard to facilities, personnel and time.

In the present discussion it is the intent to define, by means of a simplified analytical procedure, the significant materials problems for a selected group of aerospace vehicles. It is obvious that, within the limitations of this discussion, there will be no room for specific materials data. In the attempt to arrive at a fairly complete picture, a certain degree of generality cannot be avoided.

Definition of "Problems"

The term "problem" appears at first sight very vague. As a technical term it represents the gap between requirements and capabilities. Materials problems, specifically, refer to the gap between design requirements and materials capabilities. A prerequisite in the determination of materials problems is, therefore, the identification of design requirements, which necessarily include environmental as well as operational aspects. These can only be spelled out accurately for specific systems components, comprising a specific set of design requirements.

In order to arrive at an objective definition and to distinguish materials problems it is, therefore, necessary to follow the established procedure of systems design, from systems analysis, configuration and environments to component design, and, ultimately, to the assessment of materials capabilities.

Definition of Systems

The systems which have been selected for this analysis comprise "Manned Near-Space Operations". This formulation identifies a number of basic requirements: the term "near space" refers to orbital operations closest to the earth, i.e., altitudes between the lowest feasible orbits up to the closest natural radiation belt, which places the orbital regime between approximately 100 and 600 miles. The discussion of the significance of this regime is beyond the scope of this paper. It is undoubtedly the choice region for continuous operations of scientific, commercial and military nature. Continuous, active orbital operations, in turn, postulate (1) that the vehicles are not only manned, but permit the crew to live and carry out their assignments without restrictions with regard to environment, space, equipment or supplies; this, in turn, postulates (2) an adequate size of the orbital station, and (3) a continuous supply line with the ground station in the form of a scheduled earth-station traffic, for supplies as well as personnel. The requirement for this supply line differs basically from those of other space operations, as the emphasis is not so much on large payloads, as on a continuous exchange of small payloads in short intervals. For this purpose, boosters, expendable as well as recoverable, are impractical and uneconomical. The most effective system for continuous earth-station traffic is a lifting, self-propelled vehicle for horizontal takeoff and landing, or, essentially, an airplane with orbital capability. A number of concepts have been proposed for such a vehicle, which establish its feasibility. Without commitment to any specific version, it may, for the purpose of this discussion, be designated as HTOL-Orbital Vehicle.

Once several space stations have been established, operations call for additional maneuverable shuttle vehicles for interstation traffic. They may be designed for re-entry or may, like the station itself, carry emergency re-entry gear with unfoldable or inflatable drag devices.

Manned Near-Space Operations, as defined for this discussion, comprise the following vehicle types:

1. Space stations with a minimum weight of 100 tons and orbital altitudes between 100 and 600 miles.

2. HTOL-orbital vehicles for a minimum payload of 5 tons and at least 200 missions.

3. Maneuverable interstation shuttle vehicles.

Assumed Systems Schedule

The earliest feasibility of these vehicles is defined in Figure 1. The schedule is closely connected with the state-of-art as well as the booster capacity and the availability of present space projects. The 100 ton station could be placed in orbit by the advanced Saturn V, while the

300 ton station would have to await the operational status of NOVA. In both cases it is assumed that the station is entirely built on earth and deployed in space by self-erection. Numerous designs for such systems have been proposed, such as the three wing station of NASA-Manned Space Center, which is typical of the 100 ton station, or the 21 man hexagonal torroid station of Langley Research Center, which is typical of the 300 ton class.

The most significant step in space station technology is the introduction of space assembly capabilities, as it opens the door for almost unlimited orbital build-up. It could be achieved earliest in 1969 without the aid of Saturn V. Boosters could later be replaced by the HTOL-orbital vehicle, which would serve as a construction truck in the gradual assembly of large space stations.

The initial deployment of space shuttle vehicles could be accomplished with a number of boosters, such as Titan II or III. It could, likewise, be carried out later by the HTOL-0 system.

SPACE STATION

In contrast to lunar or interplanetary operations, near-space systems are concerned only with three trajectory regimes: exit, near-earth orbit and re-entry. For the near-space station only the first two apply, with the exception of the on-board emergency re-entry gear, which may be considered as a small system in itself. The exit or launch phase introduces a number of requirements which are common to all components, particularly the following three: (1) the resistance to the acceleration forces, (2) the compatibility with the aerodynamic shape restrictions and, (3) the all-important drive for light weight which has always been the guiding principle of aerospace engineering.

As soon as the station has been deployed, the necessity of human accommodation introduces a first distinction of components and design requirements with regard to the environmental exposure:

(a) Components exposed to space environment only.

(b) Components of the protective envelope, partially exposed to both space and atmospheric environments.

(c) Internal components, exposed to atmospheric environment only.

The related materials may, likewise, be classified according to their relation to space environment as:

(a) Materials with passive function, which merely have to withstand space environment.

(b) Materials which carry out an active function, primarily in the protection against space environment.

(c) Materials not in contact with space environment.

Critical Components of the Space Station

The most delicate component of a space station is the primary hull, as it provides the separation between human life and space environment. It is exposed to all constituents of space environment and has further the active function of shielding against its harmful effects, such as micrometeroid impact or high energy radiation. In order to be feasible weight-wise, it will combine all functions integrated in a single structural composite, for which a variety of design configurations have been proposed. The typical structural arrangement for such a composite is shown in Figure 2, which defines the function, environment and material requirements for each of its elements. The inner skin provides the sealed envelope for human environment and carries the pressurization loads. The outer skin acts as a meteroid bumper, primary radiation shield, substrate for the temperature control coating and as an envelope for the meteroid and radiation protection system between outer and inner wall. As the tabular listing in Figure 2 shows, the critical environments as well as the material requirements differ considerably for each element of the composite.

The dimensioning of the radiation and meteroid protection is essentially a trade-off between weight and risk. This may be illustrated for meteroid protection by the nomogram of Figure 3. The left side of the diagram represents the familiar environmental frequency - mass relationship, based on the assessment by Whipple et al, as well as on recent satellite experiments. Superimposed at the right side is the relationship between particle mass and minimum wall thickness (aluminum) to prevent puncture. For any risk level the required wall thickness can be determined as indicated by the dotted line and, vice versa, the risk for a selected weight limitation.

In Figure 4, the nomogram is further augmented by the addition of an assumed meteroid protection system and by minimum gage limitations. The cross-over or break-even points define the limitations of various protection concepts for minimum weight design. The shaded envelope identifies the minimum weight condition for the entire field, from the minimum available sheet gages up to a upper weight limit, selected for an accepted risk level.

The energy dissipation or "filler" material of a meteroid protection system may, for low orbital operations, serve simultaneously as thermal insulation, as convective heating will be encountered in addition to the radiant solar heating at altitudes below 150 miles. The magnitude of solar and aerodynamic heating in relation to orbital altitude is illustrated in

Figure 5, which places the break-even point at approximately 95 miles.

Similar requirements, as discussed for the primary hull, apply to its integral components, such as doors, windows and their sealing elements. Windows present particularly delicate requirements, as shielding against radiation and meteroids may defeat their primary purpose of transparency. For window materials it is desirable to confine the transmission to the optically useful range. This is illustrated in Figure 6: If we would use conventional glass, the ultra-violet radiation will be absorbed, however, the infra-red portion of the spectrum will be transmitted to the inside, resulting in considerable transfer of heat to the interior, where it is difficult to remove. If we attempt to avoid this by selecting a glass which absorbs infra-red radiation, the energy appears as heat which it then partially transmits to the interior by conduction. Internal heating may be further reduced by a coating which reflects most of the infra-red radiation without excessive penalty on the visible transmission. The problem of window materials is further complicated by the gradual loss of optical properties due to micrometeroid errosion and radiation effects.

For internal components the prime criterion is low density, while strength and stiffness depend on the g-level at various locations (distance from the station hub). Ultra-light weight materials, such as magnesium-lithium alloys or foamed materials are of prime interest.

For external structures, the foremost requirement is stiffness, as flexibility can lead to considerable "wobbling" in a revolving station. External components are usually large, such as antenna reflectors or solar collectors. Inflatable structures and self-rigidizing materials play a predominant role.

Erection by inflation is often difficult, particularly for intricate shapes. An expendable envelope, added to the actual structure, solely for the process of inflation is illustrated in Figure 7. While the permanent structure consists of a self-rigidizing material, type A, the expendable portion features a material, type B, which gradually sublimes in space vacuum, so that ultimately only the rigidized useful structure is left.

Material Problems of the Space Station

The results of an analysis of essential components of a space station are compiled in Figure 8. The components are divided in primary components, i.e., those which apply necessarily to any space station design; secondary components whose use depends on specific missions and are, therefore, optional, and finally, a number of typical elements which are integral parts of primary or secondary components.

The evaluation is presented in tabular form, each column representing a design criterion. The first two criteria are of a general nature: (1) the functional significance of a component and (2) its contribution to the overall weight of the system. Only those components which appear to be significant with regard to either of these two basic criteria are included in the evaluation. The next group of columns lists various material property requirements, such as strength or stiffness as well as functional material characteristics, such as sealing or transmission capability. The third group of columns refers to various critical environments, such as induced temperatures or constituents of the space environment. The significance of the individual criteria is expressed by either dark dots, identifying the most critical topics, or light dots, referring to the less severe.

The relative significance of components with regard to the combined effect of the material problems involved is indicated in the last column, using the same means of distinction. At the same time it is attempted to identify, on the basis of simple statistics, the recurrent critical material properties and environments, as shown on the bottom line.

HTOL-ORBITAL VEHICLE

The same procedure of identifying design requirements has been followed for the HTOL-Orbital Vehicle, which has been described before as an airplane with orbital capabilities. A simplified mission profile is presented in Figure 9. The system can be conceived as a single stage vehicle or a two stage configuration, with various types propulsion systems, mostly using cryogenic fuels and oxidizers. A number of proposed designs are based on the use of an air collection and enrichment system, generating the oxidizer for final boost into orbit during a cruise period at intermediate altitude levels.

Critical Design Environments

The most severe design requirement of the atmospheric portion of the trajectory is the aerodynamic heating at hypersonic velocities. The magnitude of heating of the lift surfaces during exit and re-entry is shown together with the respective flight profiles in Figures 10 and 11. During exit, temperatures depend extensively on the propulsion concept. They may reach 3000°F for supersonic combustion ramjet, or 1600°F for a prolonged time period in case of air collection cruise. During re-entry, considering the long service time of the vehicle, structural temperature limitations placed at 2000°F confine the maneuvering capability to lower altitudes, as indicated by the shaded reentry corrider. The importance of materials is clearly demonstrated by the region between the 2000 and 2500°F line: only a few hundred degrees of added high temperature capability would widen the re-entry corridor remarkably and permit maneuvering at higher altitudes and higher wing loadings.

The metallurgical significance of this temperature range is illustrated in Figure 12 which shows the high temperature usefulness of several typical alloy groups. As indicated by the dotted lines, the use of an apparently high strength is often precluded by oxidation or joining considerations, particularly in the application to minimum gage structures, where conventional technologies no longer apply.

It should be noted that the high temperature material requirements of the HTOL - 0 vehicle differ considerably from boost glide systems as this vehicle will necessarily carry its cryogenic fuels in wing tanks, precluding a hot structure. This calls for a wall design with almost perfect insulation between 2000°F and -432°F, to minimize fuel boil-off. In addition, the structure must be compatible with repeated exposure to space environment. The temperature ranges and secondary requirements of the wing and fuselage structure in comparison with the boost glide vehicle are illustrated in Figure 13.

Critical Components of the HTOL-0 Vehicle

In designing a composite structure integrating stress carrying capability, thermal protection and meteroid protection, there are three basic concepts:

(1) Internal insulation.

(2) Split insulation.

(3) External protection system.

Parametric studies indicate clearly the superiority of the external protection system, in which the stress carrying structure is at cryogenic temperatures. In assessing the weight requirements of thermal protection systems, Figure 14, we find that the metal envelope, needed in view of the poor load carrying capability of high efficiency insulation materials (including vacuum), represents more than 85% of the total weight, even by the use of foil gages. Foil gages, in turn, introduce a set of new problems, as their technology is,in many respects, completely undeveloped. A typical example of foil gage difficulties is illustrated in Figure 15 which refers to oxidation protection. The designer's idea and his basis for weight calculation is identified in the first sketch, showing the cross-section of a foil skin. In order to make it useful for high temperatures it requires oxidation protection, particularly in the case of refractory metals. Diffusion coatings are very effective, yet consume a considerable portion of the foil thickness, so that in some cases hardly any metal is left, not to speak of the unacceptable material embrittlement. If we attempt to avoid this by means of a metallic coating, the added weight defeats the original purpose of foil gage structures. The use of foil gage materials encompasses a number of other problems such as:

availability, uniformity, corrosion, forming, joining, heat treating, handling and repair.

At the leading edges and the nose, temperatures between 3000° and 4500°F may be encountered. The following potential solutions are envisioned:

(1) Refractory metal "boiler plate"

(2) Refractory metal honeycomb

(3) Convective cooling

(4) Pryolitic graphite

(5) Ceramics

(6) Ablation composites

(7) Limited ablation refractories

(8) Transpiration cooling

The features of most of these designs are well known. One promising new concept is the use of "limited ablation refractories", i.e., materials which serve as a hot structure during most of the heat input, yet have the capability of absorbing the peak heating by temporary ablation.

Additional problems are introduced by functional components which are integral parts of the heated structure, such as doors or windows. Window materials must combine high temperature capability with the previously described limited transmission for the orbital portion of the trajectory. In Figure 16 the temperature limitation of a number of optical materials is shown together with the range of light transmission. A fair compromise is obtained with the AKLO-type Pyrex which is essentially limited to visible transmission and withstands temperatures up to 1200°F; its low thermal expansion is an additional advantage. It is recognized that such delicate components as windows will be equipped with a shutter, to be closed during exposure to the most severe environments. The selection of materials should, however, be guided by the "fail-safe" principle, which precludes catastrophic effects in case of the failure of protection devices.

Materials Problems of the HTOL-O Vehicle

The analysis of the design requirements of the various components of the HTOL-O vehicle is again compiled in chart form (Figure 17). Besides the lift surfaces, leading edges, nose and windows, it places a severe problem rating on radomes, thermal protection, meteroid protection, sealant materials and the components of the air collection system.

INTERSTATION SHUTTLE

The requirements of the third type of near-space vehicles, the inter-station shuttle, are essentially covered by the preceding discussion of the space station, and are therefore not further discussed.

FINAL ANALYSIS OF MATERIALS PROBLEMS

Any problem can be solved if enough time is given for research. The initially discussed vehicle schedule, however, introduces stringent time limitations. In order to phase the time aspect in the establishment of priorities, it will be necessary to assess the time required for research and development and to compare this time with the vehicle schedule.

For this purpose the initially established time table is once more shown in Figure 18 for the systems under discussion. In addition to the initial operational capability, it includes the preceding phases of design, manufacture and testing, omitting the earlier systems phases to allow the utmost time for materials research. The target date for its completion has been selected to coincide with the start of final design, except for the space assembly capability for which research can extend to the time of initial operational capability. The resulting target dates for materials research are indicated at the bottom of the time table.

Projection of Research Accomplishments

The time required for research depends on a number of aspects, listed in Table I, together with other priority criteria. The first consideration is the present state of art and the projected rate of effort in terms of funds and manpower. There is further the question of complexity, referring primarily to the number of disciplines involved: The breakdown of a research project in individual tasks, their alignment (including the establishment of a common language between disciplines) and the final synthesis always tend to lengthen the time of research accomplishment. In addition, many topics include tests with certain minimum time requirements, such as creep or corrosion tests. In very advanced topics considerable time is often lost before they gain general recognition, a prerequisite of funding. Finally, one item which can not be overlooked is the time required for administrative contracting procedures, from the requesting and allotment of funds, issuing of the work statement, the preparation and evaluation of proposals to the final contract assignment.

The assessment of the total time requirements and the resulting projection of research accomplishments has been carried out for each component and problem, which has been termed as critical in the preceding test analysis. As an example of this procedure, the evaluation of the space station hull structure, is shown in Figure 19.

Listed at the left are the previously stated requirements, such as ultra light weight materials and designs, joining and deployment techniques, meteroid protection, radiation shielding, development of "integral" design concepts and the development of space assembly technology. The expected research trend during the applicable time period is identified by a triangular pattern, representing the gradual diminishing magnitude or severity of the problem to its final solution.

The same evaluation has been carried out for all major components. The resulting trend charts for meteroid protection systems, thermal protection systems, space assembly, seals and sealants, temperature control surfaces, optical transparencies as well as for the leading edges and nose of the HTOL-0 vehicle are presented in figures 20 through 26.

Priority Classification of Research Problems

In the final evaluation, the problems which meet the deadline for the applying vehicle system are not further considered. The severity of the remaining problems is measured by the number of years by which research exceeds the vehicle target date. Using this information they were subsequently separated into priority categories which are defined as follows:

Priority 1: "5 years behind schedule".

Priority 2: "3 to 4 years behind schedule".

Priority 3: "2 years behind schedule".

Priority 4: "1 year behind schedule".

The problems included in the first priority category are presented in Table II. A further distinction of importance is made by the sequence of topics, which has been based on the remaining priority criteria of Table I.

Table II produces a number of expected items, as well as a few surprising topics. The development of foil gage technology and the related means of oxidation protection rank on top of the list. Further termed as very critical is the integral hull structure for the space station as well as for the HTOL-0 vehicle, which combines the basic structural requirements with those of meteroid protection, thermal protection and radiation shielding. The first priority further lists the materials for emergency re-entry systems, such as unfoldable high-temperature drag devices. In the second priority, Table III, appear such topics as "integral" thermal and meteroid protection systems for service between -423 and 2200°F, required for the HTOL-0 vehicle sealants compatible with a temperature of 2200°F as well as space environment; rotating seals for the space station, required for the separation between the revolving assembly and the zero-g area; the potential contamination of the air in the space station interior resulting from materials; or even the exact establishment of the meteroid environment, which is a prerequisite to materials development rather than a materials problem per se.

This final representation is admittedly rather crude. However, the purpose of this discussion was not solely the identification of critical problem areas, but also the demonstration of the procedures employed in their determination.

These procedures, likewise, are somewhat crude. With some added refinement, however, they may provide a helpful tool in the wise use of our resources in terms of funds, manpower and materials, and ultimately, in the orderly achievement of new aerospace capabilities.

FIG. 1
Potential Schedule for Manned Near-Space Operations

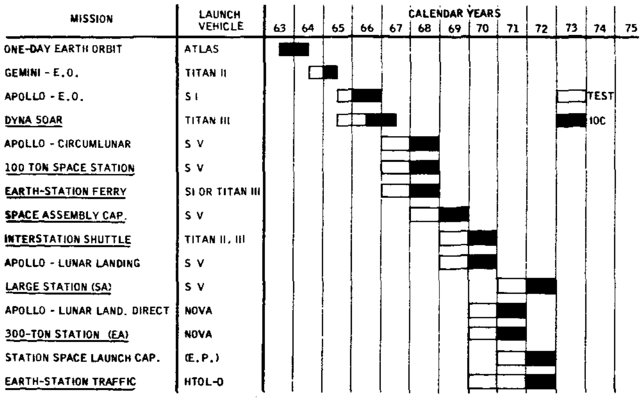


FIG. 2
Composite Wall Design for Manned Space Systems

CONFIGURATION	COMPONENT	FUNCTION	CRITICAL ENVIRONMENT	MATERIAL REQUIREMENTS
	COATING	THERMAL CONTROL	SOLAR HEATING RADIATION METEOROIDS VACUUM	α/ε RATIO RETAINED AT ALL ENVIRONMENTS
	SKIN	METEROID BUMPER RADIATION SHIELD STRUCTURAL SUBSTRATE	METEOROIDS RADIATION SOLAR HEATING VACUUM	LOW ATOMIC NO. HIGH DENSITY COMPATIBILITY WITH COATING
	FILLER INT. BUMPERS	METEOROID ENERGY ABSORBER INSULATOR RADIATION SHIELD	METEOROIDS PRIMARY AND SECONDARY RADIATION	HIGH METEOROID ENERGY ABSORPTION INTERMEDIATE ATOMIC NO.
	INTERNAL WALL	STRUCTURAL RADIATION SHIELD AIR-VACUUM SEAL.	SECONDARY RADIATION INTERNAL PRESSURE HUMAN ENVIRONMENT	HIGH ATOMIC NO. STRENGTH NO CONTAMINATION

FIG. 3
Nomogram for Correlation of Risk Level and Wall Thickness (AL) (HYPERVELOCITY REGION 25,000 FT/SEC)

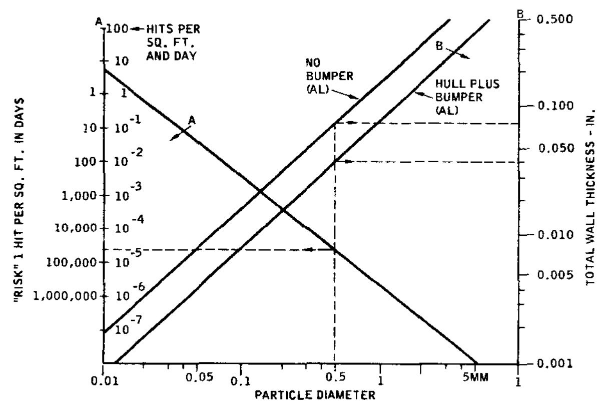


FIG. 4
Weight and Risk Assessment of Various Design Concepts for Meteoroid Protection

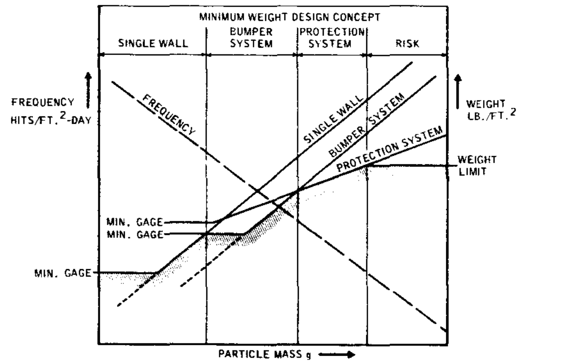
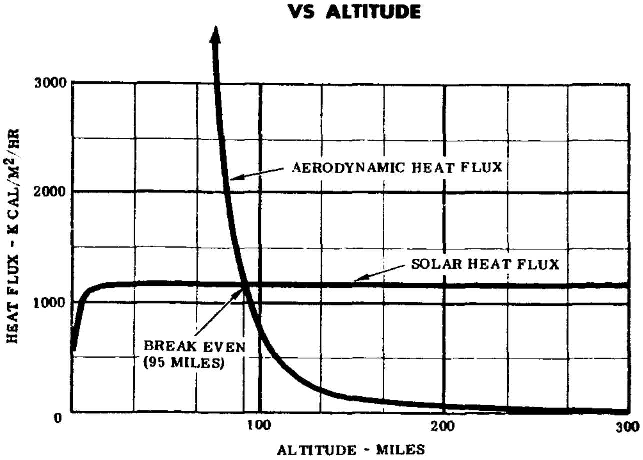
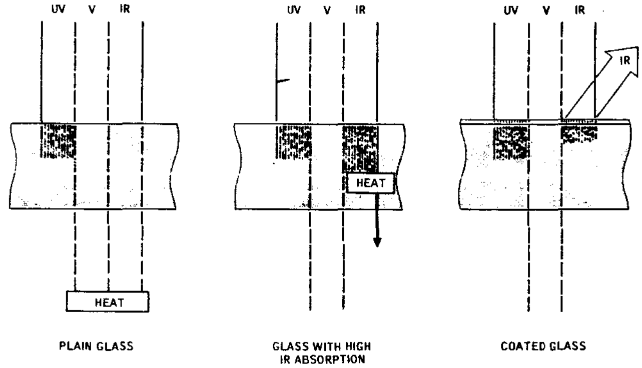


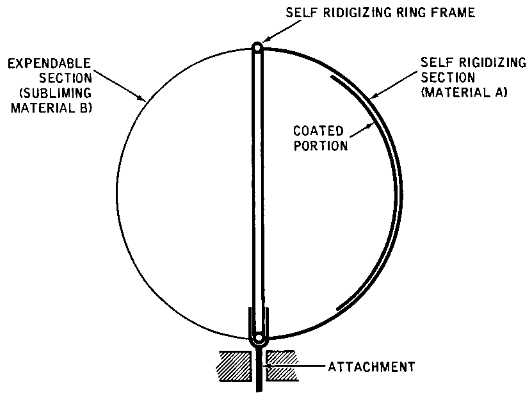
FIG. 5 ORBITING VEHICLE SOLAR & AERODYNAMIC HEAT FLUX VS ALTITUDE



Space Window Spectra FIG. 6



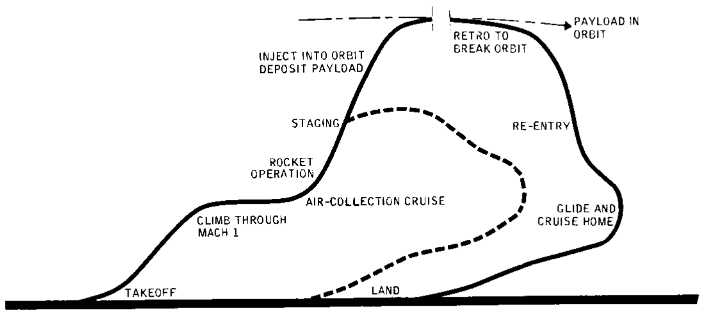
Self-Rigidizing Structure FIG. 7



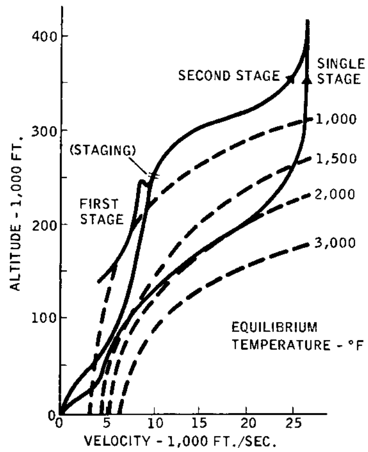
Space Station FIG. 8

	FU	W	STR	STF	ELA	TRA	SLG	FRI	COR	FAT	OTH	LT	HT	VAC	MET	RA	P
PRIMARY																	
HULL STRUCTURE	○	●	●	○			●			○			○		●	●	●
INTERNAL STRUCTURE		●							○		●						
EXTERNAL STRUCTURE		○		●				○		○			○		○		
INFLAT. STRUCTURE	●	○	●	○	●		○						○	●	●	●	●
WINDOWS	●		○			●	○						●	○	●	●	●
DOORS	●			○			●	○							○		
DOCKING DEVICES	○	○		○			●	●						●		○	●
PENETRATORS	○						●				○			○			
SECONDARY																	
POWER PLANTS	●	●					○	○	○	○	●		●			●	●
RADIATORS	●	●	●	●			○		●	○	○		●	○	●		●
SOLAR CELLS	●			○							○		○		●	●	○
SOLAR COLLECTOR		●		●							○		○	○		○	
ANTENNA ASSEMBLY		●		●							○			○		○	
CRYOGENIC TANKAGE	○	●	●	○			●					●	●		●		●
EMERGENCY RE GEAR	●	●	●	○			○				●		●				○
CRITICAL ELEMENTS																	
SEALS	●		○		●		●	○			○			●		●	●
MOVING SURFACES	●						○	●	○	○	○			●		○	○
TEMPERATURE CONTROL SYSTEM	●										○		○	●	●	●	●
METEOROID PROTECTION SYSTEM	●	●	○				●							○	●	○	●
RADIATION SHIELDING	●	●									○				○	●	●
SPACE TPS	●	●					○				○	●	●				○
PRIMARY REQUIREMENTS			●	●	○		●	○					●	●	●	●	

HTOL - Orbital Vehicles Single and Two-Stage Mission Profiles FIG. 9



HTOL - Orbital Vehicles Exit Profiles - FIG. 10



Orbital Lift Re-entry Profiles - FIG. 11

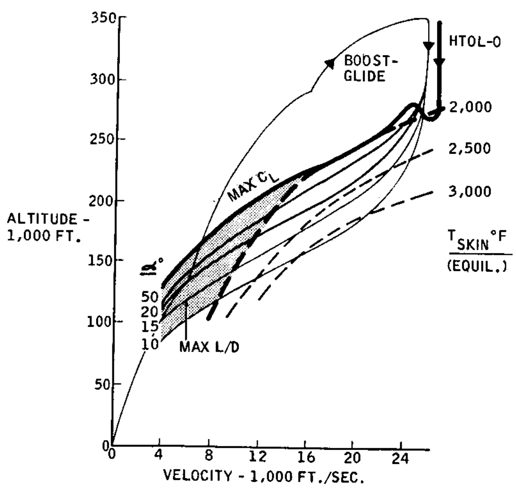


FIG. 12 Limitations of Metals for Heated Structures

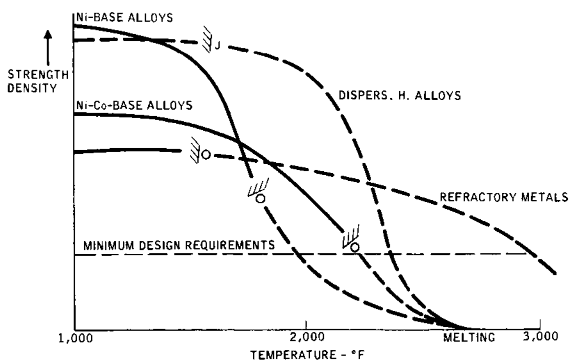


FIG. 13
HTOL-O and Boost-Glide Vehicles Temperature Environments and Associated Problems of Primary Structure

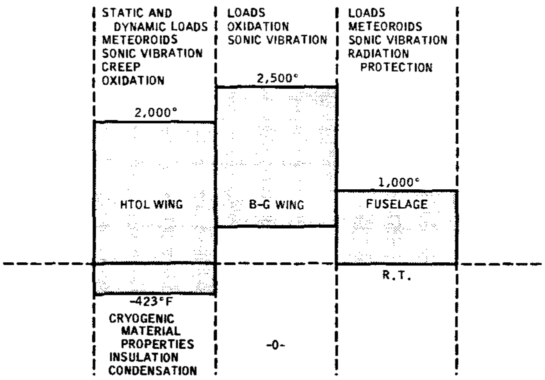


FIG. 14
Weight and Thermal Efficiency of Structural Insulation Systems

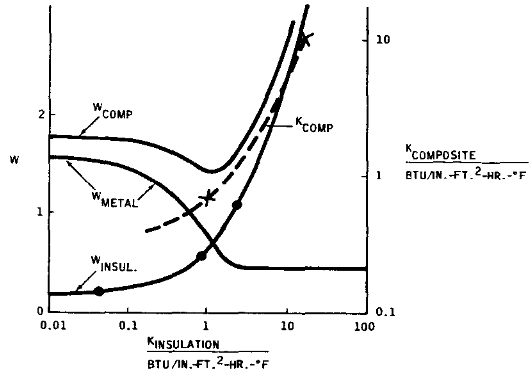


FIG. 15
Oxidation Protection of Foil Metals

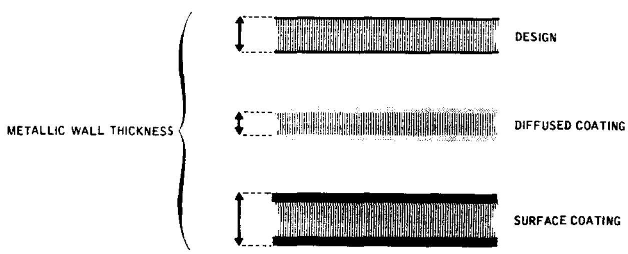


FIG. 16
Transmission and Temperature Limitations of Window Materials

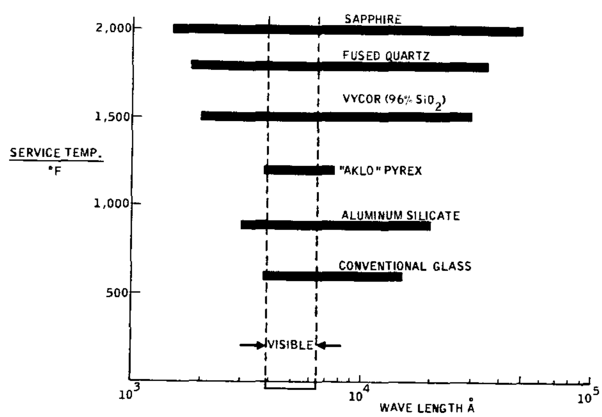


FIG. 17
HTOL - Orbital Vehicle

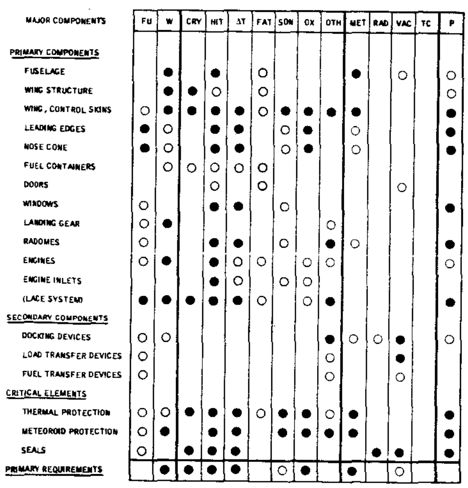


FIG. 18
Determination of Time Limitation for Materials Research

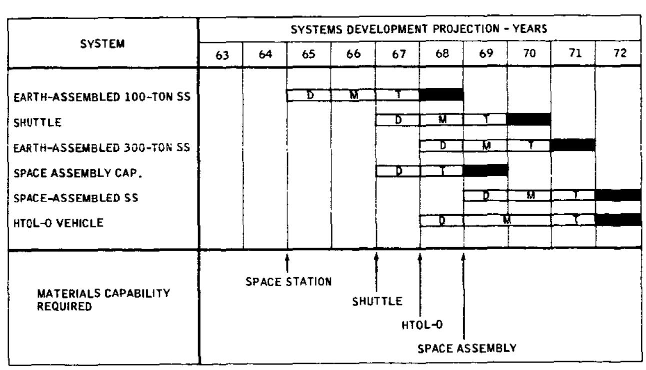


FIG. 19
SS Hull Structure

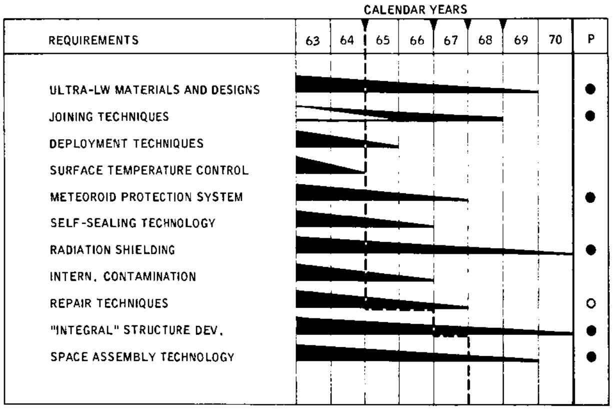


FIG. 20
Meteoroid Protection Systems

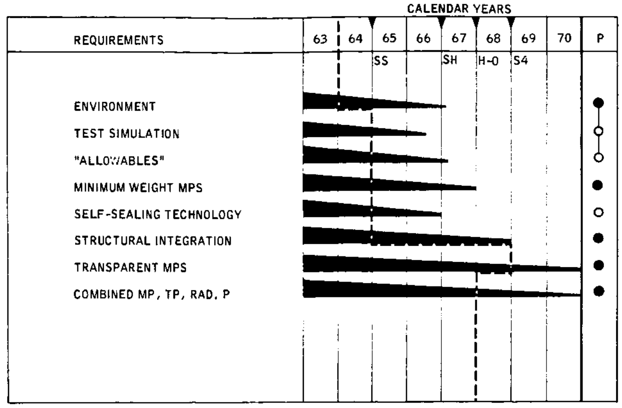


FIG. 21
Thermal Protection Systems

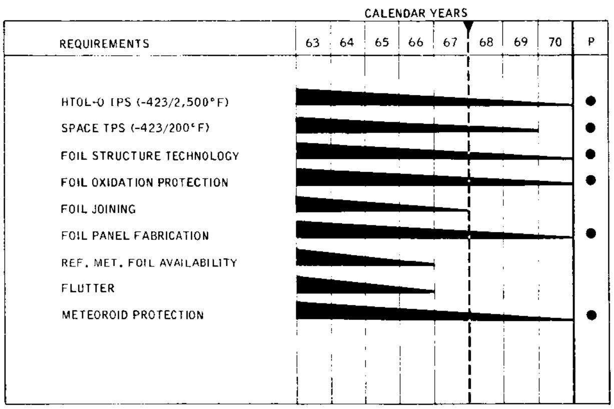


FIG. 22
Space Assembly

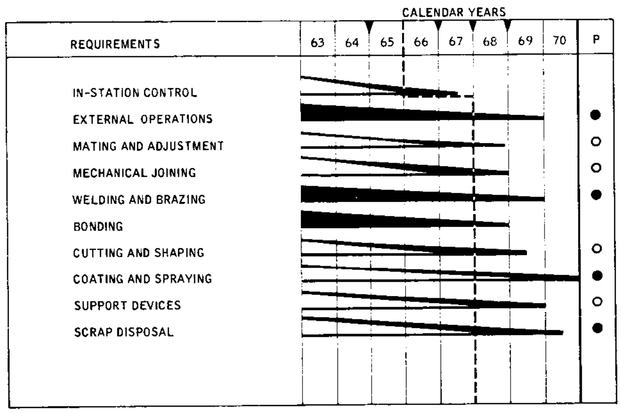


FIG. 23
Seals and Sealants

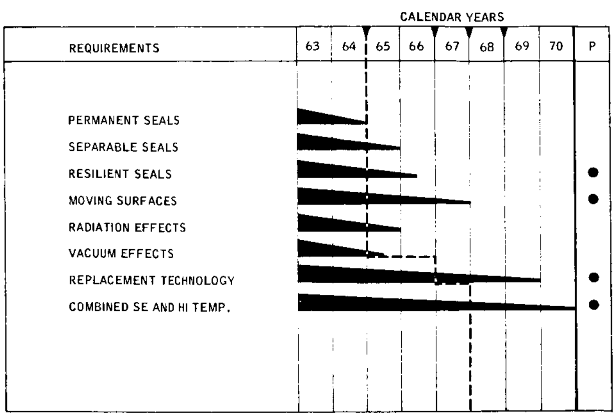


FIG. 24
Temperature Control Surfaces

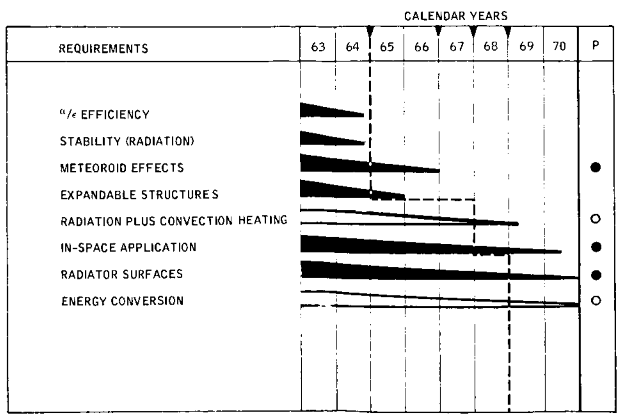


FIG. 25
Optical Transparencies (Windows)

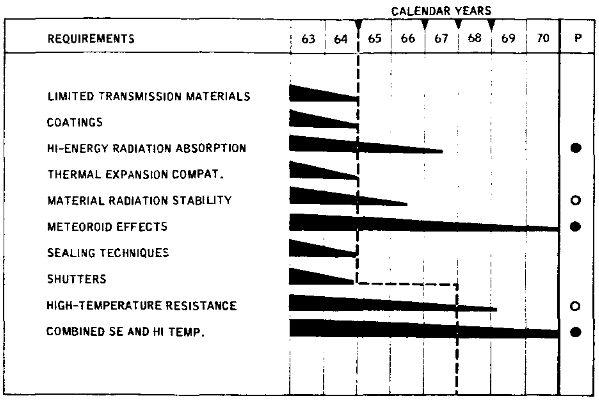
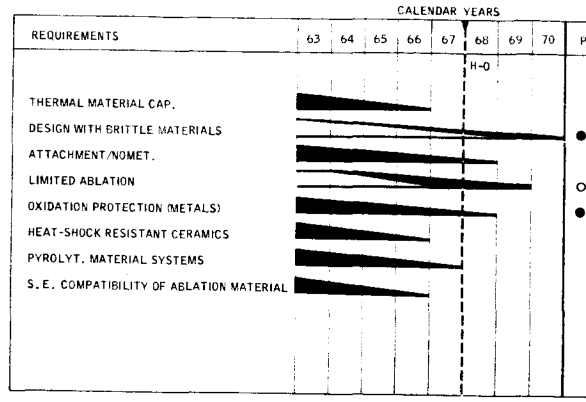


FIG. 26
HTOL-O Leading-Edge, Nose (3,000-4,500 °F)



Priority Criteria TABLE I

- REQUIRED RESEARCH TIME VS. SCHEDULES
  - SYSTEMS TARGET DATES
  - APPLICABLE SYSTEMS DEVELOPMENT PHASE
  - PRESENT STATE OF ART
  - PROJECTED EFFORT RATE
  - COMPLEXITY
  - MINIMUM TIME TESTS
  - RECOGNITION OF PROBLEM
  - CONTRACTING PROCEDURES
- FUNCTIONAL IMPORTANCE
- EXCLUSIVENESS
- TYPICALITY
- AVAILABILITY OF SUPPORT INFORMATION

Priority I - ("5 YEARS BEHIND SCHEDULE") TABLE II

- DEVELOPMENT OF FOIL GAGE TECHNOLOGY
- OXIDATION PROTECTION OF FOIL GAGES
- "INTEGRAL" SPACE HULL MATERIAL SYSTEM
- IN-SPACE HULL-JOINING TECHNIOUES
- MINIMUM WEIGHT METEOROID PROTECTION SYSTEMS
- MINIMUM WEIGHT RADIATION PROTECTION SYSTEMS
- TRANSPARENT METEROID PROTECTION SYSTEMS
- MATERIALS FOR EMERGENCY RE-ENTRY DEVICES
- STRUCTURAL MATERIALS FOR 2,500° - 3,500°F

Priority II - ("3 - 4 YEARS BEHIND SCHEDULE") TABLE - III

- "INTEGRAL" THERMAL/METEOROID PROTECTION SYSTEM (-423°/2,200°F)
- SEALANTS FOR 2,200°F AND SPACE ENVIRONMENT
- ROTATING SEALS FOR SPACE ENVIRONMENT
- IN-SPACE SEAL REPLACEMENT TECHNIQUE
- WINDOW MATERIALS FOR SPACE AND HIGH TEMPERATURES
- STABLE RESILIENT SEALS FOR SPACE ENVIRONMENT
- HIGH E/ρ MATERIALS FOR SPACE STRUCTURES
- SELF-RIGIDIZING, INFLATABLE STRUCTURES
- EXTERNAL SPACE ASSEMBLY TECHNIQUES
- AIR CONTAMINATION FROM INTERNAL MATERIALS
- EXACT METEOROID NEAR-SPACE ENVIRONMENT